

SECTION 1
TACAN

1.1 INTRODUCTION

The tactical air navigation (TACAN) is a military L-band air navigation (NAV) system with data computational capability that determines slant range and magnetic bearing of the Orbiter with respect to a ground station (fig. 1-1). The desired ground station is selected by choosing a channel number (specific frequency) which can be verified by the crew via a periodic transmission of a coded station identification (ID) signal (resulting in an audio tone to the crew). There are three TACAN's on the Orbiter, located in the forward avionics bays. Two types of TACAN's are available for installation; an older Gould model, or a newer Collins model. The solid state Collins model is a mechanically and functionally transparent replacement for the Gould model, however it weighs less, uses DC (versus AC) electrical power, and does not require forced air cooling.

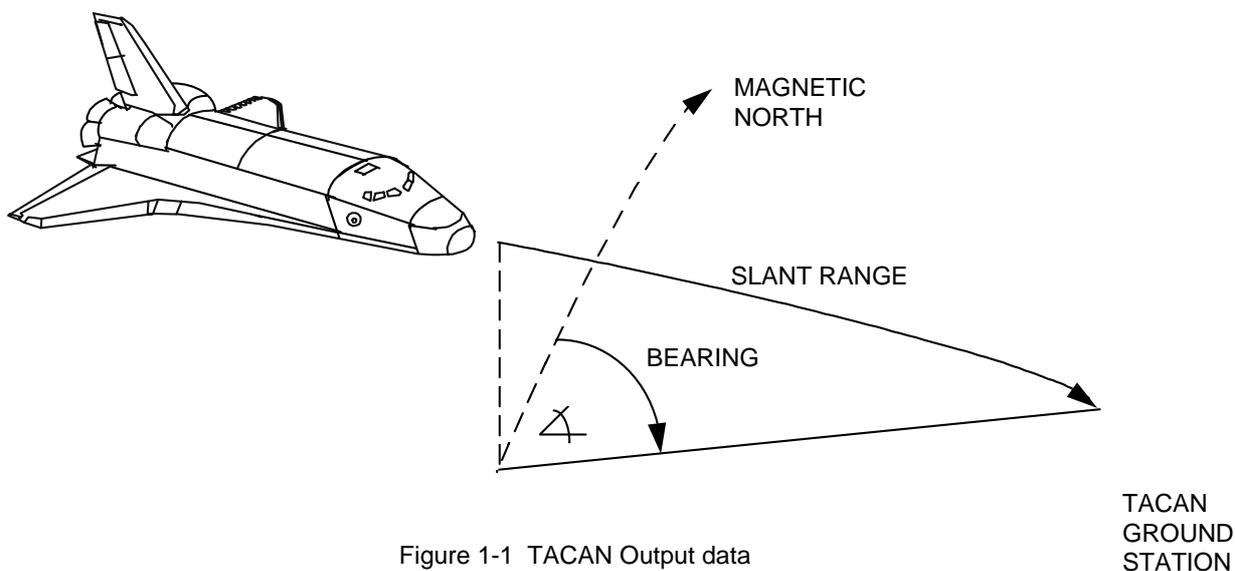


Figure 1-1 TACAN Output data

Each TACAN has an L-band antenna located on the upper and lower portions of the vehicle nose (fig. 1-2). Antenna selection can be made either automatically by computer control or manually by a select switch (fig. 1-3). The TACAN range and bearing data are used to update and increase the accuracy of the NAV state vector during the entry and terminal area energy management (TAEM) flight phases. After transformation to the runway coordinate system, the range and bearing data can also be displayed on the horizontal situation indicators (HSI's). Nominally, the three TACAN's operate simultaneously on 1 of 252 L-band channels. Ultimate control of the TACAN subsystem is via the control head on panel 07 (fig. 1-3). The control

head has a four-position rotary function switch, four thumbwheels for channel selection, and

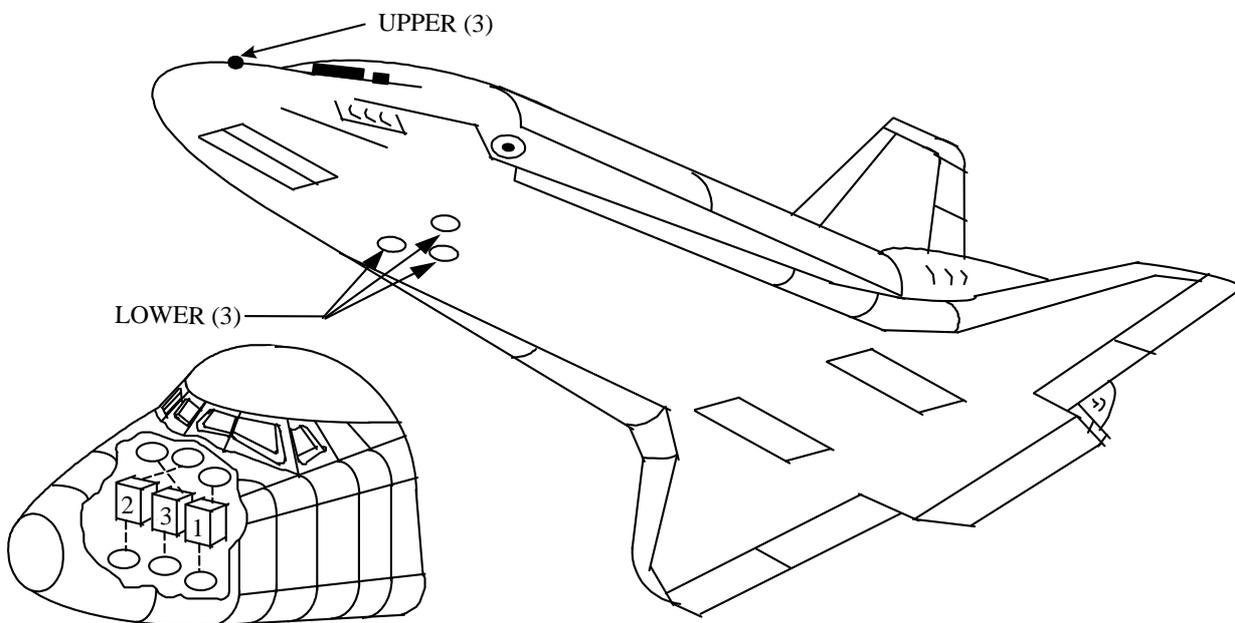


Figure 1-2. -TACAN Antenna locations

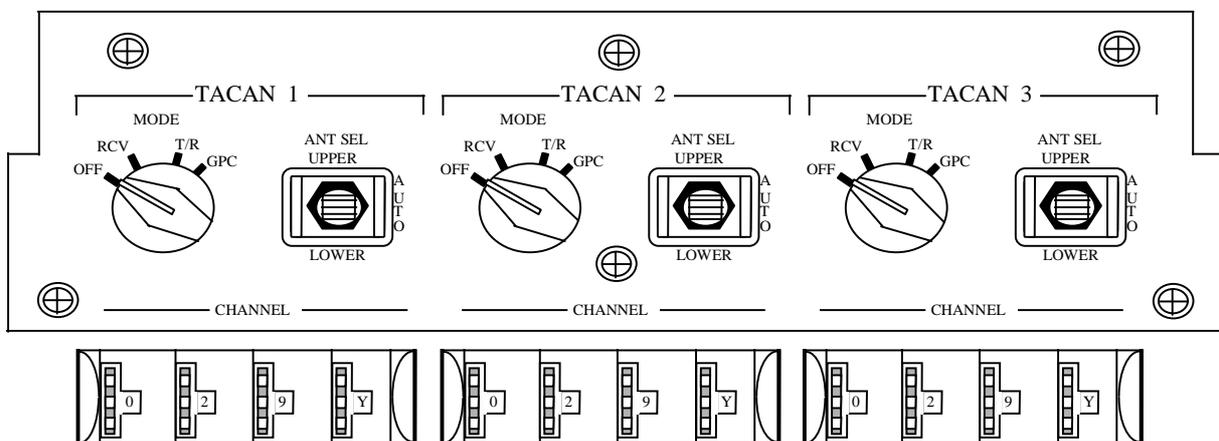


Figure 1-3. TACAN System Controls (Panel 07)

an antenna select switch. The four positions of the function switch are as follows: OFF, RECEIVE, TRANSMIT/ RECEIVE, and general purpose computer (GPC). AC power (Gould model) or DC power (Collins model) is applied to the receiver/transmitter (R/T) when the rotary switch is in any position except OFF (fig. 1-4).

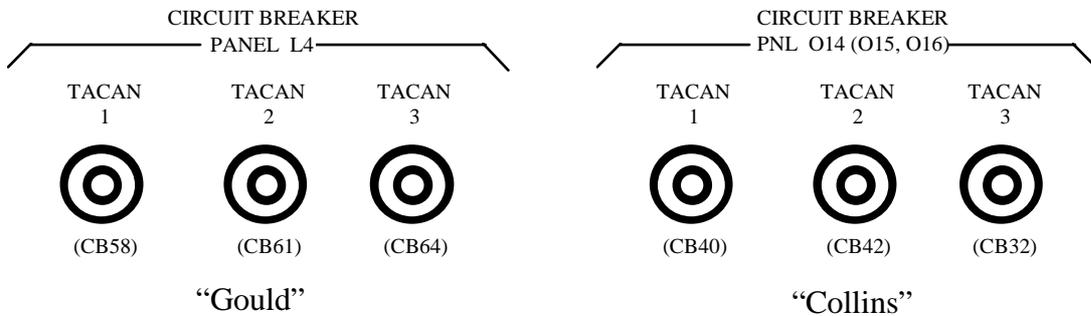
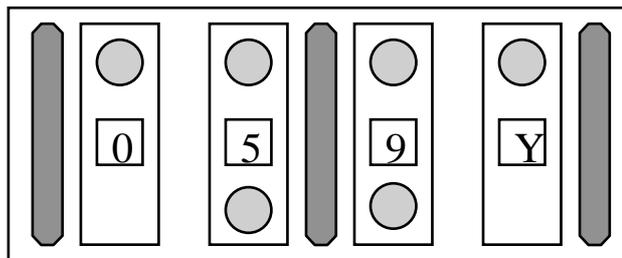


Figure 1-4 TACAN power

When in receive (RCV) or transmit/receive (T/R) positions, the commanded channel is determined by the thumbwheel positions. Channel numbers 1 through 126 are recognized by the R/T. The fourth thumbwheel determines the mode of the ground beacon (X or Y - this essentially doubles the available frequencies from 126 to 252) that completes the channel selection definition. When in RCV mode, the R/T outputs only bearing data. When in T/R mode, the R/T outputs both bearing and range data. When the switch is in the GPC position, the channel number, mode, and self-test command are determined by the GPC. Placing the antenna select switch in the AUTO position allows the GPC to automatically switch antennas, based upon successful range lock-on. The auto function is inhibited by selecting the UPPER or LOWER positions.

NOTE:

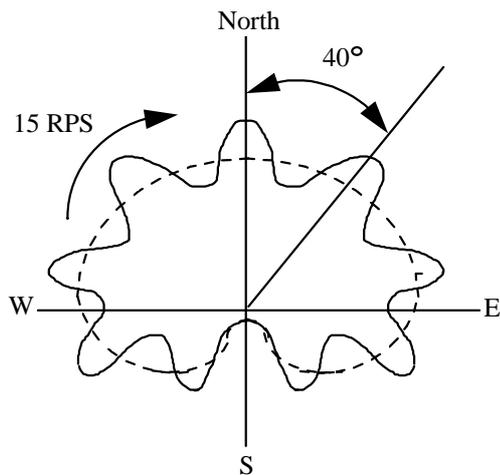
PUSHBUTTONS



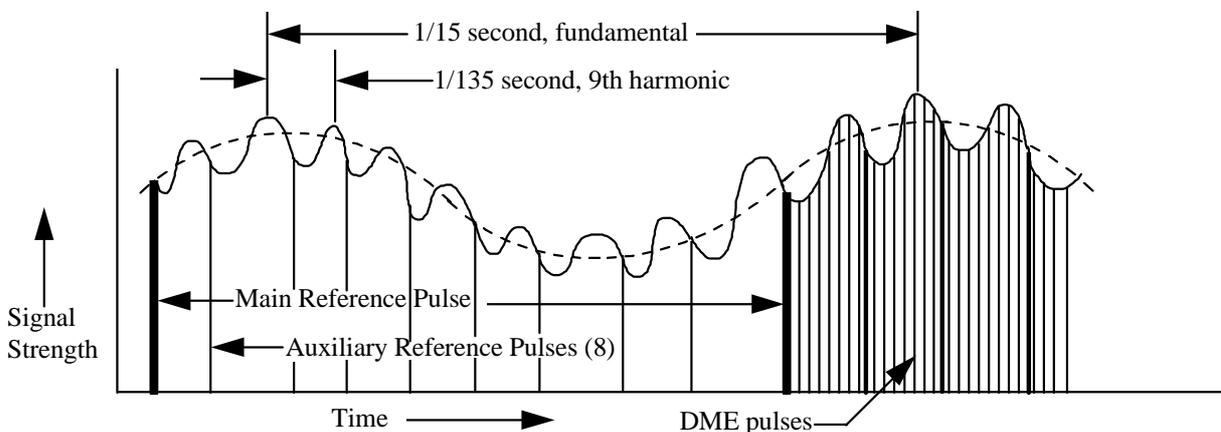
Pushbuttons will replace Thumbwheels when failures occur (mixed fleet). Depressing pushbutton will index the wheel one digit up (for upper button) or down (for lower button). Where only two digits apply, e.g. X/Y (only upper pushbutton), each digit will necessarily be repeated on alternate positions of the wheel; the switch will be wired accordingly to account for all (10) positions.

1.1.1 Ground Station Operation

The ground station transmits on one of 252 (126X and 126Y) preselected frequencies (channels) which correspond to the frequencies the NAV set is capable of receiving. Each of these frequencies is 63 MHz away from a corresponding R/T transmitting frequency. In X-pulse coding, the ground station continuously transmits pulse pairs, spaced 12 μs apart, at a repetition rate of 3600 pulse pairs per sec. The pulse pairs consist of the following:



(a) Radiation pattern - effect of both cylinders



(b) Detected signal - pulse envelope only indicated

Figure 1-5. - Bearing pattern

- a. A main reference pulse group (15 per sec), each consisting of 12 coded pulse pairs
- b. Eight auxiliary reference pulse groups between each main reference pulse group, each consisting of six coded pulse pairs

- c. Reply pulse pairs, transmitted only when the ground station is interrogated by an R/T (representing range data)
- d. Randomly spaced pulse pairs (squitter), that fill in the spaces between the other pulses

Signals transmitted by the ground station are formed into a modified cardioid (fig. 1-5) radiation pattern with rotating lobes and nulls, caused by the rotating antenna system. Thus, at any given point in space, the R/T receives a signal which, due to the rotating lobe and null pattern, is amplitude modulated. The modulation frequencies are 15 Hz and 135 Hz. The main and auxiliary pulse groups are transmitted at fixed intervals in direct relationship to the direction of the main lobe of the radiation pattern, and serve as a reference point from which the R/T extracts bearing data. At intervals of 37.5 ± 2.5 sec, the ground station transmits a three-letter identification signal consisting of a series of pulse pairs in international Morse code (fig. 1-6). Distance measurement equipment (DME) pulses are added to the signal when an interrogation signal is received from the airborne R/T.

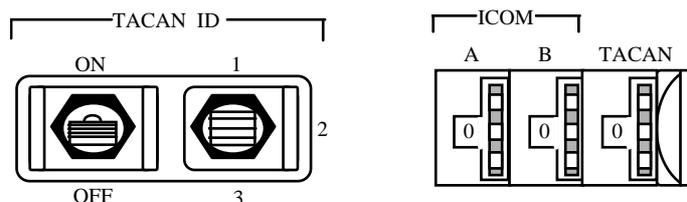


Figure 1-6 TACAN ID controls (Panel 05/09)

1.1.2 TACAN Operation

From the received radio frequency (RF) signal, the R/T computes the bearing of the Orbiter to the ground station relative to magnetic north. This is accomplished by comparing the phase of the bearing modulation of the main and auxiliary reference pulses with the positive inflection of the 15 Hz amplitude modulation.

Because of the horizontal directivity of the ground station antenna radiation pattern, a volume of weak and/or poor RF energy is created. The volume has the shape of an inverted cone whose included angle varies from 60° to 120° - typically 90° (fig. 1-7). It is usually termed the "cone of confusion"; when in the cone, the bearing will generally break lock.

When the mode selector switch is set to GPC or T/R, the R/T will emit interrogation pulses. This causes the specific ground station to respond with DME pulses. The slant range (Orbiter to ground station) is computed by the R/T by measuring the elapsed time from interrogation to valid reply and subtracting known system delays. The maximum range of the R/T is 399.9 n. mi. at a maximum speed of 6500 knots. When approaching a ground station, the range will decrease. Inversely, the range will increase when flying away from the station. When directly

over the ground station, the range will be approximately equal to the altitude. Whenever the R/T is transmitting, a blanking signal is generated which is used by all of the TACAN's to inhibit reception of the transmitted signals.

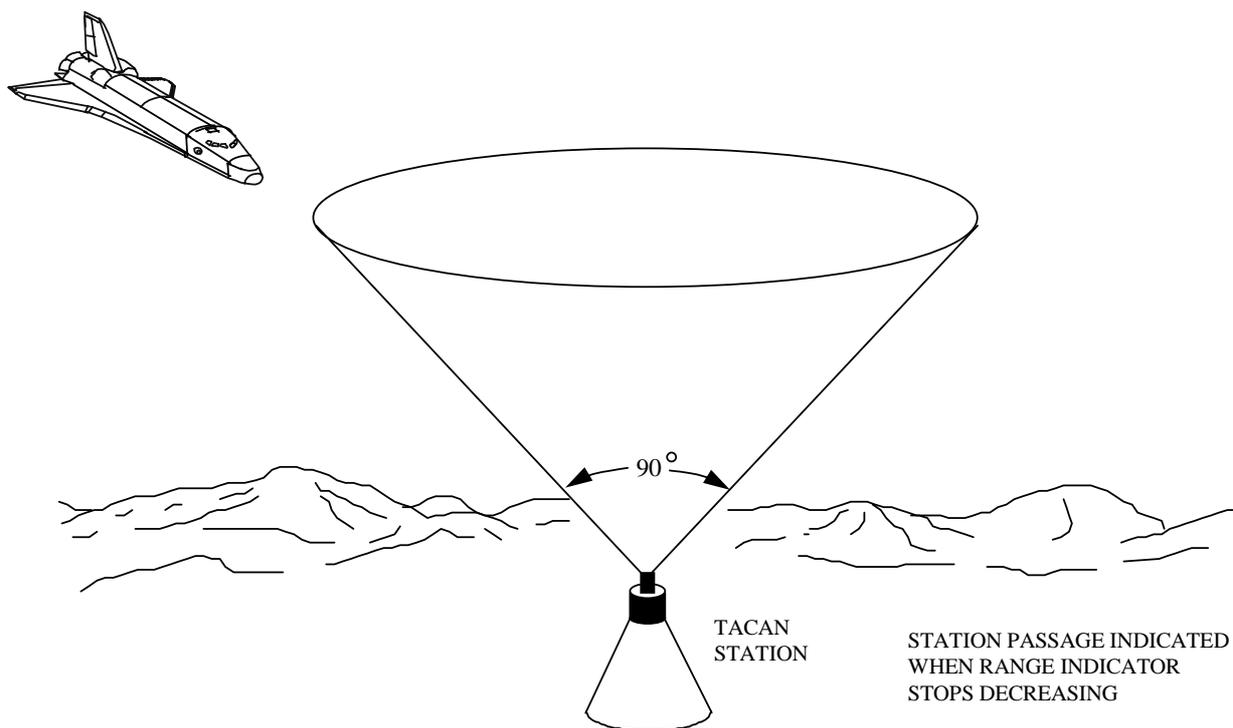
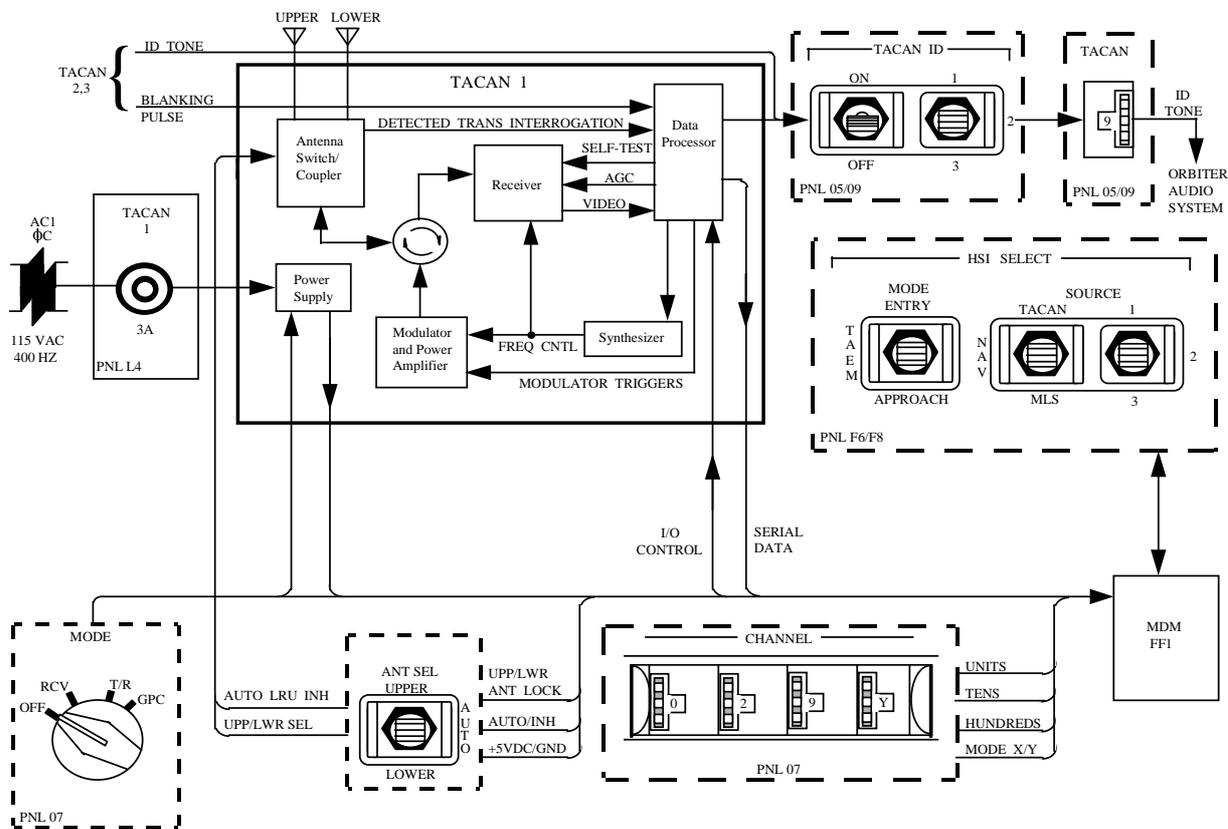


Figure 1-7. - Cone of Confusion

1.2 FUNCTIONAL DESCRIPTION

Figure 1-8 is a block diagram of the TACAN subsystem. A brief functional description follows.

- a. Receiver - The receiver amplifies the received RF pulses, performs frequency conversions, and detects the envelope of the converted signal. The signal is amplified and demodulated to obtain the composite received video signal, and is applied to the data processor for processing. The data processor decodes the pulses to recognize valid pulse pairs and times a counter to provide bearing computation. An internal blanking pulse, to blank out the receiver and other TACAN's, is also generated. The receiver composes the signals into a serial data word.



NOTES: 1. AUTO POSITION OF SELECT SWITCH ALLOWS GPC SELECTION OF ANTENNA BASED UPON RANGE LOCKON (PERIODIC SWITCHING AT AN 11 SECOND INTERVAL).

Figure 1-8. - TACAN Block Diagram

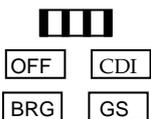
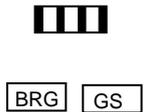
- b. Data processor - The data processor is the brain of the R/T. It decodes all command signals and produces the necessary signals to control other modules in the R/T. The data processor decodes the selected channel frequency and sends that data in coded form to the receiver and frequency synthesizer. In addition, the data processor sends a transmitter trigger pulse to initiate transmitter pulses. The transmitter RF signals are applied directly through the coupler-filter switch.
- c. Transmitter - The signal to be transmitted is generated by the power oscillator/modulator, which pulses a continuous wave from the frequency synthesizer. The frequency is determined by the data processor. The pulsed output signal from the power oscillator/modulator is applied through the circulator and coupler-filter switch to the antennas. The transmitting frequency depends on the selected operating mode and channel, and is 1 of 252 frequencies.

The TACAN R/T will break lock (return to a search mode) under the following conditions:

- a. Whenever the ground-to-air transmissions are obstructed by natural terrain or aircraft attitude (loss of signal (LOS)).
- b. Bearing will periodically break lock in the cone of confusion.
- c. Whenever the channel (frequency) or operating mode is changed.

When break lock occurs, the data is marked invalid and the flags on the HSI will be displayed, corresponding to the parameter (range or bearing) that has broken lock. (Table 1-1 illustrates the flag conditions.) In addition, the horizontal situation display (HSD) and the guidance, navigation, and control (GNC) entry systems summary display will display M's.

TABLE 1 - I. - HSI - FLAG FUNCTIONS

| Indication | Cause | Effects |
|---|---|---|
|  | <ul style="list-style-type: none"> ▪ HSI voltage low | <p>BRG pointers, compass card, range & course indicators frozen.</p> <p>If voltage less than 17 V, range flags may appear.</p> |
|  | <p>IF: </p> <ul style="list-style-type: none"> ▪ TACAN channel not in software (site lookup) | <p>Bearing pointers and CDI frozen, GSI out of view</p> |
|  | <p>IF: </p> <ul style="list-style-type: none"> ▪ TACAN range or bearing invalid (Commfault/lock-on) | <p>Bearing pointers frozen, GSI out of view, CDI may be in error.</p> |
| | <p>IF: </p> <ul style="list-style-type: none"> ▪ MLS range or bearing invalid (Commfault/lock-on) | |
|  | <p>IF: </p> <ul style="list-style-type: none"> ▪ BARO altitude invalid (air data RM) | <p>GSI out of view, constants or NAV altitude used in bearing and range calculations.</p> |
| | <p>IF: </p> <ul style="list-style-type: none"> ▪ MLS elevation invalid (Commfault/lock-on) | <p>GSI out of view, one of two elevation constants used in bearing and range calculations: prior to final flare- 0.33281 MM; after final flare- 0 MM.</p> |
|  | <ul style="list-style-type: none"> ▪ HSI internal failure; SERVO AMP feedback miscompare | <p>Compass card and course pointer frozen (bearing pointers continue to drive but may be in error).</p> |

1.3 DISPLAYS

All the displays used by the crew for TACAN monitoring and systems management are documented and described in detail in the Data Processing Systems (DPS) Dictionary (JSC-48017). A brief summary of these displays is documented below.

1.3.1 Horizontal Situation Display (HSD)

The HSD allows control and assessment of the NAV filter performance, as well as the three TACAN subsystems. The display is available in OPS 3 and OPS 6 and is called by a SPEC 50 PRO on any CRT (Figure 1-9). The TACAN data presented in the lower right corner of the display represents actual slant range and magnetic bearing if ABS (Item 34) is selected. If DELTA (Item 35) is selected, the difference between the TACAN measured data and the navigation derived data is displayed. In addition, the current status of the TACAN's is displayed adjacent to the parameter. If TACAN self-test has been initiated by redundancy management (RM), the word "TEST" is displayed.

The HSD contains the edit controls which determine usage of TACAN data by navigation. For TACAN, the edit occurs on a parameter-by-parameter basis.

RESID = RESIDUAL = SELECTION FILTER_{OUT} - NAV FILTER_{est}

$$\text{RATIO} = \frac{\text{RESIDUAL}}{\text{MAX ALLOWABLE RESIDUAL}} < 1.0$$

A RATIO greater than 1.0 will cause a:

- NAV EDIT TAC message to be generated on all active Cathode Ray Tubes (CRT's)
- Class 3 alarm
- "down arrow" to be displayed in the RATIO status column

The EDIT condition is reset if two out of four ratio checks are < or = 1.0.

AUTO = Automatic = allows use of TACAN data that passes the EDIT function

INH = Inhibit = does not allow incoming TACAN data to update the state vector

FOR = Force = overrides the EDIT function; forces TACAN data into the state vector

The HSD also provides the capability to select either of two TACAN channels associated with a selected landing area. When a landing area is selected via Item 2, the primary TACAN channel is automatically selected. Subsequent execution of Item 5 alternately selects the secondary

and primary TACAN channels. The selected channel ID is displayed next to Item 5. Selection of the primary or secondary runway within a landing area (Item 3 and 4) does not affect the selected TACAN channel. The TACAN mode switch must be in the GPC position in order for Item 5 to actually change the channel to which a line replacement unit (LRU) is tuned. If the mode switch is in any other position, the TACAN LRU will be tuned to the channel selected on the TACAN channel thumbwheels, regardless of the channel selected by Item 5 on the HSD. Table 1-II shows a typical list of available landing areas and their associated primary and secondary channel ID's.

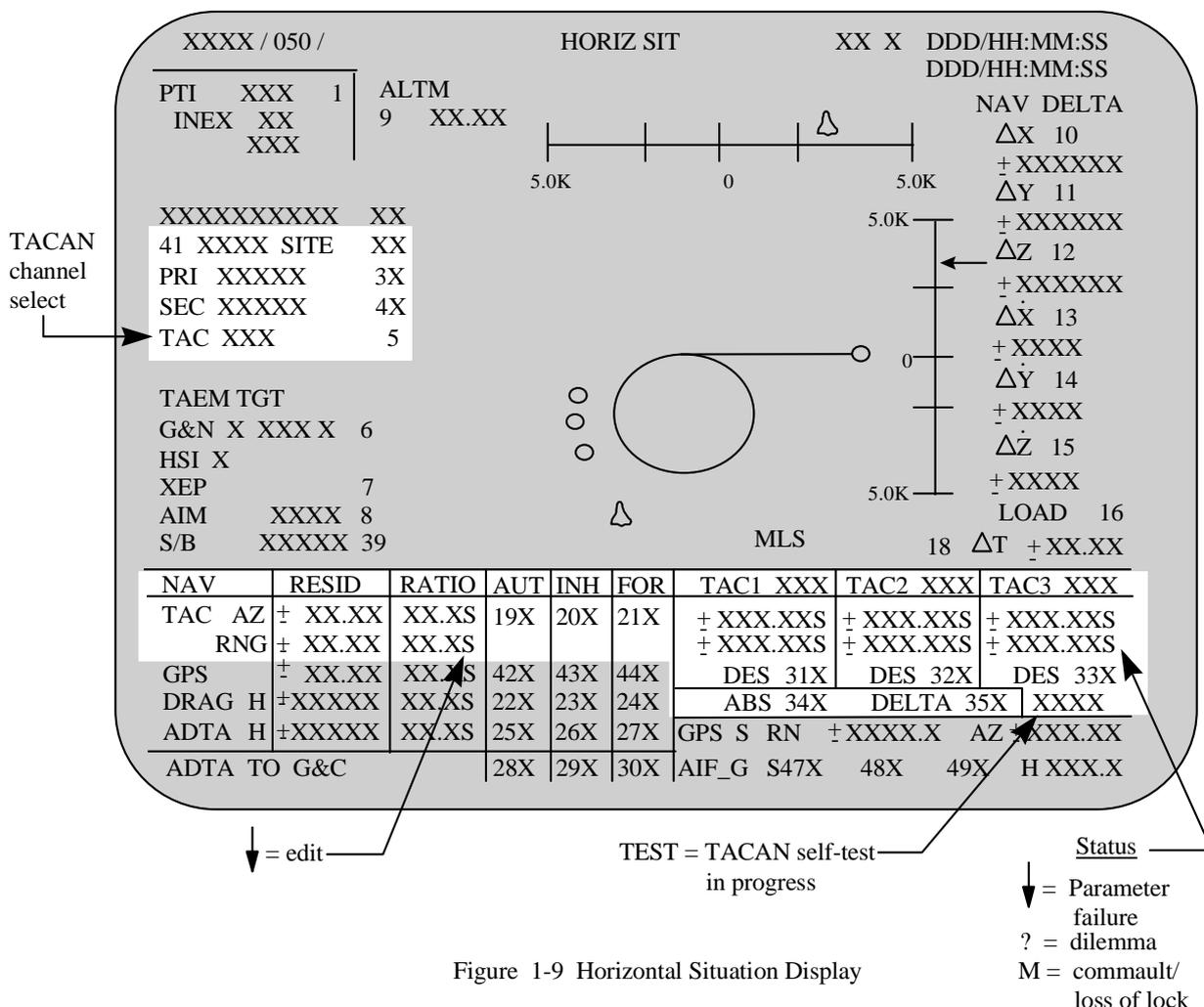


Figure 1-9 Horizontal Situation Display

TABLE 1-II.- LAND SITE DATA CHART

ETR low inclination

| A r e a | Location | RWY | TACAN | | MLS | Rwy Length (feet) |
|------------------|------------------------|--------|------------|-----------|-----|-----------------------------|
| | | | (PRI) | (SEC) | | |
| 1 | KSC | KSC 15 | TTS59Y | COF97 | 8 | 15000 |
| | | KSC 33 | TTS59Y | COF97 | 6 | 15000 |
| 2 | Dakar Amilcar CBRL | YOF 01 | YOF68 | YF78(DME) | 8 | 11500 |
| | | AML 02 | YOF68 | YF78(DME) | | 10700 |
| 3 | | MRN 21 | MRN100 | AOG23 | | 11840 |
| | | MRN 03 | MRN100 | AOG23 | | 11840 |
| 4 | Las Palmas Kinshasa | GDV 03 | GDV76(DME) | | | 10200 |
| | | KIN 25 | KW70(DME) | | | 15500 |
| 5 | Darwin | DDN 29 | DN84 | | | 11040 |
| | | DDN 11 | DN84 | | | 11040 |
| 6 | Edwards | EDW 22 | EDW111 | PMD92 | 8 | 15000 |
| | | EDW 17 | EDW111 | PMD92 | 6 | 15000 |
| 7 | Honolulu | HNL 08 | HNL100 | CKH86 | | 12000 |
| | | HNL 26 | HNL100 | CKH86 | | 12000 |
| 8 | Northrup | NOR 17 | SNG-121 | HMN92 | 8 | 15000 |
| | | NOR 23 | SNG-121 | HMN92 | | 15000 |
| 9 | Edwards | EDW 23 | EDW111 | PMD92 | | 14880 |
| | | EDW 15 | EDW111 | PMD92 | | 15000 |
| 10 | Edwards Northrup | EDW 04 | EDW111 | | | 15000 |
| | | NOR 05 | SNG-121 | | | 15000 |
| 11 | Diego Garcia | JDG 31 | NKW57 | | | 12040 |
| | | JDG 13 | NKW57 | | | 12040 |
| 12 | Riyadh | RIY 01 | RIY92 | EKI80 | | 10750 |
| | | RIY 30 | RIY92 | EKI80 | | 11780 |
| 13 | Guam | GUA 06 | UAM54 | GUM58 | | 11220 |
| | | GUA 24 | UAM54 | GUM58 | | 11220 |
| 14 | Souda | SUD 29 | SUD28 | SUD49 | | 11810 |
| 15 | Hao Hoedspruit | HAO 12 | HA80 | | | 11090 |
| | | HDS 18 | HS73 | | | 13120 |

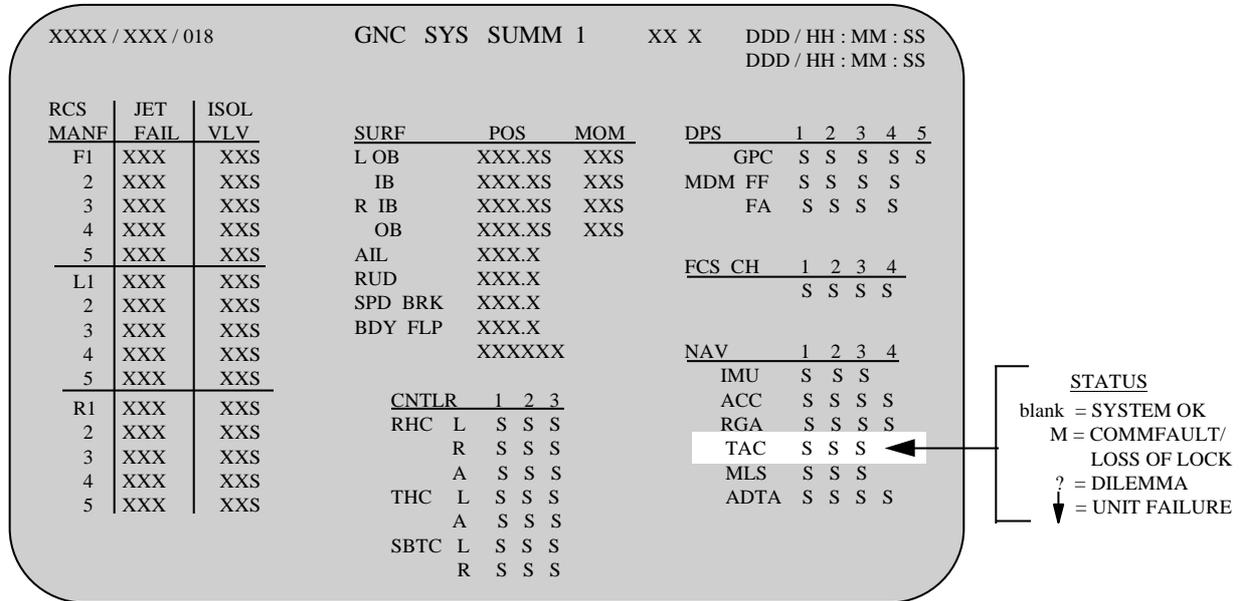


Figure 1-10 GNC SYSTEM SUMMARY 1 display

1.3.2 GNC Entry Systems Summary (GNC SYS SUMM)

The Systems Summary display (Figure 1-10) provides overall monitoring of the GNC systems and supports the GNC caution and warning (C&W) system. This display is available in OPS 3 or OPS 6 by using the SYS SUMM key on the crew's keypad or by a SPEC 18 PRO on any CRT.

1.4 TACAN CONTROL WORD

The multiplexer/demultiplexer (MDM) plays a major role in the formatting of the TACAN control register format. In the GPC mode, the MDM formats the control word from the GPC inputs. While in the T/R mode, it uses inputs from the control head thumbwheels. The control word contains channel number, X/Y mode, T/R or RCV mode, and the self-test

| TACAN Control Register Format (MDM-to-TACAN) | | |
|--|-------------------|--------------------|
| <u>Bit no.</u> | <u>Function</u> | <u>Comment</u> |
| 0 | Sync | Always 1 |
| 1 | Spare | * |
| 2 | X/Y mode | 1 = Y, 0 = X |
| 3 | A/A | 0 = Not air to air |
| 4 | T/R, R | 1 = T/R, 0 = R |
| 5 | Spare | * |
| 6 | Spare | * |
| 7 | Channel selection | 1 Units |
| 8 | Channel selection | 2 Units |
| 9 | Channel selection | 4 Units |
| 10 | Channel selection | 8 Units |
| 11 | Channel selection | 10 Tens |
| 12 | Channel selection | 20 Tens |
| 13 | Channel selection | 40 Tens |
| 14 | Channel selection | 80 Tens |
| 15 | Self test | 1 = Test |
| 16-31 | Set to 0 by MDM | |

* Don't care bits in hardware.

| TACAN Discrete Word Format (TACAN Control Head-to-MDM) | | |
|--|---------------------|----------------|
| <u>Bit no.</u> | <u>Function</u> | <u>Comment</u> |
| 0 | Channel Selection | 100 hundreds |
| 1 | Power On discrete | 1 = Power On |
| 2 | X/Y mode | 1 = Y, 0 = X |
| 3 | Test Status Monitor | 1 = Good |
| 4 | T/R, R | 1 = T/R, 0 = R |
| 5 | Radar Altimeter | |
| 6 | Radar Altimeter | |
| 7 | Channel selection | 1 Units |
| 8 | Channel selection | 2 Units |
| 9 | Channel selection | 4 Units |
| 10 | Channel selection | 8 Units |
| 11 | Channel selection | 10 Tens |
| 12 | Channel selection | 20 Tens |
| 13 | Channel selection | 40 Tens |
| 14 | Channel selection | 80 Tens |
| 15 | Spare | |

Note: For TACAN channel numbers ≥ 100 , bits 11-14 of the control word differ from bits 11-14 and 0 of the discrete word.

Figure 1-11 TACAN Control Word

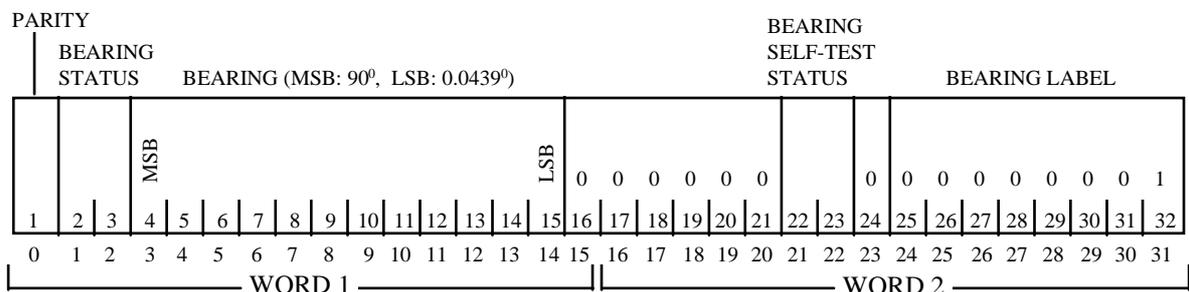
command (fig. 1-11). An illustration of this control word can be found in the "TACAN Control Register Format" and "TACAN Discrete Word Format" figures in the NAVAIDS FSSR (STS-83-0014F, Volume 1).

1.5 TACAN OUTPUT DATA

Range/bearing data - A simplex serial data channel exists between the TACAN unit and the MDM input/output module (IOM) for transmission of range and bearing information to the MDM IOM. Both range and bearing are transmitted as 32-bit words, 3.162 ms/word, 96 us/bit (11 kHz ± 3.5 kHz bit rate). The bits appear in ternary-level, return-to-zero format. A logical 1 consists of a positive signal level during the first half of the bit time; a logical 0 consists of negative signal level during the first half of the bit time. In both cases, the second half of the bit time consists of a zero-volt level.

Bearing words are transmitted to the MDM IOM at a maximum rate of 15 words per second; range words are transmitted at a maximum rate of 27 words per second. The range and bearing words are transmitted independently of one another. The TACAN unit ensures that for near simultaneous use of the serial data channel for range and bearing transmissions, a gap of at least four bit times exists between range and bearing words. The range and bearing words are transmitted by the TACAN unit without initiation by the MDM IOM. The MDM IOM accepts the words whenever they are transmitted. The format of the bearing word is illustrated in the "TACAN-to-MDM Bearing Word" figure in the NAVAIDS FSSR (STS-83-0014F, Volume 1), as well as in the "MDM-to-GPC Bearing Word" illustration in the same document (figure 1-12). There are slight differences in the bit numbering between these words, but the information transmitted by the words is the same.

BEARING WORD (TACAN UNIT-TO-MDM IOM)



Note: This word is actually transmitted bit 32 first.

Figure 1-12 Bearing Word

BIT

0 Parity. This bit assigns odd parity to all 32 bits

1, 2 Bearing Status.

- 00 - Valid bearing, less than 180 deg (track or short-term memory)
- 11 - Valid bearing, 180 deg or greater (track or short-term memory)
- 10 - Invalid bearing (no computed data; search mode)
- 01 - TACAN built-in test equipment (BITE) failure (bearing)

When the TACAN unit has locked onto a beacon, bearing measurements are processed at the rate of 15 samples per second. The TACAN unit is referred to as being in track mode. (Initial lock-on for bearing determination requires up to 3 seconds).

When the TACAN unit has been in bearing track mode for 15 seconds or more, LOS from the beacon does not necessarily result in a halt of signal transmission (bearing words) from the TACAN unit to the MDM IOM. During this time period, the TACAN unit is referred to as being in short-term memory mode. At the end of the short-term memory period, the TACAN unit enters long-term memory mode for a period of three to eight seconds. Bearing words continue to be transmitted to the MDM IOM with invalid data indication (in the absence of incoming signals). At the end of the long-term memory period, the TACAN unit enters search mode. Bearing words are no longer transmitted to the MDM IOM.

3-14 Bearing. These 12 binary bits define bearing in magnitude format. The high order bit (bit 3) has a weight of 90 degrees. The low order bit (bit 14) has a weight of 0.0439 degrees. Bits 1 and 2 above establish whether or not the value of 180 degrees should be added to the information contained in bits 3-14.

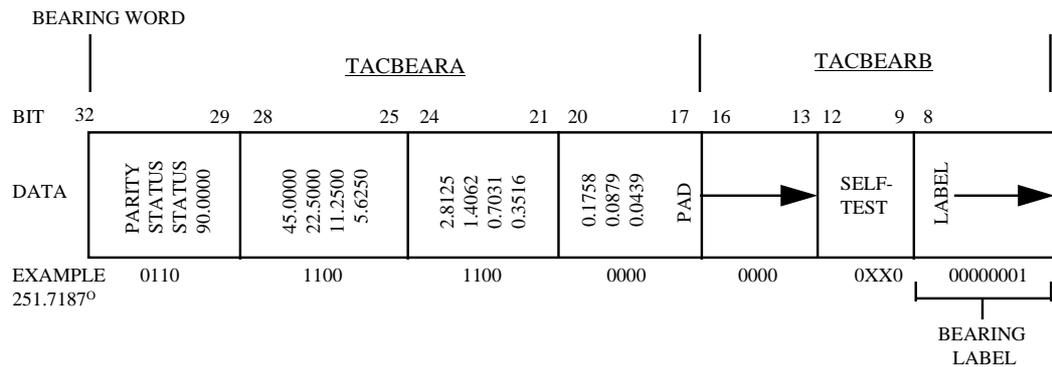
15-20, 23 These bits are always logical 0's.

21, 22 Bearing self-test status. These two bits define whether or not the TACAN unit is in self-test mode:

- 00 - Normal operation
- 11 - In bearing self-test
- 10,01 - Undefined

When the TACAN unit is successfully performing self-test, it continuously transmits a valid bearing value between 177.5 and 182.5 degrees. It is required that IOP or government-supplied equipment (GSE) units conducting self-tests verify that the bearing is within the above limits. Bearing values outside of the limits are to be interpreted as a failure of TACAN bearing circuitry.

24-31 Bearing label. Bits 24-31 define the bearing label. The label serves to allow discrimination between bearing and range words.

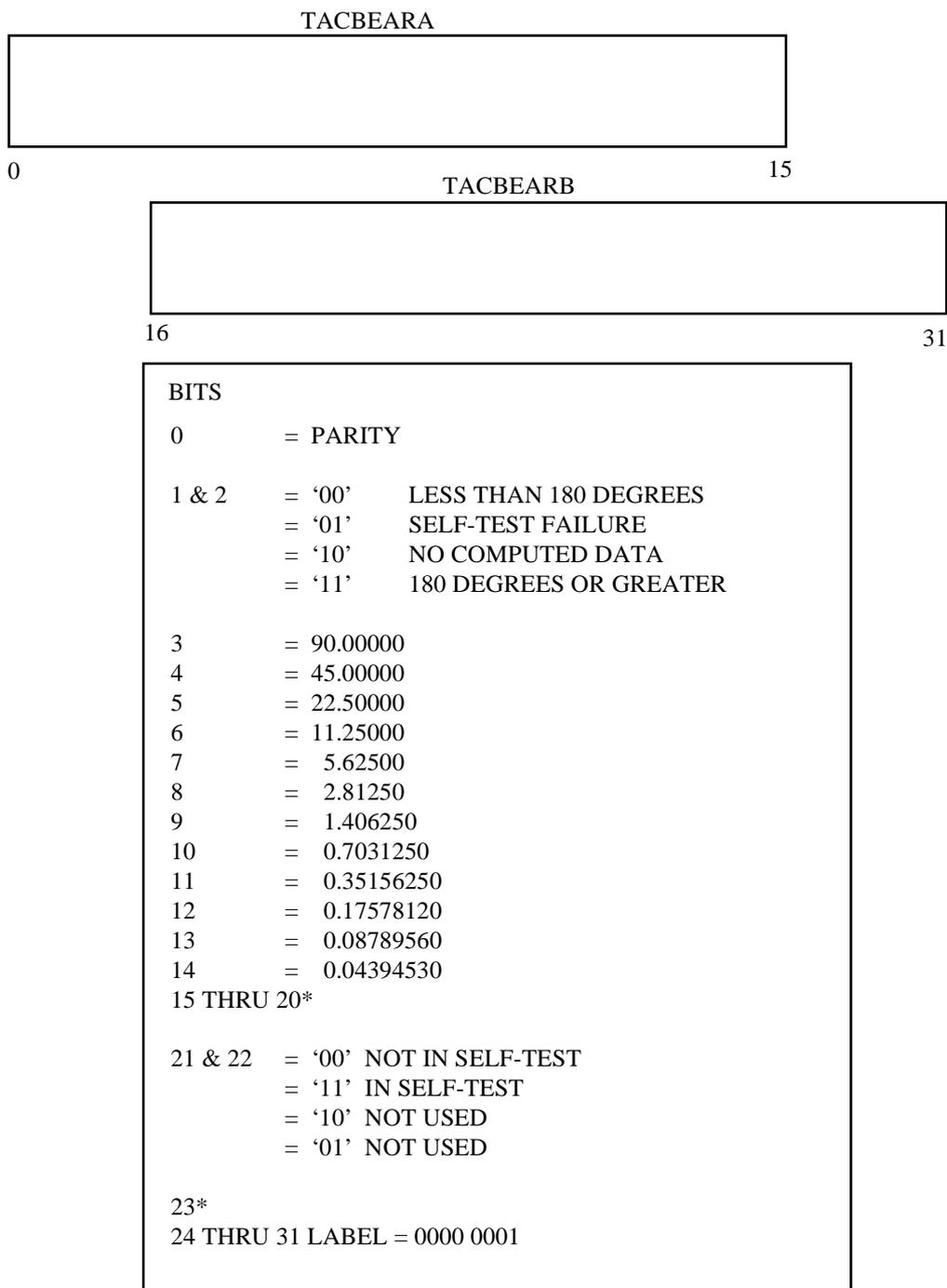


| 31 | 30 | 11 | 10 | BIT |
|----|----|----|----|---|
| 0 | 0 | — | — | LESS THAN 180 DEGREES (TRACK OR SHORT-TERM MEMORY) |
| 0 | 1 | — | — | SELF-TEST FAILURE |
| 1 | 0 | — | — | NO COMPUTED DATA (NO OR UNUSABLE SIGNAL) |
| 1 | 1 | — | — | 180 DEGREES OF GREATER (TRACK OR SHORT-TERM MEMORY) |
| — | — | 0 | 0 | NOT IN SELF-TEST |
| — | — | 1 | 1 | IN SELF-TEST |
| — | — | 0 | 1 | NOT DEFINED |
| — | — | 1 | 0 | NOT DEFINED |

NOTE: BIT 32 IS ALWAYS SET FOR ODD PARITY
BINARY VALUES ROUNDED

Figure 1-13 TACAN -to-MDM Bearing Word

BEARING WORD



* DON'T CARE BITS IN H/W.

Figure 1-14 MDM-to-GPC Bearing Word

The format of the range word is illustrated in the "TACAN-to MDM Range Word" figure in the NAVAIDS FSSR (STS-83-0014F, Volume 1), as well as in the "MDM-to-GPC Range word illustration in the same document (figure 1-15).

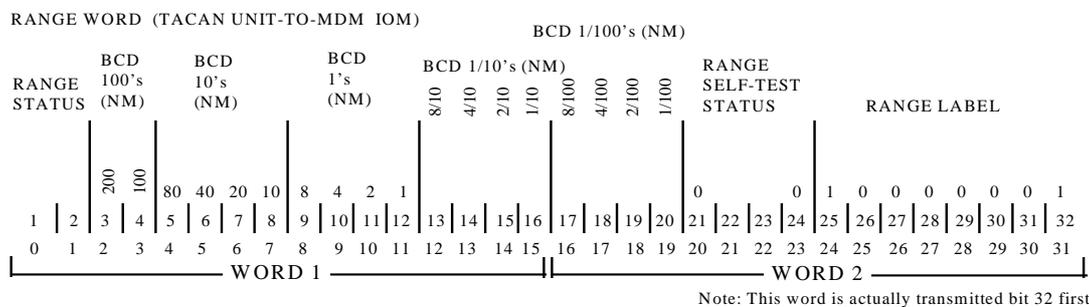


Figure 1-15 Format of the Range Word

BIT

0, 1

Range Status.

- 00 - Valid data (track or short-term memory)
- 10 - No compute data (search or long-term memory)
- 01 - TACAN built-in test equipment (BITE) failure
(range circuitry)
- 11 - Spare

Up to 2 seconds are required for initial lock-on and processing of the first valid range measurement after the RF channel of the beacon has been selected. RF channel selection requires an additional 1 second (maximum). The 1 second time period is required by the TACAN unit to establish the RF transmitter/receiver pair of frequencies.

When the TACAN unit is performing valid measurements of range, the TACAN unit is referred to as being in track mode. Upon the LOS from the beacon, the TACAN enters short-term memory mode and remains in this mode of operation (in the absence of incoming signals) for approximately 0.5 to 1.0 seconds. During this time period, the TACAN unit continues to output valid range values to the MDM IOM.

At the end of the short-term memory period, the TACAN unit enters the long-term memory period. The range status code is changed from 00 (valid data) to 10 (no computed data). The long-term memory period is at least 8 seconds and as much as 12 seconds in duration (provided the TACAN unit has been processing valid data for at least 15 seconds). The long-term memory mode allows TACAN signal processing to continue immediately upon beacon signal return.

The TACAN unit incorporates a range filter of the velocity type to filter range. The MDM IOM receives range values which are the output of this filter. When the beacon signal is lost, the output of this filter continues to generate range values. The range values exhibit constant range rate (velocity), zero acceleration.

Range words are transmitted from the TACAN unit to the MDM IOM whenever the output of the range filter changes by 0.05 nm/sec (4860 nm/hr).

At the end of the long-term memory period, the TACAN enters the search mode. In search mode, up to 2 seconds are required for lock-on (when signal are present).

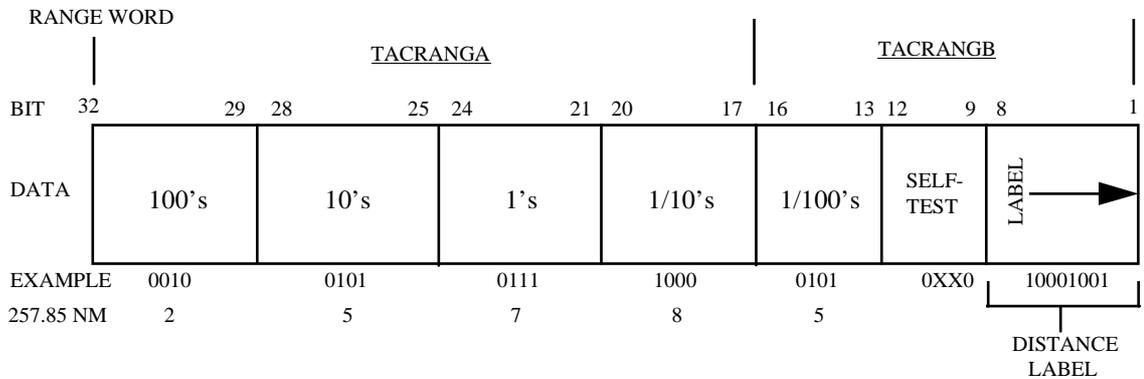
2-19

Range data. These 18 bits provide range in binary coded decimal (BCD) format. The maximum possible indicated range is 399.99 nm.

- 20 Collins BITE Status. Reference section 1.9.1.2 for details of what sets this BITE discrete.
0 - No BITE present
1 - BITE present
- 21, 22 Range self-test status. These two bits define whether or not the TACAN unit is in self-test mode.
00 - Normal operation
11 - In range self-test
10,01 - Undefined
When the TACAN unit is successfully performing self-test, it continuously transmits a range value between 0 and 0.5 nm. It is required that IOP or GSE units conducting self-test verify that the range is within the above limits. Range values outside of the limits are to be interpreted as a failure of the TACAN range circuitry.
- 23 Collins Suppression Pulse Present. Reference section 1.9.1.1 for details of this discrete.
0 - Suppression pulse not present
1 - Suppression pulse present
- 24-31 Range label. Bits 24-31 define the range label. The label allows discrimination between bearing and range words.

The foregoing completes the description of the range and bearing words from the MDM to the GPC. The GPC receives two additional signals from the TACAN unit. These signals (TACAN power-on indication, TACAN test monitor) are described as follows:

- a. TACAN power-on indication - This discrete signal indicates that the TACAN is powered ON.
- b. TACAN test monitor - This signal is used to indicate the progress of the self-test. The signal is at a logical 1 level (+5 V dc) whenever the TACAN unit is in normal operation.



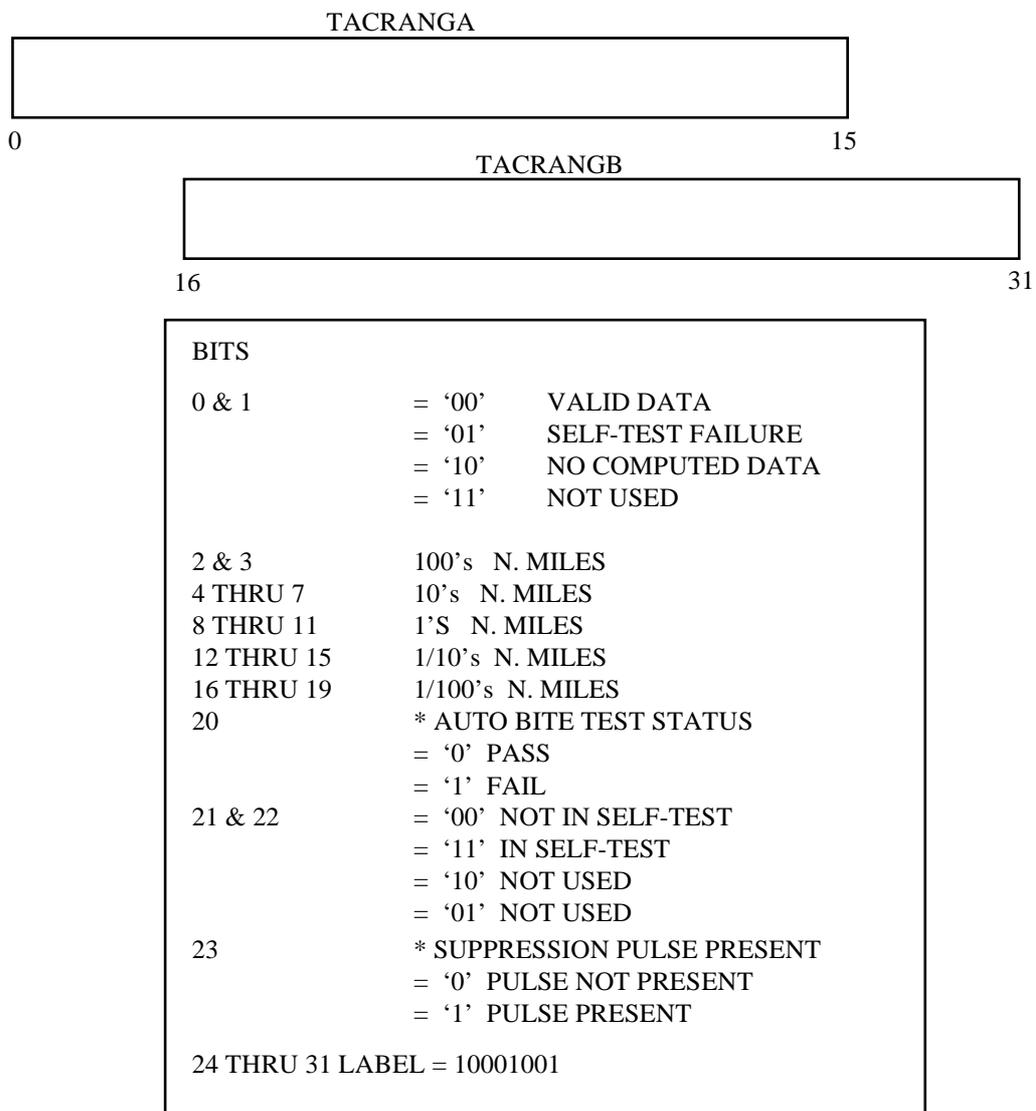
| 31 | 30 | 12 | 11 | 10 | 9 | BIT |
|----|----|----|----|----|---|---|
| 0 | 0 | — | — | — | — | VALID DATA (TRACK OR SHORT-TERM MEMORY) |
| 0 | 1 | — | — | — | — | SELF-TEST FAILURE |
| 1 | 0 | — | — | — | — | NO COMPUTED DATA (SEARCH OR LONG-TERM MEMORY) |
| 1 | 1 | — | — | — | — | NOT USED |
| — | — | — | 0 | 0 | — | NOT IN SELF-TEST |
| — | — | — | 1 | 1 | — | IN SELF-TEST |
| — | — | — | 0 | 1 | — | NOT DEFINED |
| — | — | — | 1 | 0 | — | NOT DEFINED |
| — | — | 0 | — | — | — | PASSED AUTO BITE TEST * |
| — | — | 1 | — | — | — | FAILED AUTO BITE TEST * |
| — | — | — | — | — | 0 | SUPPRESSION PULSE NOT PRESENT * |
| — | — | — | — | — | 1 | SUPPRESSION PULSE PRESENT * |

* VALID FOR MC409-0184 TACAN ONLY (COLLINS)

Figure 1-16 TACAN-to-MDM Range Word

When self-test is initiated, the signal drops to a logical 0 level. It remains at a logical 0 level throughout the self-test period and beyond if a failure is encountered. The signal returns to a logical 1 state upon the completion of self-test if self-test is successful. Self-test requires a minimum of 4 sec and a maximum of 18 sec for completion.

RANGE WORD



* VALID FOR MC409-0184 TACAN ONLY (COLLINS)

Figure 1-17 MDM-to-GPC Range Word

1.6 40-DEGREE BEARING GLITCH

There have been false lock-ons of TACAN's since the beginning of the TACAN system. One particular false lock manifests itself by a bearing error of 40 degrees, or a multiple of 40 degrees. This particular false lock is usually referred to as the 40-degree bearing glitch. Its duration can be either momentary or prolonged. In general, momentary errors do not exceed several seconds. A prolonged false lock can last indefinitely.

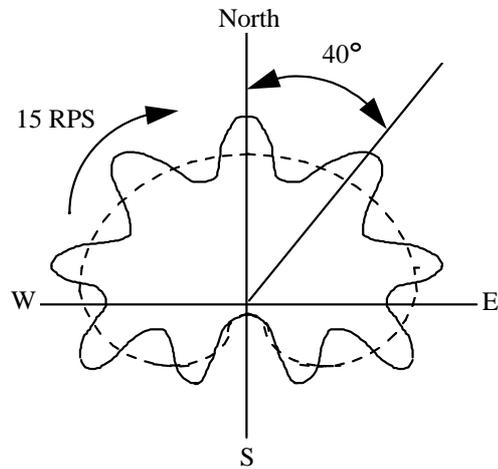
The momentary false lock (assuming correctly aligned and adjusted system equipment) is usually attributable to RF multipath (site reflection) effects. The prolonged glitch is generally caused by incorrect equipment alignment.

The TACAN azimuth system uses a 15 Hz modulation for a coarse bearing and a 135 Hz modulation for fine bearing. One period of the 15 Hz cycle is equal to 360 bearing degrees while one period of the 135 Hz cycle equals 40 degrees (fig. 1-18). An improper phase determination of the 15 cycle modulation by as much as 20 degrees (one-half sector) or more would result in the 15 Hz cycle information being positioned in the incorrect sector resulting in incorrect bearing information.

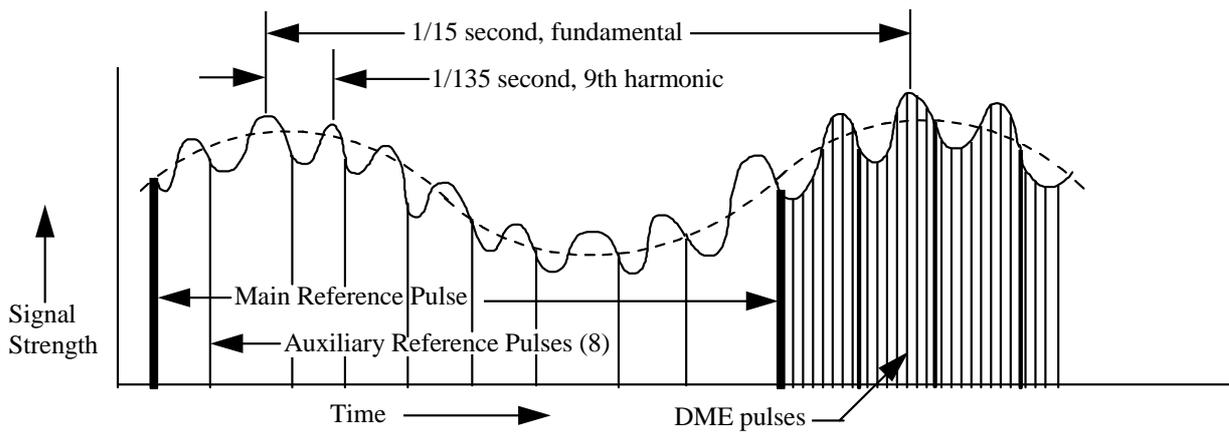
Possible causes of the glitch include the following:

- a. Incorrect internal co-alignment of the parasitic elements inside the surface beacon antenna
- b. Incorrect rotational speed of the surface beacon antenna
- c. Incorrect alignment of the airborne equipment
- d. Deterioration of component characteristics in the airborne and/or ground equipment
- e. RF reflections of the radiate beacon signal from combinations of horizontal and vertical surfaces
- f. False modulations produced in a low-signal, high-noise environment

Since this phenomenon is a recognized possibility, onboard RM accounts for the 40-degree bearing glitch as discussed below.



(a) Radiation pattern - effect of both cylinders



(b) Detected signal - pulse envelope only indicated

Figure 1-18. - Bearing pattern

The source for the discussion of this anomaly was taken from TECHNICAL INVESTIGATION OF FALSE AZIMUTH LOCK-ONS IN THE TACAN NAVIGATION SYSTEM dated August 19, 1968 (Project No. 67-57 Serial number 20).

1.7 RM

Selection filtering (SF) utilizes standard midvalue select (MVS) at the three level, averaging at the two level, and prime select at the one level (fig. 1-19). Selection is performed on a parameter basis of candidate units.

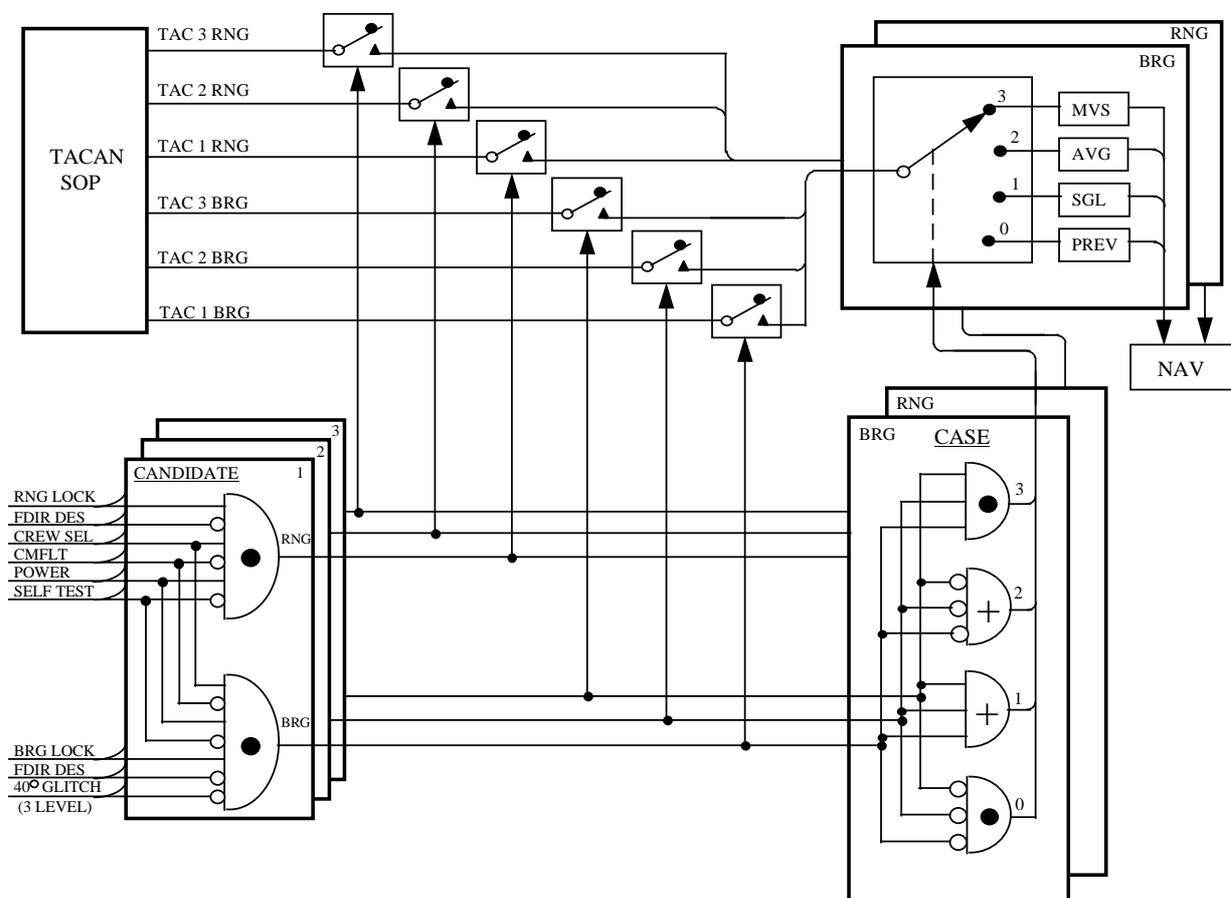


Figure 1-19. - TACAN RM-SF

Fault detection, isolation, and reconfiguration (FDIR) determines the eligibility of a parameter for SF candidacy based upon two out of three comparison at the three level, and an automatic self-test (if in GPC mode) at the two level (fig. 1-20). An unresolved self-test leaves that parameter in a dilemma situation (not used by NAV) (fig. 1-21). Aside from an FDIR declared failure, candidacy for SF will also be dependent on lock-on, crew selection, communication fault (CF), power status, 40° glitch check at the three level, and participation in self-test at the two

level. The 40° glitch check logic determines if a TACAN's bearing data differs from the previous selected bearing value by an increment of 40° ($\pm 4.5^\circ$). If it is determined that a 40° glitch has occurred on one TACAN at the three level, then that LRU's bearing will not be a candidate for SF until the 40° glitch clears; however, it will not be declared failed by FDIR.

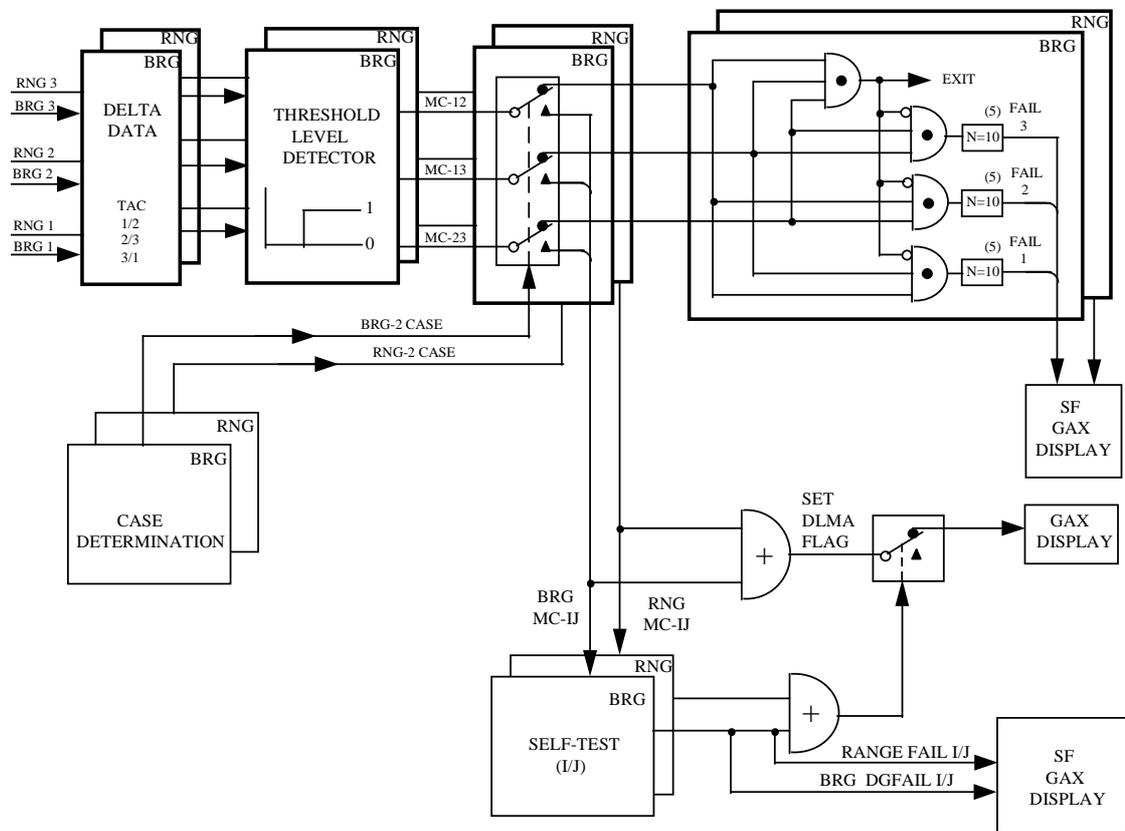


Figure 1-20. - TACAN RM-FDIR

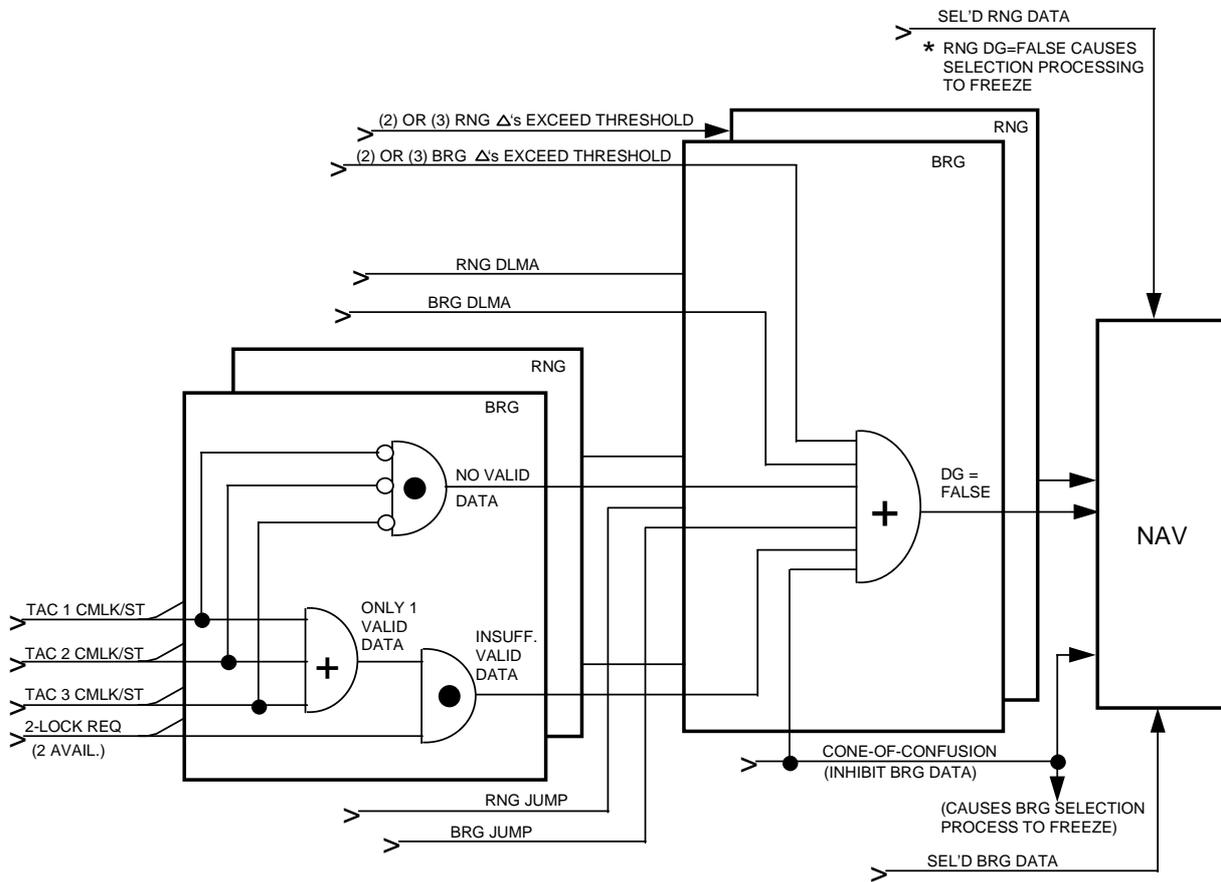


Figure 1-21. - Inhibit use of TACAN data

Backup flight system (BFS) RM (fig. 1-22) does not contain any FDIR software; however, the other candidacy requirements do apply. The major SF difference is that the lowest numbered LRU is selected at the two level. A form of fault detection exists in that a parameter discompare at the two level will inhibit NAV use of that data.

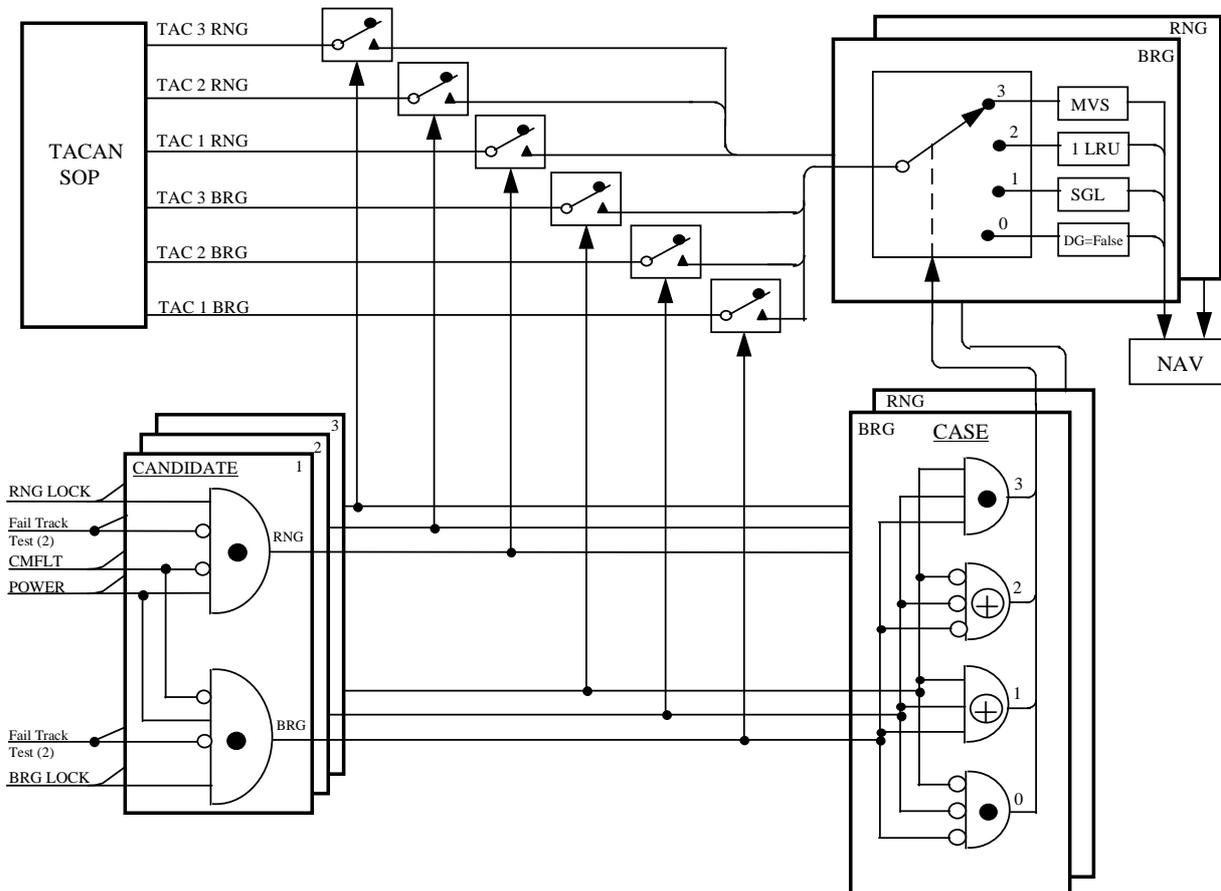


Figure 1-22. - TACAN RM-SF BFS

1.8 SELF-TEST

The TACAN LRU has an internal self-test feature whereby range and bearing circuits can be exercised independently. However, an external self-test command is required to initiate the test. This command may be generated automatically by the GPC's or manually from the keyboard.

- a. Manual - TACANS may be self-tested onorbit in OPS 8 via a keyboard entry. LRU's that exceed test limits will be flagged on the sensor self-test CRT display (fig. 1-23) with up or down arrows and a BITE indication. Manually deselecting a unit (due to only one parameter failing) will inhibit both parameters from use.

- b. Automatic - During active entry/return to launch site (RTLS) phases, TACANS that miscompare (range or bearing) at the two level (while rotary switches are both in GPC mode) will automatically be commanded to self-test as a vote in breaking a dilemma situation. Following approximately 20 sec, the failed parameter (only) will be deselected by RM; however, during the test period both units are unavailable for SF.

The TACAN self-test feature provides a 0.90 probability of failure detection at the LRU level. Self-test does not check the antennas, the antenna switch assembly, the duplexer assembly, the power amplitude/modulator assembly, portions of the power supply or the mixer in the receiver assembly. The TACAN unit does not radiate during self-test.

| 8011 / 040 / | | SENSOR TEST | | | | XX X | DDD / HH : MM : SS |
|--------------|---|-------------|------|-------|---------|-------|--------------------|
| | | | | | | | DDD / HH : MM : SS |
| | | DES | STAT | RNG | AZ | EL | |
| MLS | 1 | 1 | | 15.0 | -2.96 | 5.92 | |
| | 2 | 2 | | 15.2 | -3.01 | 6.05 | |
| | 3 | 3 | BITE | 15.4↑ | -3.13↑ | 6.17↑ | |
| | | DES | STAT | RNG | AZ | | |
| TAC | 1 | 4 | | 0.2 | 177.51 | | |
| | 2 | 5 | | 0.5 | 178.33 | | |
| | 3 | 6 | BITE | 0.4 | 182.54↑ | | |
| | | DES | STAT | ALT | | | |
| RA | 1 | 7 | | 1100 | | | |
| | 2 | 8 | | 900 | | | |
| | | DES | | Y | Z | | |
| AA | 1 | 9 | | +14.4 | -82.8 | | |
| | 2 | 10 | | +15.0 | -64.4 | | |
| | 3 | 11 | | +15.2 | -58.3 | | |
| | 4 | 12 | | +15.6 | -45.9 | | |
| | | START | 13* | STOP | 14 | | |

Figure 1-23 On-orbit Sensor Self-Test

1.9 COLLINS TACAN

The Collins TACAN (model TCN-252) is an off-the-shelf unit (model TCN-250), modified to be transparent to the Orbiter interfaces. This is true for the mechanical, functional, and most of the electrical interfaces. However, the unit weighs less than the Gould TACAN, uses dc (versus ac) electrical power, and does not require forced air cooling. These units are expected to have improved reliability, maintainability, and accuracy over the Gould TACANs.

1.9.1.1 TACAN Hardware

The Collins unit is a derivative of an existing TACAN (TCN-250) built by Collins Government Avionics Division (CGAD), and includes a modified shock mount to provide the proper interface of this smaller unit to the Orbiter. The receiver/transmitter (R/T) does not require forced air cooling, which has resulted in the elimination of the cooling duct on the Orbiter Collins TACAN shock mount. This unit is powered by 28 volts dc, and requires modifications to the Orbiter overhead panels (014, 015 and 016, row C, for TACANs 1, 2, and 3 respectively) to include the appropriate dc circuit breakers (see figure 1-4).

Because of greater isolation between the units, the Collins TACAN is less susceptible to the 40-degree Bearing glitch that is inherent in the Gould TACAN. In addition, this greater isolation is expected to reduce the effect of a blanking pulse failure, and a Collins TACAN should lock on very close to nominal altitude with such a failure present on a unit.

The Collins TACANs also have a blanking pulse (BP) monitor discrete. This TACAN Range Suppression Pulse Present discrete is output in bit 23 of the MDM-to-GPC Range word and is monitored prelaunch in the Launch Commit Criteria. The suppression pulse present bit is the value of 1 if any TACAN (Gould or Collins) is transmitting a blanking pulse, and the value of 0 if no TACAN is transmitting a blanking pulse. For example, when at least one TACAN (Gould or Collins) is in the T/R mode, blanking pulses are sent which should be detected by all TACANs. The Collins TACANs should display the value of 1 for the suppression bit. If a Collins does not indicate a 1, then a failure in the suppression circuitry exists affecting at least one TACAN. If this occurs prelaunch, all TACANs will be taken one at a time to T/R mode, with the remaining TACANs in RCV mode in order to determine which TACAN is not sending or receiving the suppression. Similar troubleshooting can also be performed on-orbit and during entry to isolate the failure. When all TACANs are in RCV only mode, no blanking pulses are being sent and all Collins TACANs should indicate the value of 0.

1.9.1.2 Self Test and BITE

The Collins TACAN has self-test and built-in-test (BITE) capabilities that check approximately 96 percent of the electronics. The BITE is referred to by the vendor as simply BIT (built-in-test), but is very similar to other BITE functions that we have in other Orbiter systems. One major difference is that the amount of realtime insight is very limited for this TACAN BITE. There are four BITE modes:

- 1) PERIODIC, periodically (once every 30 seconds) monitors performance and detects faults when in the search mode,
- 2) CONTINUOUS, continuous monitor of performance during all conditions (locked and search) ,
- 3) INITIATED, performs end-to-end testing and requires operator intervention (this is the hardware Self Test), and
- 4) POWER-UP, performs functional tests upon LRU power-up.

The functions which are monitored by the BITE in the various BITE modes are as follows:

- CPU - Exercises representative subset of instructions.
- RAM - Writes test values to each RAM address and reads them back.
- ROM - Checksum ROM and CRC ROM verification.
- TWO PORT RAM - Writes test values to each address and reads them back.
- TWO PORT DATA - Verifies minimum required activity (new writes) occurs.
- NOVRAM - Tests stored parameters for reasonableness.
- AUDIO - Checks amplitude of ID signal generated by processor.
- MDM WRAP-AROUND - Loops good command word followed by bad word thru MDM, and verifies correct return.
- POWER SUPPLY - Verifies presence of 5 volt monitor output.
- RECEIVER - Checks gain of receiver and AGC circuits.
- SYNTHESIZER - Checks monitor signal for frequency lock-up.
- TRANSMIT POWER - Checks amplitude of transmit power monitor, which is proportional to peak transmitted power.

The following table shows which BITE checks are performed for the different BITE modes. Note that the PERIODIC BITE checks are performed every 30 seconds in the SEARCH mode only, and the CONTINUOUS checks are always performed (during SEARCH, TRACK, PWR UP, and during the H/W Self Test (INITIATED)).

| | Periodic (Search Mode) | Continuous (Search/Track) | Initiated (H/W Self Test) | Power Up |
|-----|---------------------------|------------------------------|------------------------------|----------|
| CPU | X | | X | X |

| | | | | |
|----------------|---|---|---|---|
| RAM | X | | X | X |
| ROM (Checksum) | X | | X | X |
| ROM (CRC) | | | | X |
| TWO PORT RAM | X | | X | X |
| TWO PORT DATA | | X | | |
| NOVRAM | X | | X | X |
| SYNTHESIZER | X | X | X | X |
| RECEIVER | X | | X | X |
| POWER SUPPLY | X | X | X | X |
| XMIT POWER | | | X | |
| AUDIO | | | X | X |
| MDM | | | X | X |

The INITIATED BITE checks are those which are performed during the manually initiated hardware Self Test. It is important not to confuse the BITE function with the Self Test function. Both of them use the same paths to check the hardware, but the results are completely separate. The results of the Self Test are seen in the Range and Bearing data, and in the Test Status Monitor (TSM), no different than the Gould Self Test monitoring. The Collins Self Test is only different in that the actual checks which are performed during the Self Test are different than in the Gould units, since the hardware is different. The TSM discrete toggles from a high (true) state to a low (false) state within 100 ms of the Self Test being initiated. This discrete returns back to the true (or high) state when the test is complete 4.2 seconds later, if all checks pass. The TSM is valid only during and at the end of Self Test.

The BITE results are available in the BITE pass/fail discrete. This binary discrete is a 1 when the BITE is present (failed condition), and is a 0 when the BITE is not present (nominal). The status of this discrete is output in bit 20 of the MDM-to-GPC Range word and is monitored prelaunch in the Launch Commit Criteria.

It is worth noting that it would be possible to pass the Self Test, and at the same time have a BITE indication in the MCC. For example, if a Self Test is initiated in OPS 8 (via SPEC 40, SENSOR TEST), and the test is passed (the 'BITE' in the status clears), and at the same time the BITE is set (MCC TLM only, not visible to the crew), then the problem is one of two things. Either there is a problem with the TWO PORT DATA that is currently not affecting the TACAN, or the BITE indication is false. In other words, the TWO PORT DATA check is done as part of the continuously monitored BITE, and so it is performed even during the initiated Self Test (see table above). If there was a problem with the TWO PORT DATA, it would not cause Self Test to fail (unless it adversely affected the TACAN performance, which it most likely would), but a BITE indication (MCC TLM) would be expected. This example failure mode is no more probable than any other, but rather it is used here to illustrate the separate function of the BITE and the Self Test.

The vendor engineers have expressed that intermittent BITE indications are possible and are somewhat expected, due to noise in the system. If the BITE persists, then they recommend

performing a Self Test if possible (OPS 8 or OPS 9), and in any case, the actual LRU performance should be monitored to determine if the LRU is actually failed.

1.9.2 Operational Notes

The operational concerns associated with the loss of an avionics bay fan during ascent are eliminated with the implementation of the Collins TACANs because these units do not require forced air cooling. It is important for the operators (GNC and EECOM) to be aware of which TACANs are being flown in the respective slots, in order to manage these concerns while flying a mixed fleet. The ascent cue card procedure (AV BAY Fan Delta P) requires the crew switch fans if no comm. This means that if we have comm, we will tell the crew if/when it's necessary to switch fans during powered flight, dependent on whether the affected av bay has a new or old TACAN in it. If we have no comm, then the crew will switch fans. Switching fans is required only if the affected av bay contains a Gould TACAN.

Some problems could arise in prelaunch if it becomes necessary to replace a failed Gould unit with a Collins (or vice versa) while on the launch pad. This might not be possible, since it requires changing the Orbiter TACAN shock mount. This is a difficult task to perform, and is further complicated by the Orbiter being in the vertical position. As a note, it is easier to change out TACAN 1 than either of the other two TACANs, because of the avionics bay configurations.

The Collins TACAN shows a distinct data signature during periods of loss of lock, or any time it is in the SEARCH mode. The Range data freezes at a predefined value of 0.0 nautical miles when in the SEARCH mode, or within 3 seconds of losing a valid signal (loss of TRACK). The Bearing data scans between 0° to 360°. This differs from the Gould TACAN, which gives indications of noisy data for both parameters during periods of loss of lock.

The Gould units require 2 to 15 minutes warm-up time (15 minutes is required on some units with older electronic components). The Collins TACANs have solid state components and require only a few seconds (2 to 3 seconds) to warm-up. In the worst case, the Collins requires 30 seconds to reach thermal stability when powered ON if the temperature is between -40 degrees F and 80 degrees F.

1.9.3 Power Management

Each vehicle* has a set of three ac circuit breakers (cb's) for Gould TACANs, and a set of three dc cb's for Collins TACANs. In this way, any vehicle will be able to accept either a Gould or a Collins TACAN in any one of its three TACAN slots. The cb's that are not used for a particular flight will remain open for the entire flight, and will be marked as such with special cb booties which indicate to the crew that they are not to be closed.

1.9.4 Miscellaneous

All of the other major items of interest are the same for the Collins TACANs as for the Gould units. The Orbiter interfaces (control head switches, antenna switches), the crew interfaces (PASS and BFS SPEC 50, Horiz Sit Display, and SPEC 40, SENSOR TEST), the antenna configuration, the TACAN SOP and RM, and the ground station function are all the same for the two units. The Collins TACAN specifications (SPECS) are also the same as those for the Gould units. (Reference SODB, Vol I, Systems Constraints, Sections 4.5.2.1.4.c, and 4.5.2.3.4.2.)

1.9.5 Collins SPECS

| | |
|-------------------|----------------------------------|
| Power Supply | 28 V dc |
| Power Consumption | 75 Watts (max) |
| Transmitter Power | 800 Watts (min @ 70 +/-10 deg F) |
| Dimensions | 4.12 x 6.78 x 12.0 (inches) |
| Weight | 14 lbs |
| Cooling | Passive |

SECTION 2
MICROWAVE LANDING SYSTEM

2.1 GENERAL

Upon switchover from tactical air navigation (TACAN), the Shuttle microwave landing system (MLS) provides the Orbiter with precise azimuth angle (lateral offset), elevation angle, and distance (slant range (RNG)) information with respect to the runway during the final approach and landing phase through rollout. MLS data is the primary navigation (NAV) sensor input for autoland and is also made available for crew display for both manual control stick steering (CSS) and autoland.

Various MLS nav aids have been built for aircraft using a variety of concepts. The MLS which the Shuttle uses is the microwave scanning beam landing system (MSBLS). It is a derivative of a landing system which is used extensively on aircraft carriers but tailored specifically with the accuracy, coverage, and reliability required for the Space Shuttle's steep, unpowered, automatic approach, landing, and rollout. Unlike fixed-glide slope beam-rider systems such as the instrument landing system (ILS), the MSBLS provides explicit and more accurate positional information throughout the entire volume of coverage and can, therefore, accommodate precision curved approaches at various glideslope angles.

This system brief provides a general overview of the Shuttle MLS. More detailed information is presented in the MSBLS handbook, manuals, and other flight operations documents and drawings.

Figure 2-1 illustrates the transmitted microwave beams and measurement geometry of the MSBLS. The MSBLS consists of a ground station (MSBLS-GS) and an airborne navigation set (MSBLS-NS or NAVSET). The NAVSET is composed of two line replacement units (LRU's): the radio frequency (RF) assembly and the decoder assembly. The MSBLS-GS is a split-site configuration, with azimuth information and ranging replies transmitted from the azimuth and distance measurement equipment (DME) transmitter beyond the stop end of the runway and elevation information transmitted from near

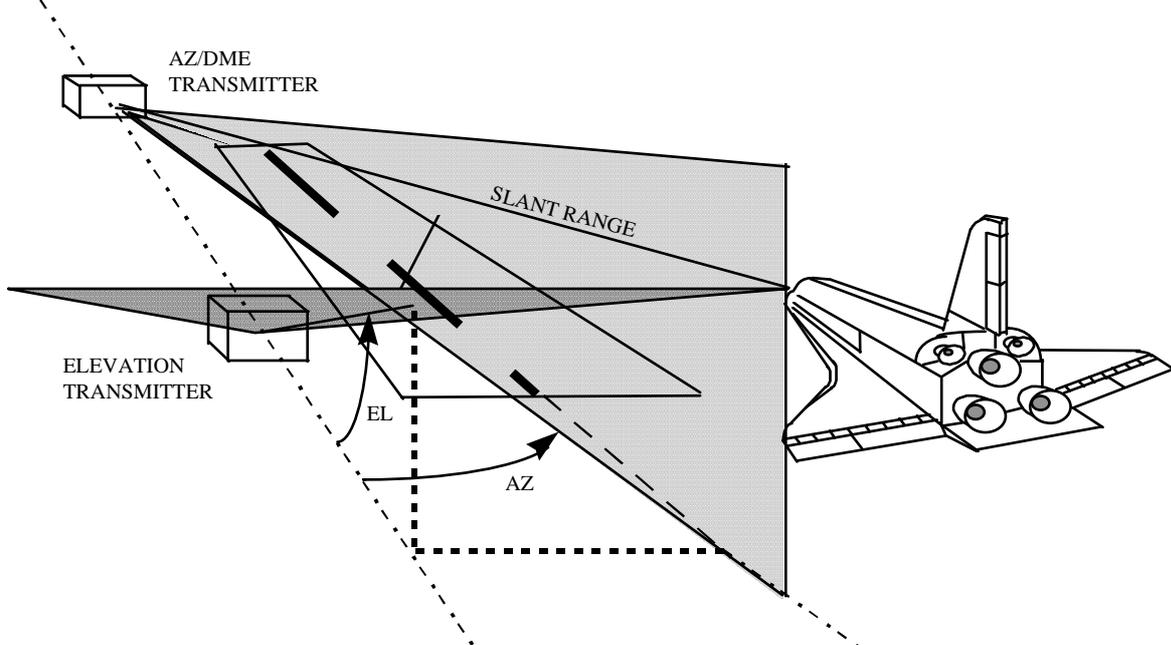


Figure 2-1. MSBLS beams and measurement geometry

the touchdown point. All three parameters (azimuth, elevation, and range) are referenced to the MSBLS-GS locations which are laterally offset from the runway centerline by 300 ft at Kennedy Space Center (KSC) and 500 ft at Edwards and White Sands (Northrup Strip). These surveyed locations are stored in memory onboard the Orbiter and used in the NAV computations in conjunction with the azimuth, elevation, and range data received and decoded by the NAVSET to determine the Orbiter position with respect to the runway.

The MSBLS has 10 frequency channels between 15.4 and 15.7 gigahertz (GHz); however, since there are other users of this generic system, only channels 6 and 8 have been cleared for Orbiter landing operations. Each Shuttle ground station is permanently assigned to either channel 6 or 8. For entry and landing, the channel number for the designated runway is selected on the Orbiter with a set of thumbwheels. The transition from TACAN to MLS typically occurs at about 20K ft altitude when certain conditions are satisfied. These conditions and other information concerning performance monitoring are discussed later.

2.2 MLS OPERATION

The major elements of the MSBLS, their level of redundancy, and the associated RF links are shown schematically in figure 2-2.

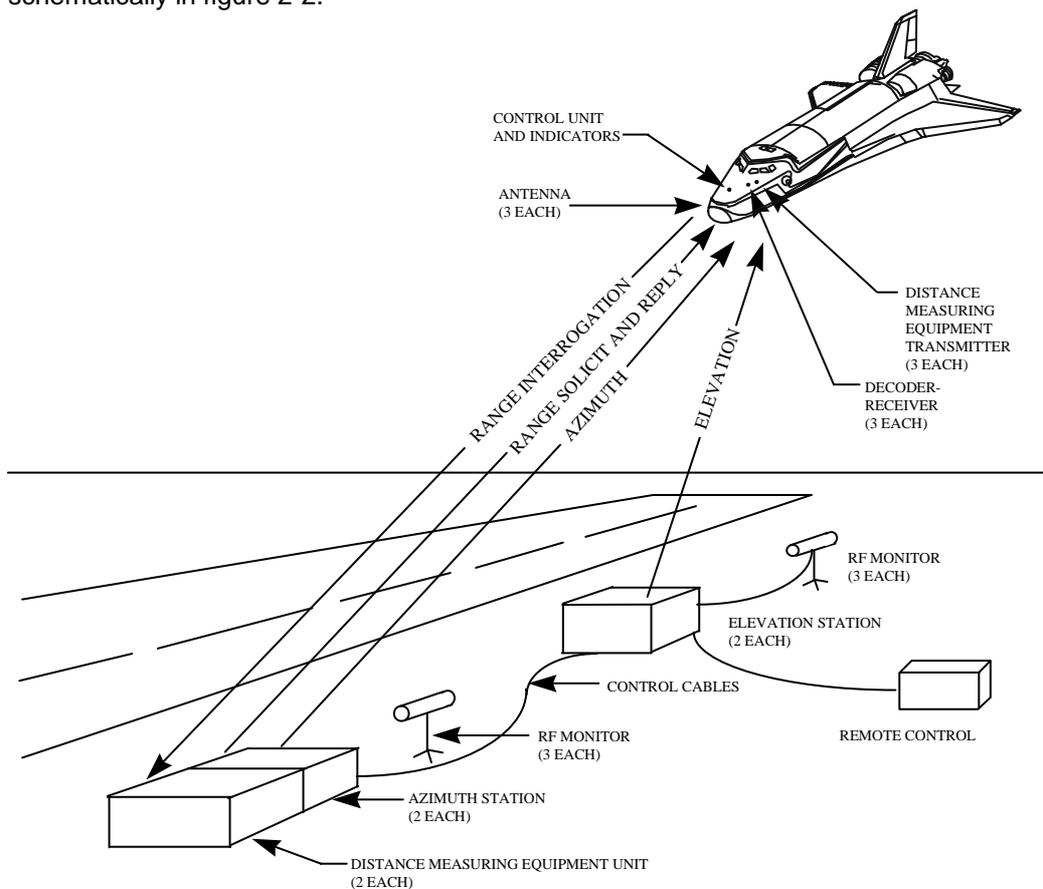


Figure 2-2. MSBLS major components and RF links

All three information parameters (azimuth, elevation, and range) are transmitted sequentially (time division multiplexed) from a given ground station on the same vertically polarized RF between 15.4 and 15.7 GHz. Figure 2-3 shows the transmission time allotted to each measurement within the 200 ms information cycle. This cycle repeats at the rate of five information cycles per sec.

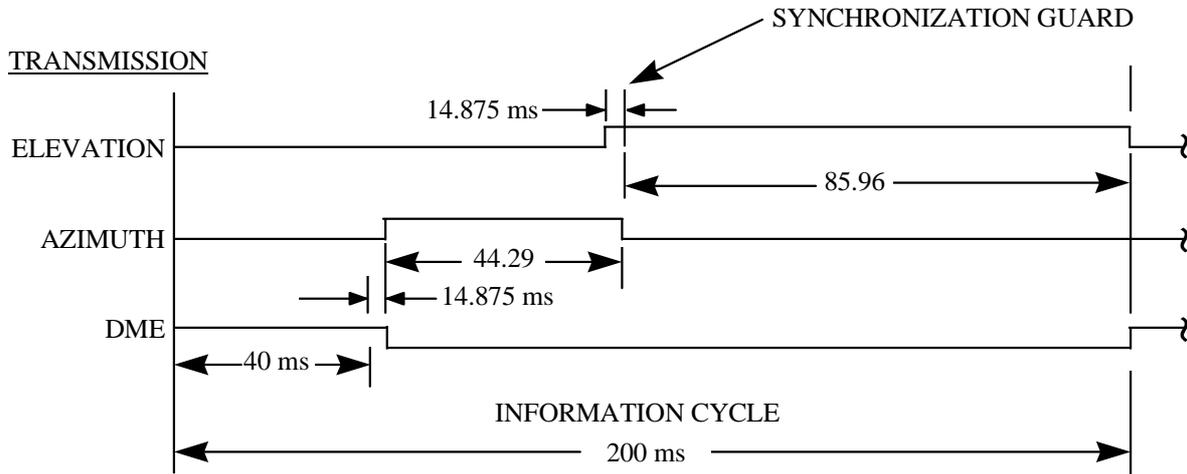


Figure 2-3. MSBLS timing and sequence format.

The angular information of azimuth and elevation is provided by two oscillating antennas on the ground (fig. 2-4 and 2-5). The antennas produce fan-shaped radio beams which are pulse pair encoded with the present antenna position. The azimuth beam contains the antenna's azimuth angle from 0° azimuth at the center of the scan. By varying the time between the pulse pairs (60 μs corresponding to 0° and 2 μs/deg delta from 0°), the angular distance from the centerline is described. Spacing between pulses within the pair describes whether or not the azimuth angle information is in the fly-left portion of coverage (10 μs between pulses) or fly-right portion (14 μs between pulses).

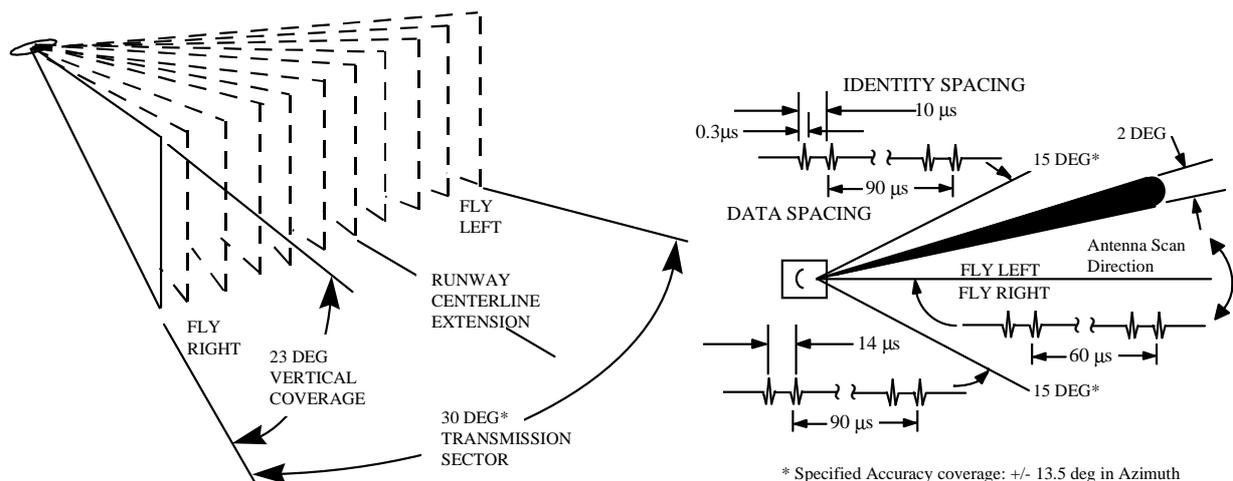


Figure 2-4. Azimuth coverage and pulse coding.

The elevation beam produces similar encoded information describing the angle from the horizon. The identity spacing between pulses in a pair for elevation is $12 \mu\text{s}$. Once again, $60 \mu\text{s}$ between pairs represents 0° elevation plus $2 \mu\text{s}/\text{deg}$ from 0° .

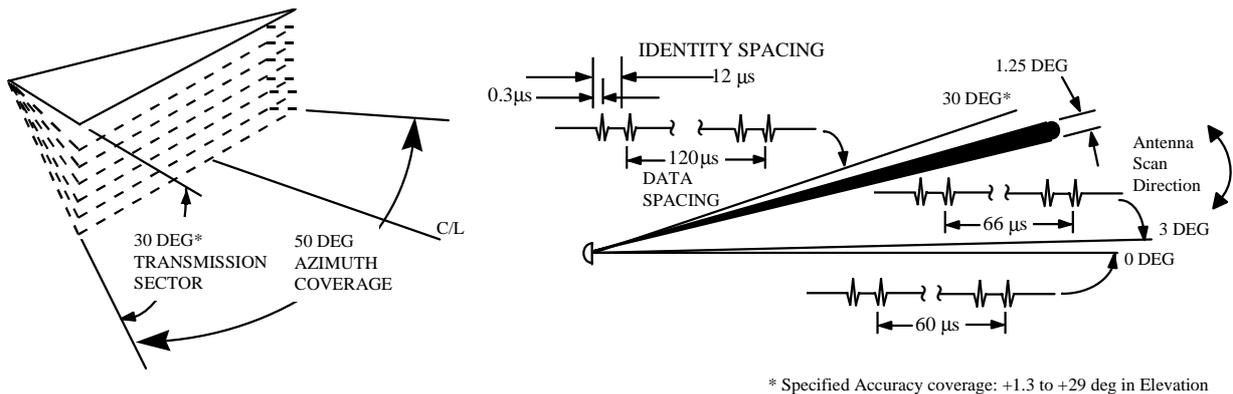


Figure 2-5. Elevation coverage and pulse coding.

The azimuth and elevation antennas are electronically synchronized so that only one antenna is radiating at any particular time. The azimuth and elevation antennas are oscillated in a sinusoidal fashion by means of torsional pendulum arrangements. The azimuth antenna oscillates $\pm 44^\circ$ about the 0° azimuth centerline (parallel to the runway), but only transmits $\pm 15^\circ$ about the centerline (the most linear section of the antenna cycle) as shown in figure 2-6. The elevation antenna oscillates $\pm 24^\circ$ about a 15° elevation rest angle, but transmits only within the sector from 0° to 30° of elevation (above the local horizontal).

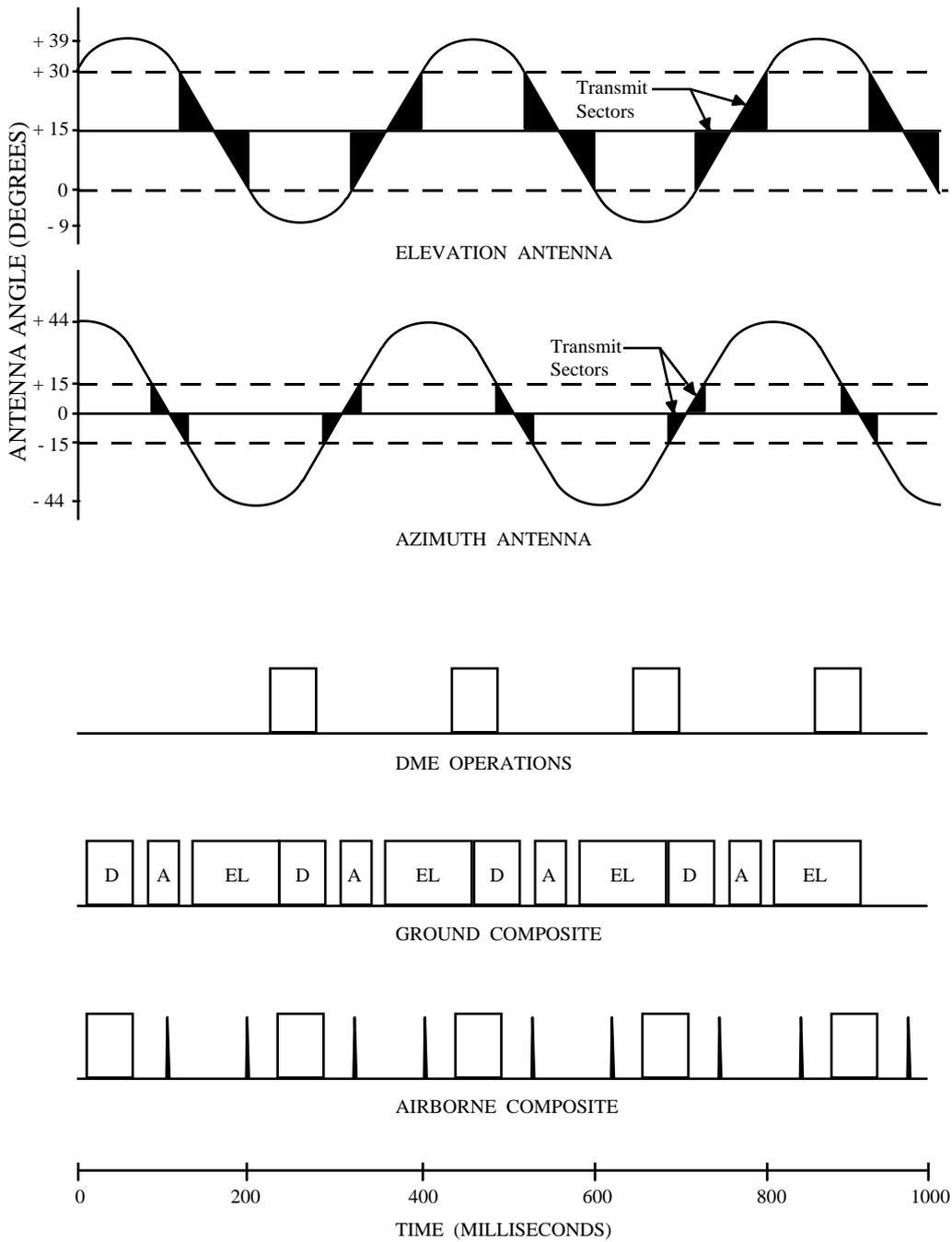


Figure 2-6.- System timing composite.

As the two directional antennas alternately scan through the approach volume, the antennas on the approaching Orbiter vehicle intercept these ground-originated signals only when the spacecraft is within the beam of the ground station antennas. This is illustrated in figure 2-7. The detection circuits within the NAVSET accept only those pulse pairs for measurement that are above a threshold amplitude level of about 4.6 dB below the peak of the beam. The NAVSET decodes the intercepted signals and averages the pulse-pair spacings to obtain the elevation angle of the Orbiter. A similar function is performed for the azimuth angle.

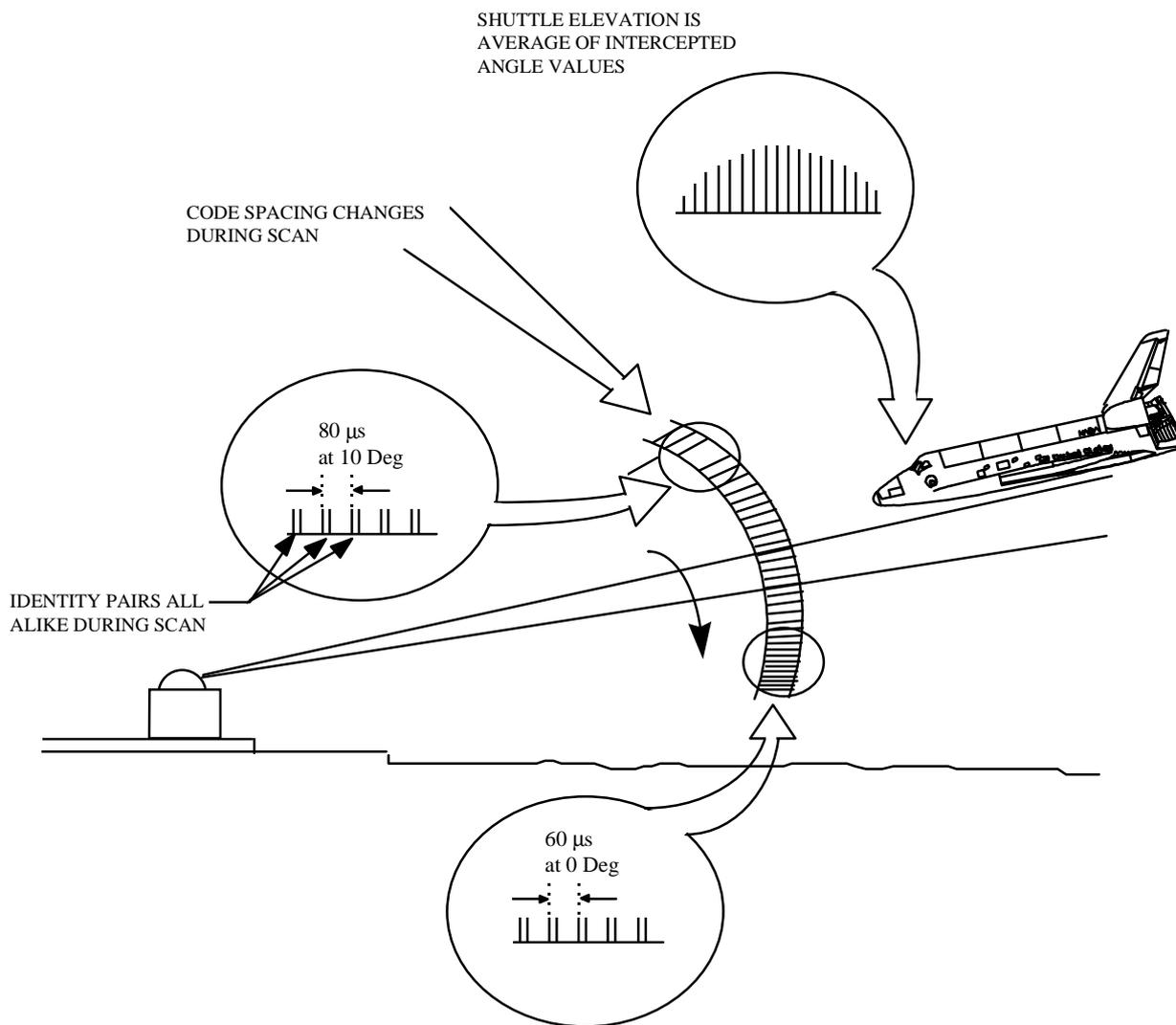
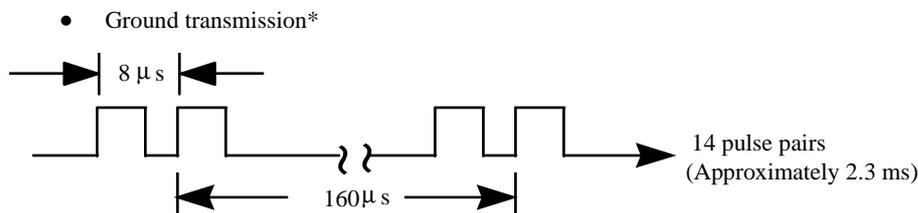


Figure 2-7.- Pulse-position coding of elevation beam.

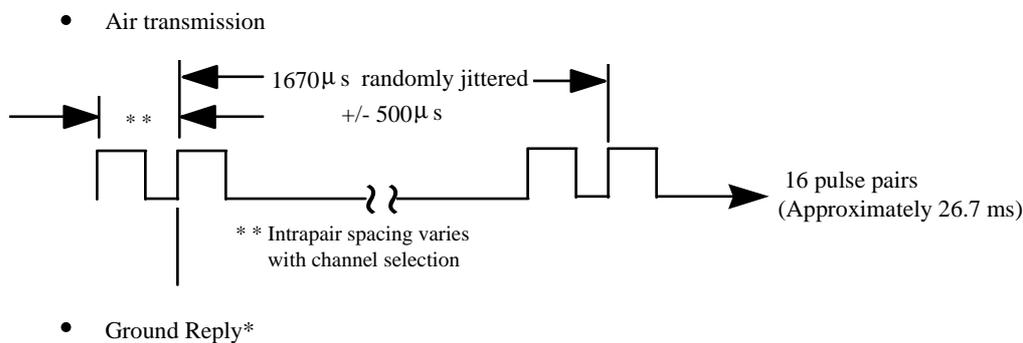
Range information is transmitted between the elevation and azimuth transmissions (fig. 2-6). Three things happen during the ranging (DME) portion of the cycle. First, the ground station transmits a series

of 14 solicit pulse pairs using the azimuth transmitter. These 14 pulse pairs, as shown in figure 2-8, have an intrapair identity coding of $8 \mu s$ and an interpair spacing of $160 \mu s$. Secondly (fig. 2-8), the NAVSET interrogates the ground station and listens for replies. The intrapair spacing for this interrogation varies from $11 \mu s$ for channel 1 operation to $29 \mu s$ for channel 10 operation. Table 2-1 lists the channels and the corresponding identity spacing as well as the uplink frequencies. The interpair spacing is randomly jittered in time by $\pm 500 \mu s$ to prevent confusion with other DME interrogations. After receiving each interrogation, the ground station waits $80 \mu s$ (corresponding to 0 nautical miles) and then transmits a reply. The airborne decoder unit measures the delay between interrogation and reply, and converts the measurement into units of distance to the ground station. Unlike the azimuth and elevation antennas, the DME antenna does not scan, but radiates over the entire coverage volume.

a. Solicit interval - Ground set initiates DME period



b. Air set interrogates - Ground set replies



*The azimuth transmitter is used for DME transmission.

Figure 2-8.- DME functions and coding.

TABLE 2-I. - DME INTERROGATION IDENTITY SPACINGS AND
CORRESPONDING UPLINK FREQUENCIES.

| Channel | Identity spacing (μs) | Frequency (GHz) |
|---------|------------------------|-----------------|
| 1 | 11 | 15.412 |
| 2 | 13 | 15.436 |
| 3 | 15 | 15.484 |
| 4 | 17 | 15.508 |
| 5 | 19 | 15.532 |
| 6 | 21 | 15.568 |
| 7 | 23 | 15.592 |
| 8 | 25 | 15.616 |
| 9 | 27 | 15.664 |
| 10 | 29 | 15.688 |

2.3 MLS PERFORMANCE

The MSBLS accuracy and coverage capabilities are summarized in table 2-II. Most of the information is self-explanatory; however, some further explanation is warranted for certain items.

Coverage (fig. 2-9) is actually provided beyond 13.5° azimuth and 29° elevation and below 1.3° elevation, but accuracy is degraded. At the track level, the azimuth fan beam is about 3° thick and the elevation fan beam is about 1.5° thick. The ground transmissions stop when the peak of the azimuth beam is at 15° and the peak of the elevation beam is at 0° and 30°. Truncated beam envelopes, therefore, are received between 13.5° and 16.5° azimuth, between 29° and 31° elevation, and between 0.75° and 0° elevation. Since the NAVSET determines angle by averaging all received pulse intervals above the track level, the angle data beyond 13.5° azimuth and 29° elevation are biased low, and the elevation data below 0.75 are biased high. In addition, elevation data below 1.3° are corrupted by multipath reflections as the bottom of the beam strikes the ground.

The 23° elevation angle limit on the azimuth beam coverage and the 25° azimuth angle limit on the elevation beam coverage are practical limits for normal Orbiter flight. The beams actually exist beyond those limits but their strength diminishes so rapidly beyond those angles that the NAVSET normally cannot track the signal. As the Orbiter flies past the elevation ground station, the elevation data will normally cease at about 25° azimuth from the elevation station.

The azimuth data angle at this point is about 1.3° or 2° depending on whether the ground station offset distance from the runway centerline is 300 or 500 ft.

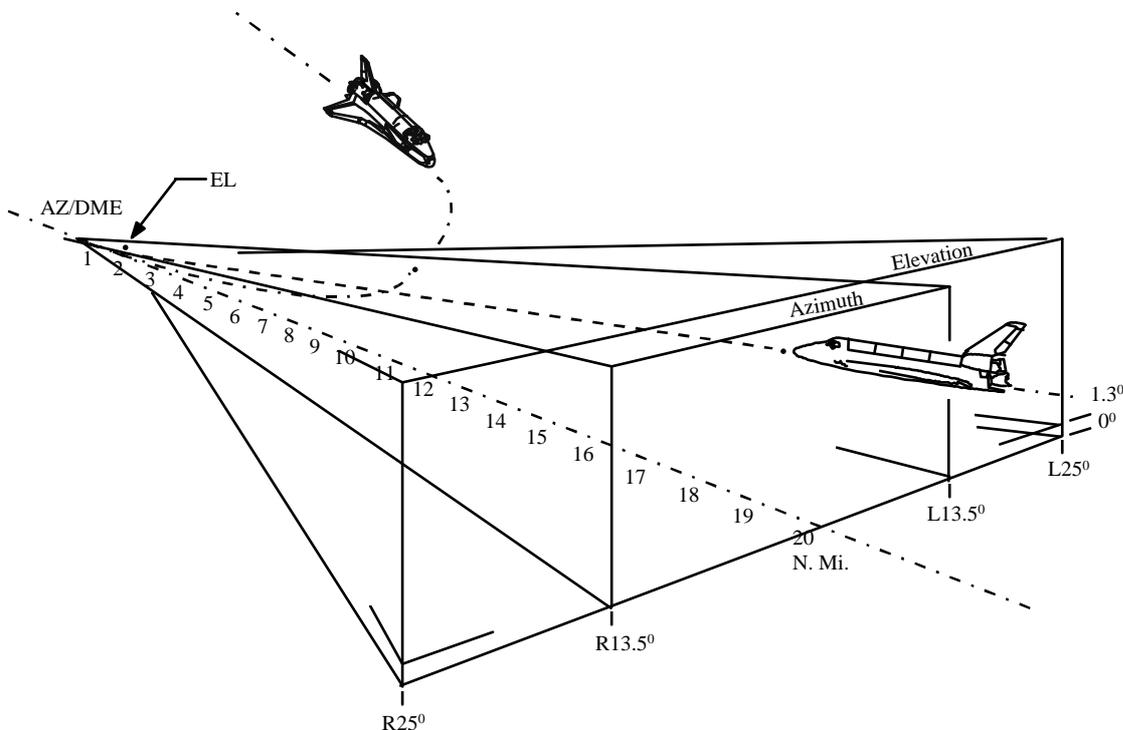


Figure 2-9.- MSBLS coverage volume.

The maximum range capability of the MSBLS is determined by signal strength which is a function of the various RF link parameters including transmitter powers, antenna gains, Orbiter heading with respect to the ground station, receiver sensitivity, and atmospheric attenuation. For a straight-in approach within the MSBLS coverage volume limited only by signal strength, azimuth and elevation acquisition could occur at ranges of up to 180 n. mi. The DME range is limited by a NAVSET digital counter to 19.9 n. mi. (This varies slightly from unit to unit.) As is discussed later, however, the MLS data are not incorporated into the NAV computations until certain conditions are satisfied, one of which is that the DME range be not greater than 15 n. mi. (altitude less than 35K ft). Water particles (especially rain) anywhere along the RF path between the ground station and the Orbiter will attenuate the signal and decrease the range capability of the MSBLS. Figure 2-10 shows the predicted effect on link margin of various uniform rain rates over the RF path length for a typical Orbiter approach and landing as a function of time before touchdown. A negative margin means insufficient signal and no MSBLS data at that time (i.e., delayed acquisition).

Since the MSBLS-GS alignment is allowed to be in error by 0.015° and there is normally some alignment difference between the primary and backup scanners, a small step change in the received angle data normally occurs if a ground station failure causes a switchover from one scanner to the other. In order to achieve the greatest possible accuracy as the Orbiter gets close to the runway, the ground station alignment accuracy is allowed to degrade somewhat at the lateral and upper extremities of the coverage volume (i.e., outside 10° azimuth and above 12K ft altitude). Refraction is the principal contributor to the elevation error at higher altitudes and long range, while ground station antenna deformation is the main contributor to azimuth errors at high elevation and azimuth angles. At low grazing angles, reflection of the beams impinging on the ground cause multipath errors in the elevation data and occasionally invalid

azimuth and range data, although this is minimized by screen fences in front of the azimuth/DME ground station.

Excessive azimuth (lateral) errors, which were experienced on early Shuttle flights due to cross-polarized emissions from the ground stations, are to be eliminated by a ground station antenna modification for operational flights.

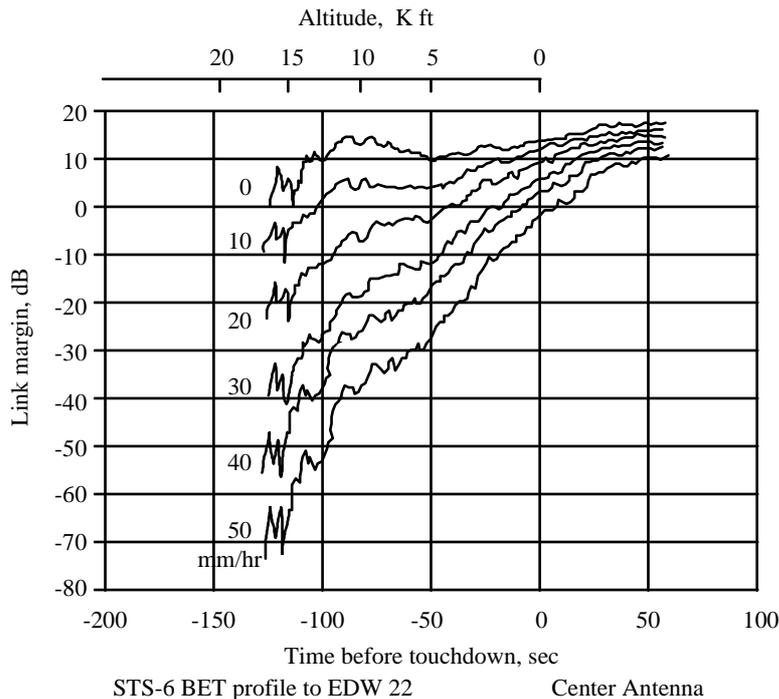


Figure 2-10.- DME link margin at rainfall rates of 0 to 50 mm/hr uniformly distributed over the RF path.

2.4 MLS GROUND STATION

The MSBLS-GS hardware configuration is illustrated in figures 2-11, 2-12, and 2-13. Functional block diagrams of the MSBLS-GS hardware are shown in figures 2-14 and 2-15. Table 2-III shows the landing sites and runways at which MSBLS-GS's are installed and their assigned channels.

2.4.1 MSBLS-GS Hardware

The MSBLS-GS consists of two equipment shelters, two external monitor poles, and a remote control unit. One shelter contains the azimuth and DME assemblies; the other contains the elevation assemblies. Each shelter contains dually redundant sets of equipment, one set forming a primary string, the other a backup string. The external monitor poles house triply redundant field sets. One set monitors azimuth/DME performance, and the other set monitors elevation equipment performance. The remote control unit allows the operator to concurrently monitor the operation of the primary and backup strings, and to intervene and override the automatic controls as necessary.

Each ground station string is powered by batteries in a separate uninterruptible power supply (UPS) which keeps the batteries continuously charged from the ac input power. If ac input power is lost, the ground station will continue to function from the batteries for at least 15 min. Backup ac input power is also available at all installations. At lakebed installations, primary and standby motor generators are used.

Each ground station is aligned and electrically boresighted with respect to the runway to within 0.015° in azimuth and elevation, and periodically verified by ground tests and flight inspections using an MSBLS test NAVSET.

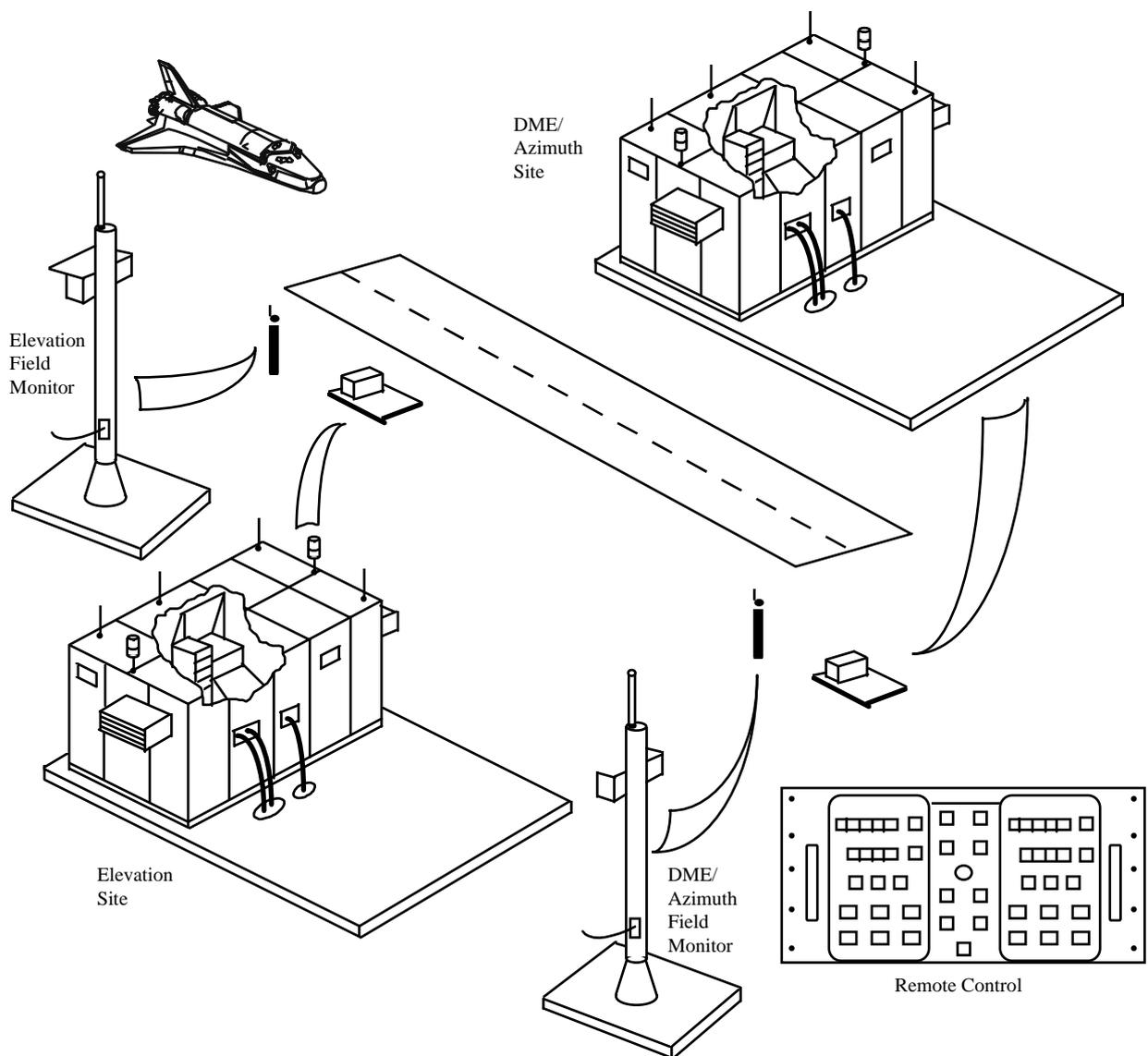


Figure 2-11.- MSBLS-GS installation.

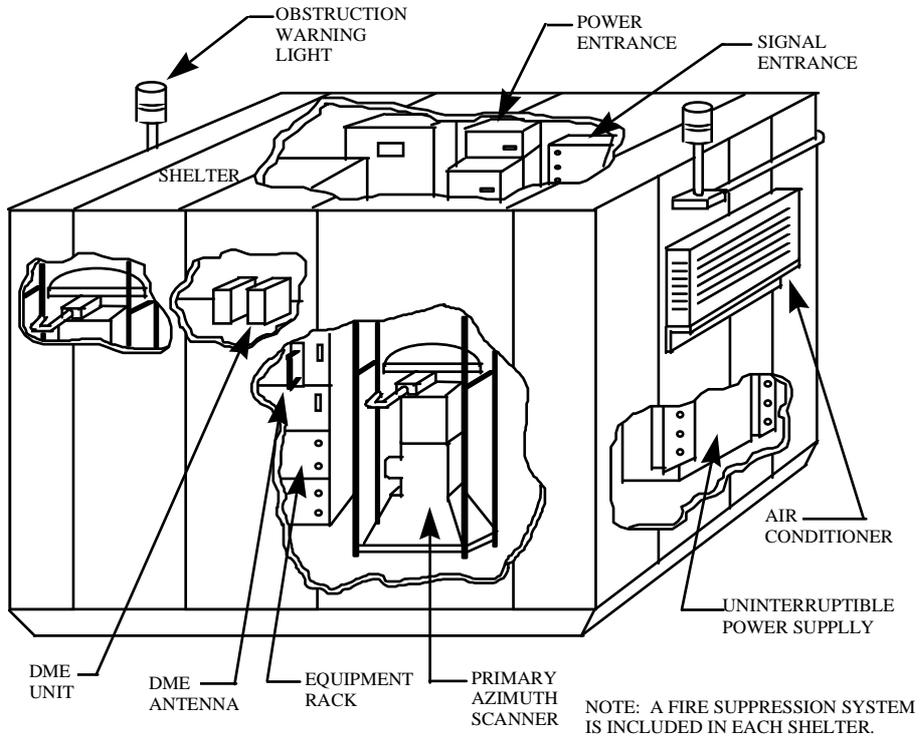


Figure 2-12.- MSBLS-GS AZ/DME shelter and equipment.

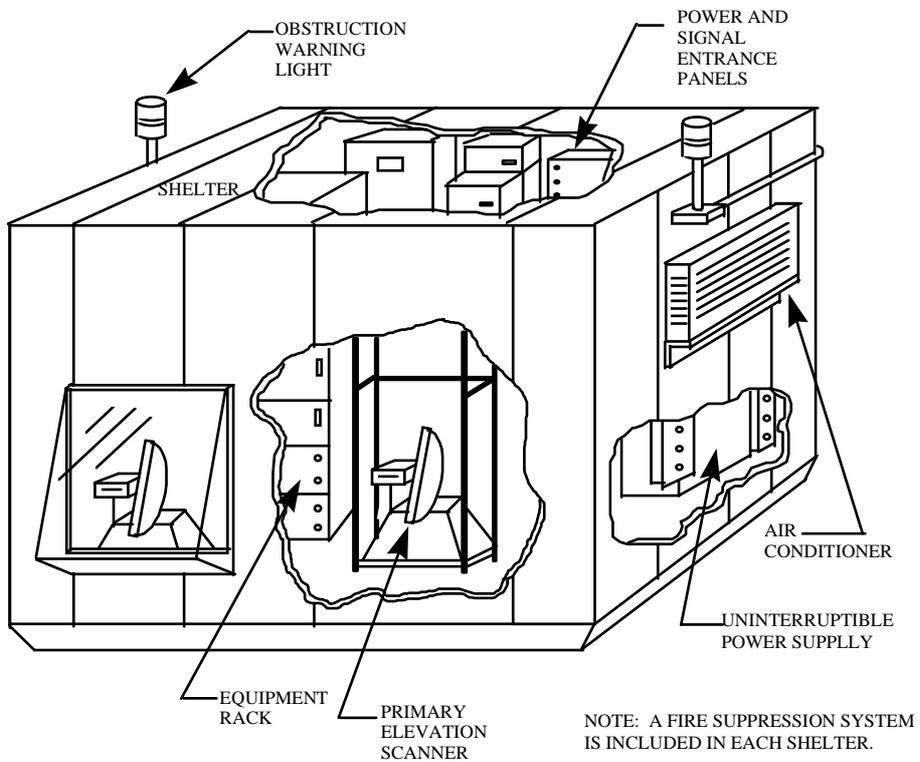


Figure 2-13.- MSBLS-GS elevation shelter and equipment.

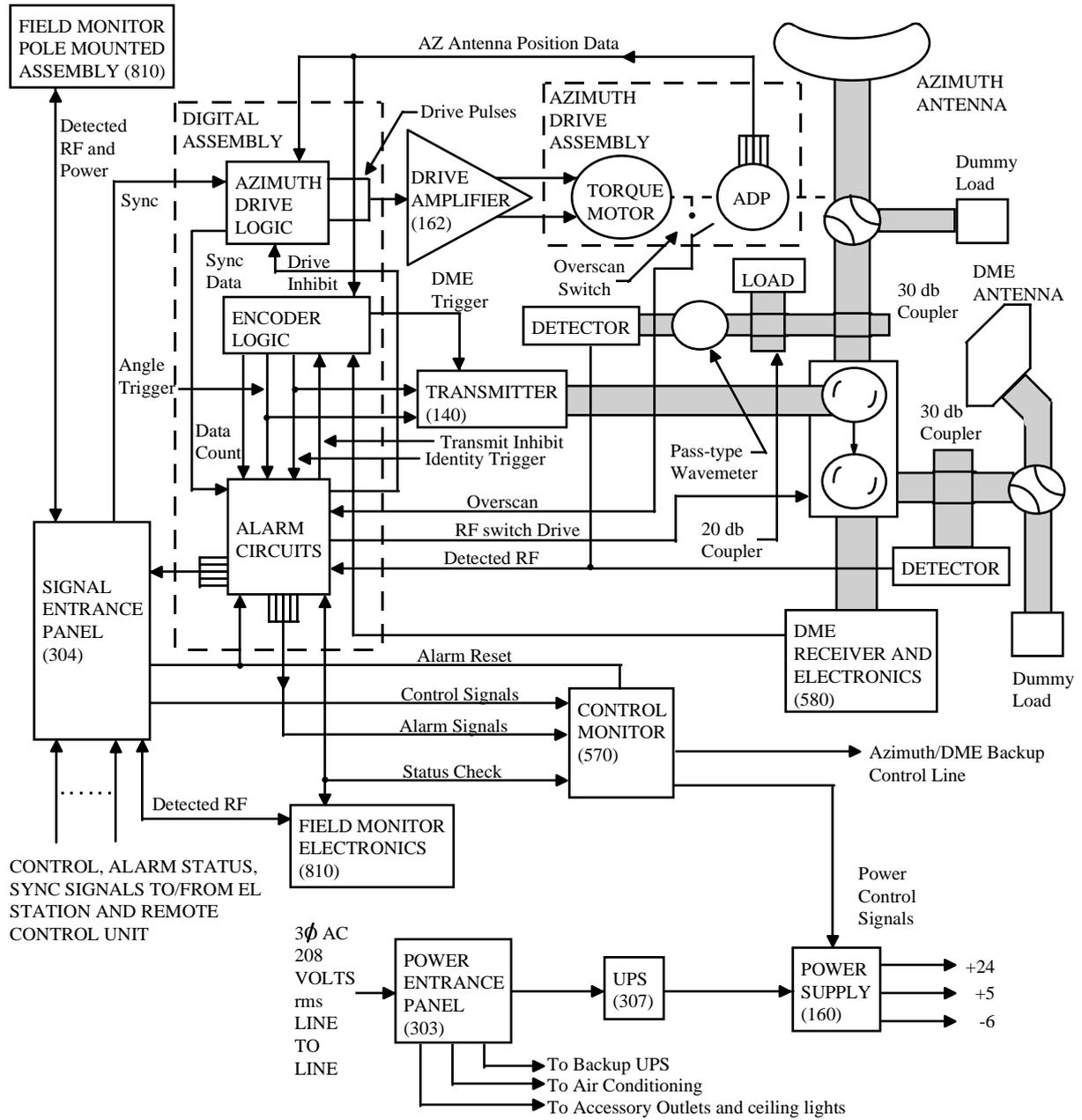


Figure 2-14.- Functional configuration of equipment housed at the MSBLS-GS AZ/DME shelter (only primary string equipment is shown for simplicity).

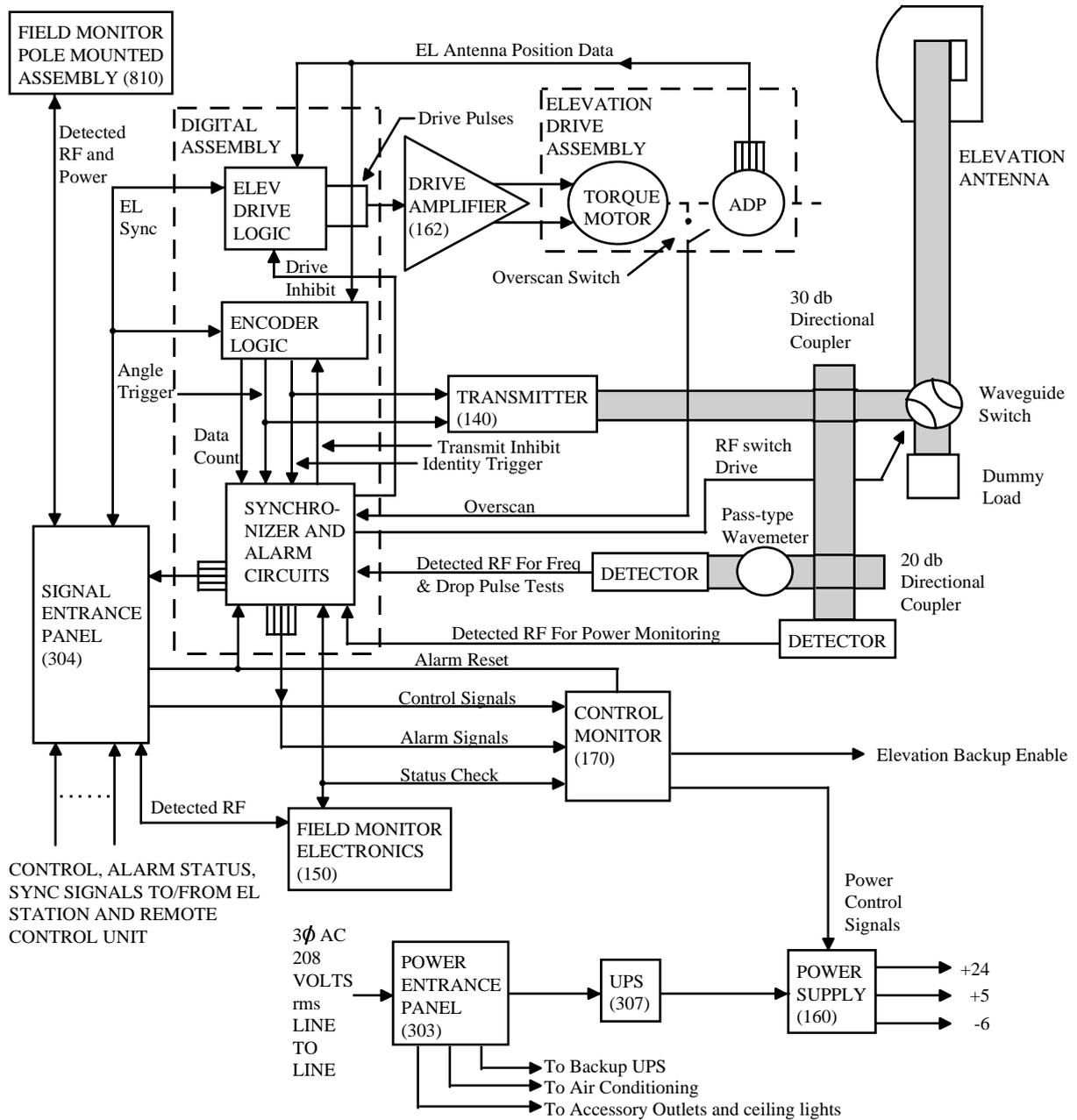


Figure 2-15.- Functional configuration of equipment housed at the MSBLS-GS elevation shelter (only primary string equipment is shown for simplicity).

TABLE 2-III. - LAND SITE DATA CHART
ETR low inclination

| Area | Location | RWY | TACAN | | MLS | Length (ft) |
|------|------------------------|--------|-------------|------------|-----|----------------|
| | | | (PRI) | (SEC) | | |
| 1 | KSC | KSC 15 | TTS 59Y | COF 97 | 8 | 15,000 |
| | | KSC 33 | TTS 59Y | COF 97 | 6 | 15,000 |
| 2 | Dakar Amilcar CBRL | YOF 01 | YOF 68 | YF 78(DME) | 8 | 11,500 |
| | | AML 02 | YOF 68 | YF 78(DME) | | 10,700 |
| 3 | Moron | MRN 21 | MRN 100 | AOG 23 | | 11,840 |
| | | MRN 03 | MRN 100 | AOG 23 | | 11,840 |
| 4 | Las Palmas Kinshasa | GDV 03 | GDV 76(DME) | | | 10,200 |
| | | KIN 25 | KW 70(DME) | | | 15,500 |
| 5 | Darwin | DDN 29 | DN 84 | | | 11,040 |
| | | DDN 11 | DN 84 | | | 11,040 |
| 6 | Edwards | EDW 22 | EDW 111 | PMD 92 | 8 | 15,000 |
| | | EDW 17 | EDW 111 | PMD 92 | | 6 |
| 7 | Honolulu | HNL 08 | HNL 100 | CKH 86 | | 12,000 |
| | | HNL 26 | HNL 100 | CKH 86 | | 12,000 |
| 8 | Northrup | NOR 17 | SNG-121 | HMN 92 | 8 | 15,000 |
| | | NOR 23 | SNG-121 | HMN 92 | | 15,000 |
| 9 | Edwards | EDW 23 | EDW 111 | PMD 92 | | 14,880 |
| | | EDW 15 | EDW 111 | PMD 92 | | 15,000 |
| 10 | Edwards Northrup | EDW 04 | EDW 111 | | | 15,000 |
| | | NOR 05 | SNG-121 | | | 15,000 |
| 11 | Diego Garcia | JDG 31 | NKW 57 | | | 12,040 |
| | | JDG 13 | NKW 57 | | | 12,040 |
| 12 | Riyadh | RIY 01 | RIY 92 | EKI 80 | | 10,750 |
| | | RIY 30 | RIY 92 | EKI 80 | | 11,780 |
| 13 | Guam | GUA 06 | UAM 54 | GUM 58 | | 11,220 |
| | | GUA 24 | UAM 54 | GUM 58 | | 11,220 |
| 14 | Souda | SUD 29 | SUD 28 | SUD 49 | | 11,810 |
| | | SUD 11 | SUD 28 | SUD 49 | | 11,810 |
| 15 | Hao Hoedspruit | HAO 12 | HA 80 | | | 11,090 |
| | | HDS 18 | HS 73 | | | 13,120 |

2.4.2 MSBLS-GS Switchover Logic

Operation of the MSBLS-GS is automatically monitored by alarm circuitry. Each of the two transmitter groups, primary and backup for azimuth/DME and primary and backup for elevation, contains circuitry which observes internal signals and monitors their status in addition to the external field monitor sets which monitor transmissions. Table 2-IV lists the alarms and the out of tolerance conditions associated with them. Of particular interest is the field monitor alarm which protects against azimuth and elevation biases greater than ± 0.125 deg and DME biases greater than ± 246 ft.

Occurrence of any alarm on the primary string will cause an automatic switchover to the backup string. A subsequent alarm on the backup string will automatically inhibit all transmission from the ground station. The operator has the capability to bypass an alarm (other than an overscan alarm) and enable transmission on the primary or backup string. All out of tolerance conditions are continuously displayed to the operator on the remote control unit.

Switchover will normally cause small step changes in received angle data and momentary NAVSET flagging because the primary and backup scanners run asynchronously.

2.4.3 MSBLS-JR

The MSBLS-JR ground station is a low cost MLS ground station planned for use at several Shuttle landing sites. The MSBLS-JR hardware configuration is illustrated in figure 2-16. The MSBLS-JR consists of a collocated azimuth/elevation/DME shelter located near the runway threshold. The MSBLS-JR does not provide coverage during rollout.

2.4.4 MCC MLS Ground Station Status Reports

The landing site officer (LSO) will be immediately notified in the event of an MLS ground station failure, and he will then report the failure to the flight dynamics officer (FDO) who will alert all Mission Control Center (MCC) operators.

TABLE 2-IV. - TRANSMITTER GROUP ALARM CRITERIA

| Alarms | Criteria | Shelter group |
|---------------|---|--------------------------|
| Overscan | The elevation (azimuth/DME) antenna scanner exceeds 28° (48°) travel. | Azimuth/DME Elevation |
| Field monitor | Two or more field monitors indicate that the angle signal monitored deviates from the nominal angle by greater than ±0.125° or DME transmission deviates by greater than ±246 ft. | Azimuth/DME Elevation |
| Transmitter | Four or more detected RF pulses are absent during a sector gate. | Azimuth/DME Elevation |
| Identity | Spacing between identity and data pulse pair is not within +¼ µs to -¾ µs tolerance or transmitter power is less than 1.6 kW. | Azimuth/DME |
| Identity | Identity pulse and data pulse spacing error is greater than +¼ µs to -¾ µs, three or more times during a sector gate, or transmitter power is less than 1.6 kW, or transmitter frequency is out of tolerance. | Elevation |
| Coding | Selected angle trigger pulses are spaced in error by more than ±¼°. | Azimuth/DME Elevation |
| Data | The number of angle data pulses from the angle data pickoff in any one scan is in error by more than ±1 pulse. | Azimuth/DME Elevation |
| Sync | An overlap exists between the azimuth, elevation, and DME sector gates. | Azimuth/DME |
| Switch | Five or more indications of a switch failure during a DME or azimuth sector gate or transmitter power is less than 1.6 kW. | Azimuth/DME |
| UPS* | (1) Batteries are drained to below 10% of capacity, or (2) UPS has reverse transferred directly to the ac input lines, or (3) the ac input lines to the UPS have failed. | Azimuth/DME Elevation |
| Temperature* | Temperature inside the shelter exceeds the range of ±15° F around 75° F (60° F to 90° F) | Azimuth/DME Elevation |

* UPS and temperature alarm conditions do not cause switchover from primary to backup operation.

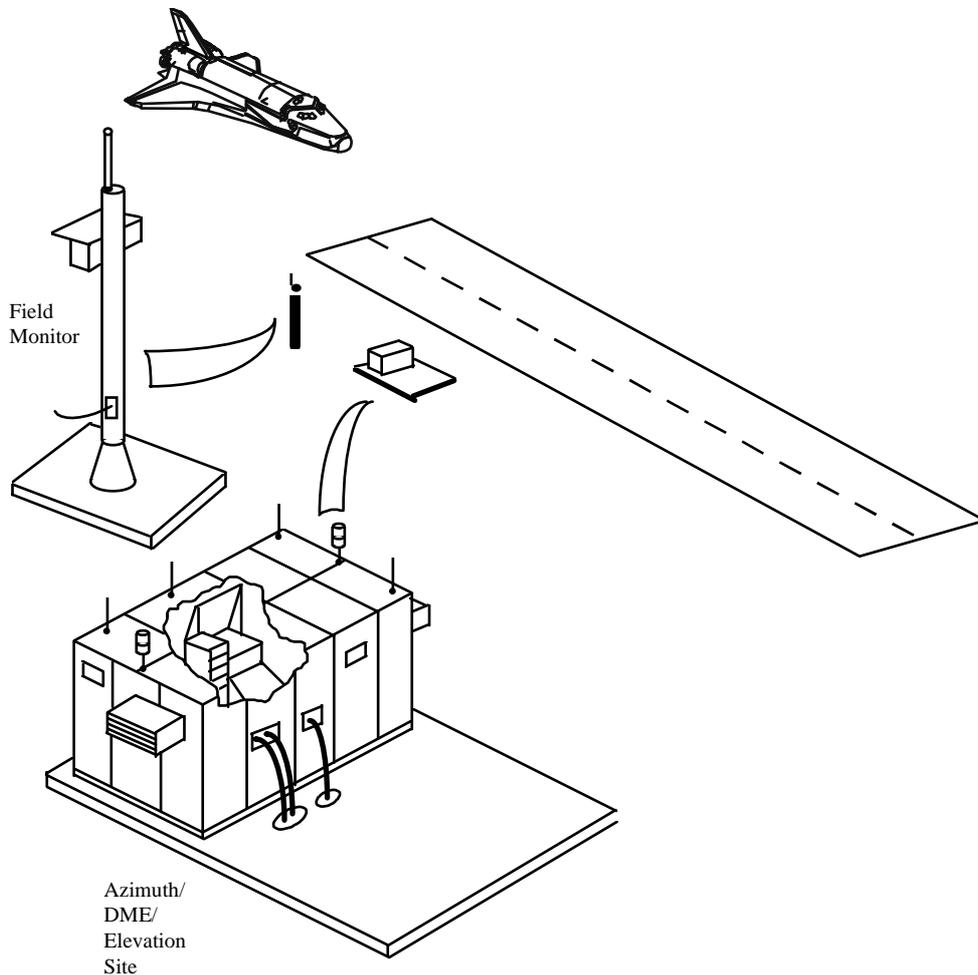


Figure 2-16.- MSBLS-JR installation.

2.5 ORBITER MLS-RELATED HARDWARE

The Orbiter MLS-related hardware and interfaces are illustrated in figure 2-17.

Vertically polarized uplink and downlink MLS RF signals are coupled simultaneously to each of the three redundant NAVSETS through three dedicated forward-looking antennas and separate flexible waveguide runs, shown in figure 2-18. Separate antennas and waveguide runs permit each NAVSET to operate independently. A typical antenna pattern is shown in figure 2-19.

The MSBLS NAVSET RF and decoder LRU assemblies (fig. 2-20) are triply-redundant units which are shock-mounted in Orbiter forward avionics bays 1 and 2 (fig. 2-21). A simplified block diagram of the NAVSET is shown in figure 2-22. A more detailed block diagram is contained in Space Shuttle Systems Handbook (SSSHB) drawing 9.4. Table 2-V summarizes the hardware characteristics of the MSBLS.

The NAVSET RF and decoder assemblies are discussed in sections 2.5.1 and 2.5.2. Crew controls and displays are discussed in section 2.5.3.

2.5.1 RF Assembly

The RF assembly has two basic functions, a receiver input coupler and a transmitter. All input signals are coupled through a circulator to the decoder assembly, which contains the receiver. When the RF assembly receives six tracked DME solicit pulses from the ground stations, the decoder triggers the fixed-tuned magnetron via the modulator in the RF assembly to transmit DME interrogation pulses back to the ground transponder. Orbiter thumbwheel channel selection establishes the interpair identity spacing relationship as mentioned previously. The ground station recognizes this identity coding and will respond only when the channels are the same.

2.5.2 Decoder Assembly

Signal reception, down-conversion, demodulation, and decoding are accomplished in the decoder assembly. The airborne NAVSET is capable of receiving over the range of frequencies from 15.412 to 15.688 GHz. Received signals outside this range are attenuated by an RF filter. Within-band signals are passed through a mixer, whose local oscillator's frequency depends on the channel selected, and are translated to an intermediate frequency (IF) centered at 460 ± 0.5 megahertz (MHz).

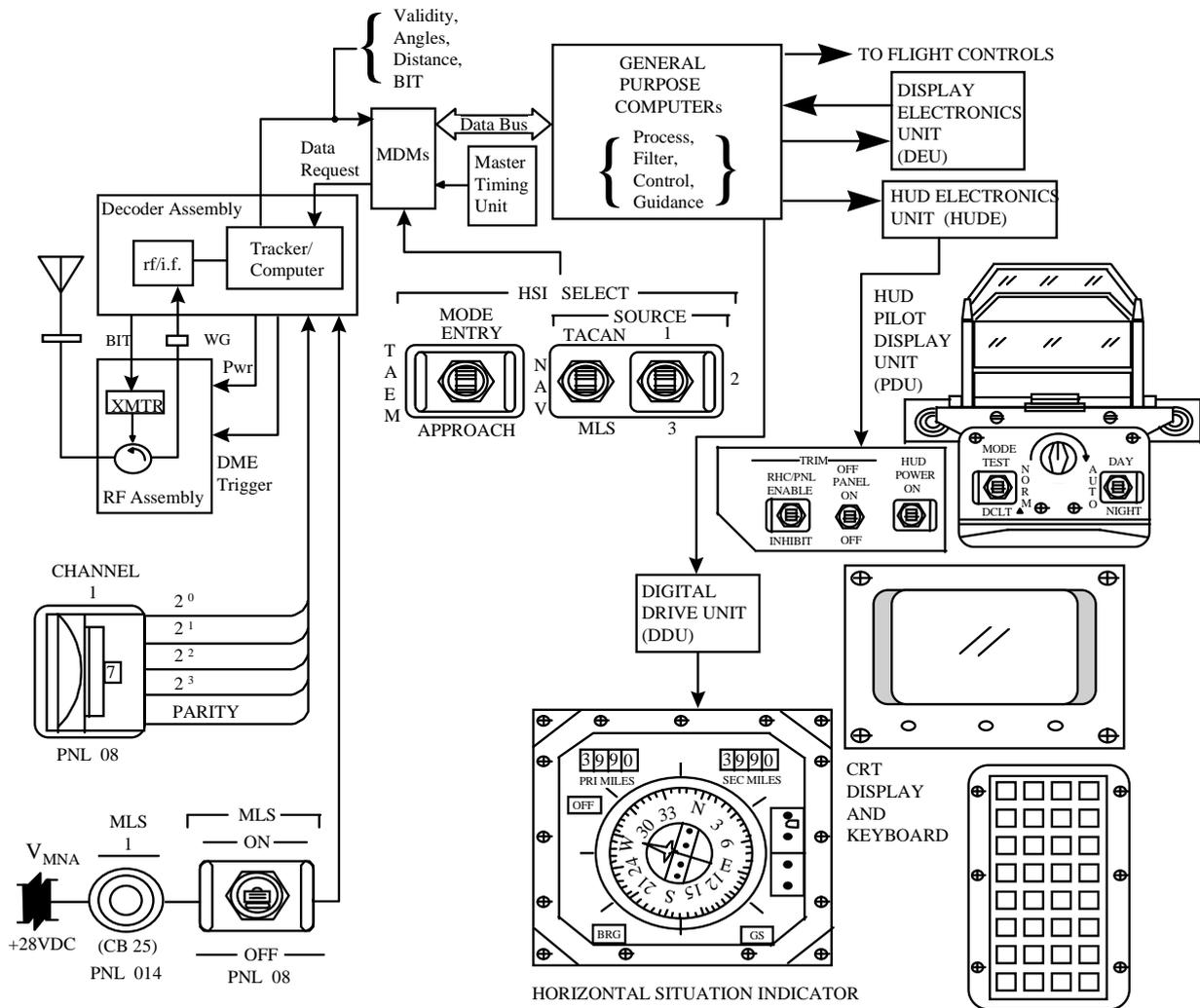
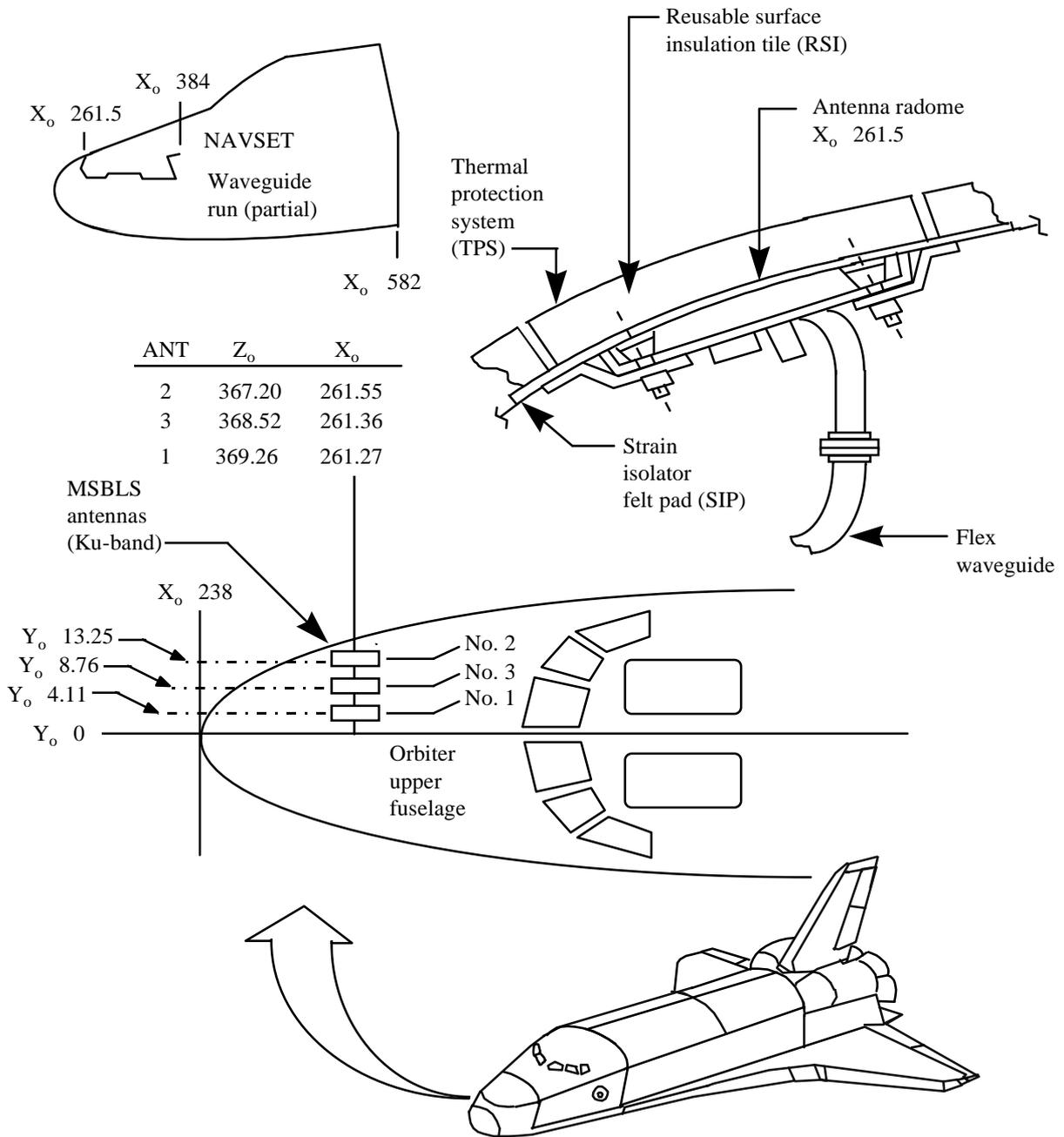
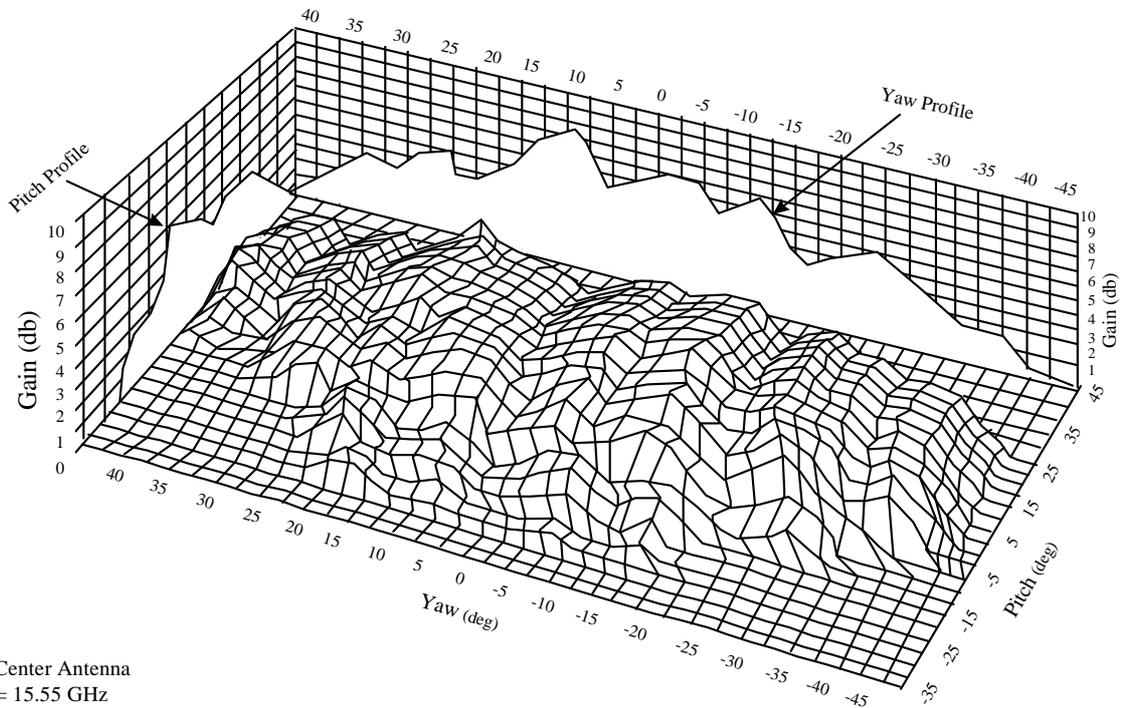


Figure 2-17.- Orbiter MLS-related hardware and interfaces (single string).





NOTES

Combined Center Antenna
Frequency = 15.55 GHz
Pitch increment = 2 deg.
Yaw increment = 1 deg.

Figure 2-19.- Orbiter MLS antenna gain pattern.

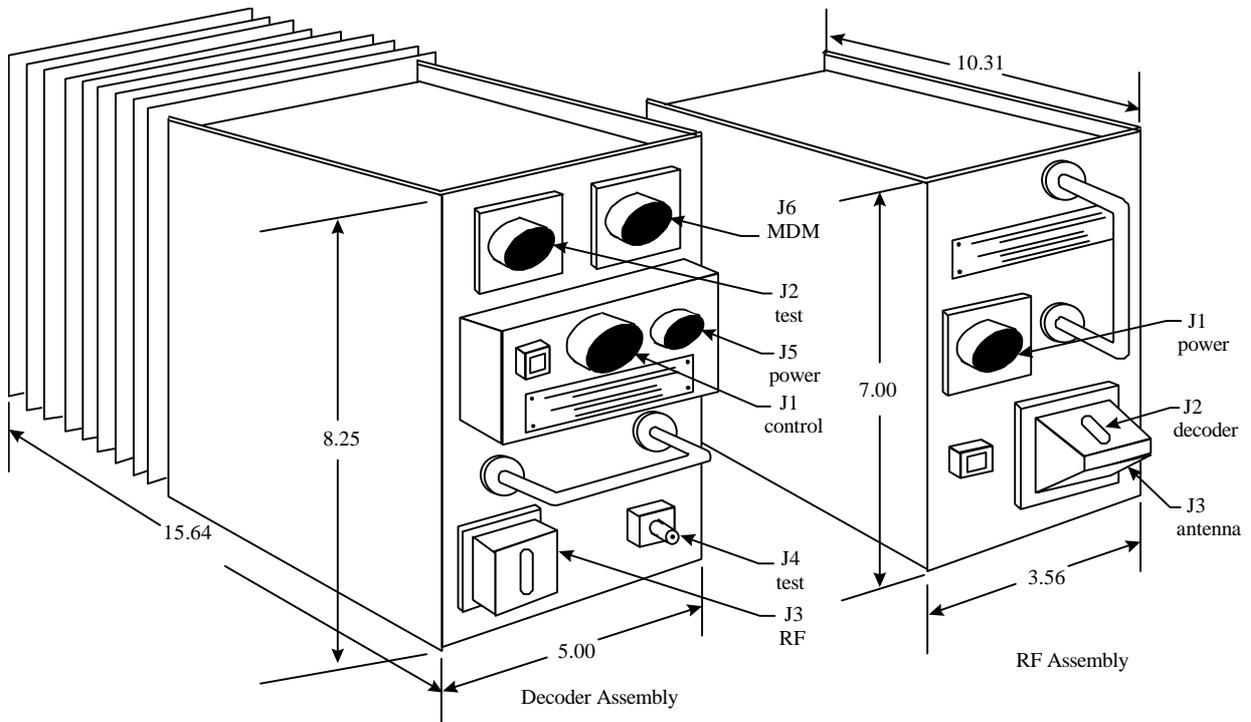


Figure 2-20.- MSBLS navigation set LRU assemblies.

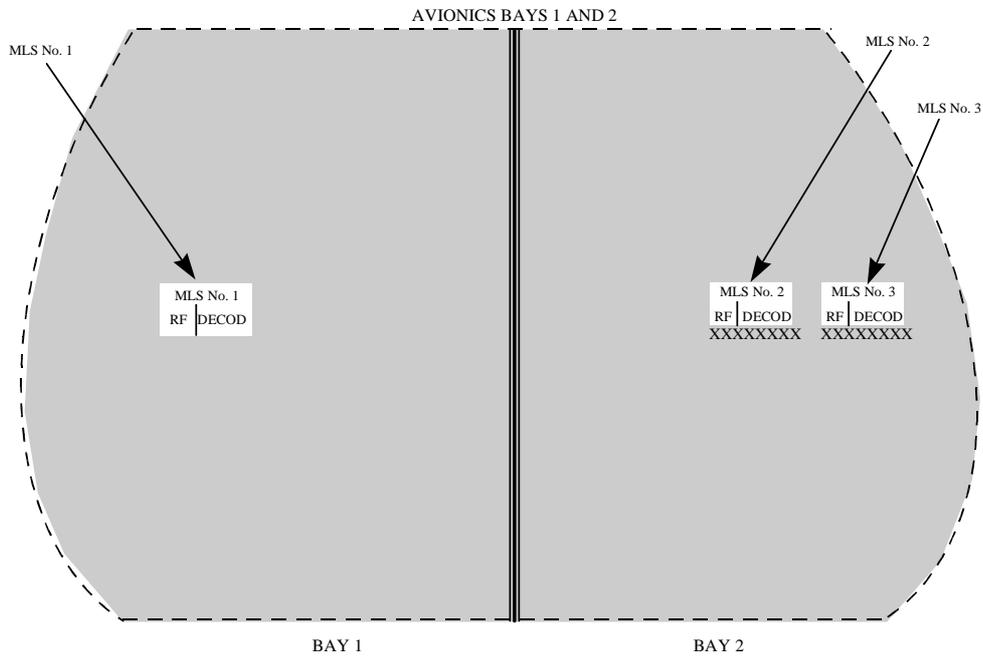


Figure 2-21.- MLS navset locations.

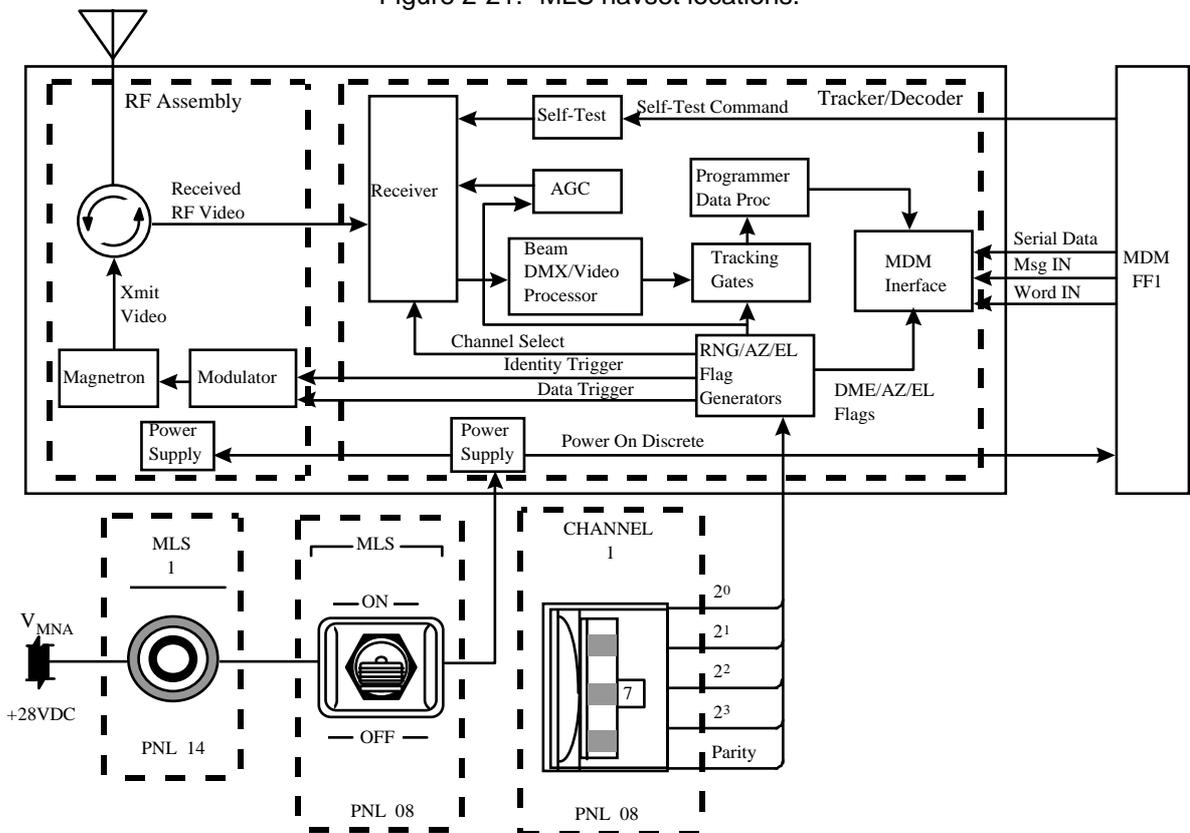


Figure 2-22.- MLS NAVSET simplified block diagram.

TABLE 2-V. - NOMINAL CHARACTERISTICS OF THE MSBLS-NS

| | |
|---|---|
| Frequency | 15.412 to 15.688 GHz (10 channels), uplink 15.460 GHz, DME downlink |
| Receiver | |
| Sensitivity | Angle, -74 dBm DME, -77 dBm |
| Noise figure | 12 dB |
| IF bandwidth | 15 MHz |
| Transmitter (DME) power output | 2 kW peak minimum |
| Accuracy* | Angle: $\pm 0.01^\circ$, 1s Range: ± 50 ft, 1s (.0082 n. mi.) |
| Resolution** | Angle: $\leq 1/128^\circ$ (.0078°) Range: ≥ 25 ft (.004 n. mi.) |
| Data code | Manchester II biphase level |
| Mean time between failures (MTBF) | 2000 hr (specified) |
| Input voltage | 28 V dc |
| Power consumption | |
| Specification | 100 watts |
| Actual | 80 watts |
| Warmup time | |
| Specification | 3 min |
| Actual | 100 sec |
| Size/weight | |
| RF assembly | 10.31 x 3.56 x 7.0 in/6 lb |
| Decoder assembly | 15.64 x 5 x 8.25 in/17.5 lb |
| Cooling | Convection |

* Attributable to NAVSET alone.

** Resolution is defined as the minimum change in receiving antenna position that can be detected by the NAVSET. This is not the same as the weight of the least significant bit in the NAVSET output data word because of the NAVSET averaging process.

Linear video pulses from the IF amplifier are processed by complex logic circuitry to accomplish the following:

- a. Establish/maintain proper signal and tracking levels
- b. Reject interference
- c. Identify beam presence (main lobe)
- d. Identify data type (azimuth fly left/right, elevation, distance)
- e. Determine data values
- f. Identify invalid signals or conditions

This video and data processing involves measurement of pulse amplitudes, timing, counts, and generation of feedback signals and flags.

2.5.2.1 Video and Data Processing

During an azimuth or elevation beam scan, the received signal amplitude varies as the beam passes across the Orbiter MLS antenna. The video pulses from the receiver are monitored for a minimum pulse width of 0.0625 μ s and are quantized in width to 0.25 μ s and in amplitude to one of three logic levels based on pulse amplitude: 1-volt low level video (LLV), 2-volt track level video (TLV), and 3.4-volt high level video (HLV).

The LLV pulses are used to identify the information channel as left/right azimuth, elevation, or range based on the intrapair identity code spacing.

When three consecutive pairs of one type of video pulses are received, a beam gate is generated which addresses the corresponding multiplexer/demultiplexer (MDM) interface memory to receive the computed data.

The angle and range data are computed from measurements of the time duration between TLV pulse pairs. These measurements are averaged for each channel and placed in the designated output buffer as data words.

Automatic gain control (AGC) of the IF amplifier, based on the number of HLV pulses, is employed to establish and maintain a video signal amplitude of 3.4 volts at beam peak for accurate signal processing and decoding. Variation of the signal level from scan to scan results in a variation of the number of HLV pulses during a beam passage. If eight or more video pulse pairs exceed the 3.4-volt HLV threshold, the AGC voltage increases to desensitize the IF amplifier at a 0.8 dB/scan rate (nominal); if less than eight HLV pulses have been received, the AGC sensitizes the IF amplifier at 0.4 dB/scan (nominal). Each information channel (azimuth, elevation, and distance) has its own AGC memory to immediately reestablish the sensitivity level for that parameter from scan to scan. Between beam passages, the sensitivity is increased by 20 dB; however, if an interfering pulse is received, squitter AGC causes the receiver to desensitize again until the next beam is received.

Figure 2-23 is a logic flow chart of the acquisition, search, and track mode criteria. Signal sequences are shown in figure 2-24 to depict events which occur during acquisition, search, and track modes for a typical azimuth beam scan. During acquisition, the high-gain AGC setting causes received pulses to reach the

saturation level. Some of these may be noise pulses. As a ground station beam approaches the NAVSET antenna, three consecutive pulse pairs with the same identity spacing will be decoded. The corresponding AGC and beam gate will be enabled; this reduces the signal amplitude to a level that is considerably below saturation (below threshold for some pulses). The AGC memory contains information from the previous scan so that the gain to which it sets the IF amplifier, on the current scan, is near the level desired for proper beam tracking (0.2 to 10 dB below the beam peak). In the example illustrated, the signal strength was not sufficient to overcome the AGC action during the initial period of pulse reception. In this case, the NAVSET returns to maximum gain to decode intercepted RF pulses again. The AGC and beam gate stay enabled for only 512 μ s unless a low level video pair is detected within this time. The process of decoding the three consecutive pairs, enabling the AGC memory, and looking for low level video is reiterated until low level video overcomes the AGC action. This low level video, coupled with the fact that the AGC memory is on, is indicative of the fact that the main part of the sweeping beam is approaching the Orbiter antenna position. When track level video is received, the NAVSET acquisition and search operation is completed; the NAVSET then enters the track mode. The nominal track levels are -4.6 dB for azimuth and elevation, and -10.6 dB for DME.

2.5.2.2 Flag Generation

Validity checks are made continuously on all information channels being processed.

For azimuth and elevation, if less than 8 or more than 127 pulse pairs are received for elevation or azimuth angle data, or they have been static for the last 380 ms, or 2 consecutive identical type beams (azimuth or elevation) are received, the corresponding azimuth or elevation validity flag will be set. Once declared, valid data must again be received for eight consecutive beams before it can be cleared.

For DME, a range validity flag is generated when valid data has not been received for 0.2 sec or there are less than 8 replies to the 16 interrogations. The flag will be cleared as soon as a single beam of valid range data is received.

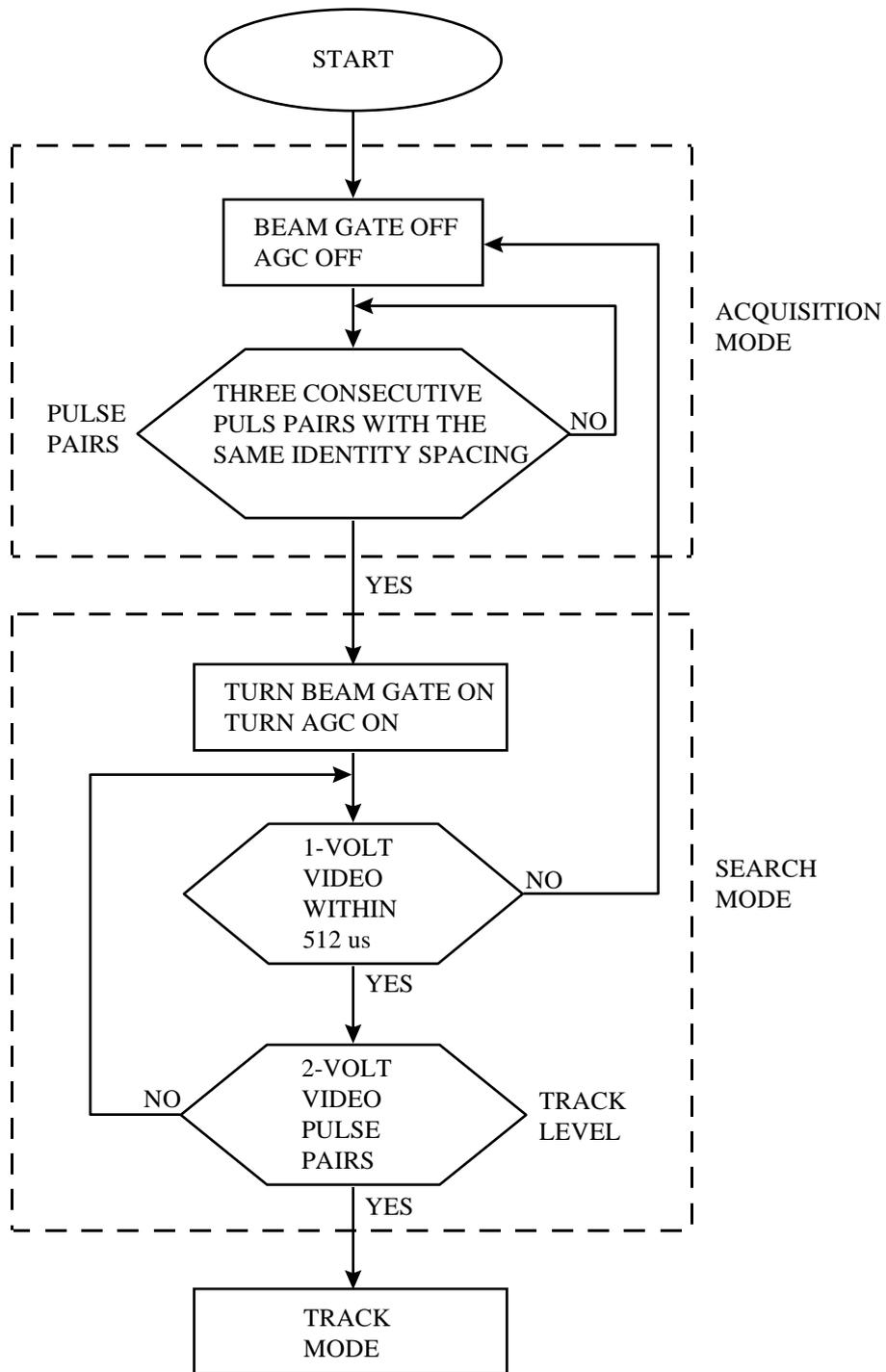


Figure 2-23.- Logic flow of acquisition, search, and track mode criteria.

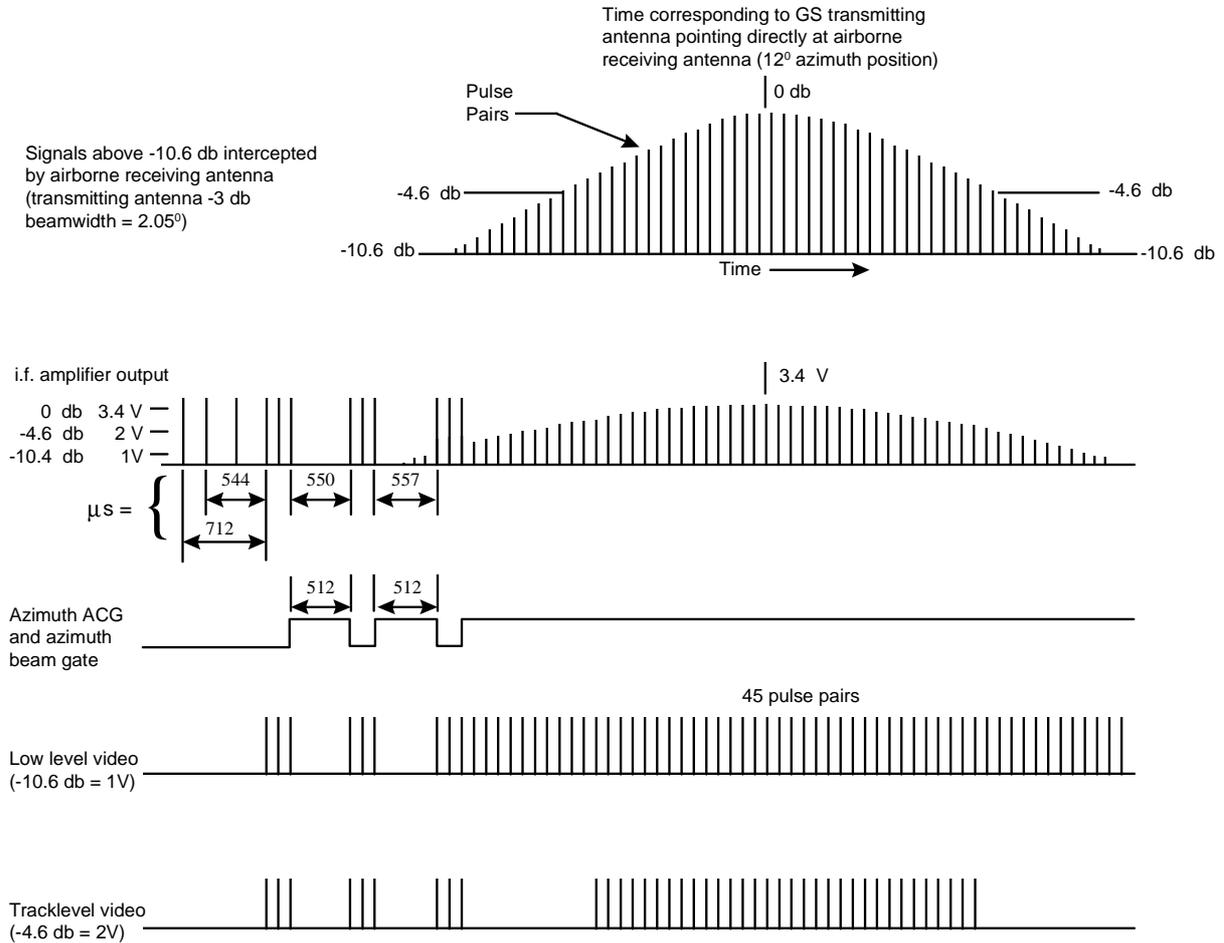


Figure 2-24.- Signal sequences during acquisition, search, and track modes for an azimuth beam pass.

2.5.2.3 MDM Interface

Data transfer between MLS 1, 2, and 3 and the data processing system (DPS) is via flight-critical forward MDM's FF1, FF2, and FF3, respectively. Raw MLS data are also downlinked to the ground on pulse code modulation (PCM) telemetry via operational instrumentation (OI) MDM's at 1 sample/sec. For azimuth, elevation, and range data to be transmitted from a NAVSET to an MDM, proper reception of message and word discretes from the MDM are required. Message and word discretes and serial data are differential signals with timing as shown in figure 2-25.

When the request for data from the MDM is detected, the interface replies with the most recent data available. The data word is processed and loaded serially into the MDM interface. Each information channel is processed as a separate data word. Figure 2-26 illustrates the contents of each data word which is transmitted to the MDM at a 1-megabit rate.

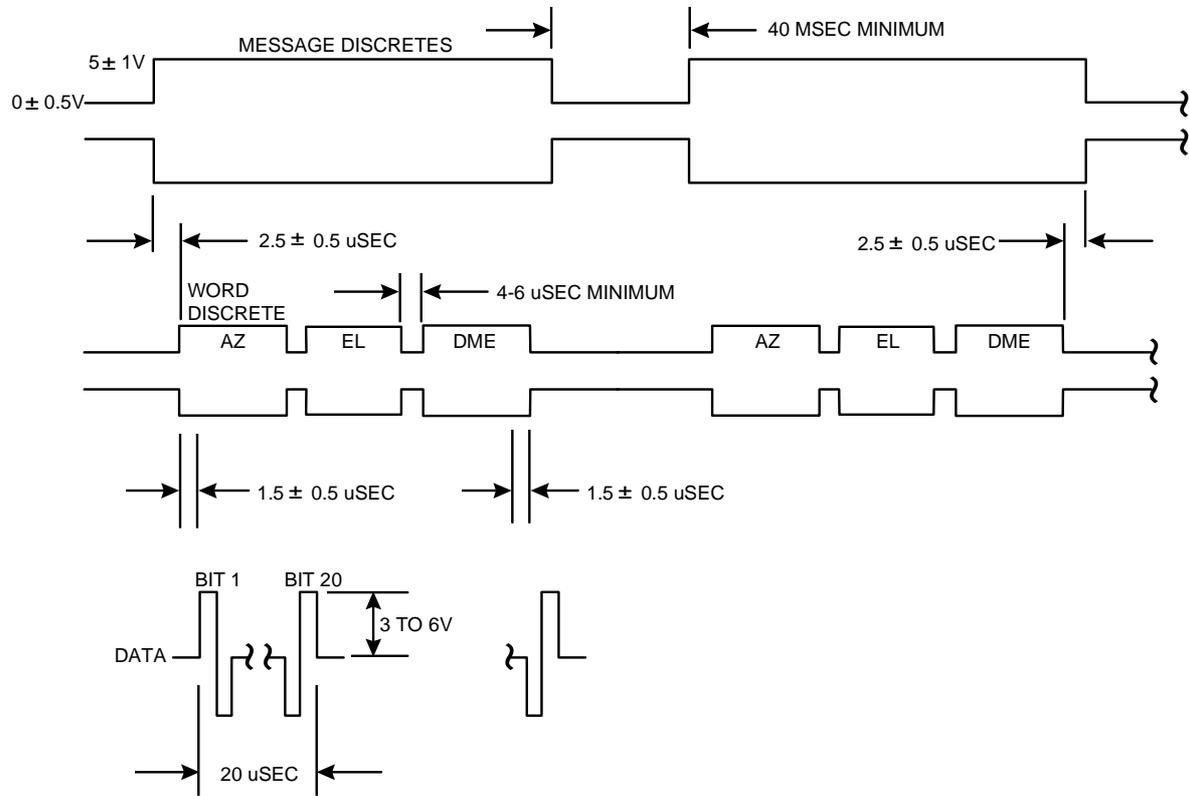


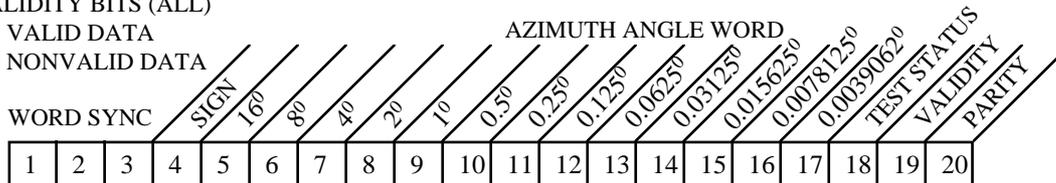
Figure 2-25.- MDM data request discrettes.

TEST STATUS BITS (ALL)

0 = NOT IN TEST
1 = PERFORMING BIT

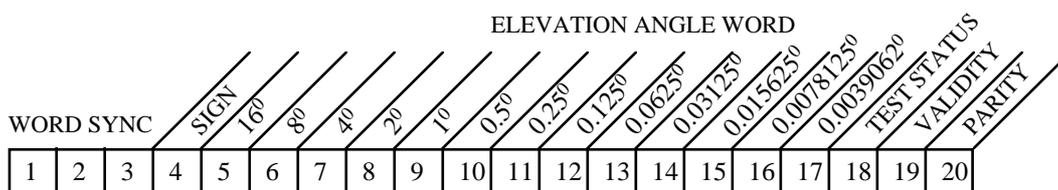
VALIDITY BITS (ALL)

0 = VALID DATA
1 = NONVALID DATA



SIGN BIT

0 = PLUS OR FLY LEFT (RIGHT OF CENTERLINE)
1 = MINUS OR FLY RIGHT (LEFT OF CENTERLINE)



SIGN BIT FOR ELEVATION AND RANGE WORDS

0 = NORMAL DATA
1 = AGC FAILURE DURING SELF TEST, OR MIXER CRYSTAL CURRENT FAILURE DURING NORMAL OPERATION

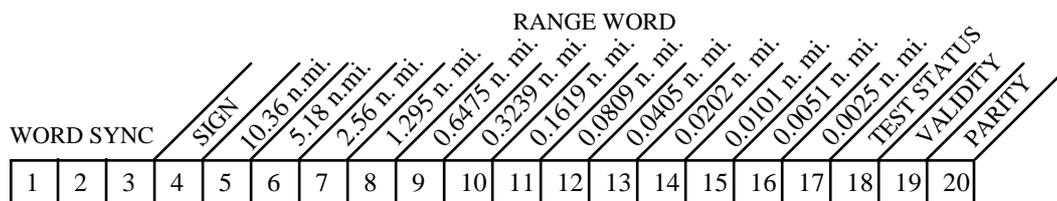


Figure 2-25.- Serial word format for each information channel

2.5.2.4 Built-in test equipment

NAVSET built-in test equipment (BITE) circuitry provides a method to check for failures throughout the functional path of the system continuously during normal operation and during a specially commanded self-test routine. Transmitting, receiving, tracking, and processing functions are checked by the BITE. The self-test can be initiated on the ground or in-flight by the crew. Simulated azimuth, elevation, range-solicit, and reply signals are generated within the NAVSET to provide a total end-to-end check. A total of 32 beam passages are generated, including a data pulse for all three information channels. The first 16 passages provide a test for the fly-left decoding while the last 16 passages are used to test the fly-right circuitry. Table 2-VI describes the predetermined hardware state generated when self-test is initiated. Since flags and AGC voltages must be allowed to reach the final state, monitoring of the output words is not performed during the first two sec. After the first 3.2 sec, if less than 12 of the required 16

interrogation pulses and their replies are tracked by the decoder assembly, replies to subsequent interrogations will be inhibited, causing a flag condition (data invalid). In addition to MDM monitoring of the serial data output, AGC voltage is monitored during on-orbit self-test, and mixer crystal currents are monitored continuously (during both on-orbit self-test and normal operation) by circuitry internal to the decoder assembly. A failure of either will result in an improper indication in the sign bit for elevation and range (-). Detection of a parity error in the channel selection lines at any time will result in the generation of parity errors in the serial data output words.

TABLE 2-VI. - CONDITIONS ESTABLISHED BY THE NAVSET BITE

| Information channel (function) | Intrapair timing (μs) | Interpair timing (μs) | Serial output data | Bit status output interval (sec) |
|--------------------------------|-----------------------|-----------------------|-------------------------------------|----------------------------------|
| Elevation | 12 | 72 | $6.0 \pm 0.1^\circ$ | 2 to 6.4 |
| DME (solicit) | 8 | 155.75 | None | 2 to 6.4 |
| DME (reply) | 8 | 155.75 | 15.2 ± 0.2 n. mi.* | 2 to 6.4 |
| Azimuth | 10 | 66 | $+3.0 \pm 0.1^\circ$ (fly left) | 2 to 3.2 |
| Azimuth | 14 | 66 | $-3.0 \pm 0.1^\circ$ (fly right) | 3.2 to 6.4 |

* Interrogation-to-reply interval = 267.625 μs

2.5.3 Crew Controls and Displays

2.5.3.1 Power Control

The MLS receives power from main buses A, B, and C via the circuit breakers (cb) on panels 014, 015, and 016 (fig. 2-27). When the cb's are activated, power is applied to the MLS power ON/OFF toggle switches on panel 08. When the toggle is in the ON state, the MLS will receive dc power (+28 V dc) and a power on discrete (POD) is sent to the MDM.

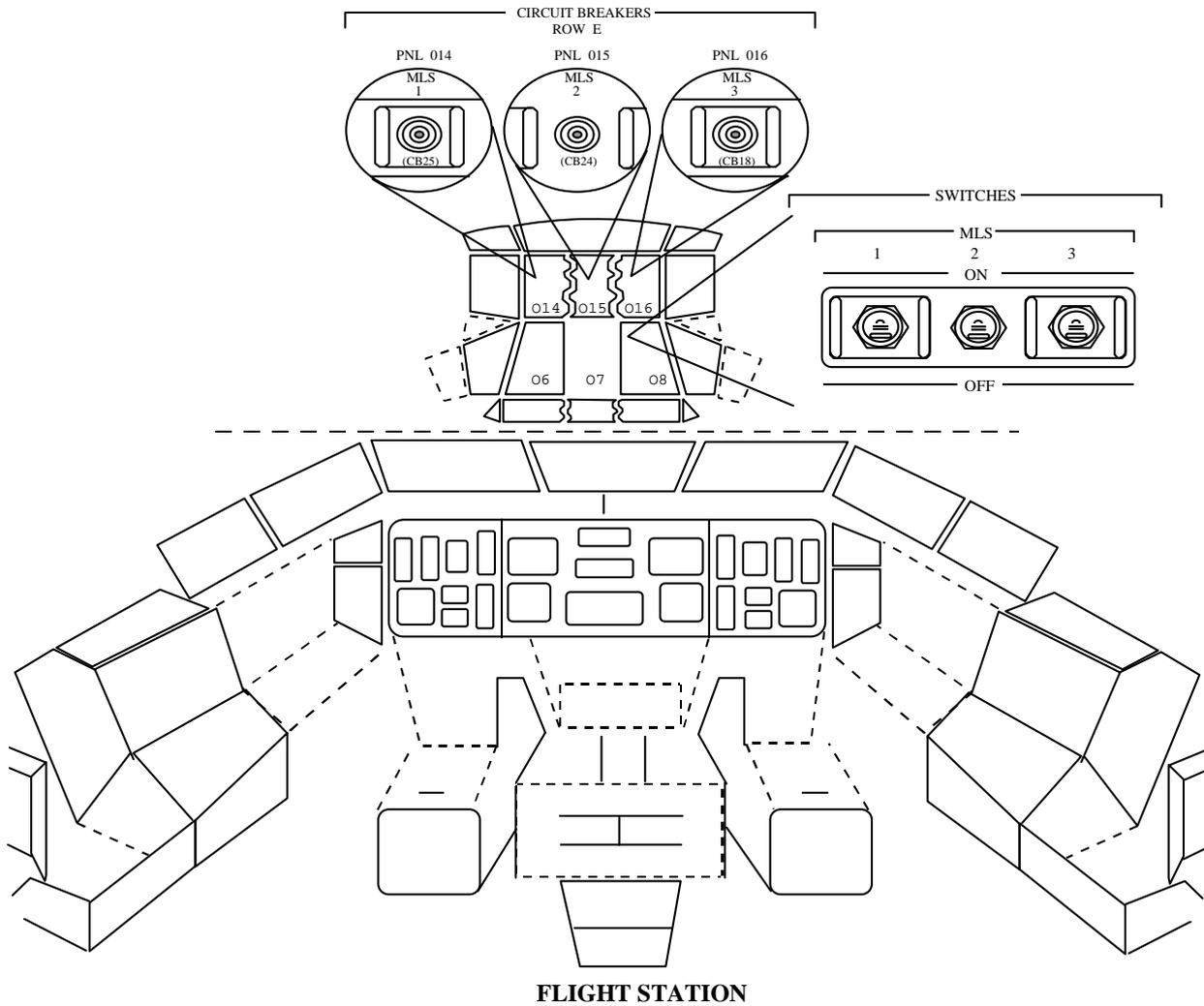


Figure 2-27.- MLS power control.

2.5.3.2 Channel Select

The CHANNEL select switches are located on panel 08 (fig. 2-28). A separate CHANNEL select switch is dedicated to each MLS. The CHANNEL select switch is a 10-position thumbwheel switch (0-9) that allows the crew to select the MLS channel for the ground station located at the selected runway.

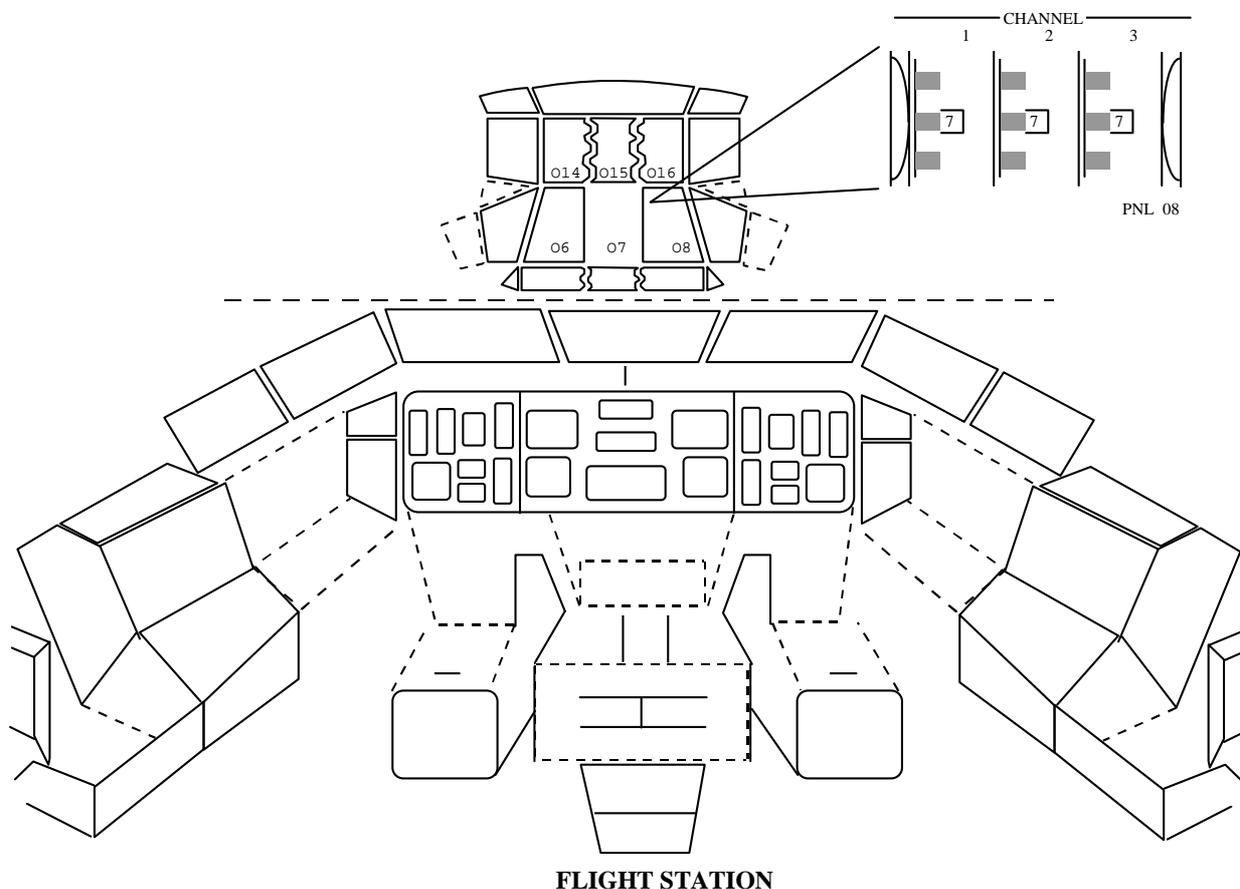


Figure 2-28.- MLS channel select.

2.5.3.3 Horizontal Situation Indicator Select

The horizontal situation indicator (HSI) SELECT switches, which are located on panels F6 and F8 under each HSI (fig. 2-29), control the data input to the HSI's. The MODE switch provides a signal to the GPC's which select data for display for the entry, terminal area energy management (TAEM), or approach phase of flight. When the SOURCE switch is set to MLS, the HSI will display area NAV data calculated using raw MLS data (as opposed to TACAN) from one of the three MLS NAVSETS, as selected on the adjacent toggle switch. When the SOURCE switch is set to NAV, the HSI will display data from the NAV filter.

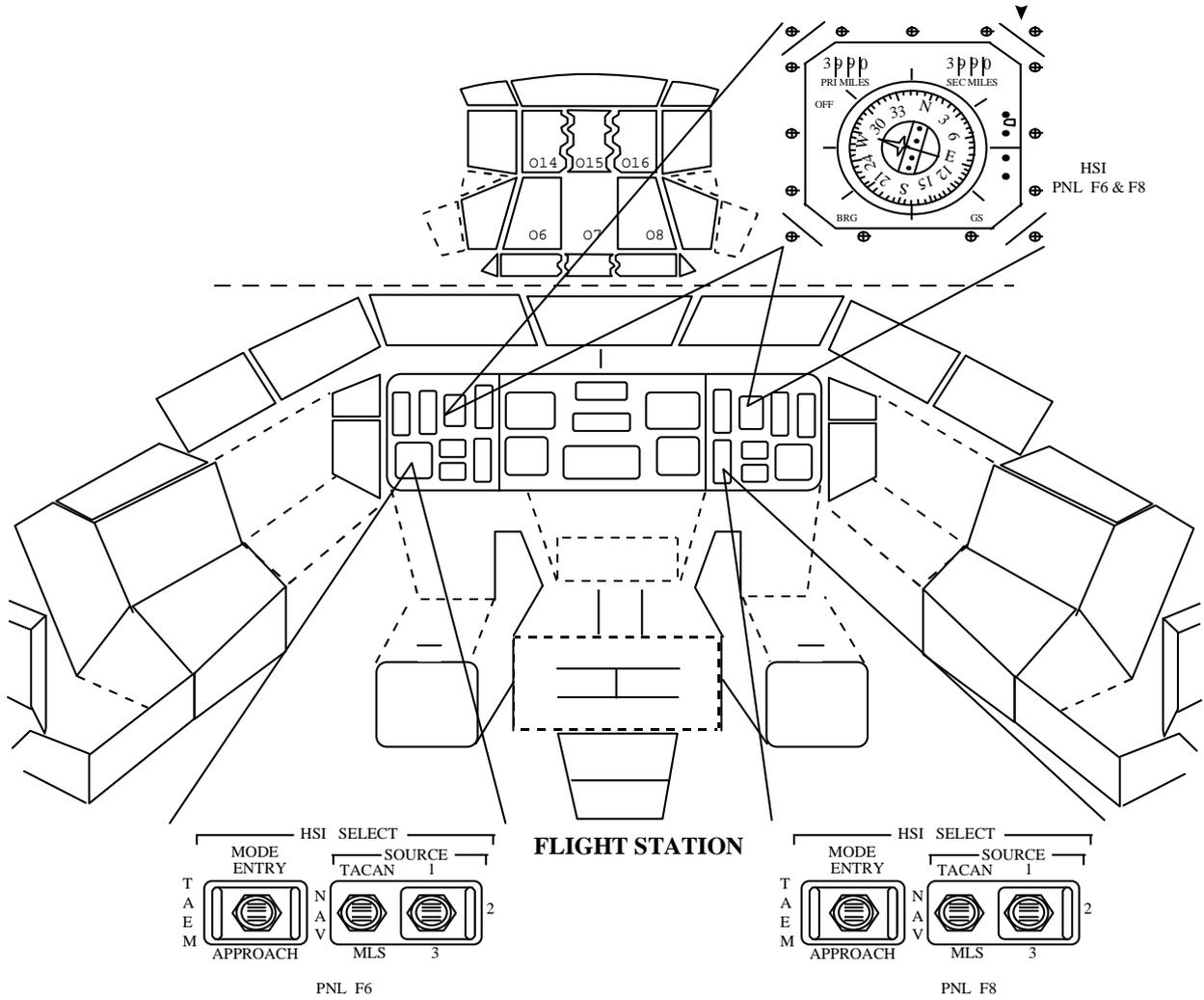


Figure 2-29.- HSI and controls.

2.5.3.4 Horizontal Situation Indicator

Raw MLS data or MLS derived information from area NAV or NAV filter are presented to the crew by two dedicated display instruments, the HSI's (depending on the position of the source switch) shown below in figure 2-30.

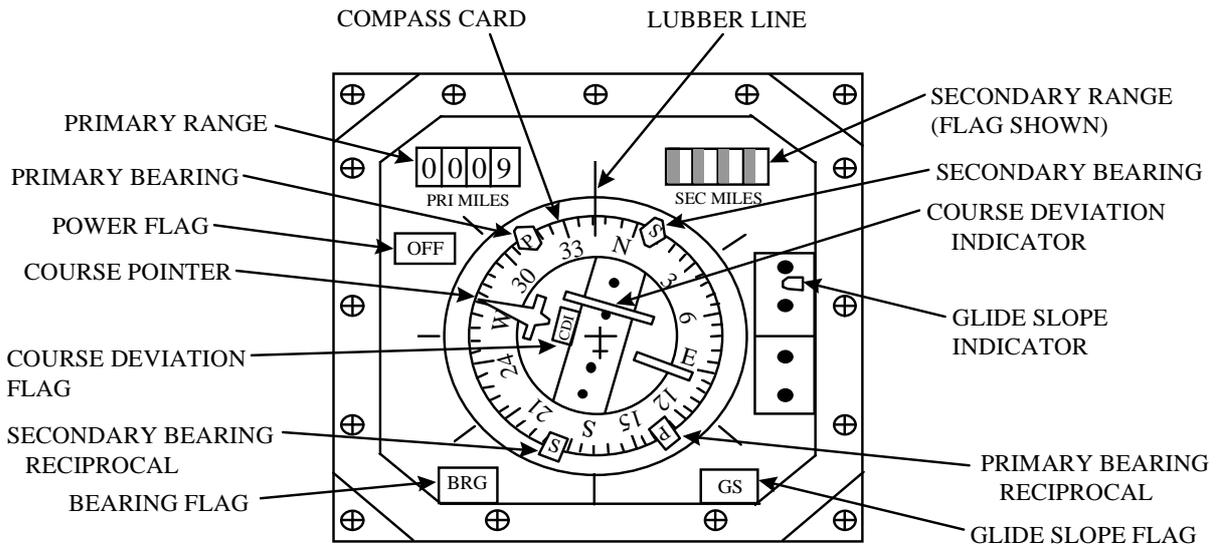


Figure 2-30.- HSI

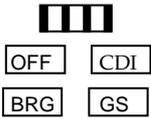
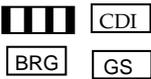
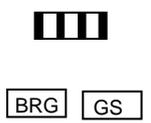
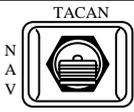
The course deviation indicator (CDI) is displaced from the center of the HSI by MLS azimuth data. Displacement towards the dots on the right tell the crewman to fly toward the right. Corrections are always made by flying in the direction of displacement.

Elevation data drives the glide slope deviation indicator. Elevation angle is represented by a pointer and accompanied by a scale. When the Orbiter is below the desired flight path angle, the pointer indicates this by moving to a position above the HSI centerline. In this manner, the pointer is a fly-to indicator.

Range information is displayed in the upper right and left corners. This range is the distance to the primary and secondary waypoints.

HSI flag functions are summarized in table 2-VII for the entry phase.

TABLE 2-VII. - HSI - FLAG FUNCTIONS (ENTRY PHASE)

| Indication | Cause | Effects |
|---|---|---|
|  | <ul style="list-style-type: none"> HSI voltage low | <p>BRG pointers, compass card, range & course indicators frozen.</p> <p>If voltage less than 17 V, range flags may appear.</p> |
|  | <p>IF: </p> <ul style="list-style-type: none"> TACAN channel not in software (site lookup) | <p>Bearing pointers and CDI frozen, GSI out of view</p> |
|  | <p>IF: </p> <ul style="list-style-type: none"> TACAN range or bearing invalid (Commfault/lock-on) | <p>Bearing pointers frozen, GSI out of view, CDI may be in error.</p> |
| | <p>IF: </p> <ul style="list-style-type: none"> MLS range or bearing invalid (Commfault/lock-on) | |
|  | <p>IF: </p> <ul style="list-style-type: none"> BARO altitude invalid (air data RM) | <p>GSI out of view, constants or NAV altitude used in bearing and range calculations.</p> |
| | <p>IF: </p> <ul style="list-style-type: none"> MLS elevation invalid (Commfault/lock-on) | <p>GSI out of view, one of two elevation constants used in bearing and range calculations: prior to final flare- 0.33281 MM; after final flare- 0 MM.</p> |
|  | <ul style="list-style-type: none"> HSI internal failure; SERVO AMP feedback miscompare | <p>Compass card and course pointer frozen (bearing pointers continue to drive but may be in error).</p> |

2.5.3.5 Head Up Display

Guidance information is presented to the crew on two head up displays (HUD's) as shown in figure 2-31. Images of flight critical information in various formats from a cathode ray tube (CRT) within the HUD are projected into the commander's and pilot's line of sight, allowing the crew to see the displayed information while looking out the window. Display format 1 (approach and landing) is shown in figure 2-32. Display format 2 (autoland monitor) is shown in figure 2-33. Event cues and crew alerts are displayed (flashing) in the upper right windows of the HUD displays. Cue/alert messages include (in order of priority):

| <u>Cue/alert message</u> | <u>Meaning of message</u> |
|--------------------------|---------------------------------------|
| CSS | CSS in all axes |
| MLS NV | NAV not processing MLS data |
| B/F | Body flap (B/F) not at trail position |

At an altitude of 12,000 ft (I-LOAD) and below, NAV is continuously checked for refreshed MLS data. If MLS data are lost for more than 6 sec after having once been acquired, the MLS NV alert message will be displayed.

Messages on the situation line are left-justified and nonflashing. For the above cue/alert messages, the following rules apply:

- a. Flashing message displayed in upper right window until condition ceases or 5 sec elapse.
- b. If message persists for 5 sec, it is blanked in the upper right window and placed on the situation line where it is retained until the situation ceases.
- c. If a third message is set while both halves of the upper right window are occupied, the lower priority message will wait until a space is vacant.
- d. These messages are blanked at touchdown.

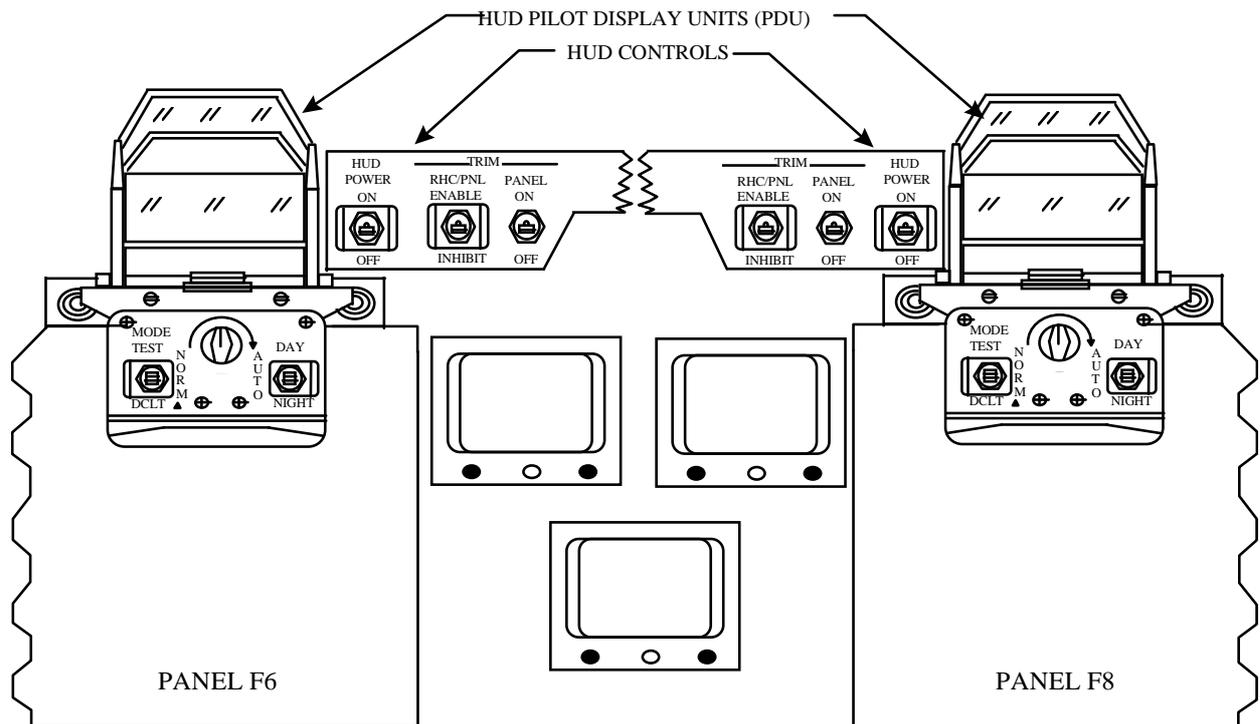


Figure 2-31.- Head up displays.

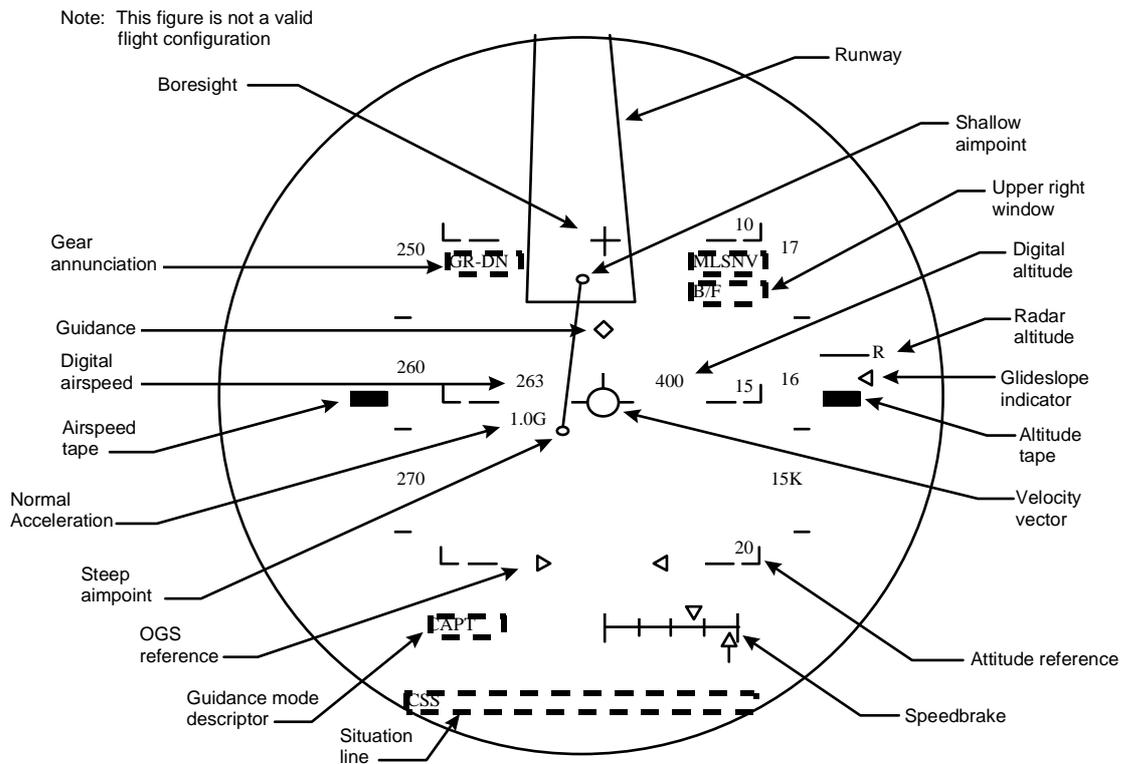


Figure 2-32.- HUD display format 1 (approach and landing).

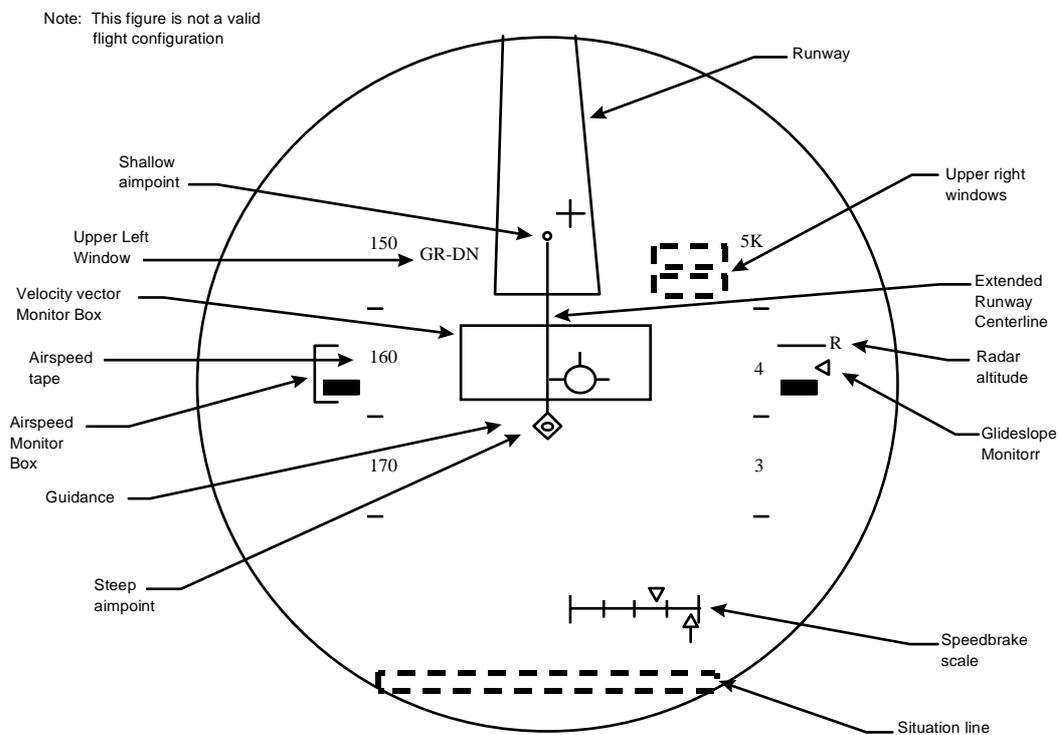


Figure 2-33.- HUD display format 2 (autoland monitor).

2.5.3.6 CRT Display

The MLS data are displayed on two CRT displays in the forward crew station (fig. 2-34). A description of the display formats and information presented is contained in section 2.6.

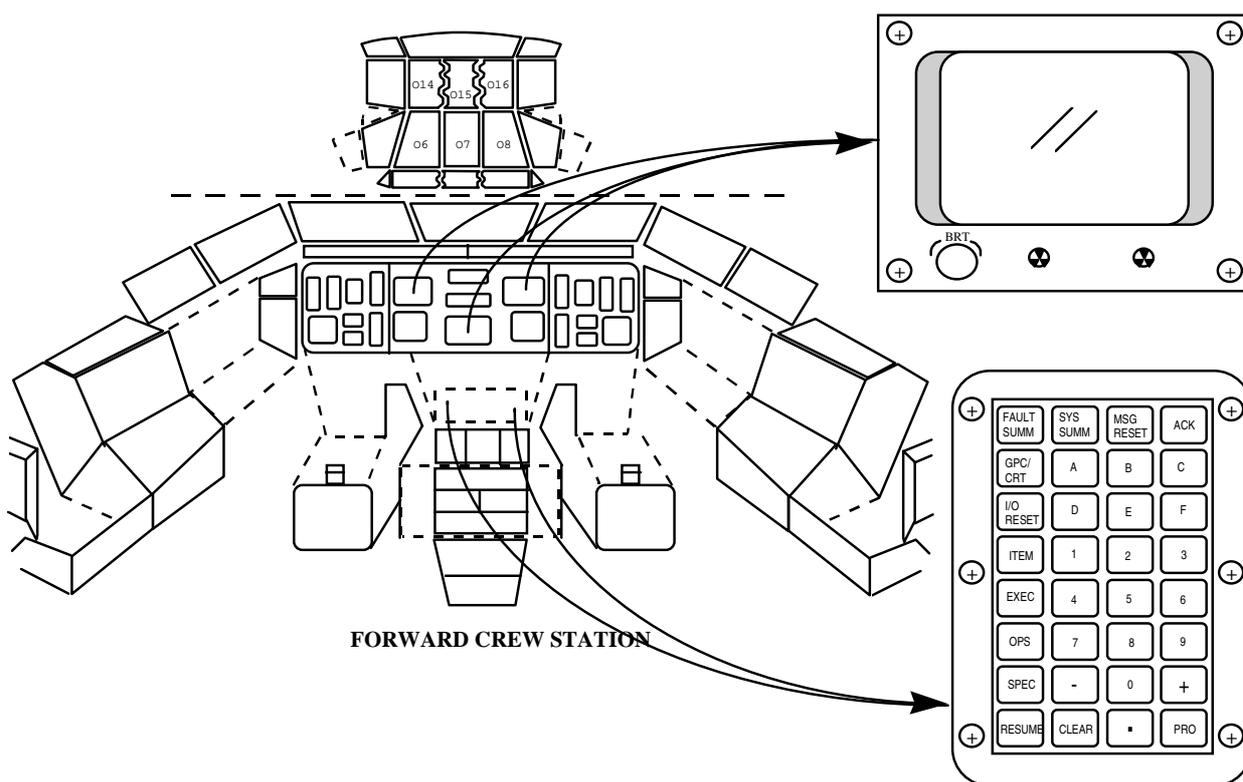


Figure 2-34.- CRT display and keyboard.

2.6 ORBITER MLS SOFTWARE

The three MLS units provide azimuth, elevation, and range data referenced to the antennas located near the runway for NAV updates and crew display. The MLS is operational in the primary avionics software system (PASS) during the operational sequence (OPS) 3 TAEM and approach and landing phases of the flight, and in OPS 6 in the case of return to launch site (RTLS) abort. It is also tested in the on-orbit self-test (GNC OPS 8) prior to deorbit. MLS data are not processed by the backup flight software (BFS). The MLS software begins processing upon transition to major mode (MM) 305 from MM 304 and MM 603 from MM 602. All processing stops upon transition to OPS 9 (postlanding).

MLS units can be selected as candidates for operational software processing only in OPS 8 by crew action (see sec. 2.6.4.2). Also, the capability for manual deselection (downmoding) of MLS units via software is only provided in OPS 8; however, MLS units deselected in OPS 8 remain deselected across OPS transitions. Automatic deselection capability is only provided by redundancy management (RM) starting in MM 305 and once deselected cannot be reselected. The MLS data flow in the Orbiter is shown in figure 2-35.

2.6.1 Subsystem Operating Program

The MLS subsystem operating program (SOP) routes the MLS data to other software modules. The SOP sends three sets of range, azimuth, and elevation with validity and communication fault (CF) flags to RM and the RM-NAV display. The three sets of data are also sent to the area NAV module of NAV software for HSI display processing at a 3.125 Hz rate. RM sends to NAV the selected range, azimuth, and elevation for consideration as a source in state vector and covariance matrix updates. Additionally, the SOP routes various discretets (power-on, etc.) to appropriate users and performs various operations on the data, as indicated in figure 2-35.

The SOP input processor module evaluates the CF status of each MLS. The azimuth, elevation, and range data are compared to the previous cycle. If there has been a change greater than the least significant bit (LSB) of each data word, the new input is stored and timetagged. Additionally, the azimuth data are monitored, updated, and timetagged if the last update occurred more than 320 ms ago. This corresponds to being refreshed every other pass through the update module which runs at 6.25 Hz.

These updated data words are processed through a preselection filter which provides the integerized raw data, data good (DG) flags, and COMM-lock flags for all three information channels at a 3.125 Hz rate.

To minimize errors due to beam truncation, multipath, and weak signal strength, a limit check is performed on the raw data to establish a second level COMM-lock status for azimuth and elevation data. As shown in table 2-VIII, COMM-lock requires the azimuth reading to be less than 13.5°. Elevation readings must fall between 0.5° and 29°. NAV software latches out elevation data below 1.2° as described in section 2.6.3. Additionally, range must be within 15 n. mi. and the Orbiter heading with respect to the runway must be within ±55° to satisfy the azimuth and elevation COMM lock.

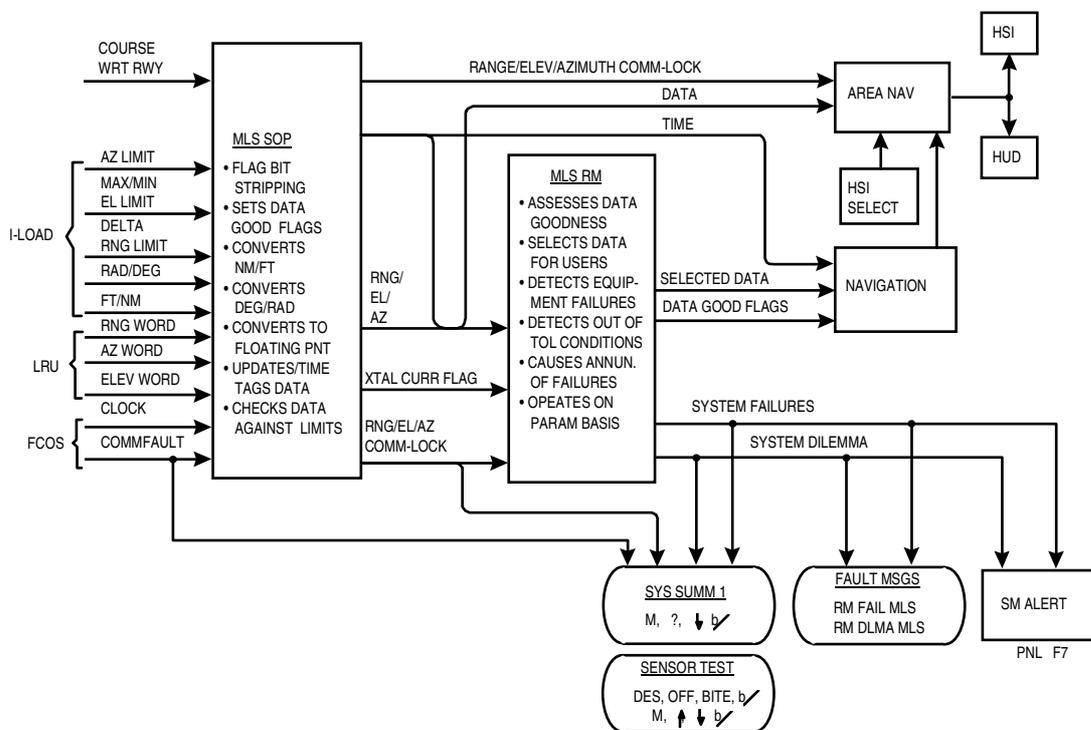


Figure 2-35.- MLS data flow.

TABLE 2-VIII.- MLS DATA LIMIT CHECKS

| Required values for NAV usage | AZ | EL | RNG | Hdg |
|-------------------------------|---------------------|-------------------------------|------------------|------------------------------|
| | $\leq 13.5^{\circ}$ | 0.5° to 29° | ≤ 15 n. mi. | $<\pm 55^{\circ}$ wrt rwy |

2.6.2 Redundancy Management

The MLS RM software performs two major functions: (1) fault detection, isolation and reconfiguration (FDIR) and (2) selection filter (SF). The FDIR function runs in OPS 3 and OPS 6 in the PASS and not in any other OPS nor in the BFS. The SF runs in OPS 3 and OPS 6. MLS RM information is displayed on the GNC SYSTEM SUMMARY 1 CRT display, the HSI dedicated display and the HUD. An RM functional block diagram is shown in figure 2-36.

2.6.2.1 Fault Detection, Identification, and Reconfiguration

The MLS FDIR function detects errors and faults in the azimuth, elevation, and range data and determines which MLS unit(s) (MLS 1, 2, or 3), if any, should be declared failed. Declaring a parameter failed results in fault annunciation and FDIR downmoding of that MLS unit.

The FDIR function is accomplished by monitoring COMM-lock status, data validity flags, BITE flag status, SF/FDI fail status, crew select/deselect status, and comparison of data values between the MLS units. The flag status determines which MLS units are available for processing. A parameter is temporarily withheld from FDIR (but not declared failed) for loss of lock or CF.

Based on availability, the absolute data difference between units (for each information channel) is compared against a miscompare tolerance. The miscompare tolerance limits (also called tracking test limits) for azimuth, elevation, and range are listed in table 2-IX.

TABLE 2-IX.- MLS FDIR MISCOMPARE TOLERANCES

| Maximum MLS parameter differences | AZ | EL | RNG |
|-----------------------------------|---------------|---------------|---------------------------|
| | 0.5° | 0.4° | 2000 ft (0.329 n. mi.) |

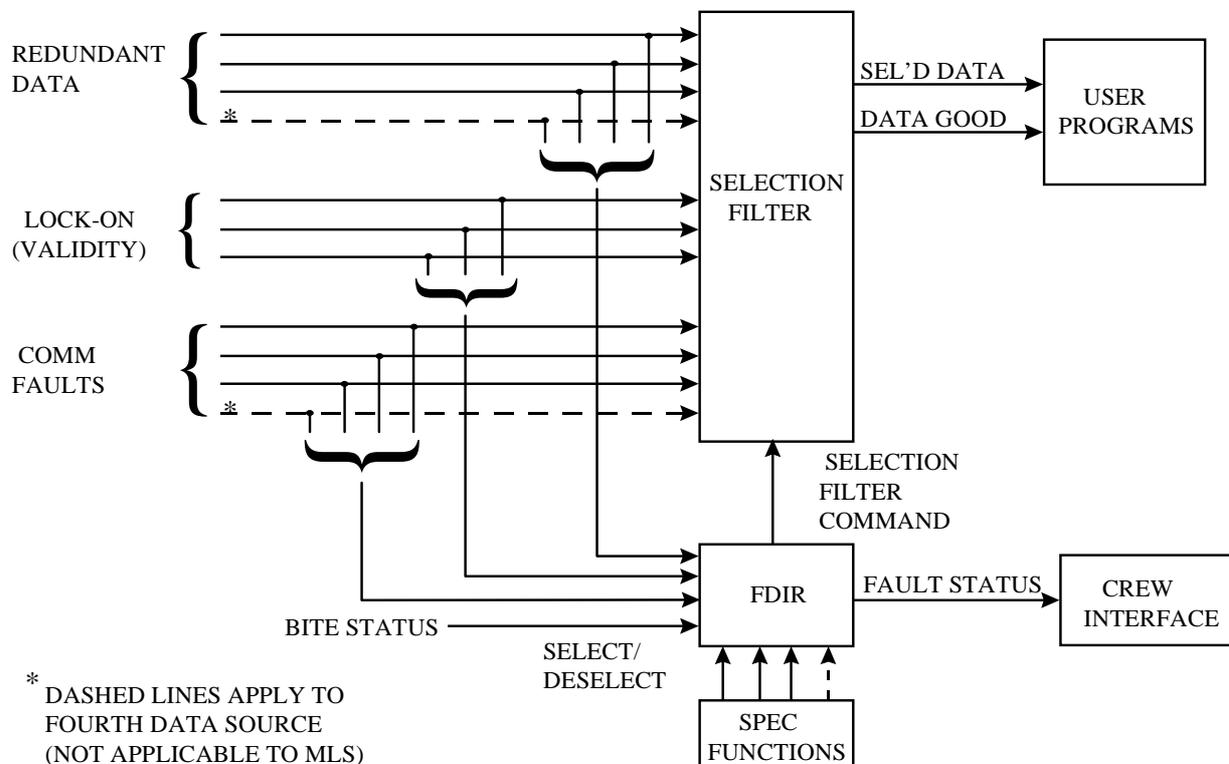


Figure 2-36.- General RM block diagram.

a. Three level

For the case of three candidate MLS units, the following parameter comparisons are made by FDIR:

$$(P_1 - P_2) > k?, (P_1 - P_3) > k?, (P_2 - P_3) > k?,$$

where k is the miscomparison tolerance for that particular parameter. A parameter is declared failed if it is the common parameter in two miscomparisons for three consecutive FDIR cycles, and that MLS unit is removed from further processing. At an FDIR rate of 0.25 Hz (4 sec/cycle), three consecutive miscompares take 12 sec. Downmoding also causes down arrows (⌵) to be displayed on crew CRT displays (SPEC 18, 40, and 101).

b. Two level

When one MLS unit has been downmoded, the following MLS parameter comparison is made by FDIR between the two remaining units:

$$(P_i - P_j) > k?.$$

If the tolerance is exceeded, the presence of BITE is checked to resolve the dilemma. If the BITE indication is present for an MLS corresponding to one of the parameters, that MLS unit is declared failed, and FDIR is downmoded to the case of one good MLS. If no BITE or more than one BITE indication is present, the disagreement is annunciated, and FDIR continues in the dilemma situation at the two level. FDIR sets the azimuth and Rng BITE status equal to the elevation BITE status since there is only one mixer crystal.

c. Zero and one levels

No FDIR.

2.6.2.2 Selection Filter

The purpose of the MLS SF is to select single values of range, azimuth, and elevation to provide to users. The common SF module, which runs at a rate of 3.125 Hz, determines the selected data to be used by navigation software. It utilizes the same COMM-lock and status bits as the FDIR module to determine the available MLS. The SF is temporarily downmoded for loss of COMM-lock (loss-of-validity or CF). The SF processes the MLS data based on the case number as described below. The SF responses to these cases are summarized in table 2-X.

a. Three level

Selected output data is midvalue selected from the three available units. In this case, the DG flag is set to a true state to notify users that good data are available to be processed. If one of the MLS units starts putting out data outside the miscompare tolerance, midvalue selection protects against its use until it is downmoded after three FDIR cycles.

b. Two level

At the two level, the selected data are the average of the two available units. The comparison test performed in FDIR is repeated here to determine whether the units disagree. If the difference between the two units exceeds the tolerance for only one count, the DG flag will be set to false to alert other users. Meanwhile, FDIR will continue to attempt to isolate the failure. The rationale for immediately setting the DG flag to false is to not risk averaging bad data. If the miscompare goes away with subsequent data, the DG flag will be set to true again.

c. One level

At the one level, data is prime selected and the DG flag is set true automatically.

d. Zero level

When there are no MLS units available, no new data will be sent to the users. The DG flag is set false.

TABLE 2-X.- SELECTION FILTER RESPONSE

| | | |
|-------------|--------------|---|
| Three level | MVS | DG = true |
| Two level | Avg | DG = false if miscompare tolerance exceeded for one count |
| One level | Prime select | DG set true |
| Zero level | No data | DG = false |

2.6.3 Navigation Software

During approach and landing, NAV uses MLS inputs to a Kalman recursive NAV filter to estimate the Orbiter's position and velocity state vector relative to the runway. Since the MLS range, azimuth, and elevation data are with respect to the MLS ground station antennas, the latitude/longitude/altitude locations, pointing angles, and range/azimuth/elevation bias corrections (if any) for each runway at which MLS is available are I-loaded into a site lookup table together with runway information. Corresponding landing areas, runways, MLS channels, and TACAN channels are contained in software and depicted in the landing site data chart in the Flight Data File (Deorbit Prep) shown in Table 2-III. The horizontal situation display (HSD) (SPEC 50), shown in figures 2-37 and 2-37a, provides the flight crew with the capability to select landing areas, primary or secondary runways for a landing area, and primary or secondary TACAN channels for a landing area. When a landing area and runway are selected, all associated data is retrieved by NAV for use during entry. NAV then processes received MLS data using geometric values for the selected runway.

Under the current FSW release (OI-27), a number of new MLS features have been implemented. When three TACAN's are manifested (with zero or one Global Positioning System (GPS) installed), processing of valid MLS data is signified by a double overbright display of "MLS" directly above the TACAN headers (see figure 2-37), in addition to the nominal blanking of TACAN and ADTA Ratios and Residuals. When three GPS units are manifested (zero TACANs), the Crew has the added ability to either Inhibit or Force MLS data into Navigation, via Item Entries (see figure 2-37a).

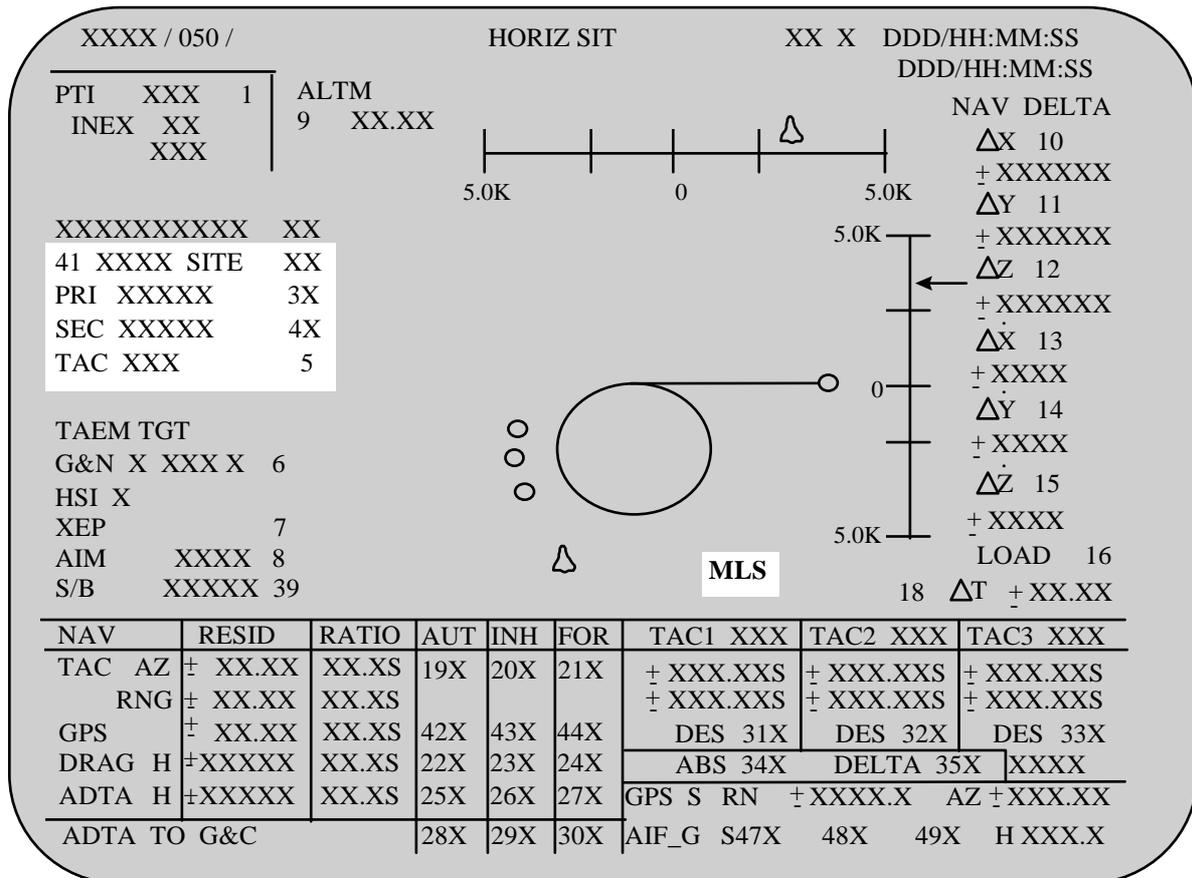
The crew must dial up the MLS channel number corresponding to the selected runway for the proper MLS data to be supplied to NAV. To illustrate the importance of this, suppose Edwards runway 17 (MLS channel 6) is selected for landing, but MLS channel 8 is dialed up. MLS data from Edwards runway 22 (MLS channel 8) will be erroneously processed by NAV and displayed on the HSI and HUD whenever the range, azimuth, and heading constraints shown in table 2-VIII are satisfied with valid range and azimuth data and heading within 55° of runway 17. MLS channel selection is not downlinked on telemetry, so ground controllers have no visibility if the wrong channel is selected; therefore, it is imperative that the crew dial up the correct MLS channel corresponding to the selected runway.

If a runway is selected which does not have MSBLS, NAV sets the MLS_AVAIL flag to off which inhibits NAV processing of any MLS data which may be received from another runway. In this case, if NAV is selected as the HSI/HUD source, no MLS data will contribute to the displays since NAV feeds area NAV which drives the HSI and HUD.

When MLS azimuth and range are acquired, NAV stops processing TACAN and baro data and starts processing MLS data (in addition to the IMU data which are always used) to update the state vector. Elevation data are incorporated into NAV if valid, but elevation data are not necessary for azimuth and

range data to be used; however, neither azimuth nor range data are incorporated into NAV without the other.

When MLS data is first available NAV executes an edit test on the azimuth, elevation, and range data. MLS data must be within the limits shown in table 2-XI when compared to the onboard state vector in order to pass the NAV edit test at data acquisition. These limits become tighter as more MLS data is processed by NAV.



Item 41 allows the selection of the landing site in OPS 1, 3, or 6 from a possible index of 1 of 25 landing areas. Each site consists of one primary (PRI) and one secondary (SEC) runway, two TACANs, and an MSBLS (where available).

Items 3 and 4 allow selection of either the primary or secondary runway of a selected landing area. The ID's of the primary and secondary runways are displayed next to items 3 and 4, respectively. Primary is automatically selected (indicated by an '*') once a landing area is specified. Items 3 and 4 are mutually exclusive.

Item 5 provides the capability to select either of two TACAN's associated with a selected landing area. When a landing area is selected, the primary TACAN is automatically selected, and subsequent executions of item 5 alternately select the secondary and primary TACAN's. The channel ID of the selected TACAN is displayed next to item 5.

Figure 2-37.- HSD (with TACANs).

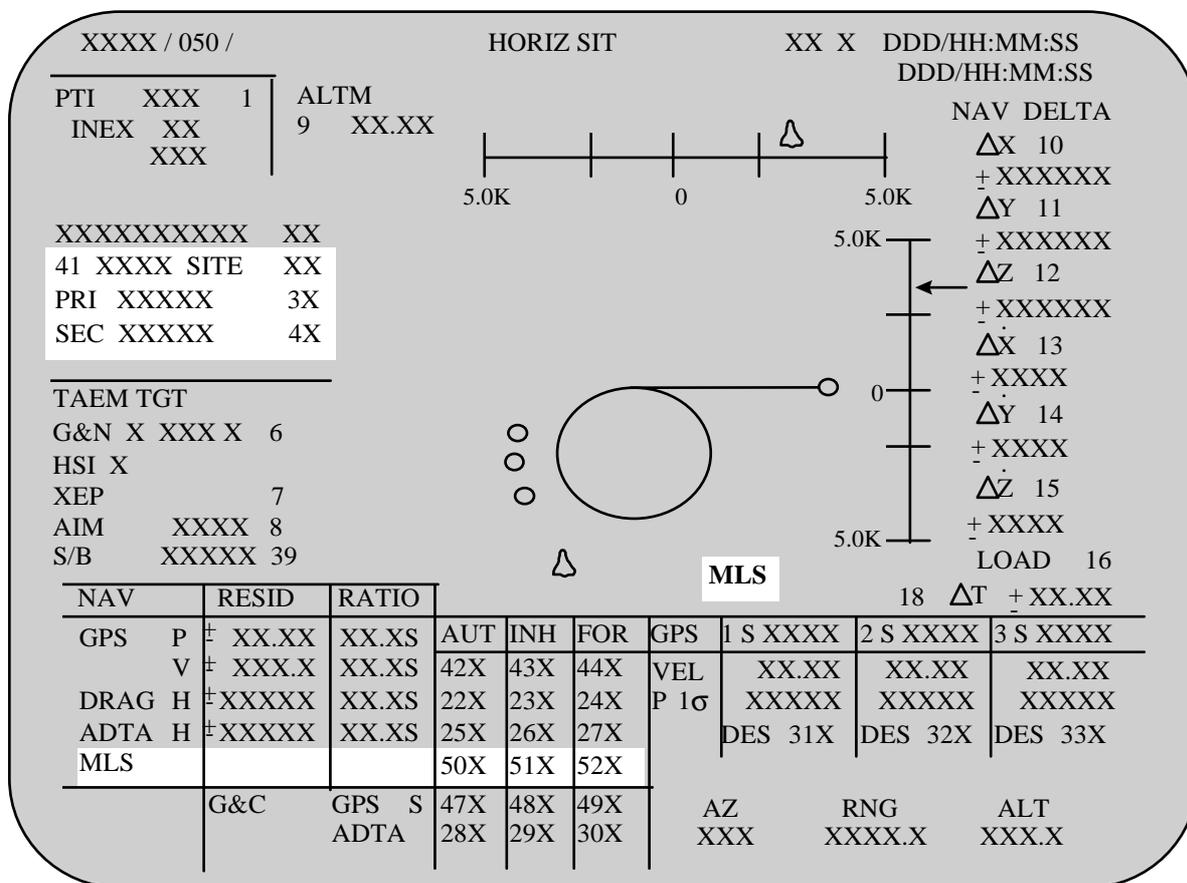


Figure 2-37a.- HSD (with three GPS, no TACANs).

TABLE 2-XI.- MLS NAV EDIT LIMITS

| Limit | Azimuth | Elevation | Range |
|-------|---------|-----------|------------|
| | 5.7° | 8.6° | 0.8 n. mi. |

If MLS azimuth or range data are lost for more than 6 sec, NAV reverts back to processing TACAN data at altitudes above 1500 ft and barometric data. However, NAV will only revert back to baro data if MLS data are lost above 1500 ft and then will only use baro data down to 500 ft. This is to prevent transients at low altitudes where the barometric calibration is adjusted for landing gear deployment. Figure 2-38a and Figure 2-38b summarize NAV requirements for the loss of MLS data, and use of MLS data, respectively.

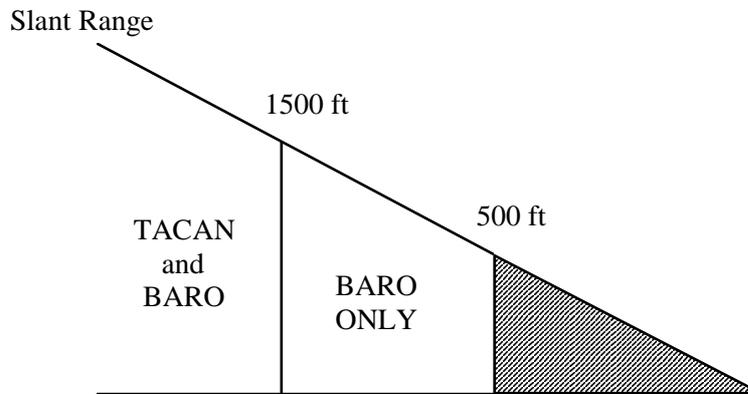
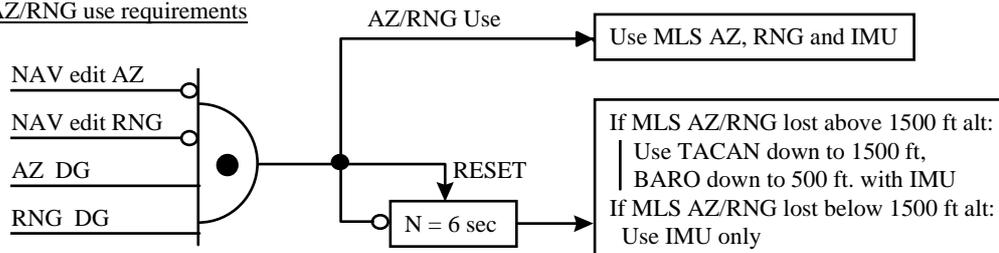


Figure 2-38a. - Loss of MLS

Near touchdown, NAV latches out all further MLS elevation data when the selected MLS elevation data drops below 1.2°. This is to prevent reincorporation of erroneous data as the elevation angle increases again. To understand this, it is necessary to recognize that the planar MLS elevation beam makes a wedge angle with the ground. Although the Orbiter does not gain altitude again, the wedge angle from the elevation station increases to 90° and beyond as the Orbiter passes the elevation station. Also, as the Orbiter gets closer to the elevation station, the azimuth angle rate from the elevation station increases to a high rate. As a result, the Orbiter flies out the top of the elevation beam at a high rate and the line of sight to the elevation station moves rapidly out the side of the Orbiter antenna coverage. Consequently, the received signal strength drops faster than the MLS NAVSET AGC can track and only a few pulse pair intervals are averaged over a beam passage, causing increased errors. These errors are coupled with the already high multipath errors caused by the elevation beam striking the ground. If NAV were to use that erroneous data, the altitude computation could cause an improper pitch maneuver near touchdown.

-AZ/RNG use requirements



-EL use requirements

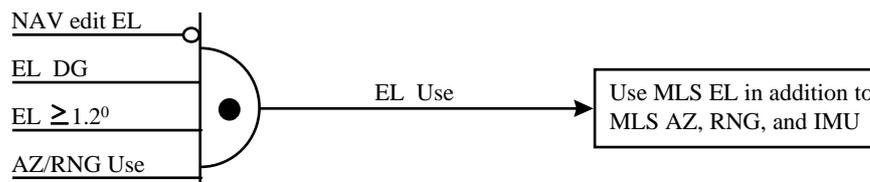


Figure 2-38b.- MLS NAV use requirements.

2.6.4 BITE Software

As mentioned previously, the capability exists to perform a check on the hardware system via ground control or on orbit. On orbit, this process is called a self-test and can only be performed in OPS 8. The software limits imposed on the self-test are listed in table 2-XII.

TABLE 2-XII.- SOFTWARE SELF-TEST LIMITS

| Information channel | Limit |
|---------------------|------------|
| +Azimuth (deg) | +3.0 ± 0.1 |
| -Azimuth (deg) | -3.0 ± 0.1 |
| Range (n. mi.) | 15.2 ± 0.2 |
| Elevation (deg) | 6.0 ± 0.1 |

2.6.5 CRT Displays

MLS information is displayed on three CRT displays:

- a. GNC system summary 1 (GNC SYS SUMM 1) (OPS 3 and 6)
- b. SENSOR TEST (OPS 8)
- c. SENSOR SELF-TEST (OPS 9)

2.6.5.1 GNC SYS SUMM 1

This display (fig. 2-39) shows the operational status of the GNC systems and supports the GNC C&W system. The display is available in OPS 3 and OPS 6 by using the system summary (SYS SUMM) key or by SPEC 18 PRO. The lower right corner of the display shows the status of the NAV systems including the MLS. The status columns for each unit may show blank (system OK), - (unit failed), M (data missing due to CF or loss of lock), or ? (dilemma). The example below shows MLS 1 and 2 in a dilemma situation and MLS 3 failed. GNC SYS SUMM 1 is a display only (DISP 18) with no interactive command capability.

EXAMPLE:

```

NAV   1   2   3   4
-----
IMU
ACC
RGA
TAC
MLS   ?   ?   ↓
ADTA
    
```

2.6.5.2 SENSOR TEST

This display (fig. 2-40), which is available in OPS 8 and via SPEC 40 PRO, provides a means of initiating, monitoring, and terminating the self-test of the MLS and three other sensors. It also provides control of the SF. Execution of a deselect (DES) item number changes the selection as a candidate for the SF. An * appears next to the item number if the corresponding unit is deselected as a candidate. The SF configuration from this display is carried over into OPS 3. Status (STAT) column indications are defined below. If the BITE indicator is set, it will be blanked if the condition is corrected. The MLS STAT runs at 2 Hz. RNG in n. mi., azimuth in deg, and elevation in deg are displayed for each MLS. These values are compared to the self-test limits (table 2-XII) and a status indicator is displayed beside each test parameter indicating the results of the tolerance check. The self-test controls are shown at the bottom of the display. Execution of item 13 starts the self-tests on all the displayed sensors. Execution of item 14 stops the self-tests. An * appears next to item 13 or item 14 to indicate, respectively, tests underway or tests stopped. Items 13 and 14 are initialized blank.

EXAMPLE:

| | DES | STAT | RNG | AZ | EL |
|-------|-----|--------|---------|----------|---------|
| MLS 1 | 1 | | 15 . 0 | -2 . 96 | 5 . 92 |
| | 2 | | 15 . 2 | -3 . 01 | 6 . 05 |
| | 3 | 3 BITE | 15 . 4↑ | -3 . 01↑ | 6 . 05↑ |

STATUS:

OFF power not applied to NAVSET

BITE AZ or EL AGC flag (only in self-test)
or
AZ or EL mixer crystal current flag
or
AZ, EL, or RNG validity flag

BLANK valid data

PARAMETER INDICATOR:

| | |
|-------|---------------------|
| 'M' | data missing |
| '↑' | test value too high |
| '↓' | test value too low |
| blank | normal |

2.6.5.3 Sensor Self-Test

This specialist function (SPEC) display (fig. 2-41) is available in OPS 9 via SPEC 101 PRO for preflight and postlanding checkout of the MLS and other sensors. It is the ground test equivalent of the on-orbit SENSOR TEST (SPEC 40).

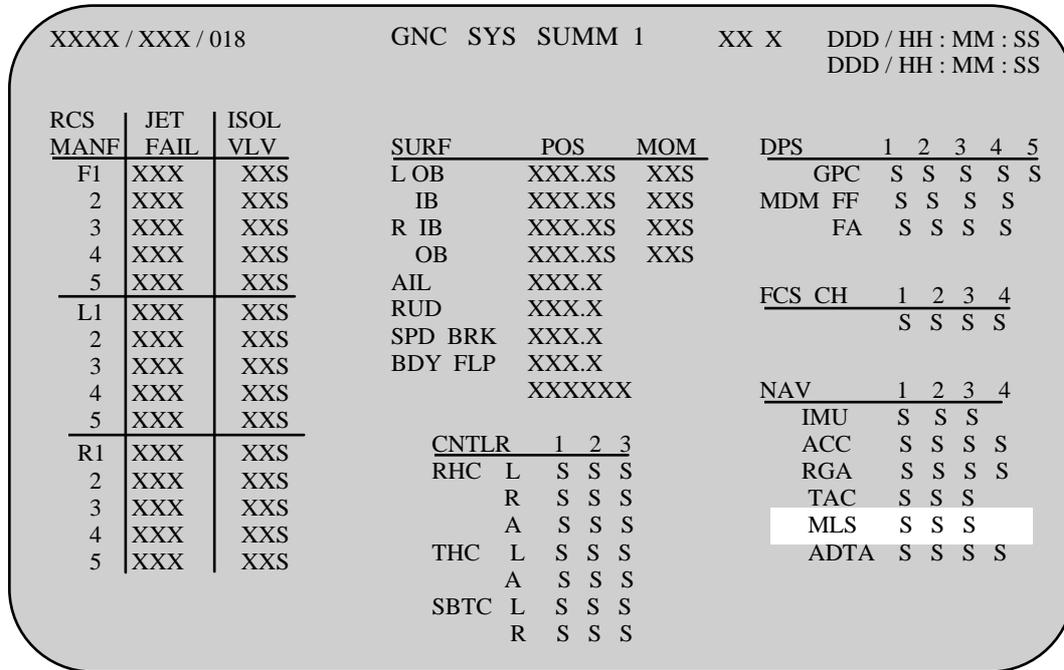


Figure 2-39.- GNC SYSTEM SUMMARY 1 CRT display.

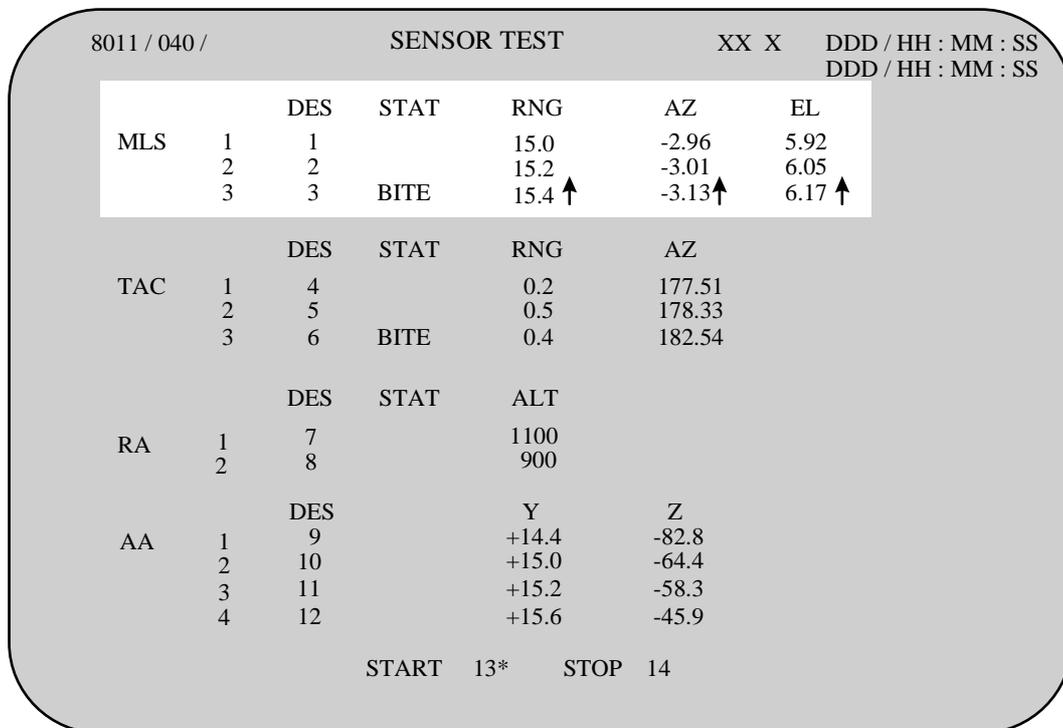


Figure 2-40.- SENSOR TEST CRT display.

MLS GOODIE SHEET

Ground Station (MSBLS-GS)

- a. Dual redundant EL and AZ/RNG ground stations
- b. Dual redundant ac power/15 min battery backup
- c. Continuous internal performance monitoring
- d. Triply redundant external EL and AZ/RNG field monitors
- e. Manual monitoring and control capability
- f. Allowed EL and AZ alignment error: ± 0.015 deg
- g. Alarm limits --

| | <u>AZIMUTH</u> | <u>ELEVATION</u> | <u>RANGE</u> |
|-----------------------------|----------------|------------------|--------------|
| - Field monitor | 0.125^0 | 0.125^0 | 0.041 n. mi. |
| - Internal (coding) monitor | 0.25^0 | 0.25^0 | NA |

- h. Automatic alarm actions –
 - (1) Alarm on primary → switchover to backup
 - (2) Subsequent alarm on backup → inhibit all transmissions
- i. Ground station failures reported to LSO (FDO)

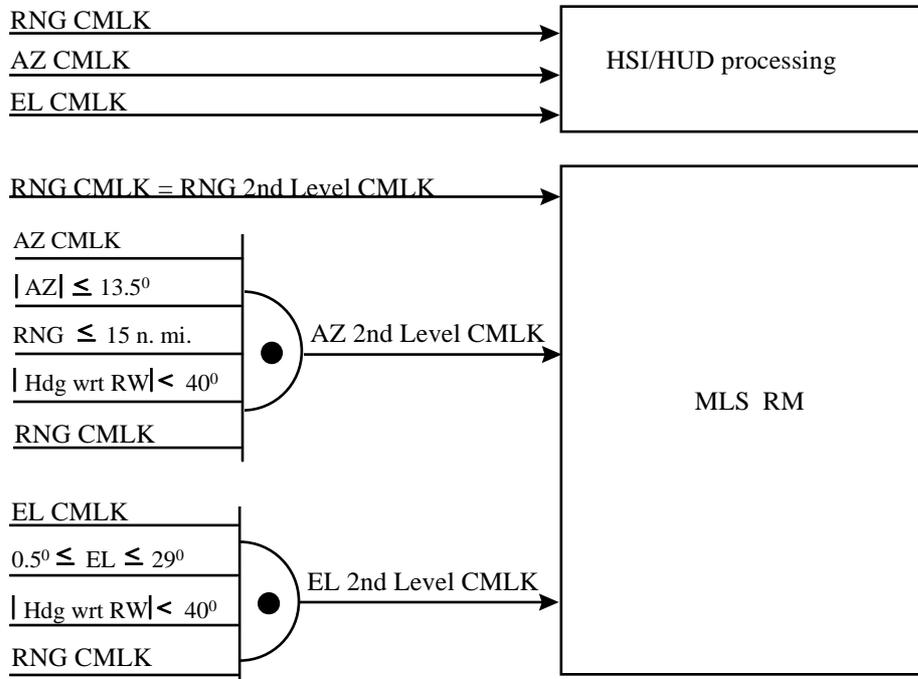
Orbiter Hardware

| | <u>MLS 1</u> | <u>MLS 2</u> | <u>MLS 3</u> |
|---------------------------------|--|-----------------|-----------------|
| a. Location | Avionics Bay 1 | Avionics Bay 2 | Avionics Bay 3 |
| b. MDM | FF 1 | FF 2 | FF 3 |
| c. Power | MNADA1 O14 | MNBDA2 O15 | MNCDA3 O16 |
| d. Power ON discrete | ESS 1BC | ESS 2CA | ESS 3AB |
| e. Switch | PNL O8 sw 8 | PNL O8 sw 9 | PNL O8 sw 10 |
| f. Circuit Breaker | PNL O14:E cb 25 | PNL O15:E cb 24 | PNL O16:E cb 18 |
| g. BITE | - Continuously monitor mixer crystal current | | |
| h. Thermal | - Convective air cooling | | |
| | - Min operating temp: -65 deg F | | |
| | - Max operating temp: 165 deg F | | |
| i. Power requirement | -- 100 watts/MLS | | |
| j. Warmup time | -- 3 min | | |
| k. Nominal acquisition altitude | -- 15,000 ft - 20,000 ft | | |

| | <u>AZIMUTH</u> | <u>ELEVATION</u> | <u>RANGE</u> |
|----------------------------|----------------|------------------|--------------|
| l. Coverage limits | $\pm 13.5^0$ | $1.3^0 - 29^0$ | 20 n. mi. |
| m. Accuracy (1σ) | 0.05^0 | 0.05^0 | 0.016 n. mi. |
| n. LSB | 0.004^0 | 0.004^0 | 0.003 n. mi. |

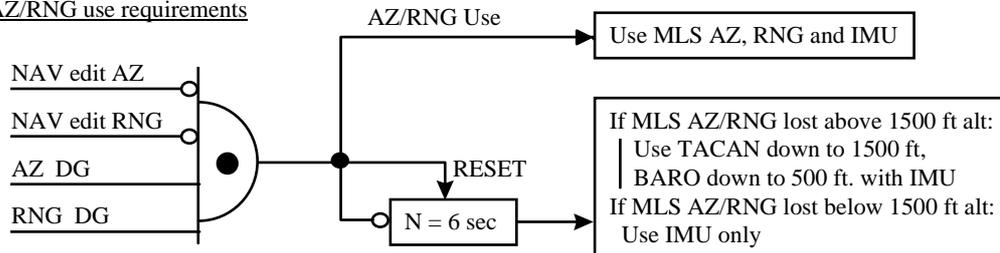
SOFTWARE (PASS Only)

- a. SOP (MM 305, 603)
 - CMLK flag generation for each MLS LRU

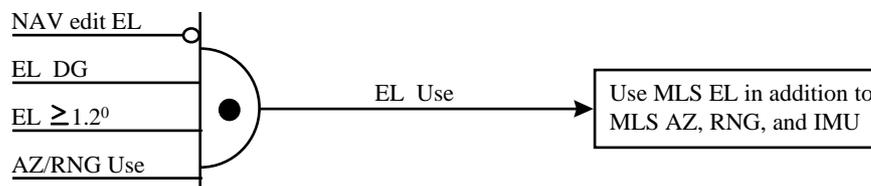


- b. FDIR (MM305, 603)
- | | <u>AZIMUTH</u> | <u>ELEVATION</u> | <u>RANGE</u> |
|---|------------------|------------------|--------------|
| - Miscmpare limits | 0.5 ⁰ | 0.4 ⁰ | 0.329 n. mi. |
| - Fail after 12 sec miscmpare (and one BITE at 2 level) | | | |
- c. Selection Filter (MM 305, 603)
- Loss of 2nd level CMLK downmodes parameter
 - Deselect (OPS 8 only)/fail downmodes LRU
- | 3 MLS's | MVS | DG = true |
|---------|---------|-------------------------|
| 2 | avg | DG = false if miscmpare |
| 1 | use | DG = true___ |
| 0 | no data | DG = false |
- d. NAV (MM 305, 603)
- | | <u>AZIMUTH</u> | <u>ELEVATION</u> | <u>RANGE</u> |
|---|------------------|------------------|--------------|
| - NAV Edit limits (at MLS acquisition) | 5.7 ⁰ | 8.6 ⁰ | 0.8 n. mi. |
| - NAV uses MLS data to supplement IMU data. | | | |

-AZ/RNG use requirements



-EL use requirements



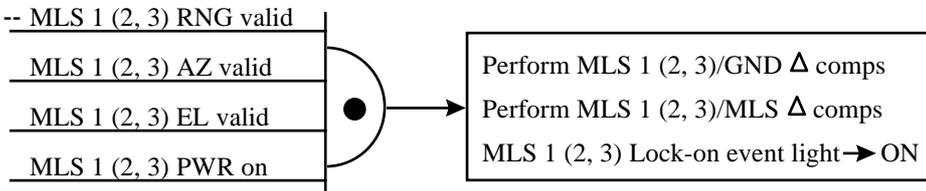
FCS Checkout

- a. Test limits

| | <u>AZIMUTH</u> | <u>ELEVATION</u> | <u>RANGE</u> |
|---------|--------------------|------------------|--------------|
| Low | ± 2.9 ⁰ | 5.9 ⁰ | 15.0 n. mi. |
| Nominal | ± 3.0 ⁰ | 6.0 ⁰ | 15.2 n. mi. |
| High | ± 3.1 ⁰ | 6.1 ⁰ | 15.4 n. mi. |
- b. Sensor test history tabs, hardcopies
 - before START (AA bias data)
 - START +3 sec (MLS + AZ data)
 - START +19 sec (MLS - AZ, EL, RNG data)
 - after STOP (AA bias data)

DISPLAYS

- a. Crew displays – HSI (determined by HSI source switch), HUD
- b. Ground comps -- MLS 1 (2, 3) RNG valid



Flight Rules/LCC

- a. Flight rules -- One MLS is required for use of autoland in emergencies (#8-21B).
- b. LCC -- There is no LCC for MLS since it is powered off prelaunch.

SECTION 3

RADAR ALTIMETER

3.1 OVERVIEW

The radar altimeter (RA) is a low-altitude (0 to 5000 ft) tracking radar system (Figure 3-1). The RA is used on the Orbiter in the final minute of landing to sense altitude above the terrain. This altitude information is read out by the pilot (PLT) to the commander (CDR) during landings and provides precise altitude data which verifies the CDR's estimated altitude based on visual cues. The RA altitude data is also checked against navigation (NAV) altitude, which is calculated in the flight software from microwave landing system (MLS) data.

The RA altitude data is dependent on the time delay between the transmitted and received pulses. The transmitted pulse, which is bounced off the ground and received, is converted from an analog ramp voltage to a digital altitude by the flight software and displayed on the AVVI's and HUD's. This RA altitude data is also transmitted to the Mission Control Center (MCC) where it is monitored by guidance, navigation and control (GNC). GNC compares RA altitude to NAV altitude data on RTPLOT.

There are two RA's on each of the Orbiters for redundancy. The units are independent of each other, there are different transmitting and receiving antennas for each RA, and the PLT and CDR can select either RA for display. The RA is the primary source of low-altitude data, with the MLS providing redundancy.

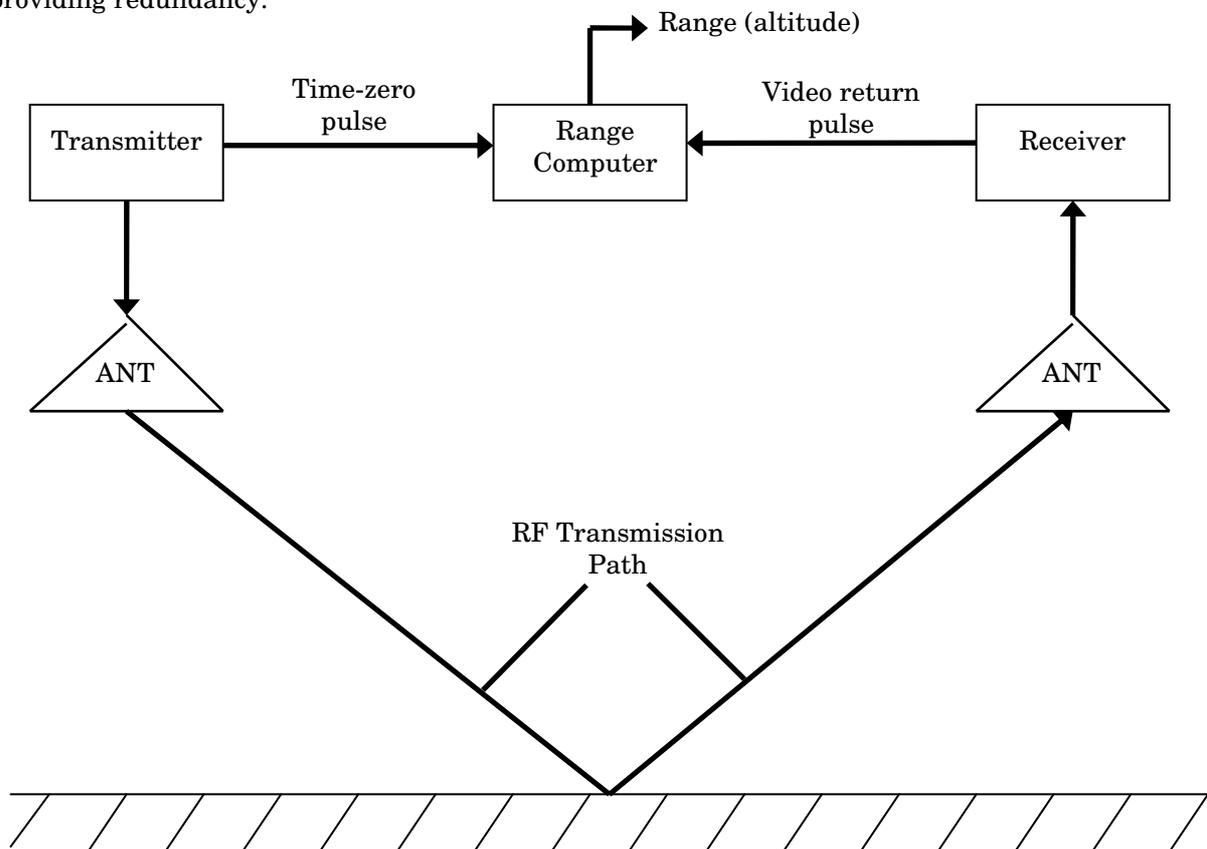


Figure 3-1. - Radar altimeter system.

3.2 RA HARDWARE

The following RA hardware components will be discussed in this section: power supply, transmitter, receiver, range computer, and built-in test equipment (BITE). Reference the RA hardware drawing (drawing 9.3) in the Space Shuttle Systems Handbook (SSSH). Figure 3-2 provides an overview of the RA hardware interfaces.

3.2.1 Power Supply

The RA power supply operates off the Orbiter 28 V main (MN) dc buses. RA 1 is fed by MN A; RA 2 is fed by MN B. The power supply consists of a switching regulator followed by an inverter operating at a frequency of 15 kHz. The switching regulator and inverter are used in conjunction with the transformer and filter components to generate the other voltages used by other circuits within the RA. The switching regulator output is controlled by a sensing circuit to maintain a 3 percent voltage regulation on the power supply.

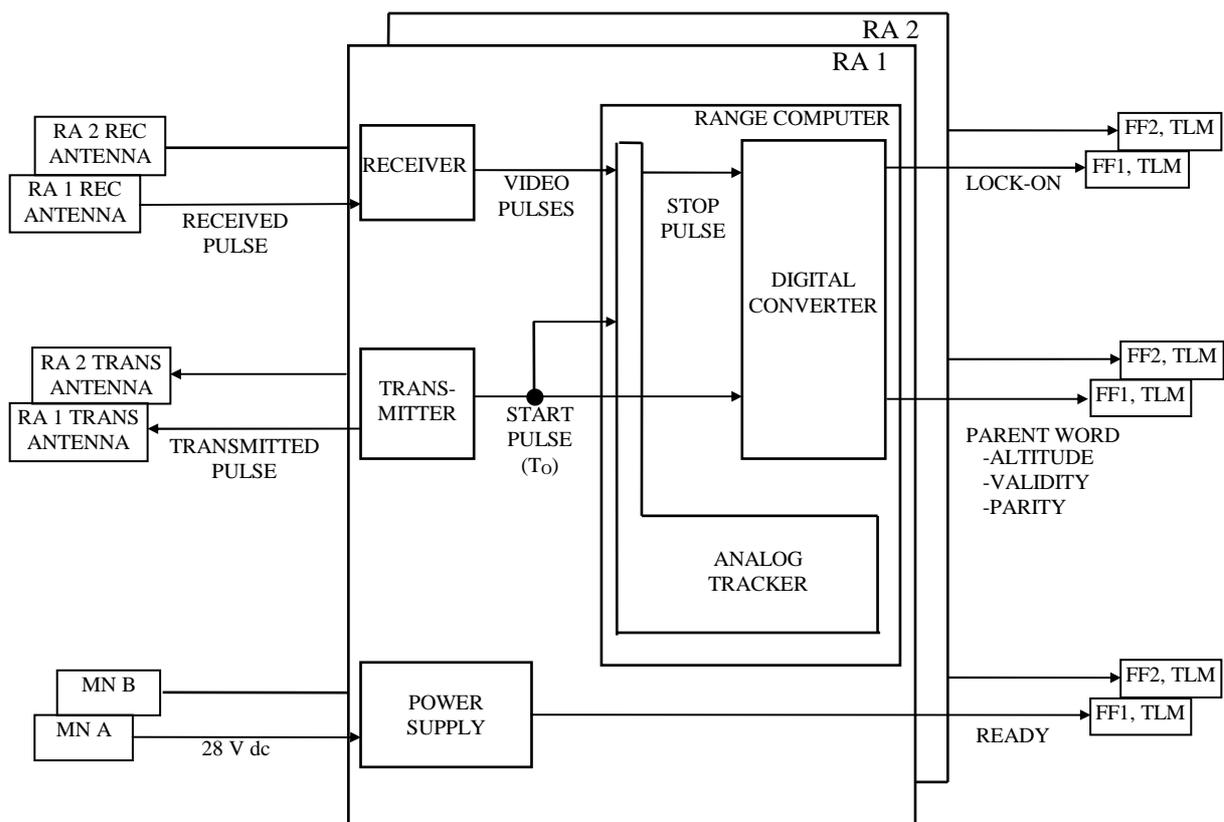


Figure 3-2. - RA hardware interface.

3.2.2 Transmitter and Antenna

The transmitter consists of the time delay circuit, pulse generator, pulse modulator, and cavity oscillator. The transmitter generates the radio frequency (RF) pulses which are radiated by the transmitting antenna.

The transmitter time delay circuit delays the +15 dc volts from being applied to the pulse generator. The time delay provides a warm-up of the transmitter tube filaments before high voltage (320 V dc) is applied to the transmitter cavity oscillator.

The output of the pulse generator is a classified signal which, after being amplified, drives the modulator pulse-forming network. The pulse-forming network, through a step-up pulse transformer, drives the cavity oscillator with high-energy pulses. The cavity oscillator runs at a classified carrier frequency. This output is sent to the antenna for transmission to the ground. Each transmitted radio frequency (RF) pulse generates a time reference pulse (t_0) which is detected by the range computer for computing altitude when the return pulse is sensed.

3.2.3 Receiver and Antenna

The receiver receives the returning pulses from the receiving antenna. The local oscillator in the receiver is a solid-state, tuned resonant cavity oscillator which supplies a classified signal to the balance mixer. The mixer contains a pair of matched crystal diode detectors and a bandpass filter operating in the mixer input providing a classified signal input bandpass.

The local oscillator signal heterodynes with the received pulses in the strip-line circuit are employed in the mixer. The crystal diodes in the mixer demodulate the resulting beat frequency, producing an output consisting of overlapping video pulses, with amplitude and polarity depending on the phase relationship between received echo pulses and the local oscillator signal.

The full wave detector converts the bipolar video pulses to unipolar video pulses for use in the range computer. The bandwidth of the amplifier is classified, with the gain controlled as a function of range by an input voltage (KAGC).

The intermediate frequency (IF) amplifier operates as a high gain video amplifier. It receives bipolar video output signals from the mixer and amplifies these signals through a five stage amplifier. There are two gain controls in the IF amplifier: an automatic gain control (AGC) and a sensitivity range control (SRC). These two gain controls are summed together in the receiver to control the gains of the IF amplifier stages.

The AGC controls the mean amplitude of the video pulse as it is detected in the tracker. The SRC controls the IF amplifier gain as a function of altitude (no SRC control above 140 ft, SRC control increases linearly between 140 and 28 ft, and maximum SRC control is below 28 ft). This results in a relatively constant output. The SRC control is set to permit tracking of the ground return signals at low altitudes with AGC controlling IF gain. The AGC is active from 0 to maximum range (5000 ft).

3.2.4 Range Computer

The range computer provides the tracking loop which maintains a leading edge track of the ground return pulses and produces the altitude signal. The range computer is made up of two major components, the analog tracker and the digital converter.

a. Analog tracker

As the pulse signal is transmitted to the ground, the time reference pulse (t_0) is fed back to the ramp generator which provides a saw-tooth ramp voltage to the comparator as a function of time. The comparator compares this ramp voltage with the internal range voltage from the range integrator. When the voltages are equal, the comparator sends an output signal which triggers the gate pulse generator that generates two types of pulses, a narrow track gate pulse and a wide KAGC gate pulse.

The track gate pulse is applied to a gate circuit where the amount of overlap between this pulse and the video pulse is converted to a track error current to determine the direction and rate of change of the internal range voltage. The range integrator circuit integrates the received voltages proportional to rate from the rate integrator and develops voltages proportional to range.

The KAGC gate pulse is supplied to the AGC loop where it is used to determine the IF amplifier gain necessary to maintain constant amplitude on the video target pulse that is used to overlap the track gate. The basic AGC loop consists of the track keyed AGC amplifier, the IF amplifier, and the video amplifier. An additional gain control circuit (the SRC in the receiver), controls the system gain as a function of altitude to prevent the system from locking up on antenna leakage, returns from Orbiter appendages (such as landing gear), or dense rain. The noise automatic gain control (NAGC) circuit operates as a function of noise to maintain the noise level below the tracking threshold. The AGC amplifier sums gain control inputs (SRC, NAGC, and KAGC) and produces a voltage to control the IF amplifier gain in the receiver.

The KAGC signal serves as an input source to the track/no track logic circuit which indicates the presence or absence of a target pulse suitable for tracking. Once the system is in track mode, the tracking signal would have to be lost by more than 0.1 seconds before the search mode is activated. Figure 3-3 shows the timing relationships among the transmitted, returned, and various gating pulses.

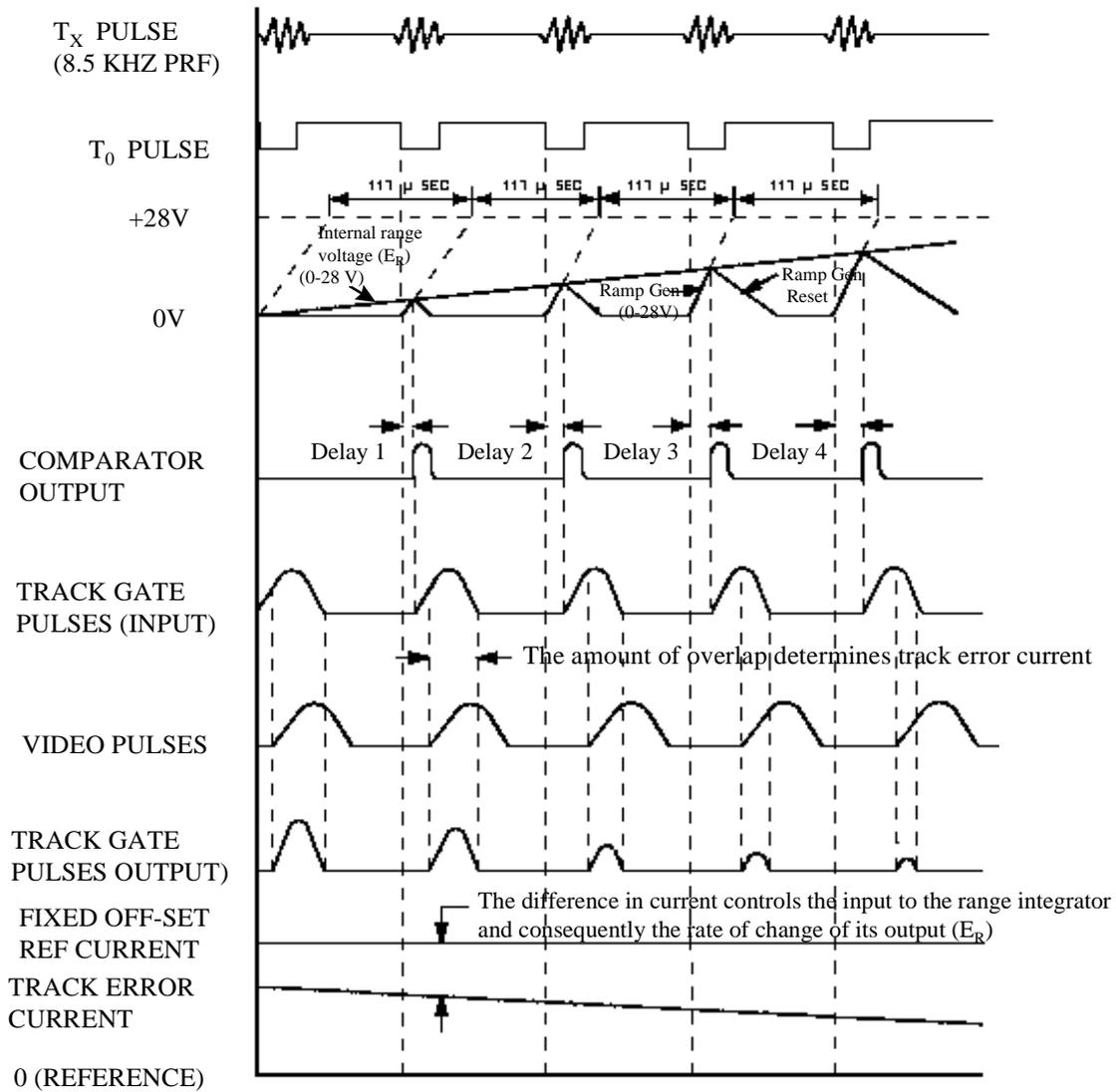


Figure 3-3. - RA analog tracker timing diagram.

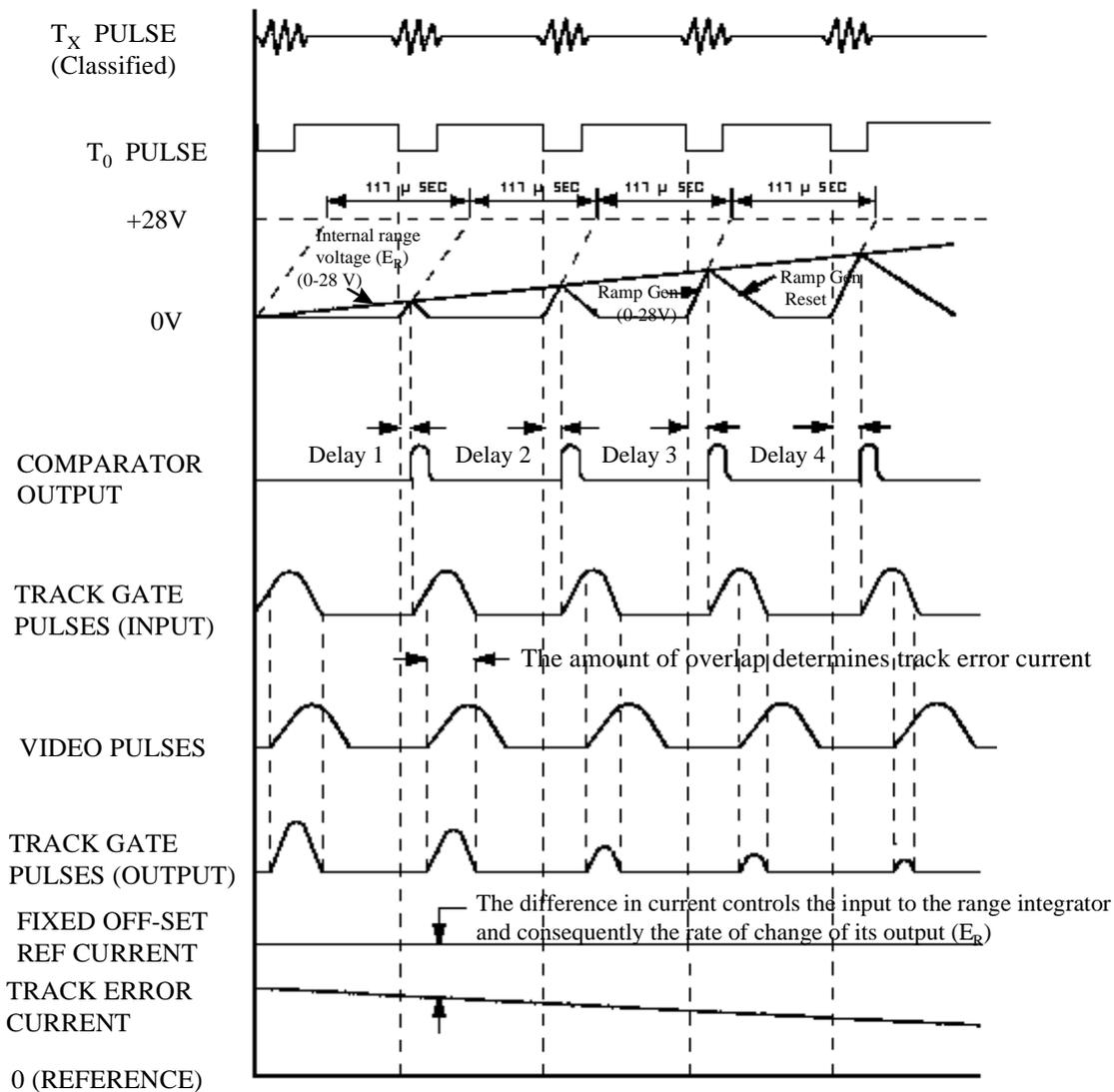


Figure 3-3. - RA analog tracker timing diagram.

b. Digital converter

The digital converter converts the time between the time reference pulse (t_0) and the track stop gate pulse to a digital serial word in which each pulse represents 1 foot altitude. The time difference between the start and stop pulses is proportional to altitude.

The clock utilized for this circuit runs at a crystal-controlled frequency of 24.58920 MHz with a scale of 20 radar feet per cycle. Each time the t_0 pulse occurs, the start-stop multi-vibrator allows the clock pulses to pass into a high speed counter. For every five pulses from the clock, a single pulse is sent to the accumulator. The stop pulses sent from the analog tracker reset the output of the multi-vibrator and stop the pulse count. At this time, the divide-by-100 counter

advances by one count. When this counter reaches 100, the output of the divide-by-5 counter is rounded off by adding one more pulse to the accumulator if the divide-by-5 counter is greater than 3. After the roundoff cycle is complete, the information in the accumulator is parallel loaded into the shift register together with the validity bit and the odd parity bit.

3.2.5 Built-In Test Equipment (BITE)

An externally initiated self-test can be performed on the RA on the Orbiters in operational sequence (OPS) 8 FCS checkout, OPS 9 prelaunch, or during ground testing. Further details of the RA self-test are discussed in another subsection of this systems brief.

3.2.6 Hardware Characteristics

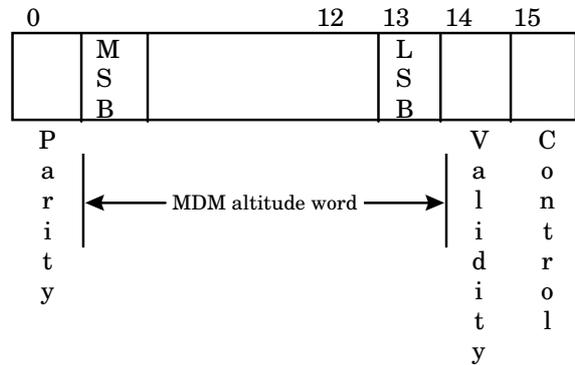
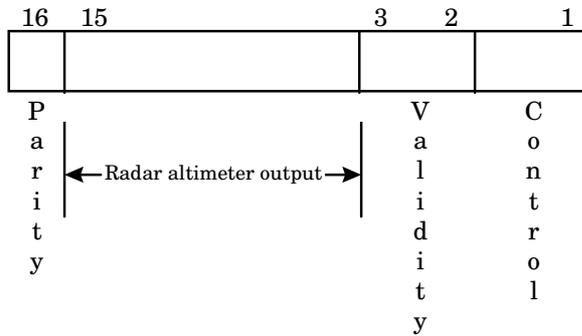
The following RA operational characteristics from the Shuttle Operational Data Book (SODB) are given here for reference:

| | |
|------------------------------------|--------------|
| Transmitter frequency | Classified |
| Transmitter pulse width | Classified |
| Pulse repetition frequency | Classified |
| Transmitter peak power, wide pulse | Classified |
| Receiver bandwidth | Classified |
| Altitude range | 0 to 5000 ft |
| Altitude rate limit | 2500 ft/sec |

The RA will provide the following outputs: analog altitude signal, digital altitude signal, and lock-on signal.

The analog altitude signal is an analog voltage from 0 to 15 V. The digital altitude signal is a 16-bit word with the least significant bit (LSB) equal to 1 foot altitude and the most significant bit (MSB) equal to 4096 feet altitude. Bit 16 is the parity bit for odd parity. The lock-on signal is logic 1 during track and logic 0 during search.

Each RA sends a 16-bit word to the GPC via the flight forward (FF) multiplexers/demultiplexers (MDM's) (RA 1 via FF1 and RA 2 via FF2). The 16-bit word consists of a data validity (lock-on), altitude data, and an odd parity bit. Figure 3-4 shows the 16-bit word from the RA to the MDM and from the MDM to the GPC. Note that the LSB and the MSB on the two 16-bit words are reversed.



| Bit | Designation | Logical value (true) |
|-----|-------------|----------------------|
| 1 | Control | 1 (always) |
| 2 | Validity | 1 = Lock-on |
| 3 | Data LSB | 1 = 1 ft |
| 4 | Data | 1 = 2 ft |
| 5 | Data | 1 = 4 ft |
| 6 | Data | 1 = 8 ft |
| 7 | Data | 1 = 16 ft |
| 8 | Data | 1 = 32 ft |
| 9 | Data | 1 = 64 ft |
| 10 | Data | 1 = 128 ft |
| 11 | Data | 1 = 256 ft |
| 12 | Data | 1 = 512 ft |
| 13 | Data | 1 = 1024 ft |
| 14 | Data | 1 = 2048 ft |
| 15 | Data MSB | 1 = 4096 ft |
| 16 | Parity | 1 = ODD parity |

RA to MDM

| Bit | Designation | Logical value (true) |
|-----|-------------|----------------------|
| 0 | Parity | 1 = ODD parity |
| 1 | Data MSB | 1 = 4096 ft |
| 2 | Data | 1 = 2048 ft |
| 3 | Data | 1 = 1024 ft |
| 4 | Data | 1 = 512 ft |
| 5 | Data | 1 = 256 ft |
| 6 | Data | 1 = 128 ft |
| 7 | Data | 1 = 64 ft |
| 8 | Data | 1 = 32 ft |
| 9 | Data | 1 = 16 ft |
| 10 | Data | 1 = 8 ft |
| 11 | Data | 1 = 4 ft |
| 12 | Data | 1 = 2 ft |
| 13 | Data LSB | 1 = 1 ft |
| 14 | Validity | 1 = Lock-on |
| 15 | Control | 1 (always) |

MDM to GPC

Figure 3-4. - RA altitude data word.

3.3 ORBITER RA-RELATED HARDWARE

3.3.1 Power

Each RA requires 28 V dc from the Orbiter main power buses. MN A powers RA1 through a circuit breaker in panel O14. MN B powers RA 2 through a circuit breaker in panel O15. Power ON/OFF switches for both RA's are located on panel O8. Figure 3-5 shows the location of the circuit breakers and power switches on the Orbiters.

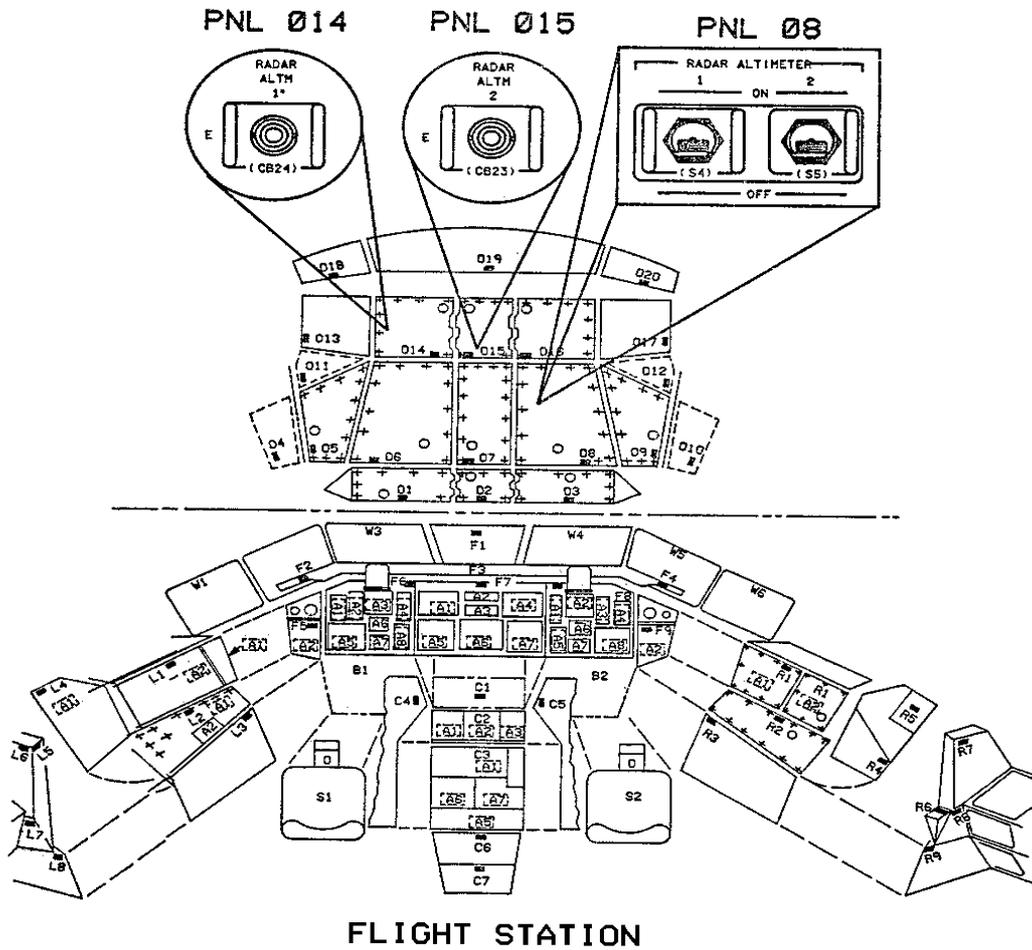


Figure 3-5. - RA power control.

3.3.2 Switch Select

The crew has the capability to manually select which RA will be used to display data on the altitude/vertical velocity indicator (AVVI) and head up display (HUD). The select switches are located on panel F6 for the CDR and on F8 for the PLT on vehicles with Multifunction Electronic Display Systems (MEDS). The select switches are located on panel F7 and F8 for the CDR and PLT, respectively, on pre-MEDS vehicles. Figure 3-6 shows the location of these switches on the Orbiters.

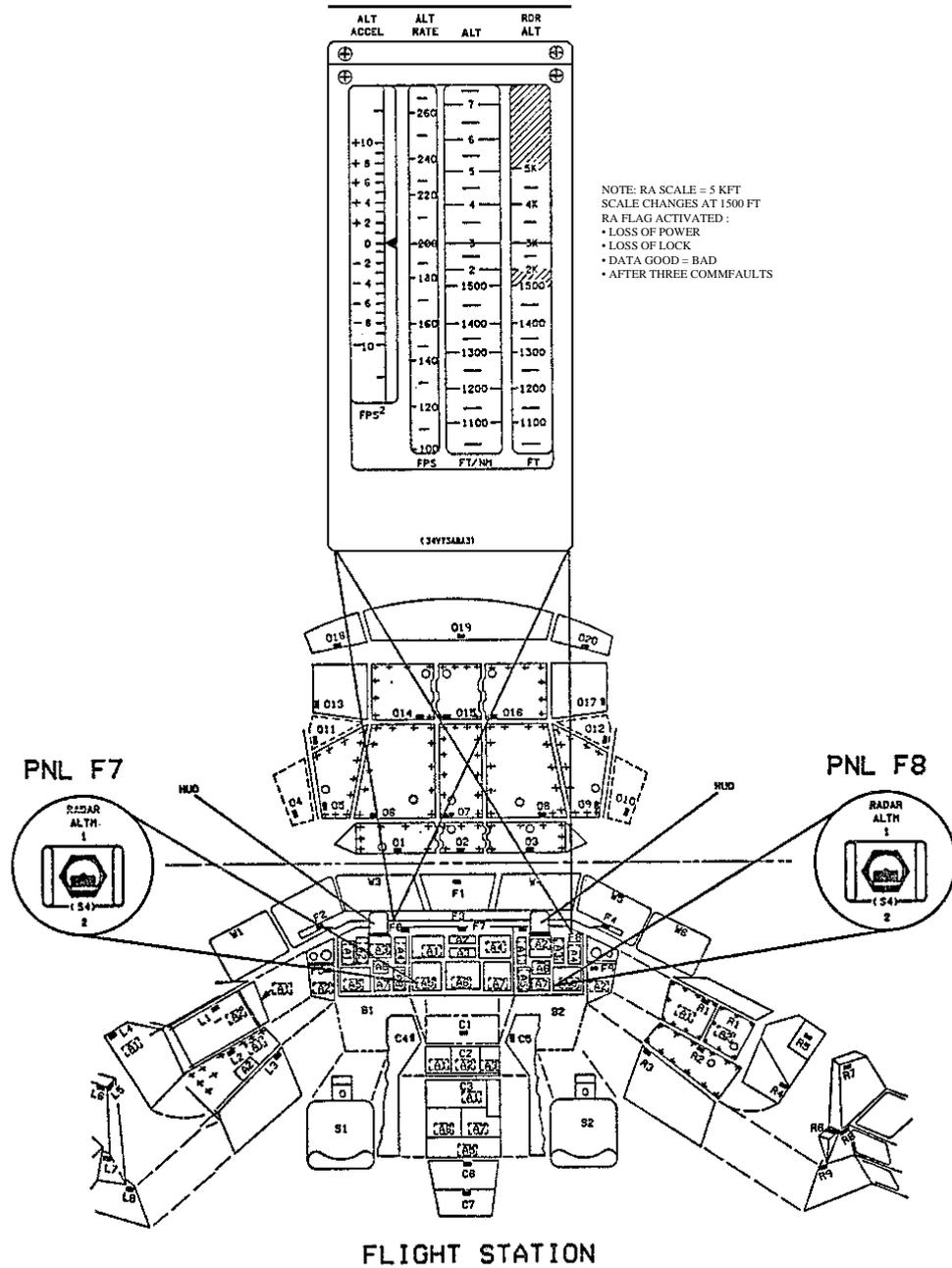


Figure 3-6.- RA select switches and displays.

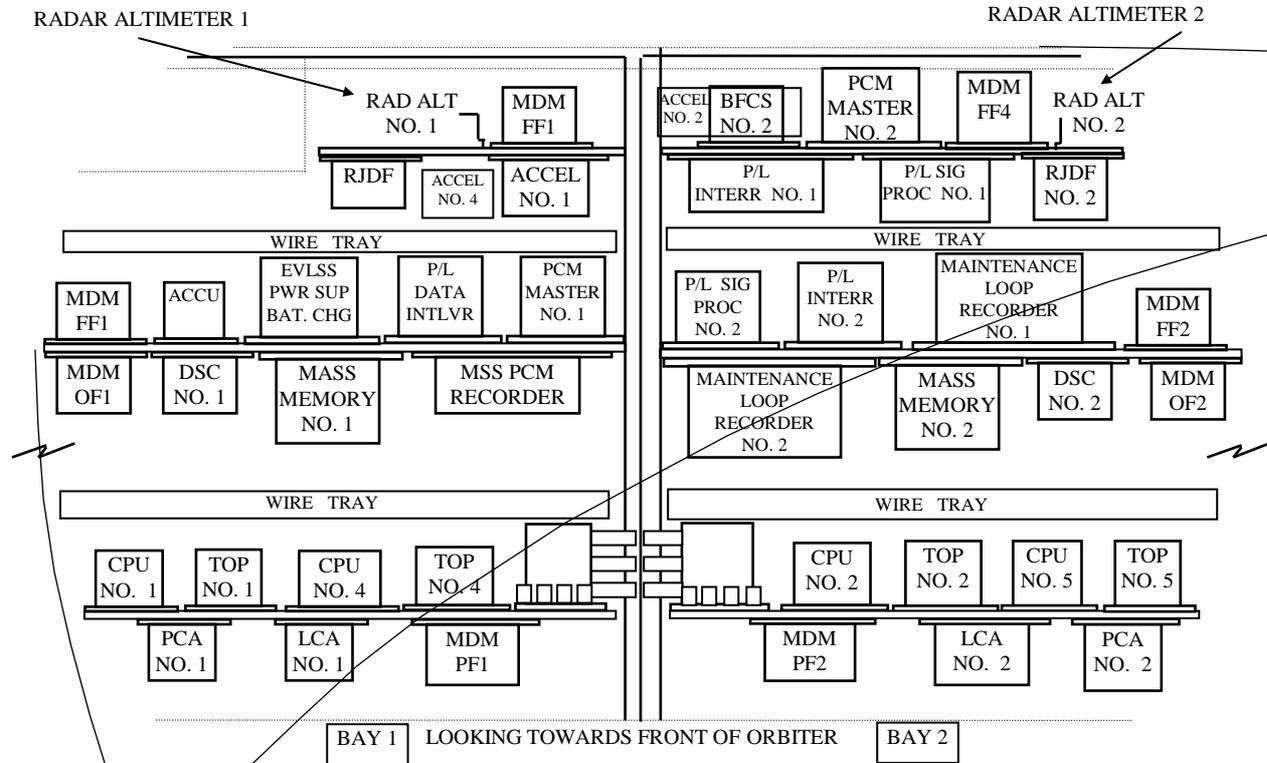


Figure 3-7. - Avionics bay installation of radar altimeters.

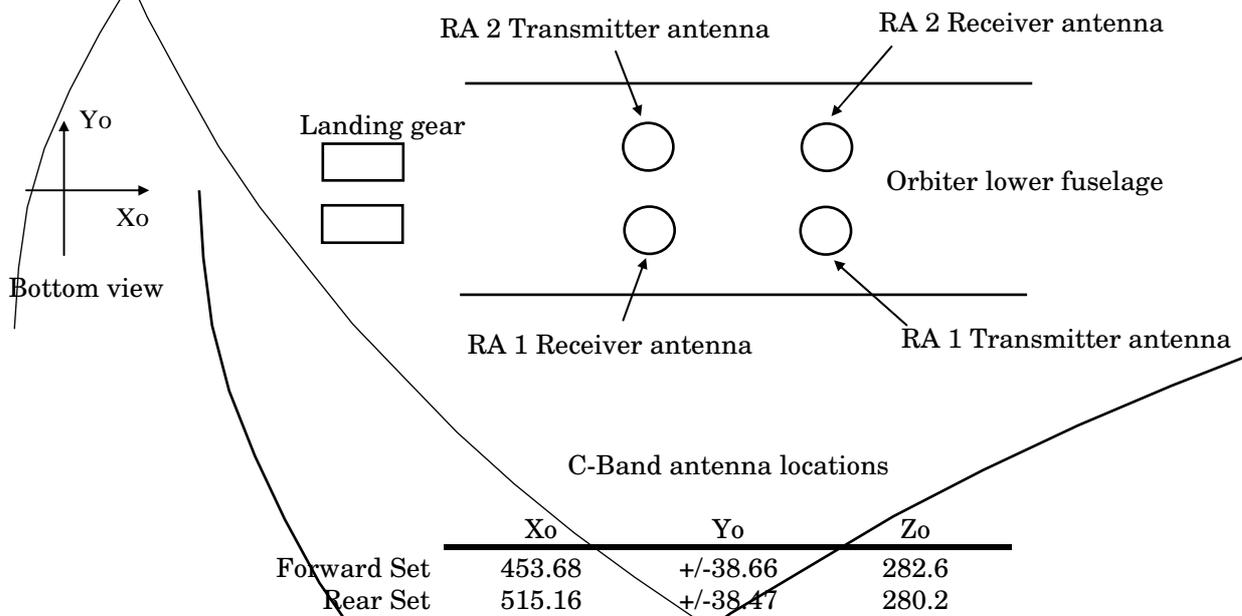


Figure 3-8. - Radar altimeter antenna locations.

3.3.4 Dedicated Displays and HUD

a. Dedicated displays

The RA altitude is displayed on the two AVVI's for use by the CDR and PLT during the final 5000 ft until touchdown. The RDR ALT tape on the AVVI is located next to the NAV ALT tape. These two sources of altitude can be compared with each other and with visual cues to provide precise altitude information during Orbiter landings.

b. Head up display

If an R is displayed after the digital altitude readout near the center of the HUD (Figure 3-9), the altitude is from the RA. If no R is displayed, the digital altitude readout is from NAV software.

Digital altitude from NAV software is displayed on the HUD second declutter level if the altitude is less than 32,767 ft. RA altitude data is available only below 5000 ft. If the declutter level one or zero is selected, digital altitude is displayed when altitude reaches 100 ft.

Displayed digital altitude is changed every 200 ft down to 1000 ft, followed by every 100 ft down to 400 ft, then every 10 ft down to 50 ft. Below 50 ft, altitude is incremented every 1 foot down to touchdown.

The radar altitude symbol (— R) consists of a double width tick mark and an R character. Radar altitude is read off the altitude scale at the tick mark. Differences between the radar altitude and the NAV altitude are indicated by the delta between the radar (— R) and NAV altitudes. The radar altitude symbol is blanked when radar altitude data is not available (above 5000 ft). Displayed data depends on the position of the RADAR ALT select switch (1 or 2). Each HUD is individually selectable.

The altitude tape on the HUD consists of a fixed rectangle against which is read a moving scale of altitude in terms of height of the main gear above the runway. The scale moves in the same direction as the AVVI altitude scale. Scaling is variable with scaling changes occurring at the same time that changes occur on the AVVI altitude scale.

The glide-slope indicator (GSI) is a triangle facing the altitude scale. The symbol is free to move with respect to the altitude scale. When no error exists between the guidance-computed reference trajectory altitude and the actual altitude, the triangle lies exactly adjacent to the altitude rectangle. The GSI is a "fly to" indicator. When actual altitude is above the reference trajectory altitude, the triangle will be below the rectangle. To correct the altitude error, fly the Orbiter towards the triangle. The sense is comparable to the glide-slope on the HSI.

Note: This figure is not a valid flight configuration

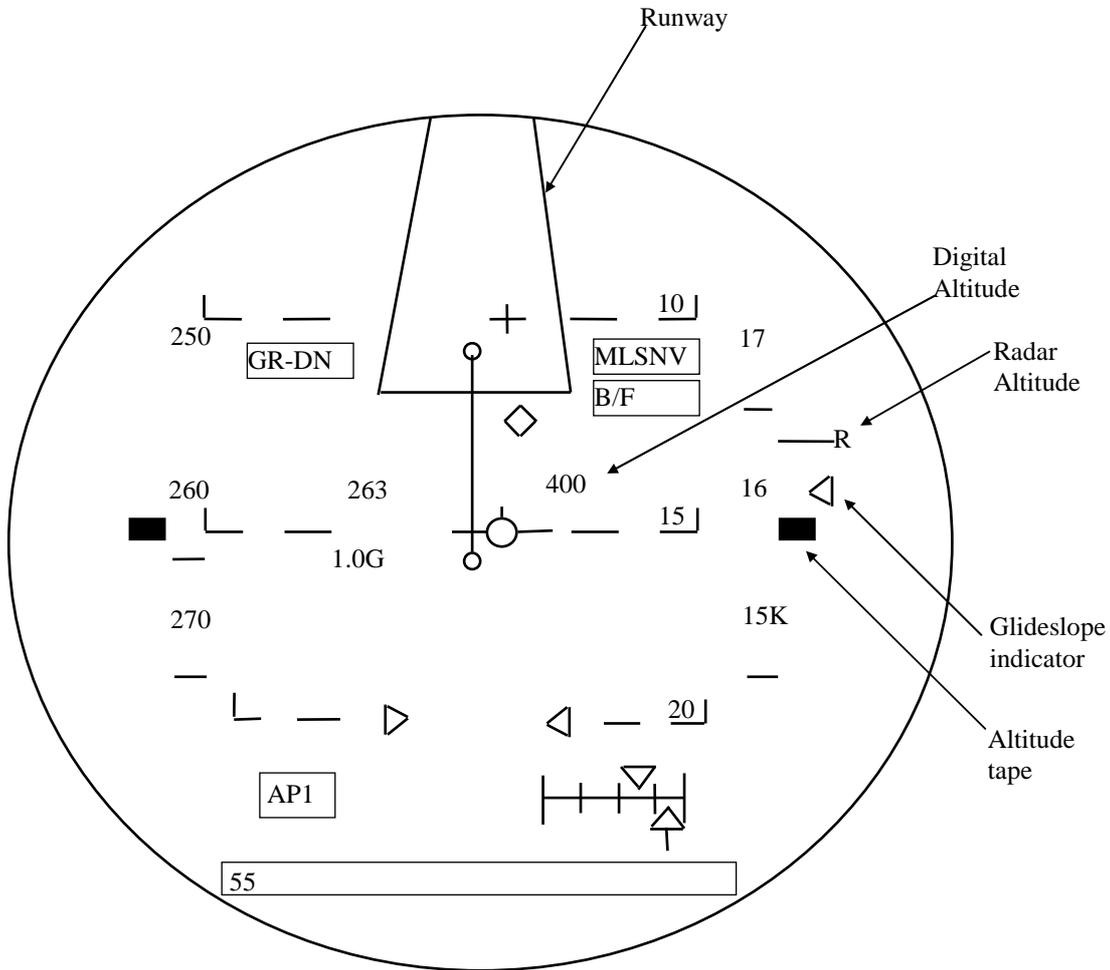


Figure 3-9. - HUD Display of RA altitude

3.4 RADAR ALTIMETER PASS SOFTWARE

Figure 3-10 shows the RA software interfaces involving the data processing module (GYR_RA_DP), the selection filter (SF) module (GRG_FIXED_SF), and the AVVI processing module (GDJ_AMI_AVVI_PROC). All RA software processing is performed at 6.25 Hz.

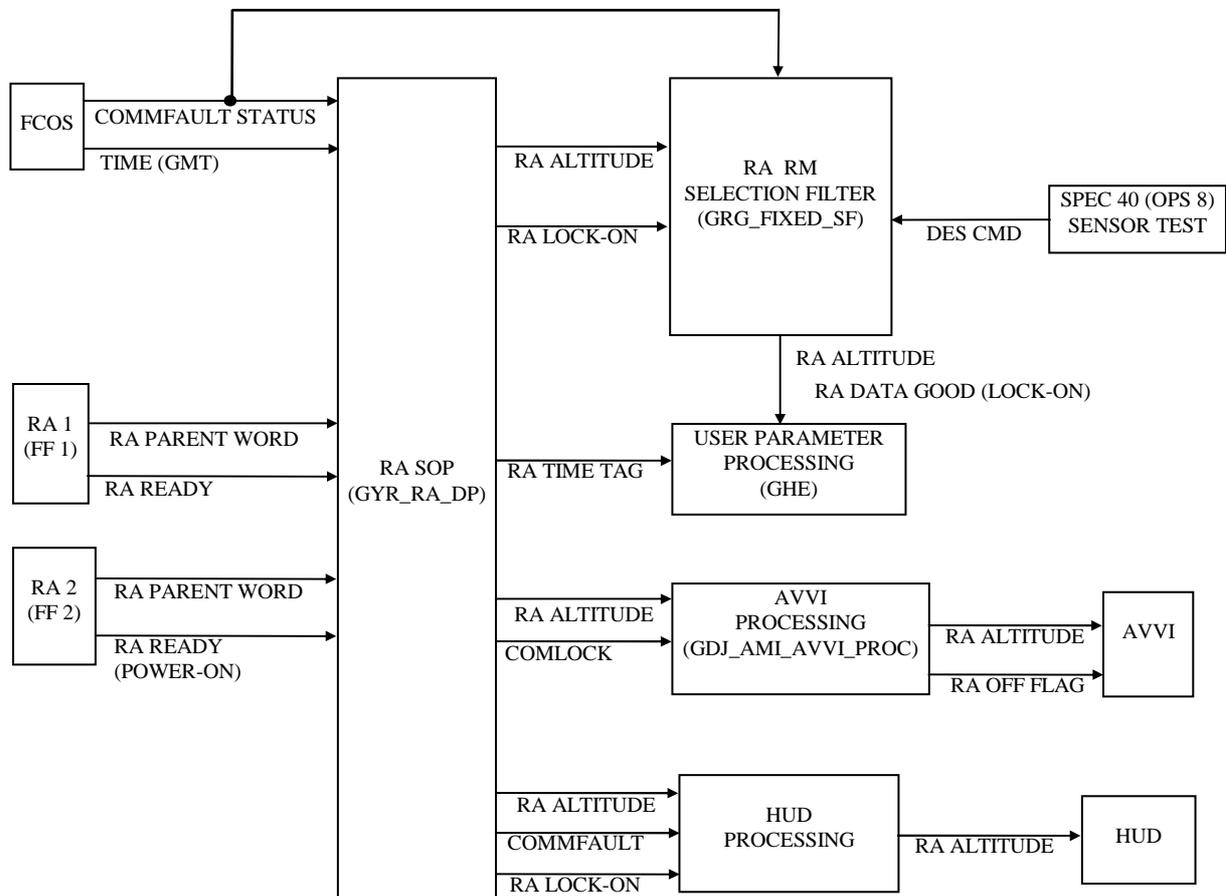


Figure 3-10. - RA software interface.

3.4.1 Radar Altimeter Data Processing

The RA subsystem operating program (SOP) (GYR_RA_DP) performs the following functions:

- a. Converts altitude data from fixed point to floating point.
- b. Initializes parity flags equal to false and strips flag bits from input data words.
- c. Time tags the data. Flight computer operating system (FCOS) provides the Greenwich mean time (GMT) to the RA SOP.
- d. Generates the lock-on and comlock flags. FCOS provides RA commfault status to the SOP and selection filter.
- e. Sends altitude data and lock-on flags to the SF module (GRG_FIXED_SF) and HUD processor for display.
- f. Sends altitude data and comlock flags to the AVVI processor. Comlock flags (RALT_CMLK) are only used by the AVVI processor at a rate of 1.04 Hz.

The RA software drawing module (GYR_RA) provides the data needed to drive the AVVI and HUD for crew display from 5000 ft to touchdown. This module provides the Orbiter altitude in terms of

feet of the Orbiter rear wheels (main landing gear) above the runway, assuming a nominal angle of attack. An I-Loaded value of 18.2 ft will be subtracted from the integer altitude data prior to being displayed on the AVVI. This 18.2 foot I-load corresponds to the difference between the Orbiter rear wheels and the nose, where the RA antennas are located.

The RA SOP is dispatched by the Ascent, return to launch site (RTLS), and Entry major functions under the following conditions: Ascent, if the external tank (ET) separation (SEP) CMD flag is ON; RTLS, if the RTLS ET SEP CMD is ON; and Entry, on transition to major mode (MM) 305 Terminal Area Energy Management (TAEM).

The RA SOP module is deactivated under the following conditions: for Ascent, when the ET SEP maneuver is complete in MM 103 or on transition to MM 104; for RTLS, when the AUTO -Z translation is complete in MM 601 or on transition to 602; for Entry, on transition to OPS 9 from OPS 3.

Even though the software is running onboard, the only time MCC receives RA telemetry is during the OPS 8 flight control system (FCS) checkout and during entry. During ascent, MM 103 to MM 104, the RA SOP module will work momentarily; however, the data will not be displayed on the crew's AVVI. To be able to observe the Orbiter distance from the ET during ET SEP, the crew would need to perform a GPC memory read of the RA altitude location in the common data pool (COMPOOL).

Once the RA is ready (powered ON), the first check performed is an odd parity check on the parent word. The raw altitude data is converted to integer data and is displayed as long as the RA is locked on, not commfaulted, and not deselected by the crew. Crew deselect can be done only in OPS 8 and carries over the OPS transitions to entry.

The RAs are powered on for ascent but MLS remain off partly because the onboard flight software does not support the MLS SOP in OPS 1 like it does the RAs. In addition to this, the MLS would not be able to lock on to a ground station because of the Orbiter's orientation and location from the runway; however, the RAs will lock on to the External Tank, giving the ground a chance to evaluate its performance prelaunch. Lastly the MLS are powered off during ascent because they emit a high powered KU-Band radiation which is considered a safety hazard. Since the MLS antennas are mounted on the nose of the vehicle looking forward, it is feasible that the MLS radiation could be reflected off of the tower prior to launch while the beanie is mounted to the top of the ET and radiate the vehicle. Although the RAs also emit radiation, they are still powered on for ascent because they emit at lower radiation levels.

3.4.2 Radar Altimeter Redundancy Management

The selection filter module (GRG_FIXED_SF) performs the following functions:

- a. Averages the altitude data from the two RA's.
- b. Performs a DATA GOOD test. The difference between the two altitude values must be less than the I-loaded limit (37 ft).
- c. Provides the data to user parameter processing (UPP) where it is made available to NAV processing but is not used by the NAV software.

The RA software does not contain any fault detection, identification, or reconfiguration (FDIR) since there are only two RA's and FDIR cannot determine the failed unit at the two level. The RA data does go through a selection filter which determines the status of the data.

At the two level (two RA's), the selected data is the average of the two values. A DATA GOOD test is performed which takes the absolute difference of the two values; if the difference is greater than 37 ft, DATA GOOD flag is set to logic 0 (BAD).

At the one level (one RA), the data from the selected RA is used. At this level, the DATA GOOD flag will always be set to GOOD. The selection filter down-modes to this level for the following reasons: if an RA is commfaulted; if an RA is deselected by the crew; or if an RA has broken lock (data validity = 0).

At the zero level, the RA DATA GOOD flag is set to BAD and no selected data is available.

3.4.3 RA Display

The AVVI processing module (GDJ_AMI_AVVI_PROC) uses altitude data from the RA SOP to drive the RAD ALT tape on the AVVI. Individual RA data, as selected by the crew, is available for display on the AVVI. The CDR and PLT can select either RA 1 or RA 2 for their display using the select switches for their displays. See section 3.3.2 for more on the location of the select switches.

Anytime the RA data is invalid, the COMMLOCK indication for that unit will go to logic 0, thereby causing the data as displayed on the AVVI to go static and the RA OFF flag to appear. The AVVI processor module will only display the RA altitude in MM's 304, 305, 602, and MM 603.

3.5 RA BFS SOFTWARE

3.5.1 RA PASS/BFS Differences

The backup flight system (BFS) processes RA data for display only on the AVVI during MM's 305 and 603. The BFS dedicated display processor will output data from RA 1 to both AVVI's if the unit has a valid lock-on. The BFS does not support RA select switch processing. In the event RA 1 does not have a valid lock-on, data from RA 2 will be used. The BFS dedicated display processor will continue to switch between the two RA's in an attempt to obtain one with a valid lock-on

3.6 RA SELF-TEST

The self-test mode occurs when the POWER-ON discrete is ON and the self-test is commanded. In this mode, the built-in test equipment tests the internal circuits and provides output data which indicates the health of the RA. An internal delay circuit delays the self-test-generated return signal, resulting in a self-test altitude of 1000 ft, plus or minus 100 ft.

The RA can be self-tested during the on-orbit OPS 8 FCS checkout and preflight during GNC OPS 9. The OPS 8 FCS checkout self-test is accomplished through use of the crew interface software, using the keyboard and the SENSOR TEST display, SPEC 40 (see fig. 3-11). A similar display, SPEC 101, SENSOR SELF-TEST (see fig. 3-12), is used for self-testing the RA in OPS 9.

| XXXX/040/ | | SENSOR TEST | | | | XX X | DDD/HH:MM:SS DDD/HH:MM:SS |
|-----------|---|-------------|------|--------|---------|-------|------------------------------|
| | | DES | STAT | RNG | AZ | EL | |
| MLS | 1 | 1X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| | 2 | 2X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| | 3 | 3X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| | | DES | STAT | RNG | AZ | | |
| TAC | 1 | 4X | XXXX | X.XS | XXX.XXS | | |
| | 2 | 5X | XXXX | X.XS | XXX.XXS | | |
| | 3 | 6X | XXXX | X.XS | XXX.XXS | | |
| | | DES | STAT | ALT | | | |
| RA | 1 | 7X | XXXX | XXXXS | | | |
| | 2 | 8X | XXXX | XXXXS | | | |
| | | DES | | Y | Z | | |
| AA | 1 | 9X | | ↑XX.XS | ↑XX.XS | | |
| | 2 | 10X | | ↑XX.XS | ↑XX.XS | | |
| | 3 | 11X | | ↑XX.XS | ↑XX.XS | | |
| | 4 | 12X | | ↑XX.XS | ↑XX.XS | | |
| | | START | 13X | STOP | 14X | | |

(XX)

Figure 3-11.- SPEC 40 Sensor Test (OPS 8).

| XXXX/101/ | | SENSOR SELF-TEST | | | | XX X | DDD/HH:MM:SS DDD/HH:MM:SS |
|-----------|---|------------------|------|----------|---------|-------|------------------------------|
| | | ENA | STAT | RNG | AZ | EL | |
| M | 1 | 1X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| L | 2 | 2X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| S | 3 | 3X | XXXX | XX.XS | ↑X.XXS | X.XXS | |
| | | ENA | STAT | RNG | AZ | | |
| T | 1 | 4X | XXXX | X.XS | XXX.XXS | | |
| A | 2 | 5X | XXXX | X.XS | XXX.XXS | | |
| C | 3 | 6X | XXXX | X.XS | XXX.XXS | | |
| | | ENA | STAT | ALT | | | |
| R | 1 | 7X | XXXX | XXXXS | | | |
| A | 2 | 8X | XXXX | XXXXS | | | |
| | | ENA | | Y | Z | | |
| A | 1 | 9X | | ↑XX.XS | ↑XX.XS | | |
| | 2 | 10X | | ↑XX.XS | ↑XX.XS | | |
| | 3 | 11X | | ↑XX.XS | ↑XX.XS | | |
| | 4 | 12X | | ↑XX.XS | ↑XX.XS | | |
| | | START | 13X | AUTO SEQ | 14X | | |
| | | TERM | 15X | INH ALL | 16X | | |

(XX)

Figure 3-12.- SPEC 101 Sensor self-test (OPS 9).

3.6.1 On-Orbit (OPS 8) FCS Checkout

SPEC 40, the SENSOR TEST display, permits self-testing the MLS, TACAN, RA and accelerometer assembly (AA) on orbit during the OPS 8 FCS checkout (reference: Orbit OPS Checklist and Data Processing System (DPS) Dictionary). These sensor tests are not individually selectable; all of the tests are started and stopped simultaneously. The self-test is initiated by an ITEM 13 EXEC (START) and terminated by an ITEM 14 EXEC (STOP).

Execution of an item number in the deselect (DES) column (items 7 and 8 for RA's) changes the selection of the LRU as a candidate for the SF. An asterisk (*) appears next to the item number to indicate when an LRU has been deselected. The selection filter configuration from this display in OPS 8 is carried over into OPS 3.

RA STAT will be displayed as BITE, OFF or blank. If the BITE status indication is set during the self-test, the BITE is latched and resets upon execution of ITEM 14 EXEC (self-test STOP).

Once the self-test is commanded, altitude for each RA will be displayed. The displayed altitude should be between 900 and 1100 feet. A status indicator beside each test parameter in the ALT column will show one of the following indications: M for missing data; a down arrow for data below the lower test limit (900 ft); an up arrow for data exceeding the upper test limit (1100 ft); or blank for self-test data within the test limits.

Should an RA fail self-test, the OPS 8 FCS checkout procedure in the Orbit OPS Checklist instructs the crew to deselect the unit and record the value in the ALT column; during OPS 3, a deselected RA will produce an OFF flag on the AVVI altitude tape unless the other RA has been selected to be displayed on both AVVI's.

3.6.2 Preflight (OPS 9) Self-Test

SPEC 101, the SENSOR SELF-TEST display, permits self-testing the MLS, TACAN, RA and AA preflight in GNC OPS 9. Execution of items 1 through 12 enables the respective LRU's for testing. Any number of these sensors can be enabled for self-test. The self-test is initiated by an ITEM 13 EXEC (START). Unlike the OPS 8 sensor test, individual sensors enabled on this display can be self-tested. An asterisk will be displayed next to the item numbers of the enabled LRU's. Item 14 (AUTO SEQ) initiates self-testing on all 12 sensors whether enabled or not.

RA STAT will be displayed as BITE, OFF, or blank. BITE is blanked whenever the condition is cleared. RA self-test altitude is displayed in feet. A status indicator beside each self-test parameter in the ALT column will show one of the following: a down arrow for data below the lower test limit (900 ft); an up arrow for data exceeding the upper limit (1100 ft); or blank for self-test data within the test limits.

ITEM 15 (TERM) terminates testing. ITEM 16 (INH ALL) inhibits all 12 sensors from self-test.

3.6.3 Self-Test Limitations

A range and rate simulator provide stimuli for the self-test. An internal delay circuit is applied to delay the return signal the appropriate time to generate a test value of 1000 ft with tolerance of plus or minus 100 ft.

The self-test circuitry develops its own timing pulse from the transmitter. Radiation is inhibited during the self-test. The self-test does not test the plate pulse cavity oscillator to transmitter antenna signal flow. Self-test does not utilize the 6.3-V ac heater voltage used by the plate pulse cavity oscillator. Self-test reduces the plate pulse modulator amplitude. Self-test does not check the receiver to mixer signal flow.

3.7 RA OPERATION

3.7.1 Onboard Monitoring (MM's 304, 305, 602 and 603)

The following dedicated displays, cathode ray tube (CRT) displays, and fault messages are used by the crew for assessing the status of the RA data.

a. Dedicated displays

| <u>Display</u> | <u>Location</u> | <u>Parameters</u> | <u>MSID</u> |
|----------------|-----------------|--------------------------------------|------------------|
| AVVI | PNL F7/F8 | Altitude on RDR ALT tape | V72H5356B (5396) |
| HUD | CDR/PLT | RDR ALT tape and digital altitude | |

b. CRT displays

SPEC 40, SENSOR TEST - Self-test altitude
SPEC 101, SENSOR_SELF_TEST - Self-test altitude

c. Caution and Warning/Fault Messages

There is no caution and warning associated with the RA. The following CRT fault message will appear in the event that all GPC's detected a problem receiving data from the indicated bus control element (BCE) of the indicated string and have bypassed that element.

CRT fault messages
BCE STRG 1 TAC FF1
BCE STRG 2 TAC FF2

Note that there is no BCE STRG RA fault message; instead, it is included in the BCE STRG TACAN message. Reference the Reference Data book, sections 2 and 3, for further details. These messages are available to the crew on the Orbiter CRT's and to MCC via telemetry.

3.7.2 MCC Monitoring

The following GNC displays, RT_PLOT, ELOG and CRANS are used in the MCC for assessing the status of the RA's.

| <u>Display</u> | <u>Parameters</u> | <u>MSID</u> |
|--|--|--|
| Ascent/Entry Sensors | RA 1 (2) ALT | V74H1804B(1854) |
| | RA 1 (2) LOCK-ON | V74X0172X (0182) |
| | R/W ALT | V95H0165C |
| | RA DATA GOOD | V90X4003X |
| | RA 1 (2) POWER | V74X0176X(0186) |
| | RA 1 (2) SWITCH ON/OFF | V74S0175E (0185) |
| Sensors Prelaunch | RA 1 (2) SWITCH ON/OFF | V74S0175E(0185) |
| | RA 1 (2) LOCK | V74X0172X (0182) |
| | RA 1 (2) POWER | V74X0176X (0186) |
| | RA 1 (2) ALT | V74H1804B (1854) |
| Sensors BFS | RA 1 (2) ALT | V98H3077C (3090) |
| <u>RT_PLOT</u> <u>SENSORS</u> | <u>Parameters</u> RA 1 & 2 ALT vs NAV ALT | <u>MSID</u> V74H1804B (1854) vs V95H0165C |
| <u>ELOG</u> <u>A/E_ GNC & SENSORS</u> | <u>Parameters</u> RA 1 (2) SWITCH ON/OFF RA 1 (2) VALIDITY RA DATA GOOD/BAD R RA SEL 1 (2) L RA SEL 1 (2) | <u>MSID</u> V74S0175E (0185) V74X1805B (1855) V90X4003X V72K8685X (8686) V72K8585X (8586) |
| <u>CRANS</u> <u>NAVAIDS MATRIX</u> | <u>Parameters</u> RA 1 (2) BYPASSED RA DATA GOOD/BAD RA 1 (2) SWITCH ON/OFF R RA SEL 1 (2) L RA SEL 1 (2) RA 1 (2) LOCK ON RA 1 (2) READY | <u>MSID</u> V91X2253XX(2254) V90X4003X V74S0175E (0185) V72K8685X (8686) V72K8585X (8586) V74X0172X (0182) V74X0176X (0186) |

3.7.3 Flight Rules

There are currently no space transportation system (STS) flight rules regarding RA requirements.

3.7.4 Launch Commit Criteria

There are two launch commit criteria (LCC) regarding RA's (CT-08 and DPS-17). Both LCCs state that one of two radar altimeters is required from RA power up to OPS-1 transition; however, two radar altimeters are required if less than three MLS available or if the planned abort or EOM landing is at night. RA's provide an increase in safety margin required at night because of the loss of visual depth perception.

3.7.5 Mission Performance

Early mission performance of the RA was characterized as marginal, prompting the need to incorporate a new modified RA and later, new RA antennas into the Orbiter which resulted in improved performance. Some of the problems associated with the older RA and antennas are included here for reference.

During STS-1 and 2, both RA's stopped tracking altitude and locked on to the nose landing gear when the landing gear was deployed. A hardware modification, which consisted of reducing the low altitude sensitivity, was implemented successfully. After that change, the new RA's performed nominally until the first flight of OV-103. In OV-103, RA 2 locked-on at a lower altitude (3000 ft) than at the nominal 5000 ft altitude level. At first, the problem was thought to be with the particular RA LRU. However, subsequent RA's installed in its place for later flights of OV-103 exhibited the same problem. Replacement of the receiving antenna did not correct the problem. The most likely cause of this RA anomaly has been attributed to a cracked/bent connector at the RA, which may be a result of having to install the cable in a hard-to-reach location. To fix this problem, the RA was rotated for easier access to the RA connections.

During STS-26 and STS-28, the RA's experienced another problem of dropping lock for a few seconds while flying over the Edwards lakebed. This was a byproduct of lowering the RA gain, to preclude locking on to the landing gear during deployment. New RA antennas with slightly less gain than the old antennas were installed on the vehicles. To compensate for the lower antenna gain, the gain of the new RA was increased, thereby fixing the problem of dropping lock while flying over a lakebed. Since the new antenna has a narrow Band-Width, lower gain, increased isolation, and an off-center directional attitude, it also helped prevent locking on to the nose gear/structure. RA's have been performing nominally..

3.8 RADAR ALTIMETER SUMMARY

a. Orbiter hardware

| | <u>RA 1</u> | <u>RA 2</u> |
|-------------------------------|---|--------------------------------|
| (1) Location: RA's | AV bay 1 | AV bay 2 |
| Transmitter antennas | Lower fuselage | Lower fuselage |
| Receiver antennas | Lower fuselage | Lower fuselage |
| (2) MDM | FF1 | FF2 |
| (3) Power supply | MN A DA1 014 | MN B DA2 015 |
| (4) Power on discrete | ESS 1 BC | ESS 2 CA |
| (5) ON/OFF switch | Panel 08 (S4) | Panel 08 (S5) |
| (6a) Select switch (pre-MEDS) | Panel F7/F8 | Panel F7/F8 |
| (6b) Select switch (MEDS) | Panel F6/F8 | Panel F6/F8 |
| (7) Circuit breaker | Panel 014: row E cb 24, 3 A | Panel 015: row E cb 23, 3 A |
| (8) Frequency | C-band (classified) | |
| (9) Thermal | Air cooled - AV bay fans | |
| | Minimum operating temp: | 35 deg F |
| | Maximum operating temp: | 140 deg F |
| (10) Power Requirement | 30 Watts, 28 V dc | |
| (11) Warm-up time | 180 sec | |
| (12) Nominal Acquisition ALT | 5000 ft | |
| (13) Altitude coverage limits | 0 to 5000 ft | |
| (14) Accuracy (1 sigma) | 0 to 10 ft: ±1.6 ft 11 to 66 ft: ±2.4 ft Above 66 ft: ± 3 percent | |
| (15) MSB | 4096 ft | |
| LSB | 1 ft | |
| (16) Expected lifetime | 1200 hours | |
| (17) Weight | RA LRU = 4.5 lb RA antenna = 1 lb 3 oz | |
| (18) Beamwidth | 50 deg | |

b. Self-test

| | <u>OPS</u> | <u>Test limits</u> | <u>SPEC</u> |
|----------------------------------|------------|--------------------|------------------------------|
| 1. OPS 8 FCS CHECKOUT (on orbit) | | 1000 ft ±100 ft | SPEC 40 SENSOR TEST |
| 2. OPS 9 (prelaunch) | | 1000 ft ±100 ft | SPEC 101 SENSOR_SELF_TEST |

c. PASS software/HAL code

| | <u>FSSR</u> | <u>DDS/HAL module</u> |
|--|-------------|-----------------------|
| (1) PASS modules | | |
| (a) RA data processor | GYR_RA | GYR_RA_DP |
| (b) RA display | | GDJ_AMI_AVVI PROC |
| (c) RA RM (fixed-point common selection filter) | | GRG_FIXED_SF |

(2) Interfaces

Inputs

RA data word
Commfault status
Lock-on status

Outputs

RA altitudes
Commfault flags
Lock-on flags

(3) Module summaries (processing rate = 6.25 Hz)

(a) RA data processing (GYR_RA_DP)

- Converts data from fixed point to floating point.
- Time tags data.
- Creates commfault/lock-on flag.
- Sends altitude data and lock-on flags to selection filter.
- Sends altitude data, comlock flag to AVVI processing.

(b) RA display (GDJ_AMI_AVVI_PROC)

Uses altitude from GYR_RA_DP to drive the radar altitude tape on the AVVI.

(c) RA RM (GRG_FIXED_SF)

- Averages altitude data.
- Data Good test is implemented: $ALT\ 1 - ALT\ 2 < 37\ ft\ (I-LOAD)$.
- Data is provided to UPP, where it is made available to the NAV processor (but is not used by NAV).

Table Of Contents

| | |
|--|-----------|
| 4.0 AIR DATA TRANSDUCER ASSEMBLY | 1 |
| 4.1 AIR DATA SYSTEM INTRODUCTION | 1 |
| 4.2 HARDWARE OVERVIEW | 1 |
| 4.2.1 ADS HARDWARE SYSTEM DESCRIPTION | 1 |
| 4.2.1.1 <i>Air Data Probes</i> | 1 |
| 4.2.1.2 ADTA | 6 |
| 4.2.2 ADTA BITE PROCESSING | 12 |
| 4.2.2.1 RADIATION INTEVENTION CIRCUITRY | 13 |
| 4.2.2.2 HIGH/LOW TEST | 14 |
| 4.2.2.3 PRESSURE TRANSDUCERS | 14 |
| 4.2.2.4 A/D CONVERTER | 14 |
| 4.2.2.5 ELECTRONICS | 14 |
| 4.2.2.6 TEMPERATURE CIRCUIT | 14 |
| 4.2.2.7 CODE MEMORY CRC | 15 |
| 4.2.2.8 POWER SUPPLY | 15 |
| 4.2.2.9 DISCRETE INPUTS | 15 |
| 4.2.2.10 RAM WRITE/READ-BACK | 15 |
| 4.2.2.11 FREQUENCY/DIGITAL CONVERTER | 15 |
| 4.2.3 PERFORMANCE DATA | 15 |
| 4.3 SOFTWARE OVERVIEW | 17 |
| 4.3.1 ADTA CRT DISPLAY OPERATIONS | 17 |
| 4.3.2 ADTA RM OVERVIEW | 19 |
| 4.3.3 ADTA PROBE RM | 19 |
| 4.3.3.1 ADTA USABILITY STATUS | 25 |
| 4.3.3.2 LEFT SF (ADTA 1, 3) | 27 |
| 4.3.3.3 RIGHT SF (ADTA 2, 4)..... | 27 |
| 4.3.3.4 ADTA DATA CONVERSION | 27 |
| 4.3.3.5 CENTER SF | 30 |
| 4.3.3.6 ADTA FDIR | 30 |
| 4.4 ADTA SOP | 36 |
| 4.4.1 ADS CALCULATION INITIATION..... | 36 |
| 4.4.2 ADTA ADS CALCULATIONS | 38 |
| 4.4.3 ADS HPC CONTROL | 41 |
| 4.4.4 NAVDAD SOP OVERVIEW | 41 |
| 4.4.4.1 NAVDAD SOP DETAILED DESCRIPTION..... | 42 |
| 4.4.4.2 NAVAIDS/DEFAULT AIR DATA TRANSITION..... | 52 |
| 4.4.4.3 LEFT/RIGHT AIR DATA CALCULATIONS FOR AVVI/AMI..... | 53 |
| 4.4.4.4 NAV SELECTED AIR DATA FOR AVVI AND AMI..... | 54 |
| 4.4.4.5 ADTA OVERRIDE DISPLAY AIR DATA CALCULATIONS..... | 54 |
| 4.4.5 ADTA SENSOR SELF-TEST | 55 |
| 4.4.5.1 ON-ORBIT (OPS-8) ADTA SENSOR SELF-TEST | 55 |
| 4.4.5.2 PREFLIGHT (OPS-9) ADTA SENSOR SELF-TEST..... | 59 |
| 4.5 BACKUP FLIGHT SYSTEM ADS (ADTA HIP) | 59 |
| 4.5.1 BACKUP FLIGHT SYSTEM ADS OVERVIEW | 59 |
| 4.5.2 BFS ADTA STATUS AND USABILITY | 61 |
| 4.5.3 BFS DATA SELECTION AND CONVERSION | 62 |
| 4.5.4 BFS ADTA ADS CALCULATIONS | 65 |

| | |
|---|-----------|
| 4.5.5 BFS NAV-DERIVED ADS CALCULATIONS (NAVDAD) | 65 |
| 4.5.6 BFS NAV/ADS AIR DATA TRANSITION..... | 65 |
| 4.5.7 BFS AIR DATA PARAMETER FOR DEDICATED DISPLAYS | 70 |
| 4.5.8 BFS AMI DATA PROCESSING | 70 |
| 4.5.9 BFS AVVI DATA PROCESSING | 71 |
| 4.6 NOTES OF INTEREST | 72 |

List of Tables

| | |
|--|----|
| Table 4- I ADTA Bus/MDM Matrix | 10 |
| Table 4 - II ADTA Built In Test (BIT) MODE/Status Word Annunciation..... | 12 |
| Table 4 - III ADTA Mode/Status Word | 16 |
| Table 4- IV ADTA GAX Annunciation | 17 |

List of Figures

| | |
|--|----|
| Figure 4 - 1 ADTA Functional Block Diagram | 2 |
| Figure 4 - 2 Air Data Probe Location | 3 |
| Figure 4 - 3 Air Data Probe Control | 4 |
| Figure 4 - 4 ADP Motor and Heater Control Logic | 5 |
| Figure 4 - 5 ADTA Probe Assembly (ADP) | 7 |
| Figure 4 - 6 Typical Probe | 8 |
| Figure 4 - 7 ADTA Interface Functional Block Diagram | 9 |
| Figure 4 - 8 ADTA Functional Block Diagram | 11 |
| Figure 4 - 9 On-Orbit RGA/ADTA/RCS Test Display | 20 |
| Figure 4 - 10 PASS HORIZ SIT display | 21 |
| Figure 4 - 11 PASS OVERRIDE Display | 22 |
| Figure 4 - 12 PASS GNC SYS SUMM 1 Display | 23 |
| Figure 4 - 13 Air Data RM Block Diagram | 24 |
| Figure 4 - 14 Probe RM and Usability | 26 |
| Figure 4 - 15 Left/Right Selection Filter | 28 |
| Figure 4 - 16 Data Conversion and Center Selection Filter | 29 |
| Figure 4 - 17 Left/Right FDI | 32 |
| Figure 4 - 18 Center FDI and Threshold Percent Computation | 33 |
| Figure 4 - 19 ADS Functional Block Diagram | 39 |
| Figure 4 - 20 ADS Calculation and Use Control | 40 |
| Figure 4 - 21 ADTA ADS Calculation | 43 |
| Figure 4 - 22 ADS HPC Control | 50 |
| Figure 4 - 23 NAVDAD Calculations | 51 |
| Figure 4 - 24 G&C ADS Transition and Control | 53 |
| Figure 4 - 25 ADTA ADS Left/Right Calculations | 56 |
| Figure 4 - 26 RM Sensor Calculations | 57 |
| Figure 4 - 27 Prelaunch RCS/RGA/ADTA Test Display | 58 |
| Figure 4 - 28 ADTA HIP Functional Block Diagram | 60 |
| Figure 4 - 29 BFS ADTA Status and Usability | 63 |
| Figure 4 - 30 BFS Data Selection and Conversion | 64 |
| Figure 4 - 31 BFS ADS Calculations | 66 |
| Figure 4 - 32 NAV-Derived ADS Calculation (NAVDAD) | 67 |
| Figure 4 - 33 BFS NAV/AIDS Air Data Transition Control | 68 |
| Figure 4 - 34 BFS NAV/ADS Air Data Transition | 69 |

SECTION 4

AIR DATA TRANSDUCER ASSEMBLY

4.0 AIR DATA SYSTEM INTRODUCTION

The air data system (ADS) provides information relative to the movement of the Orbiter in the air mass to guidance, navigation and control (GNC) for Orbiter control, and to dedicated displays for crew monitoring. The air data system functional block diagram is shown in figure 4-1.

Beginning with STS-99, the old Air Data Transducer Assembly (ADTA), was replaced with the Advanced Air Data Transducer (AADT). The ADTA Nomenclature was retained to avoid costs associated changes to on-board software, flight data file, and other documentation. A gradual deployment is planned with an AADT being flown in slot 1 of STS-99, 101, and 103, and a full four AADT deployment on STS-92. This brief will continue to refer to the LRU's as 'ADTA', even though they are technically 'AADT's'.

4.1 HARDWARE OVERVIEW

There are two deployable air data probes (ADP's), one on the left side and one on the right side of the Orbiter. When stowed, the probes are in the forward compartment underneath the Orbiter skin (fig. 4-2). Each probe contains ports (holes) which sense four pressures: PS (static), PaC (center or total), PaL (alpha lower), and PaU (alpha upper). In addition, each probe has a total temperature head, which contains dual sensors to measure temperature resistance, and three heater elements to prevent ice and/or moisture buildup. Pressures and temperatures sensed by the probe go to four independent air data transducers assemblies (ADTA's): The left probe is connected to ADTA's 1 and 3 and the right probe to ADTA's 2 and 4. Logic circuitry within each ADTA processes the pressures and temperature, and also performs internal hardware status checks for output to the data processing system (DPS). The pressures are used in software calculations to determine other air data parameters used by GNC during the terminal area energy management (TAEM), approach, and landing phases of entry.

4.1.1 ADS Hardware System Description

4.1.1.1 Air Data Probes

The ADP's are independently deployed by placing the ADP switch to the desired position (fig. 4-3). A deploy request results in closures of the dual series relays in the forward motor control assemblies (FMCA's), thus allowing three-phase ac power to the probe motors (fig. 4-4). The motors, through differential gearing, are mechanically linked to an actuator which rotates the probe outward so that it faces forward. When the probe is fully deployed, the arm of the actuator trips the deploy limit switches within the probe assembly. Closure of the deploy limit switches is fed back (inverted) to the deploy driver logic, thus opening the relays and removing power from the motors. The stow limit switches immediately open upon movement of the ADP from the stow position. The ADP is shown in figures 4-5 and 4-6.

Positioning the ADP switches to DEPLOY/HEAT allows one set of series relays in the forward power control assemblies (FPCA's) to close. When the probe is fully deployed, the same signal used to open the motor relays is also utilized to close a second set of series relays in the FPCA's, thus allowing power to the heaters.

Stowing the probes is accomplished in a similar manner. A second switch, AIR DATA PROBE STOW, must be in the ENABLE position to allow the operation of the STOW position of the probe switch. The

only other difference for stowing is that two phases of the motors are reversed. The status of each limit switch, deployed or stowed, is transmitted to the multiplexer/demultiplexer (MDMs).

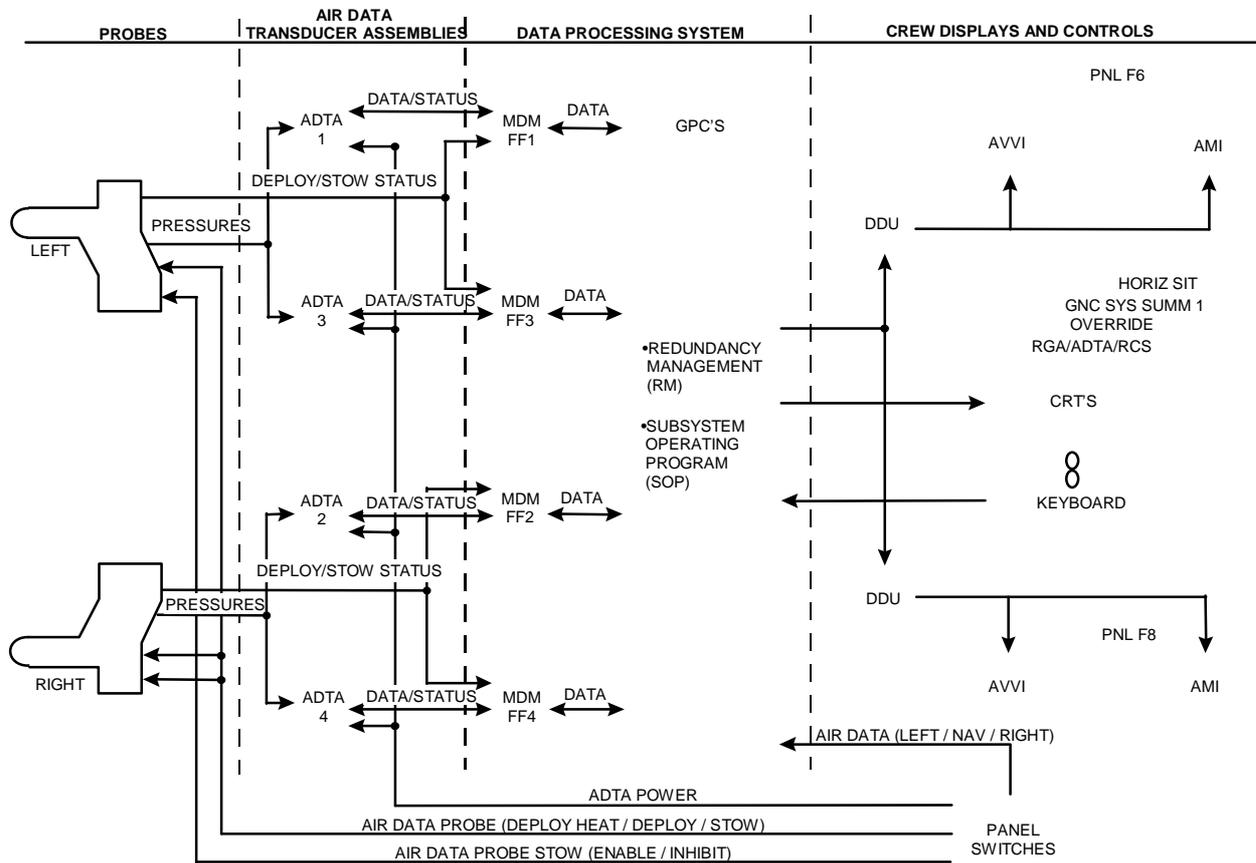


Figure 4 - 1 ADTA Functional Block Diagram

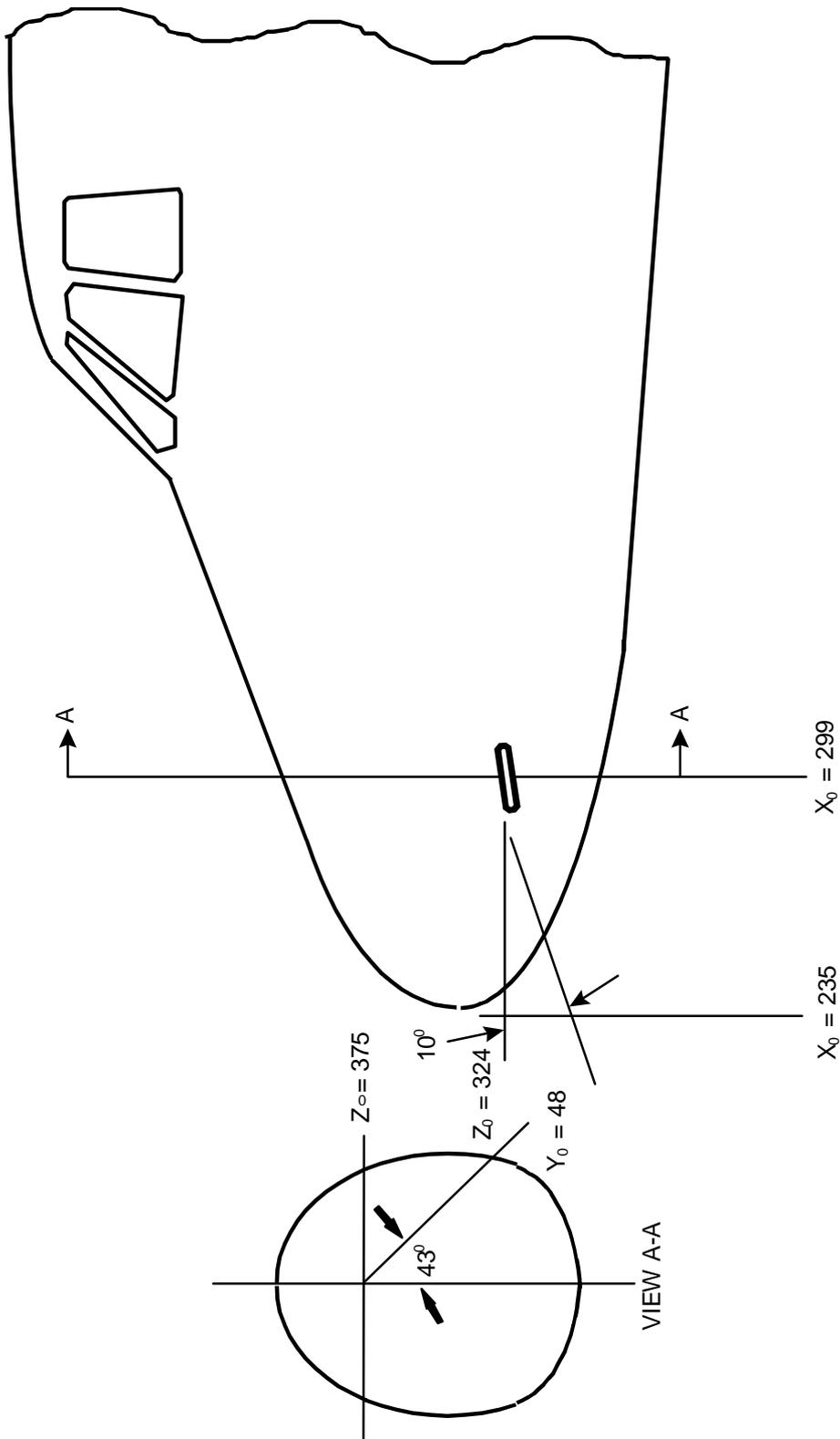


Figure 4 - 2 Air Data Probe Location

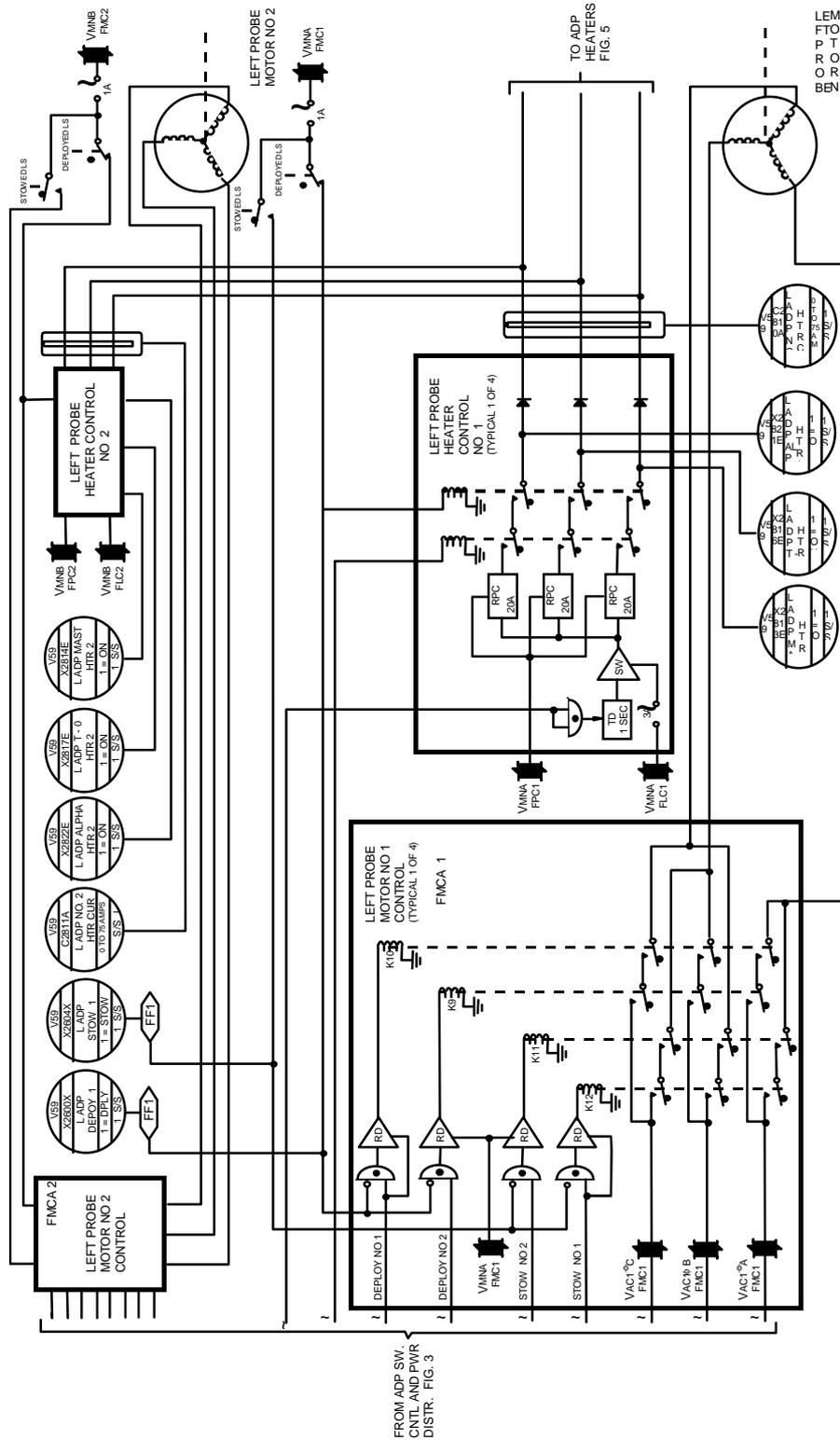


Figure 4 - 4 ADP Motor and Heater Control Logic

There is redundancy in the electronics and probe motors to ensure operation in the event of failures. Only one of the two motors is required to deploy or stow a probe. The deploy time for two motors is 15 sec; and for one motor, it is 30 sec. Other redundancy is provided in series relays in the FMCA so that if a driver fails closed, the other driver will open the path and remove motor power. The addition of the enable/inhibit switch allows power to be removed from the stow circuits in the inhibit position to eliminate the possibility of a single limit switch failure causing actuator damage.

The ADP's, one per Orbiter side, are located as shown in figure 4-2. Each probe consists of a pressure head, total temperature (T) head, mast, and heaters as shown in figures 4-5 and 4-6. The pressure head is comprised of a series of ports (holes) which sense pressures (PaC, PaU, PaL, and PS). Stagnation temperature or T is provided by dual temperature-sensitive resistive elements located in the T-head. A mast provides the structure to which the pressure and T-heads are mounted and serves as the interface with the actuator system. Each probe contains three heaters for purposes of de-icing. There is one heater in the vicinity of the pressure head, a second in the T-head, and one in the area of the mast. Heater activation, to prevent icing, is under the control of the crew. Operating on a maximum of 32 volts, the heaters will expend up to 1300 watts of power when activated.

Although the intent of the heaters is to prevent icing on the probes during entry, studies have shown that because of the aerodynamic heating effects on the probes, ice cannot form on them. With the probes in the deployed position during entry, the differences between the probe temperatures with the heaters ON and OFF is nearly negligible. However, if the heaters are used, then the ADP switch should be placed in the DEPLOY position postlanding, thus turning the heaters OFF and allowing the probes to cool. Note also that the use of the heaters will have no effect on the pressure data and the ADTA SOP calculations of the flight control parameters will be unaffected (ref. RI/D Memo No. SAS-TA-RCC-83-023, Thermal Analysis of the Air Data System (ADS) Probe with Internal Heaters; and Operational Space Shuttle Orbiter Heater Systems, JSC-18549, July 1988).

4.1.1.2 ADTA

The ADTA's receive 28 V dc power through circuit breakers shown in figure 4-7 which set a power-on discrete sent to the general purpose computer (GPC) through the flight-critical MDM's. ADTA MDM/bus loss matrix (table 4-1) provides a quick-look reference to determine the effect of power bus/MDM failures on the ADS system operation. The matrices are set up with all ADTA redundant components listed along the top in vertical columns. Along the side in horizontal rows are listed all the power buses and the flight-critical MDM's. An X in a given column and row indicates that a failure of the power bus or MDM labeled in the left-hand column for that row will result in the loss of the equipment or redundancy labeled at the top of that column. The ADTA's are located in the forward avionics bay (ADTA 1 and 3 in bay 1, ADTA 2 and 4 in bay 2) and radiantly cooled.

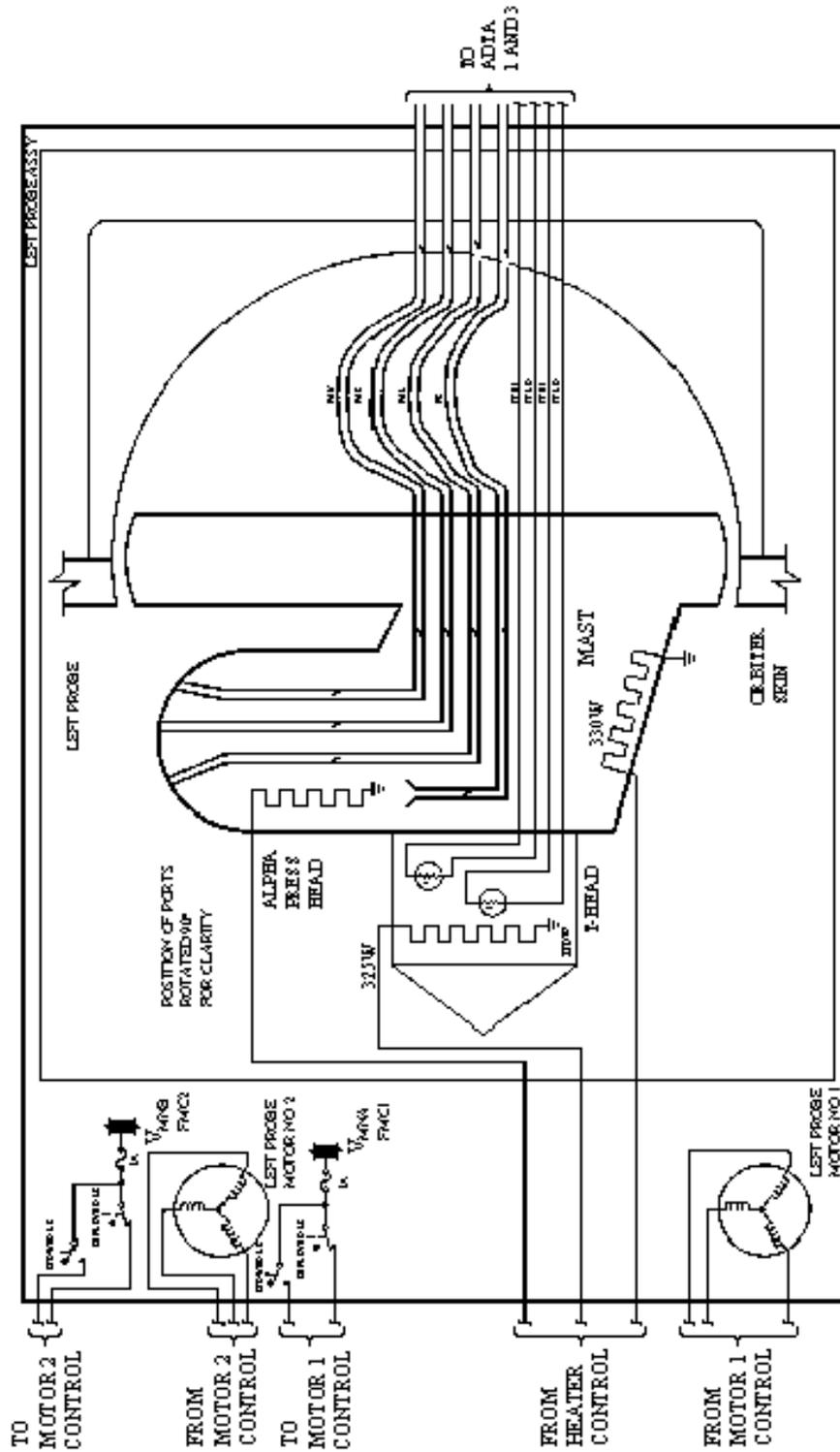


Figure 4-5. - ADTA probe assembly (ADP).

Figure 4 - 5 ADTA Probe Assembly (ADP)

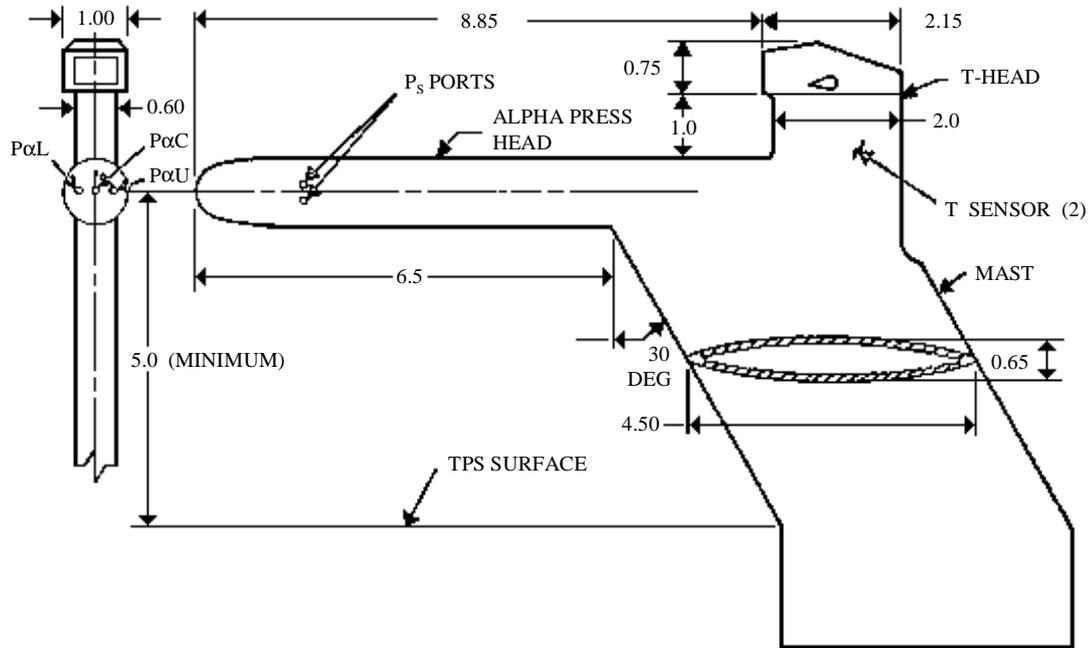


Figure 4 - 6 Typical Probe

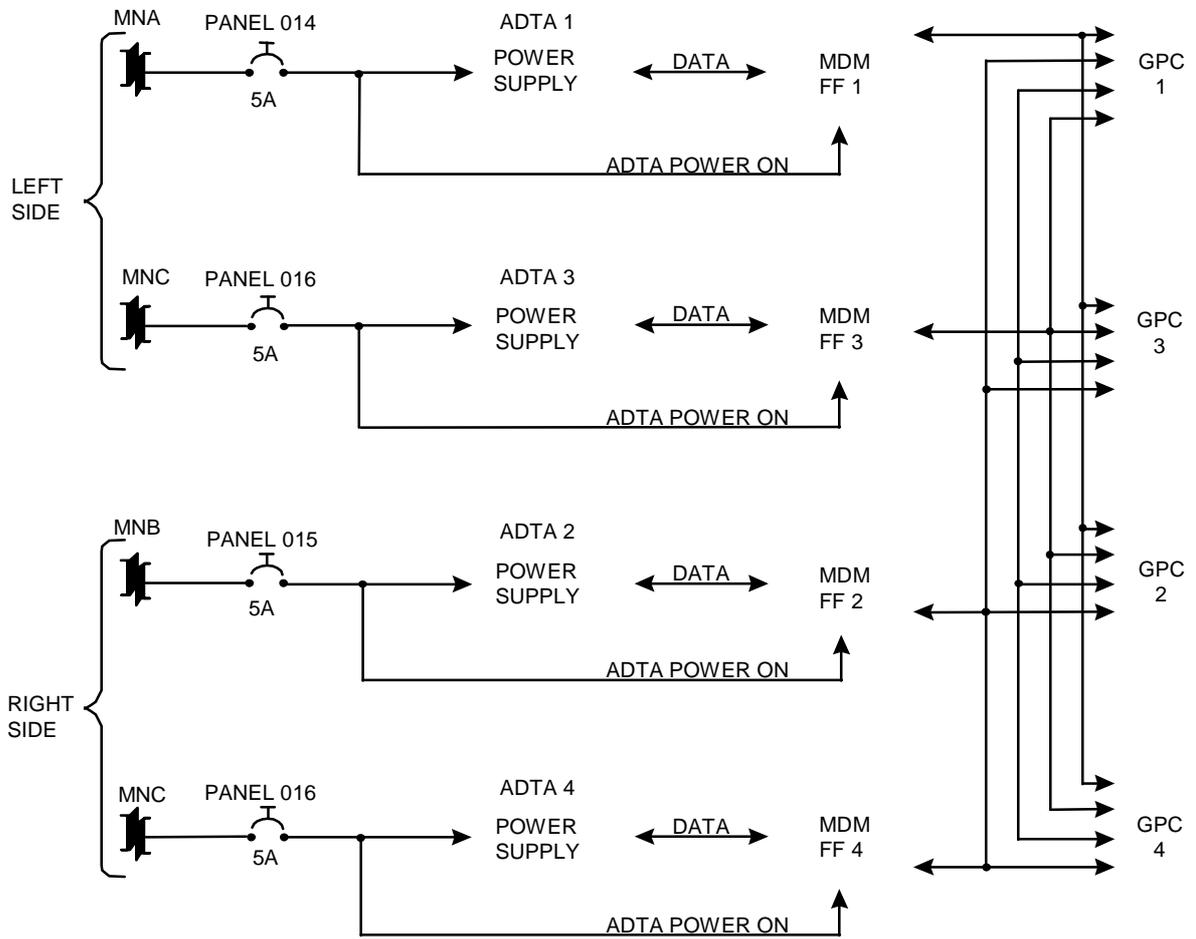


Figure 4 - 7 ADTA Interface Functional Block Diagram

Table 4- I ADTA Bus/MDM Matrix

| SSSHDWG | | 9.9 | | | | | | | | | | | | | | | |
|----------|--|-----|---|---|---|------------------------|-----------------------|-----------------------|--------|--------|--------|--------|----------------|----------------|----------------|----------------|--|
| PAGE 4 | | 1 | 2 | 3 | 4 | ADTA 18 & 3 (RMD) EF 1 | ADTA 2 & 4 (RMD) EF 1 | ADTA 2 & 4 (RMD) EF 2 | LAD M1 | LAD M2 | RAD M1 | RAD M2 | LADPHT CNTL L1 | LADPHT CNTL L2 | RADPHT CNTL L1 | RADPHT CNTL L2 | |
| MINADA1 | | X | | | X | | | | X | | | | | | X | | |
| FPC1 | | | | | X | | | | X | | | | | | X | | |
| FLC1 | | | | | | | | | | | | | | | X | | |
| FMC1 | | | | | X | | | | X | | | | | | X | | |
| AC3BC | | | | | | | | | | | | | | | | | |
| AC3BFMC1 | | | | | | | | | X | | | | | | | | |
| MPC1 | | | | | | | | | | | | | | | | | |
| APC4 | | | | | | | | | | | | | | | | | |
| ALC1 | | | | | | | | | | | | | | | | | |
| APC1 | | | | | | | | | | | | | | | | | |
| O14 | | X | | | | | | | | | | | | | | | |
| O14 & A8 | | | | | | | | | | | | | | | | | |
| MINADA2 | | X | | | X | X | | | X | X | | | X | X | X | X | |
| FPC2 | | | | | X | X | | | X | X | | | X | X | X | X | |
| FLC2 | | | | | | | | | | | | | X | X | X | X | |
| FMC2 | | | | | X | X | | | X | X | | | X | X | X | X | |
| AC3BC | | | | | | | | | | | | | | | | | |
| AC3BFMC2 | | | | | | | | | | | | | X | | | | |
| APC5 | | | | | | | | | | | | | | | | | |
| ALC2 | | | | | | | | | | | | | | | | | |
| APC2 | | | | | | | | | | | | | | | | | |
| O15 | | X | | | | | | | | | | | | | | | |
| O15 & A8 | | | | | | | | | | | | | | | | | |
| MINADA3 | | | X | X | | X | | | X | | | | | | X | | |
| FPC3 | | | | | | X | | | X | | | | | | X | | |
| FLC3 | | | | | | | | | | | | | | | X | | |
| FMC3 | | | | | | X | | | X | | | | | | X | | |
| AC3BA | | | | | | | | | | | | | | | | | |
| AC3BC | | | | | | | | | | | | | | | | | |
| AC3BFMC3 | | | | | | | | | | | | | X | | | | |
| MPC3 | | | | | | | | | | | | | | | | | |
| APC6 | | | | | | | | | | | | | | | | | |
| ALC3 | | | | | | | | | | | | | | | | | |
| APC3 | | | | | | | | | | | | | | | | | |
| O16 | | | X | X | | | | | | | | | | | | | |
| O16RJD | | | | | | | | | | | | | | | | | |
| CNTL AB1 | | | | | | | | | X | | | | | | X | | |
| AB2 | | | | | | | | | X | | | | | | X | | |
| AB3 | | | | | X | | | | X | | | | | | X | | |
| CNTL BC1 | | | | | | | | | X | X | | | X | X | X | X | |
| BC2 | | | | | | | | | X | X | | | X | X | X | X | |
| BC3 | | | | | X | X | | | X | X | | | X | X | X | X | |
| CNTL CA1 | | | | | | | | | | | | | X | | X | | |
| CA2 | | | | | | | | | | | | | X | | X | | |
| CA3 | | | | | | X | | | X | | | | X | | X | | |
| ESS 1RC | | | | | | | | | | | | | | | | | |
| 2CA | | | | | | | | | | | | | | | | | |
| 3AB | | | | | | | | | | | | | | | | | |
| FF 1 | | X | | | X | | | | | | | | | | | | |
| 2 | | X | | | X | | | | | | | | | | | | |
| 3 | | | X | | X | | | | | | | | | | | | |
| 4 | | | X | | X | | | | | | | | | | | | |
| FA 1 | | | | | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | | | | | |
| PF1 | | | | | | | | | | | | | | | | | |
| PF2 | | | | | | | | | | | | | | | | | |

The pressure and total temperature processing is shown in figure 4-8, ADTA functional block diagram. Pressures sensed at the ports are transmitted via pneumatic lines to the two ADTA's associated with the probe. The raw pressures are applied to a quartz diaphragm in the pressure sensor. The deposited metal on the transducer diaphragm acts as a variable capacitor as the diaphragm flexes with changes in pressure. An oscillator frequency, determined by the sensor capacitance, is then converted to a serial digital signal in the transducer Frequency/Digital (F/D) converter. These digital data (equivalent to sensed pressure) are sent to the Central Processing Unit (CPU). A transducer temperature measurement, calibration slope and offset data are also sent to the frequency to digital (F/D) converter and then to the CPU. The processor then linearizes and corrects the probe-sensed pressure measurement using temperature and calibration data.

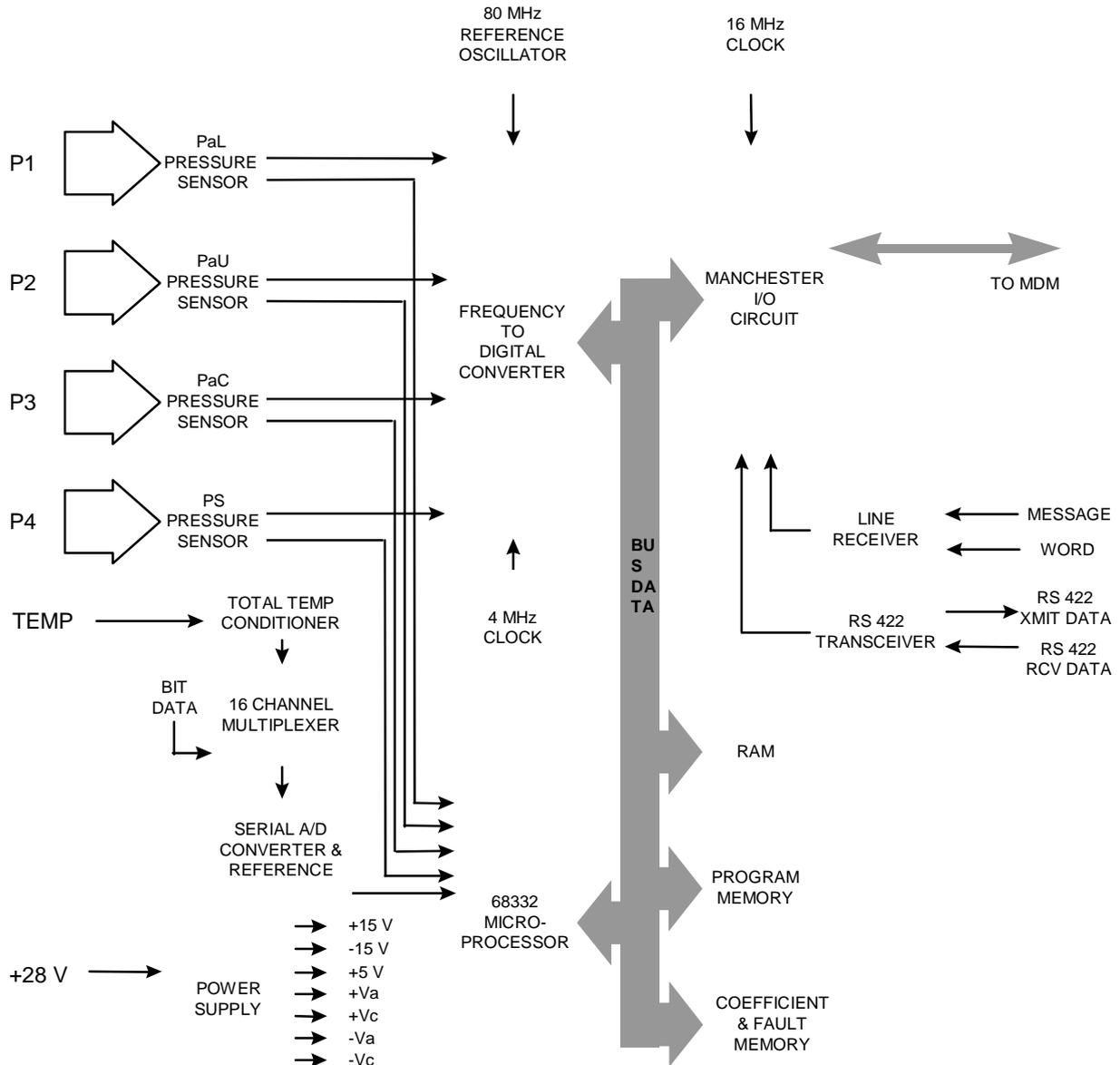


Figure 4 - 8 ADTA Functional Block Diagram

The resistive temperature measurement from the probe is applied to two identical temperature bridge circuits in the ADTA and converted to two total temperature analog signals. The two total temperature signals are sent to the A/D converter, changed from A/D, and then stored in the CPU.

4.1.2 ADTA BITE PROCESSING

Internal status checks and monitoring are also performed by the processor at power-up and during each cycle as part of the built in test (BIT) equipment circuitry. The processor performs reasonability checks on other components (e.g., power supply) and on the received pressures and temperature. In the event of an out-of-tolerance condition, a fail bit is set in the ADTA mode/status word. At the end of each computational cycle, PS, PaL, PaU, PaC, temperature, and the mode/status word are transferred to the Manchester Input/Output circuit, and the next cycle begins. By specification, the BIT failure detection reliability is at least 90%.

In addition to performing continuous BITE tests, each ADTA can be commanded to perform a high or low self-test. The self-test request from the MDM is validated in the processor and causes two actions. First, a set of constants, high or low values, are output from the processor for PS, PaU, PaL, PaC, and temperature. Secondly, the processor performs comparison checks on the sensed pressure measurements and compares them to a threshold. An out-of-tolerance condition results in a fail bit being set in the mode/status word. As is the case during continuous operation, the self-test results are transferred to the Manchester Input/Output circuit and finally to the MDM.

The structure of the ADTA mode status word is shown in table 4-II. The ADTA DG (bit 1) is utilized to drive a crew status indicator on operational sequence OPS 8 SPEC 41 and OPS-9 SPEC 102. The ADTA DG bit is set to '0' (fail) whenever any of bits 2, 5 through 10, or 11 through 16 indicate a fail condition.

Table 4-II outlines the individual BIT programs, the frequency of their execution, and what mode/status word bits will be set during a fail condition. The following sections describe the BIT sequences that drive the individual bits of the mode/status word.

Table 4 - II ADTA Built In Test (BIT) MODE/Status Word Annunciation

| Test / Mode/Status Bit | (15) 1 ADTA DG | (14) 2 Radition Ckt Good | (13) 3 High Test | (12) 4 Low Test | (11) 5 PaL Xducer Good | (10) 6 PaU Xducer Good | (9) 7 PaC Xducer Good | (8) 8 PS Xducer Good | (7) 9 A/D Converter Good | (6) 10 Electronics Good | (5) 11 Temp Circuit Good | (4) 12 Code Memory CRC | (3) 13 Power Supply Good | (2) 14 Discrete Inputs Good | (1) 15 RAM Memory Good | (0) 16 F/D Good | Periodic/Continuous | Power-up | Test Mode |
|----------------------------------|----------------|--------------------------|------------------|-----------------|------------------------|------------------------|-----------------------|----------------------|--------------------------|-------------------------|--------------------------|------------------------|--------------------------|-----------------------------|------------------------|-----------------|---------------------|----------|-----------|
| Watchdog Timer Test | X | | | | | | | | | X | | | | | | | | X | |
| Watchdog Time-out | | | | | | | | | | | | | | | | | X | X | |
| Power Shut Down Circuitry Test | X | X | | | | | | | | | | | | | | | | X | |
| Invalid Power-up State Test | X | X | | | | | | | | X | | | | | | | | X | |
| Radiation Intervention Circuitry | X | X | | | | | | | | | | | | | | | X | | |
| Current Range Test | | | | | | | | | | | | | | | | | | | |
| RAM Write/Read-back Test | X | | | | | | | | | | | | | | X | | | X | |
| NVM Write/Read-back Test | | | | | | | | | | X | | | | | | | | X | |
| Code CRC Test | X | | | | | | | | | | | X | | | | | | X | |
| Sensor Characterization Data | X | | | | X | X | X | X | | | | X | | | | | | X | |
| ROM Power-up Test | | | | | | | | | | | | | | | | | | | |
| Sensor Characterization Data | X | | | | X | X | X | X | | | | X | | | | | X | | |
| Periodic Test | | | | | | | | | | | | | | | | | | | |
| Pressure Count Test | X | | | | X | X | X | X | | | | | | | | | | | |
| Pressure Count Cross-Check | X | | | | X | X | X | X | | | | | | | | | | X | |
| Power-up Test | | | | | | | | | | | | | | | | | | | |
| Pressure Sensor Cross-Check | X | | X | X | X | X | X | X | | | | | | | | | | | X |

| | | | | | | | | | | | | | | | | | | | |
|---|---|--|--|--|---|---|---|---|--|---|--|--|--|--|--|--|--|---|---|
| Temperature Count test | X | | | | X | X | X | X | | | | | | | | | | X | |
| Temperature Count Cross-Check Test | X | | | | X | X | X | X | | | | | | | | | | | X |
| F/D Power-up Test | X | | | | X | X | X | X | | | | | | | | | | | X |
| F/D Conversion Complete | X | | | | X | X | X | X | | | | | | | | | | | X |
| Manchester Output Serial Device Power-up Test | X | | | | | | | | | | | | | | | | | | X |
| Manchester Output Serial Device Periodic Test | X | | | | | | | | | | | | | | | | | | X |
| Oscillator Cross-Check Power-up Test | X | | | | X | X | X | X | | | | | | | | | | X | X |
| A/D Test | X | | | | | | | | | X | | | | | | | | | X |
| Total Temperature Circuit Test | | | | | | | | | | | | | | | | | | | X |
| Total Temperature Circuit Cross-Compare Test | | | | | | | | | | | | | | | | | | | X |
| Discrete Input Test | X | | | | | | | | | | | | | | | | | | X |
| Power Supply Voltage Test | X | | | | X | X | X | X | | | | | | | | | | | X |
| Pressure Sensor Supply Voltage Test | X | | | | X | X | X | X | | | | | | | | | | | X |
| Linear Reference Voltage Range Test | X | | | | | | | | | | | | | | | | | | X |
| Distributed Voltage Range test | X | | | | | | | | | | | | | | | | | | X |

4.1.2.1 RADIATION INTEVENTION CIRCUITRY

During power up and periodically during operation, the ADTA BIT performs a checkout of the radiation circumvention circuitry. This circuitry is required to clear Single Event Latch-up (SEL) conditions in un-hardened circuitry, specifically the CPU, frequency/digital (F/D) converter, analog/digital (A/D) converter, and the Manchester Input/Output (I/O) circuit. SEL conditions result from ionizing radiation disrupting the previously mentioned integrated circuits. The radiation circumvention circuit performs a power cycle of the ADTA electronics (“warm start”) fast enough (35.7 msec) not to cause an I/O error at the MDM (40 msec cycle). Note that depending on the timing of the MDM polling and the SEL, a single I/O Error “hit” is possible and will be registered by the GPC. Two consecutive I/O Errors are necessary to generate a bypass condition, requiring an ‘I/O RESET’ to re-establish communication. The circuit will continue to attempt to clear the latch-up for 4 occurrences accumulated since the last power-up, and will power down the ADTA after a 5th unsuccessful attempt, resulting in a commfault. The only recovery option is to power cycle the unit in an attempt to clear the latch-up condition. Based on current draw of the monitored circuits, the Radiate Circuit Good bit will be set to ‘1’ (fail), indicating the radiation circumvention circuitry has failed to act upon a latch-up condition. Results of the failure to clear a latch-up condition depend on the affected circuit.

For a latch-up in the F/D converter, pressure transducer data will be lost, resulting in one or more of the following bits being set to ‘0’ (fail): Alpha 1 Transducer Good (bit 5), Alpha 2 Transducer Good (bit 6), Total Pressure Transducer Good (bit 7), and Static Pressure Transducer Good (bit 7). The BFS will downmode for any of these pressure BITE conditions.

For a latch-up in the CPU, response is unpredictable. The ADTA may output erroneous data.

For a latch-up in the A/D converter, total temperature data will be lost, but there will be no change in the mode/status word. Note: the temperature circuit is only used for calibration of the pressure transducers and not used in air data calculations. The A/D converter also processes BIT data, so this condition could result in a loss of some BIT processing. See A/D Converter BITE.

If the Manchester I/O circuit goes into latch-up and fails, the ADTA will not respond to a poll request from the MDM and result in a commfault.

The Radiation Circuit Good BITE in the mode status word is set whenever current levels indicate a latch-up condition is present. These currents are monitored through the A/D converter, and an A/D Converter BITE will result in this test being masked (and as a result the Radiation Good BITE since this is the only test driving that BITE). The radiation circuit is not affected by a loss of A/D Converter, and will continue operating even though BITE fault detection is lost. Note: there is a 2.5 V reference used in the radiation circuit, but it is sourced from the I/O board and independent of the 2.5 V reference voltage used in the A/D BITE tests.

4.1.2.2 HIGH/LOW TEST

High and Low test is initiated by crew command from SPEC 41 as part of FCS checkout, or during prelaunch in OPS 9. During high test ($P > 25$ in. Hg), the transducer is declared failed if it differs by more than 0.04 in Hg from the other sensors. During low test ($P \leq 25$ in Hg), failure is declared if it differs by more than 0.08 in Hg. The appropriate pressure bits (5 through 8) are set to indicate which transducer has failed.

4.1.2.3 PRESSURE TRANSDUCERS

These bits will be set during various periodic, power-up, and self test initiated BIT programs. Depending on which test is in progress, one or all can be enunciated. For F/D and oscillator failures, all pressure bits can be expected to indicate 'fail' ("0"). During pressure tests, the failed transducer(s) would enunciate fail. Unlike the previous generation ADTA, a pressure rate check was not included in the AADT design since RM is expected to detect any pressure variation and remove the appropriate ADTA from the selection filter.

4.1.2.4 A/D CONVERTER

The Analog/Digital BIT is a periodic test which performs range checking on a 2.5 V reference voltage the A/D ground reference. A raw count outside the allowed range results in a failure of the A/D test. The A/D converter processes voltage inputs to the MC 68332 microprocessor from selected locations within the ADTA circuitry. The A/D BITE itself means that the 2.5 V A/D reference, and other voltages being monitored by BIT through the A/D are suspect. These include the +15, -15, +5 V supplies, and linear reference voltages. The +5 V provides central power for all integrated circuits (Vcc input). Thus, a real +5 V supply problem would result in a bypass (commfault) of the ADTA since the Manchester I/O would lose power. The +/-15 V provides the +/-10 V input voltages for the pressure transducers. A failure of the A/D Converter BITE will inhibit the Power Supply Voltage Test and Pressure Sensor Supply Voltage Test which drive the Power Supply Good BITE (see below) and Pressure Transducer BITEs (although other built in tests will continue to drive the pressure BITEs if found out of tolerance). A self test can confirm health of the pressure circuits. The Linear Reference Voltage Range Test and Distributed Voltage Range Test also rely on the ground, 2.5 V, and 5 V indications provided by the A/D converter. These tests drive the Electronics Good BITE condition, and are also inhibited in the presence of an A/D Converter BITE.

4.1.2.5 ELECTRONICS

During power-up, the Watchdog Timer Test, Non-Volatile Memory (NVM) Test, Manchester Output Serial Device Power-Up Test, and Oscillator Cross Check Power Up Test will set this bit for a fail condition. The Manchester Output Serial Device Test, Linear reference Voltage Range Test, and Distributed Voltage Range Test are performed periodically, and will result in a '0' (fail) state. The presence of an A/D Converter BITE will inhibit the Linear Reference Voltage Range Test and Distributed Voltage Range Test.

4.1.2.6 TEMPERATURE CIRCUIT

The temperature circuit tests are performed periodically and at power-up. They do not set any of the mode/status words to fail (the old ADTA's set bit 11). This bit is not set for any of the BIT sequences.

4.1.2.7 CODE MEMORY CRC

A code memory cyclic redundancy check (CRC) is performed at power up and periodically to check utilized code storage memory locations.

4.1.2.8 POWER SUPPLY

The power supply bit is set periodically by the Power Supply Voltage Test and the Pressure Sensor Supply Voltage Test. The former checks the power supply ± 15 and $+5$ V circuits, and the latter the pressure sensors ± 10 V circuits. Allowable ranges for the tests are:

| | |
|--------------------|------------------|
| +15 V Power Supply | 12.75 to 16.50 |
| -15 V Power Supply | -16.50 to -12.75 |
| +5 V Power Supply | 4.65 to 5.30 |
| +10 V Power Supply | 9.0 to 11.2 |
| -10 V Power Supply | -11.2 to -9.0 |

Presence of and A/D Converter BITE will inhibit the tests driving this BITE, however a lack of commfault and successful completion of self test is available to verify performance of the circuit being monitored by the lost BITE condition.

4.1.2.9 DISCRETE INPUTS

The Discrete Input Test periodically reads the contents of the Manchester status register bits 0 through 3 and compares bit 0 with the complement of bit 2, and bit 1 with the complement of bit 3. If the bits do not compare with each other, the discrete input test fails and the mode/status word updated to reflect the condition (bit 14 set to '0'). The purpose of this test is to verify the ability to command self test, which requires complementary inputs to generate a valid self test command.

4.1.2.10 RAM WRITE/READ-BACK

The RAM write/read-back test is performed only at power-up and consists of a sequence of write and reads to each memory address of the static RAM (SRAM) and external RAM (ERAM). The sequence verifies the patterns of FFFF (HEX), AAAA (HEX), 5555 (HEX), and 0000 (HEX) can be successfully written to each address. Failure of any read-back results in the RAM Memory Good bit (bit 15) being set to '1' (fail). After power-up, the RAM Memory Good bit does not change.

CPU response to this failure is unpredictable and the CPU may continue to complete its processing and output erroneous data.

4.1.2.11 FREQUENCY/DIGITAL CONVERTER

The F/D circuit is tested during the Oscillator Cross-Check Test at power up. Only this test will set this BITE in the mode/status word. Other BIT sequences and failures result in other mode/status word bits being set to 'fail' (the four pressure bits), and used as a verification of the health of the F/D converter.

4.1.3 PERFORMANCE DATA

Some of the performance data and operational limitations are listed below.

- a. Performance envelope
 - Altitude 0 to 105,000 ft
 - Mach number 0.1 to 3.5
 - Angle of attack -4° to 20°
 - Maximum temperature 850° F
- b. Probe deploy time

| | | |
|----|-----------------------------|------------------------|
| | 2 motors | 15 sec |
| | 1 motor | 30 sec |
| c. | Maximum probe actuator load | 375 lb/ft ² |
| d. | ADTA warmup time | 3 sec |
| e. | ADTA operating temperature | -20° to +120° F |
| f. | Power | |
| | Probe heaters | 1300 W per probe (max) |
| | ADTA | 12 W |

Table 4 - III ADTA Mode/Status Word

| | | Status Word Bits : 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 | | | | | | | | | | | | | | | |
|--------|---------------------------|--|--|--|--|--|--|--|--|--|--|--|--|-------------------------------------|--|--|--|
| | | Nominal: 1 0 0 0 1 1 1 1 1 1 1 1 1 1 0 0 | | | | | | | | | | | | | | | |
| BIT 1 | ADTA GOOD | 0 = FAIL. DRIVEN BY SELECTED BIT FAIL CONDITIONS | | | | | | | | | | | | | | | |
| BIT 2 | ADTA RADIATE CIRCUIT GOOD | 0 = GOOD. FAILURE TO CLEAR LATCH UP CONDITION | | | | | | | | | | | | | | | |
| BIT 3 | HIGH TEST MODE | HIGH TEST CMD FROM OPS 8 SPEC 41 OR OPS 9 SPEC 102 | | | | | | | | | | | | | | | |
| BIT 4 | LOW TEST MODE | LOW TEST CMD FROM OPS 8 SPEC 41 OR OPS 9 SPEC 102 | | | | | | | | | | | | | | | |
| BIT 5 | PaL TRANSDUCER GOOD | REASONABLENESS TEST OPS 8 PaL-PaU COMPARISON TEST | | | | | | | | | | | | 0 - 45.5 in. Hg +/- 0.080 in. Hg | | | |
| BIT 6 | PaU TRANSDUCER GOOD | REASONABLENESS TEST OPS 8 PaU-PaS COMPARISON TEST | | | | | | | | | | | | 0 - 45.5 in. Hg +/- 0.080 in. Hg | | | |
| BIT 7 | PaC TRANSDUCER GOOD | REASONABLENESS TEST OPS 8 PaC-PaL COMPARISON TEST | | | | | | | | | | | | 0 - 45.5 in. Hg +/- 0.080 in. Hg | | | |
| BIT 8 | PS TRANSDUCER GOOD | REASONABLENESS TEST OPS 8 PaS-PaC COMPARISON TEST | | | | | | | | | | | | 0 - 32.5 in. Hg +/- 0.080 in. Hg | | | |
| BIT 9 | A/D CONVERTER GOOD | A/D REFERENCE VOLTAGE TEST | | | | | | | | | | | | | | | |
| BIT 10 | ADTA ELECTRONICS GOOD | OSCILLATOR, I/O, REFERENCE VOLTAGE, AND NVM TEST | | | | | | | | | | | | | | | |
| BIT 11 | ADTA TEMP CIRCUIT GOOD | NEVER SET. TOTAL TEMP REASONABLENESS TEST (TEMP NOT USED IN AD CALCULATIONS) | | | | | | | | | | | | | | | |
| BIT 12 | ADTA CODE MEMORY CRC | NVM AND ROM ADDRESS CHECKSUM | | | | | | | | | | | | | | | |
| BIT 13 | ADTA POWER SUPPLY GOOD | PRESSURE SENSOR AND RADIATION CIRCUIT SUPPLY VOLTAGE CHECKS | | | | | | | | | | | | | | | |
| BIT 14 | ADTA DISCRETE INPUTS GOOD | SELF TEST CMD CAPABILITY CHECK | | | | | | | | | | | | | | | |
| BIT 15 | ADTA RAM MEMORY GOOD | 0 = GOOD. POWER UP MEMORY WRITE/READ CHECK | | | | | | | | | | | | | | | |
| BIT 16 | ADTA F/D GOOD | 0 = GOOD. POWER UP F/D OSCILLATOR CHECK | | | | | | | | | | | | | | | |

4.2 SOFTWARE OVERVIEW

ADTA redundancy management (RM) and subsystem operating program (SOP) (fig. 4-1) are the major software modules or processes associated with the ADS. RM software validates the ADTA's to ensure that a failed component is removed and also reduces multiple data from the ADTA's to a single output set. The ADTA SOP contains the algorithms which generate air data parameters for GNC and dedicated displays. The alpha/Mach indicator (AMI) and altitude/vertical velocity indicator (AVVI) are the dedicated displays which provide air data related information to the crew. Cathode ray tube (CRT) displays rate gyro assembly (RGA)/ADTA/reaction control system (RCS), OVERRIDE, GNC SYS SUMM 1, and HORIZ SIT monitor and control the ADS. The ADTA software processing rates for major mode (MM) 304, 305, 602, and 603 are as follows.

| | |
|--------------------------------|---------|
| RM | 1.04 Hz |
| SOP | |
| ADS CALC | 1.04 Hz |
| ADS LEFT/RIGHT CALC (AMI,AVVI) | 1.04 Hz |
| RM SENSOR CALC (OVERRIDE DISP) | 0.5 Hz |
| NAVDAD | 1.04 Hz |

ADTA RM also provides a dilemma signal to the caution and warning (C&W) system and fail signals to the systems management (SM) alert system. The fault summary messages (FSM's) generated by the ADTA RM are shown in table 4-III, ADTA GAX ANNUNCIATION.

If the ADTA loses power, the transfer of the serial digital data to the GPC will be terminated (bypassed), and the following FSM will be generated in OPS G1, 3, 6, and 8.

BCE STRG 1 ADTA
BCE STRG 2 ADTA
BCE STRG 3 ADTA
BCE STRG 4 ADTA

Table 4- IV ADTA GAX Annunciation

| MML No. | LVLB | Nomenclature | Source | OPS avail | ALRM | | | MSG ID | Message-Text | |
|-----------|--------|------------------------|---------|-----------|------|-----|-----|--------|--------------|------|
| | | | | | N F | C L | I D | | Major | Min |
| V90X5001X | G15001 | ADTA 1 FAIL | ADTA RM | 36 | | | | | | |
| V90X5051X | G15002 | ADTA 2 FAIL | ADTA RM | 36 | | | | | | |
| V90X5101X | G15003 | ADTA 3 FAIL | ADTA RM | 36 | | | | | | |
| V90X5151X | G15004 | ADTA 4 FAIL | ADTA RM | 36 | | | | | | |
| V94X4737X | | ADTA 1/2/3/4 FL MSG | | 36 | | | 3 7 | 13644 | RM FAIL | ADTA |
| V90X4840X | | DELAYED ADTA DLMA FLAG | ADTA RM | 36 | | | | | | |
| V94X4738X | | ADTA DLMA MSG | | 36 | | | 2 7 | 13844 | RM DLMA | ADTA |

4.2.1 ADTA CRT DISPLAY OPERATIONS

The CRT displays that provide a means for monitoring and controlling the ADS are as follows. Numbers in parenthesis correspond to number on figures 4-9 through 4-12, respectively.

a. SPEC 41 OPS 8 RGA/ADTA/RCS (fig. 4-9)

(1) ADTA. For the ADTA's PS, PaC, PaU, and PaL in in. of mercury and total T in degrees Celsius are displayed while testing is in progress. A status parameter by each value will be driven with an (M) to indicate missing data, an up arrow () to indicate test data too high, a down arrow () to indicate test data too low, or blank if data are normal. ADTA status (STAT) shows BITE (double overbright), or blank. The BITE status is a reflection of the Data Good bit (mode/status word bit 1). If BITE is displayed, it will be blanked if condition is corrected. Whether or not testing is in progress, items 5 through 8 used to deselect or select the corresponding sensor as a candidate for the selection during ENTRY (OPS 3). An asterisk (*) beside the item number indicates that the corresponding ADTA is deselected. ADTA the error may be filter (SF)

b. SPEC 050 OPS 3, 6 HORIZ SIT (fig. 4-10)

- (1) Item 9, the ALTM data field, shall provide the capability to input the landing field barometric altimeter setting corrected to mean sea level. The display is initialized the first time with this parameter set to 29.92 in. Hg, and, thereafter, shall reflect crew inputs.
- (2) The RESID column contains the residual data value for each component of each data type that is being processed by the navigation (NAV) filter. The residual is formed in navigation by subtracting the NAV estimate of the data from the SF output of the data. (It should be noted here that NAV uses the composite data output from the SF as inputs to the NAV filter so that no distinctions are made as to the LRU source of the data to the NAV filter. In the two-LRU cases, the LRU data are averaged so that the data cannot be distinguished as to LRU. The residual value gives the crew an indication as to how well the data and the NAV estimate of the data agree. However, this data alone cannot identify whether the data or the NAV estimate of the data is in error.
- (3) The RATIO column contains the edit ratio computed in navigation as the ratio of the data residual to the maximum allowable residual. If this ratio is less than one, the residual edit test performed by navigation on the data is passed, and the data are subsequently used to update the state vector. If the data fail the residual edit test, the ratio will be displayed as a number equal to or larger than one and will not be used to update the state vector. Thus, the edit ratio and the ratio parameter status indicators give the crew an indication of whether a particular data type is being incorporated into the state vector. The residuals and edit ratios are computed by NAV each NAV cycle and are available for display. When data are not being processed by the NAV filter, the residual and ratio data fields for that parameter shall be blanked. A parameter status indicator column to the right of the RATIO will contain a down arrow when the parameter has failed the update edit test. The down arrow will disappear when the edit test is passed by a predefined percentage of good data. The parameters defining percentage of good data are I-loaded.
- (4) There are three filter control options: automatic (AUT), inhibit (INH), and force (FOR) which the crew can exercise on three data types: TACAN (TAC AZ and RNG), drag altitude (DRAG H), and baro altitude (ADTA H) to update NAV. Items 19, 22, and 25 allow navigation to automatically select data to be used if it passes the edit test. Items 20, 23, and 26 (INH) preclude updating the state vector with the RM-selected data, but these data are

used to generate the displayed residual and ratio values. Items 21, 24, and 27 (FOR) shall override the edit and force the data to be used to update the state vector if the data are being processed by the NAV filter. Items 19-21, 22-24, and 25-27 are mutually exclusive.

An asterisk (*) appears next to any Item (19-27) that has been selected. The display will be initialized with items 20 (INH), 22 (AUT) and 26 (INH) selected.

(5) Items 28-30 provide control over the source of air data parameters to guidance and flight control. Item 28 (AUT) provides auto transition from NAV-derived air data to ADTA. Item 29 (INH) inhibits the ADTA data. Item 30 (FOR) forces the use of ADTA data. These items are mutually exclusive. In "AUT", MACH must be less than 2.5 to process ADTA parameters. The display will be initialized with item 29 (INH) selected.

c. SPEC 051 OPS 304, 305, 602, 603 OVERRIDE (fig. 4-11)

(1) For each of the four ADTA LRU's, the following data are displayed to the crew cyclically to allow them to determine whether an LRU failed by RM is in agreement with the other LRU's and should be reselected or to aid in solving RM dilemmas: ADTA derived altitude in feet, ADTA-derived angle of attack in deg, and ADTA-derived Mach number. A parameter status column after the LRU ID number shall be blank for normal operation, display a down arrow (∨) for an RM declared failure or crew deselection, display a question mark (?) for an RM dilemma, or an M for missing data. There are four item numbers (34, 35, 36, 37) which allow the crew to reselect an LRU for use by the ADTA SF. These items are legal in MM 304, 305, 602, and 603. Execution of these items in OPS 1, MM 301, 302, 303, or 601 will result in an ILLEGAL ENTRY message.

d. DISP 018 OPS 3, 6 (ADTA) GNC SYS SUMM 1 (fig. 4-12)

(1) For each LRU, input/output (I/O) errors, failure, manual deselection, and dilemma status indicators shall be displayed. For all LRU's, failure and manual deselection shall be ORed. The status output for each LRU will normally be blank. For failures detected by RM or manual deselection, down arrows (∨) shall be output. Question marks (?) shall be output for each LRU in the voting logic if a dilemma (disagree and/or fault detect) is declared. If the data for any sensor is missing (I/O error), the COMM fault (CF) shall be reflected by the output of an M status indicator.

4.2.2 ADTA RM OVERVIEW

ADTA's 1 and 3 are connected to the left ADP, and ADTA's 2 and 4 are connected to the right ADP. The ADTA RM block diagram (fig. 4-13) shows the data flow from redundant inputs to the selected output. In addition to the SF, the FDIR determines the data used in the SF process. The ADP RM and ADTA usability status determine configuration for FDIR and SF. FDIR usability works on an LRU basis. RM can be broken down into left, right, and center data pairs. The data sets used in RM are PS, PaC, PaU, and PaL. The failed ADTA's and ADS dilemma may be reset by the crew select/deselect command.

4.2.3 ADTA PROBE RM

The ADTA probe RM function is shown in figure 4-14. The purpose of the ADP RM is to determine if one or both probes are deployed and to provide this status to other software functions. The ADTA SOP and RM are contingent upon probe RM software to determine when a deployed state exists for either probe.

The probe deploy status for each probe is a logical function of the two redundant deployed discrettes (D₁, D₂), two redundant stowed discrettes (S₁, S₂), and their associated CF status as shown.

| | | | | | | | |
|-------------|------|--------------|---------|---------|---------|---------|--------------|
| XXXX/041/ | | RGA/ADTA/RCS | | | XX | X | DDD/HH:MM:SS |
| | | X | | X | | | DDD/HH:MM:SS |
| DES | STAT | R | P | Y | | | |
| R1 | 1X | XXXX | +XX.XS | +XX.XS | +XX.XS | | |
| G2 | 2X | XXXX | +XX.XS | +XX.XS | +XX.XS | | |
| A3 | 3X | XXXX | +XX.XS | +XX.XS | +XX.XS | | |
| 4 | 4X | XXXX | +XX.XS | +XX.XS | +XX.XS | | |
| DES | STAT | PS | PαC | PαU | PαL | T | |
| A1 | 5X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS |
| D3 | 6X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS |
| T2 | 7X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS |
| A4 | 8X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS |
| TEST | HI | 9X | LOW | 10X | STOP | 11X | |
| RCS DRIVERS | | | | | | | |
| RJDF | 12X | TEST EXEC | 15X | TEST | XXXX | | |
| RJDA-A | 13X | CONTINUE | 16X | JET | XXX | | |
| -B | 14X | STOP | 17 | | | | |

(XX)

Figure 4 - 9 On-Orbit RGA/ADTA/RCS Test Display

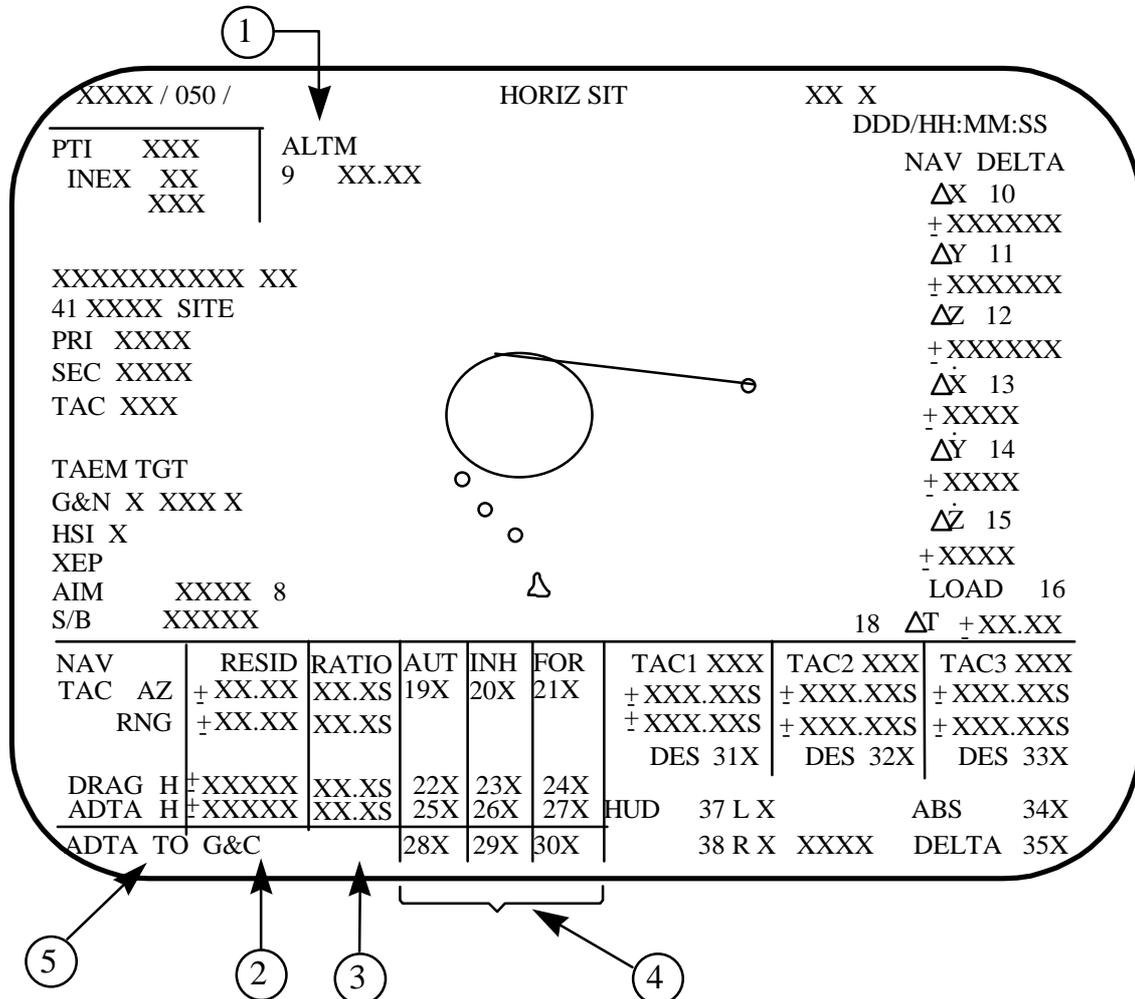


Figure 4 - 10 PASS HORIZ SIT display

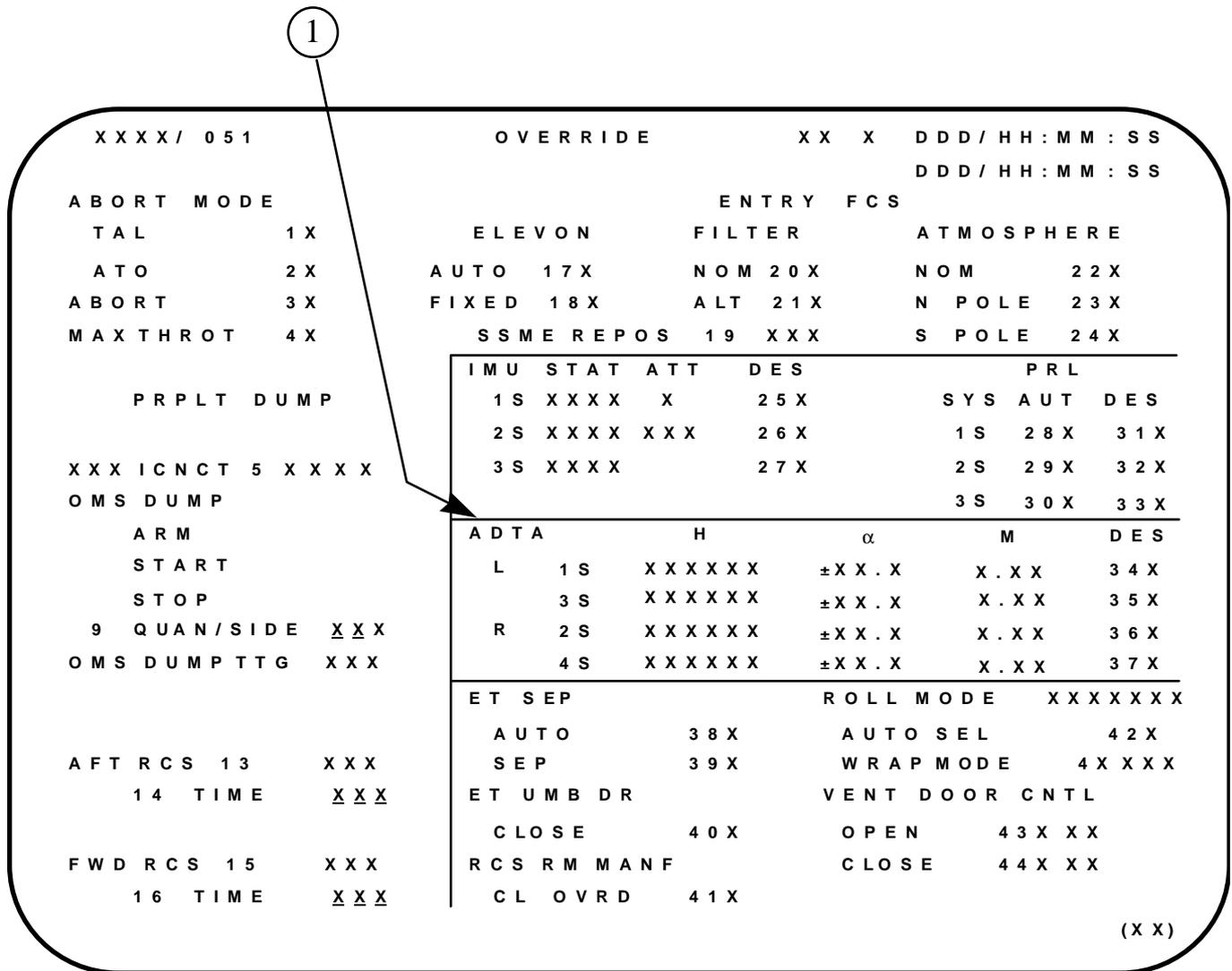


Figure 4 - 11 PASS OVERRIDE Display

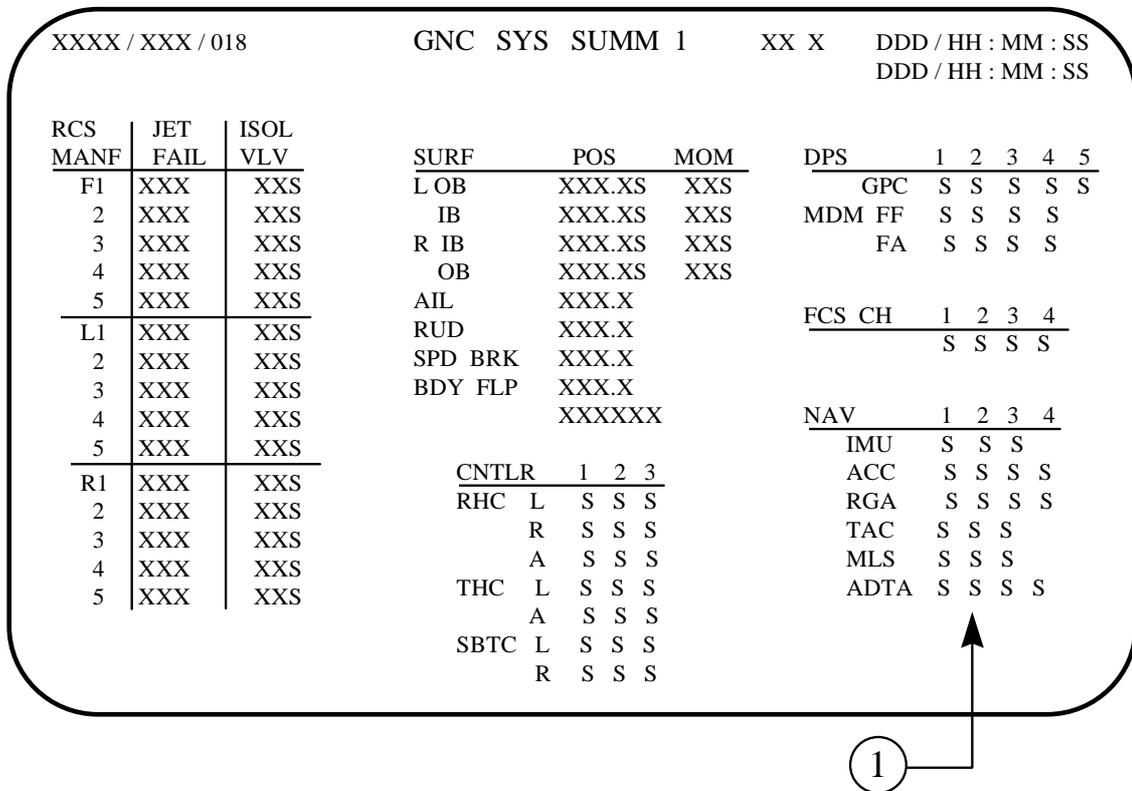


Figure 4 - 12 PASS GNC SYS SUMM 1 Display

The left and right deploy indications are sent to the event annunciation processing function. If either (left or right) ADTA probe is not deployed during $1 \leq M \leq 2.5$, an ADTA PROBE Fault Summary Message will be generated.

In addition to the L and R deploy flags used to determine ADTA data usability, the "either probe deploy" flag is generated to initiate the ADTA ADS calculations. The "either probe deploy" flag will go true when both probes are deployed or a single probe has been deployed for 15 sec. With an ADP motor failure, the ADP will take 30 sec to deploy with one motor. If a deploy limit switch fails closed after the stow limit switch opens, the ADS calculations will be delayed 15 sec for the ADTA probe to deploy. Once the either probe deploy flag has switched to the deployed state, it shall be latched for the remainder of the flight.

4.2.3.1 ADTA USABILITY STATUS

The ADTA usability state (fig. 4-14) is controlled by the following.

- a. ADTA probe deployed
 - R probe deploy - ADTA 2 and 4 usable
 - L probe deploy - ADTA 1 and 3 usable
- b. ADTA usable if no parameter has failed RM
- c. ADTA usable if no CF
- d. ADTA usability can be controlled by the crew select/deselect capability from SPEC 41 (OPS-8) RGA/ADTA/RCS display and SPEC 51 (OPS 304, 305, 602, 603) OVERRIDE display.

The parameter status indicators (PSI's) on CRT displays SPEC 41, SPEC 51, and DISP 18 allow the crew to monitor the ADTA usability.

| | |
|---|------------|
| SPEC 41 (OPS-8) "RGA/ADTA/RCS" | * = DES |
| SPEC 51 (OPS 304, 305, 602, 603) "OVERRIDE" | * = DES |
| | ↓ = DES/FL |
| | ? = DLMA |
| DISP 18 (OPS 3, 6) "GNC SYS SUMM 1" | ↓ = DES/FL |
| | ? = DLMA |

The PSI is blank if ADTA is usable, and displays "M" if ADTA is COMM faulted.

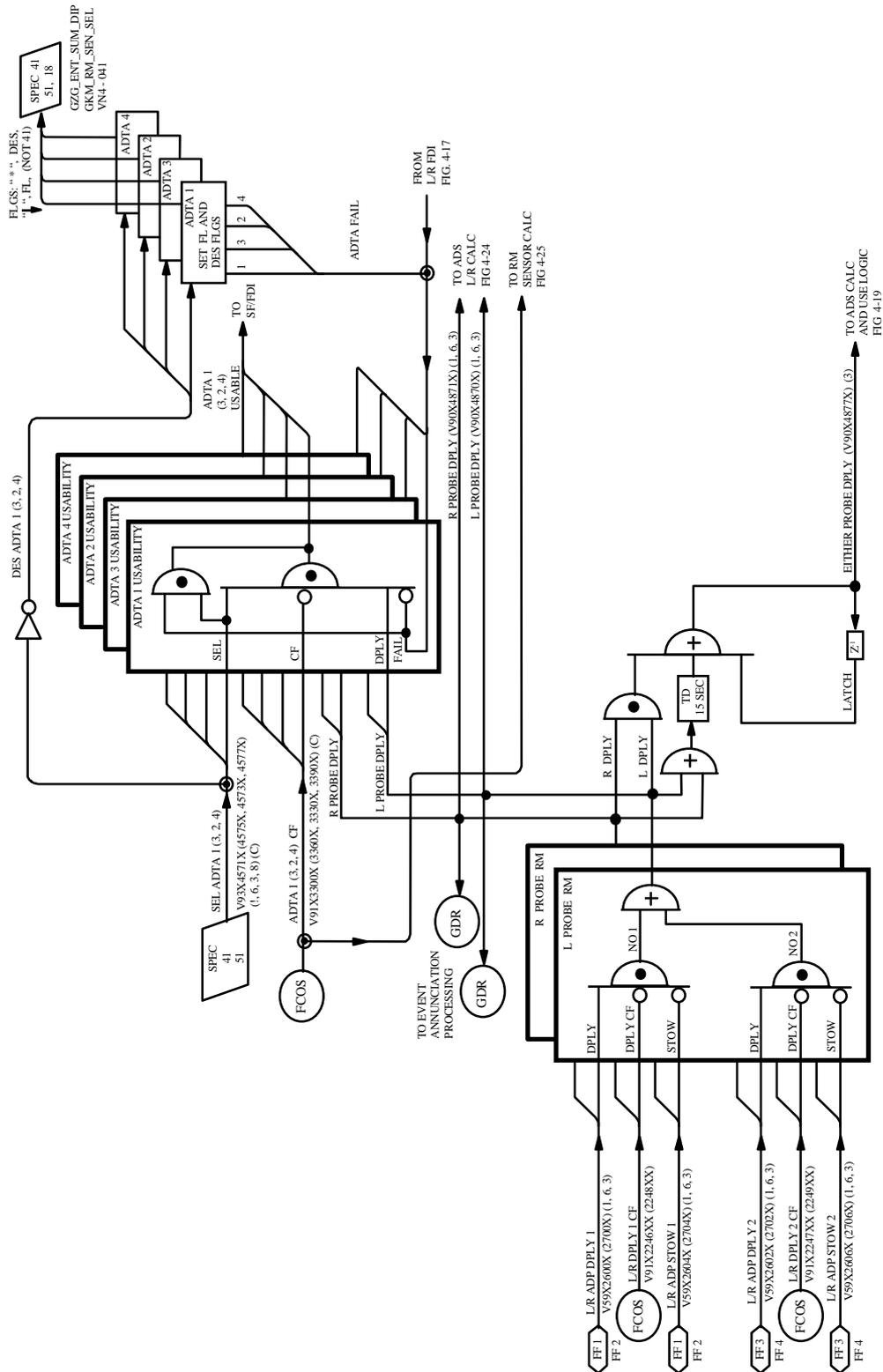


Figure 4 - 14 Probe RM and Usability

Figure 4-14. - Probe RM and ADTA usability

4.2.3.2 LEFT SF (ADTA 1, 3)

Depending upon ADTA usability status, the SF processing (fig. 4-15) performs the following functions.

- a. If ADTA 1 and 3 usable, L Sel Data is average.
- b. If ADTA 1 not usable, L Sel Data is ADTA 3.
- c. If ADTA 3 not usable, L Sel Data is ADTA 1.
- d. If ADTA 1, 3 not usable, ADTA 1 NO CF, L Sel Data is ADTA 1.
- e. If ADTA 1, 3 not usable, ADTA 1 CF, L Sel Data is ADTA 3.

The data from the left SF are in counts and come from the hardware via MDM's 1 and 3.

4.2.3.3 RIGHT SF (ADTA 2, 4)

Depending upon ADTA usability status, the SF processing (fig. 4-15) performs the following functions.

- a. If ADTA 2 and 4 usable, R Sel Data is average.
- b. If ADTA 2 not usable, R Sel Data is ADTA 4.
- c. If ADTA 4 not usable, R Sel Data is ADTA 2.
- d. If ADTA 2, 4 not usable, ADTA 2 NO CF, R Sel Data is ADTA 2.
- e. If ADTA 2, 4 not usable, ADTA 2 CF, R Sel Data is ADTA 4.

The data from the right SF are in counts and come from the hardware via MDM's 2 and 4.

4.2.3.4 ADTA DATA CONVERSION

Data that have been selected from the left side (left select data) and from the right side (right select data) must be converted from counts to engineering units (fig. 4-16) for use by the air data SOP in order to provide air data for the dedicated displays (AMI/AVVI). These converted

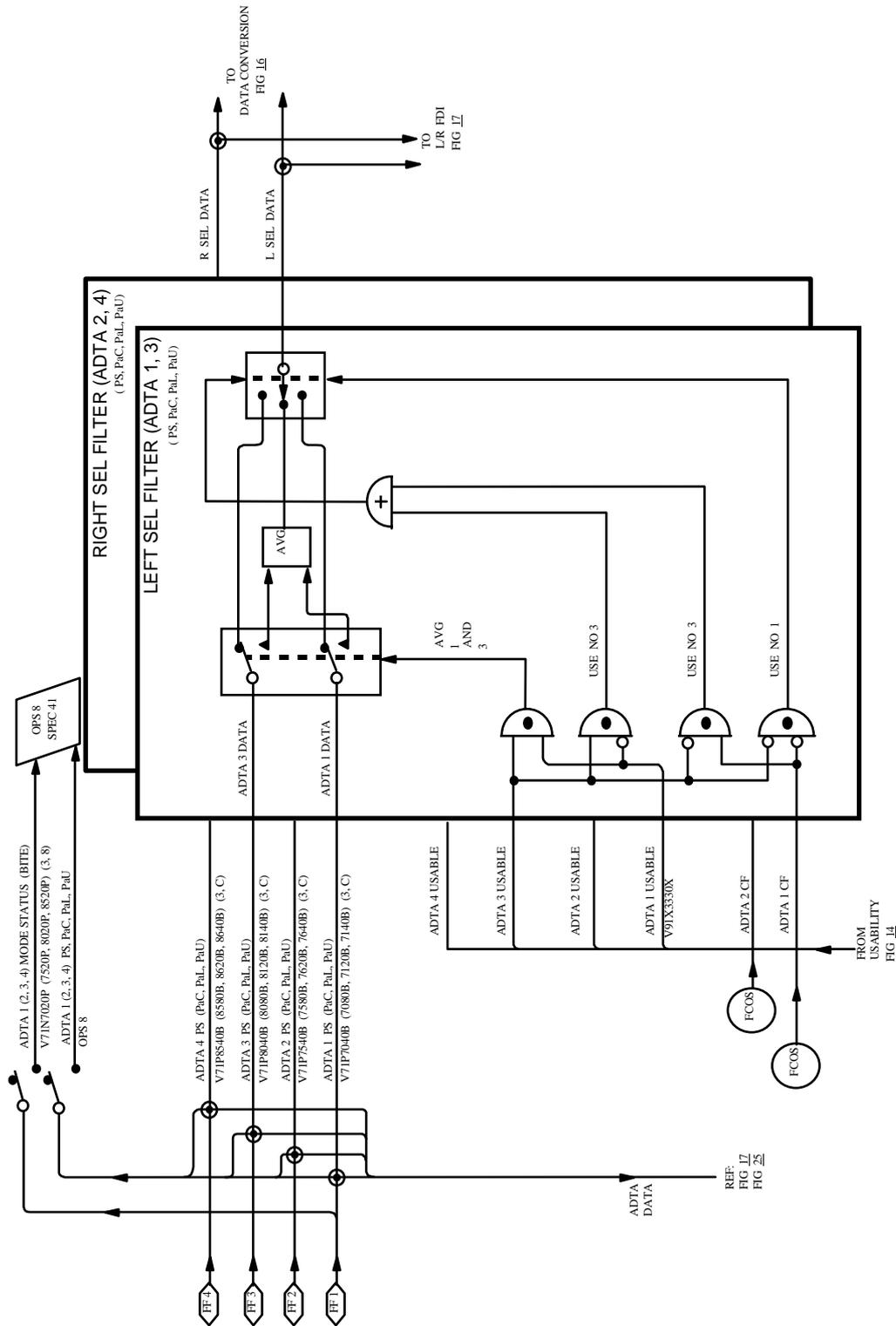


Figure 4 -15. - Left/Right Selection Filter

Figure 4 - 15 Left/Right Selection Filter

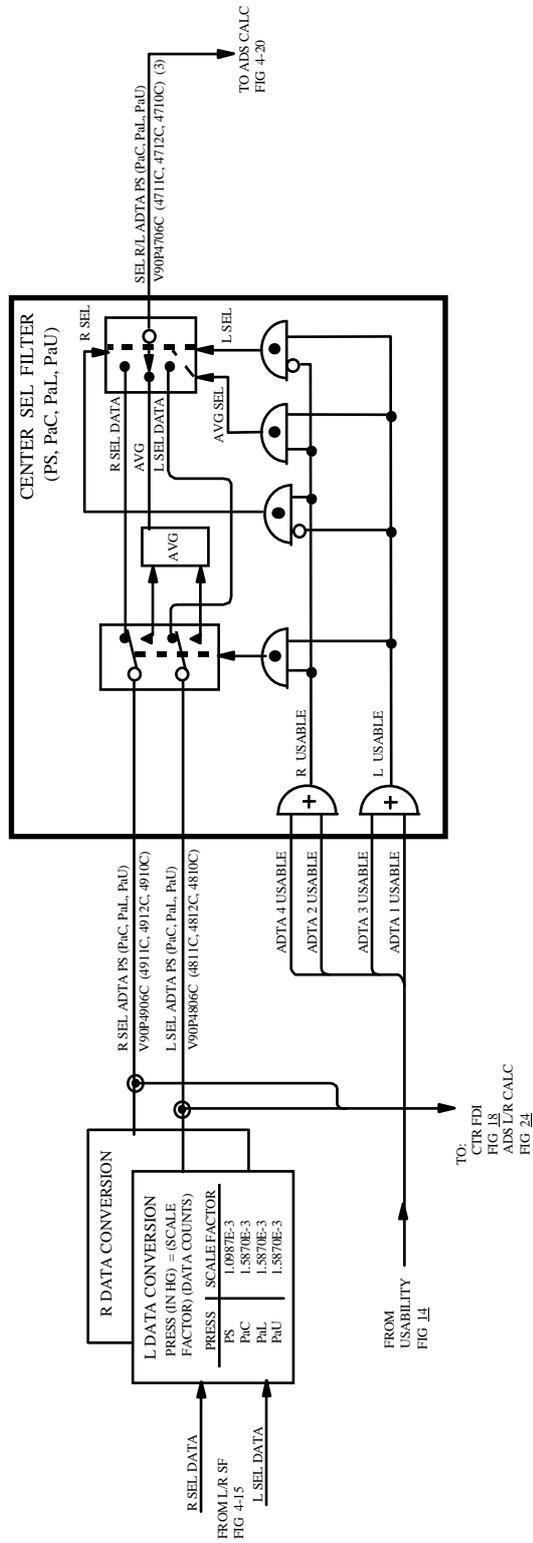


Figure 4-16. - Data conversion and center selection filter.

Figure 4 - 16 Data Conversion and Center Selection Filter

data will also be used by the center SF. Conversion is accomplished by multiplying each pressure (which is in counts) by the scale factor for that pressure. The general form of this conversion is:

$$C_LT_SEL_DATA_i = (LT_SEL_DATA_i) (ADTA SF_i)$$

$$C_RT_SEL_DATA_i = (RT_SEL_DATA_i) (ADTA SF_i)$$

$$\text{In. Hg} = (\text{counts}) \frac{\text{In. Hg}}{\text{count}}$$

i = pressures PS, PaC, PaL, PaU

4.2.3.5 CENTER SF

The center SF (fig. 4-16) nominally selects average data from left and right sides to remove the sideslip effects. If only one side is usable (L-ADTA 1 and 3 or R-ADTA 2 and 4), output is data from that side. If neither side is usable, an RM dilemma is declared by FDIR, and ADS output is switched to navigationally derived air data (NAVDAD). The output from the center SF is sent to the ADTA SOP for GNC parameter calculations and utilized by the center FDI in calculating the FDI thresholds.

4.2.3.6 ADTA FDIR

The purpose of ADTA FDIR is to determine which of the four pressures (PS, PaC, PaL, PaU) from each of the four ADTA's should be declared failed and to reconfigure the SF's accordingly. This process consists of performing a comparison of ADTA 1 and ADTA 3 pressures (C1), a comparison of ADTA 2 and ADTA 4 pressures (C2), and a comparison of the left and right pressures (C3). These comparisons are performed continuously. Additional comparisons of ADTA 1 and ADTA 3 and right side pressures (C4, C5), and ADTA 2 and ADTA 4 and left side pressures (C6, C7) are performed only if the C1 or C2 comparisons have exceeded the threshold three consecutive times. Comparators C4, 5, 6, 7 are used to isolate and identify the failed ADTA pressure. Declaring a pressure failed results in fault annunciation and FDIR downmoding (not usable) the entire ADTA. The crew controls the ADS RM via the select/deselect keyboard command capability. The ADTA FDI is shown in figure 4-17, left/right FDI, and in figure 4-18, center FDI.

4.2.3.6.1 TRANSIENT DETECTOR (DATA GOOD)

Before selected data can be output by RM and used by ADS SOP to compute GNC air data, a transient test must be performed to protect against transients in data going to GNC. This transient test is either the C3, C1, or C2 comparison test. If the transient test passes, then ADS SOP will compute air data for GNC in a normal manner. If the transient test fails, then ADS SOP will not update measured air data for GNC. The transient test does not apply to SOP air data computations for crew displays (AMI/AVVI) using selected left data and selected right data or the RM sensors SPEC page. If at least one ADTA on each side is usable, then compare left selected data with right selected data (the C3 cross-side comparison test) for the transient test. If both ADTA's on the left side (ADTA 1 and ADTA 3) are usable and neither

ADTA on the right side is usable, then compare ADTA 1 data with ADTA 3 data (the C1 comparison test) for the transient test. If both ADTA's on the right side (ADTA 2 and ADTA 4) are usable and neither ADTA on the left side is usable, then compare ADTA 2 data with ADTA 4 data (the C2 comparison test) for the transient test.

If any parameter fails the transient test, a data good (ADTA_DG) flag shall be set to the bad state. This ADTA_DG flag will cause ADS SOP to not update measured air data for GNC. The ADTA_DG flag will be reset to the good state when all parameters pass the transient test or when only one ADTA remains usable. If no ADTA's are usable, set the ADTA_DG flag to the bad state.

4.2.3.6.2 COMPARISON TESTS (FAIL/DILEMMA)

Data comparisons are performed for three purposes: transient protection, failure detection (requires a counter), and failure identification. Referring to figure 4-17, left/right FDI, and figure 4-18, center FDI, the comparators are described as follows.

a. C3 Cross-Side Comparator

The C3 comparator is performed when data are usable on both sides. It provides transient protection and detection of a side failure. The C3 comparison test is performed as follows.

IF: $|(Left\ Selected\ Data)_i - (Right\ Selected\ Data)_i| \geq (C3\ Threshold)_i$

THEN: Set DG flag to bad state.

Advance the C3 counter for the ith parameter.

ELSE: Reset the C3 counter for the ith parameter.

If, for any parameter, the C3 comparison test miscompares N=3 consecutive times, then the C3 test shall be deemed failed. The C3 failure status is used by FDI to determine if a dilemma situation has occurred.

b. C1 Left-Side Comparator

The C1 comparator is performed in FDI to detect failures between ADTA 1 and ADTA 3 on the left side. Also, if neither ADTA is usable on the right side and both ADTA's are usable on the left side, then C1 comparison tests will be performed for transient tests and dilemma tests. If C1 is used for the transient test, then its data good flag (L_DG) will be used to denote a transient test failure. The C1 comparison test is performed as follows.

IF: $|(ADTA\ 1\ Data)_i - (ADTA\ 3\ Data)_i| \geq (C1\ Threshold)_i$

THEN: Advance the C1 counter for the ith parameter.

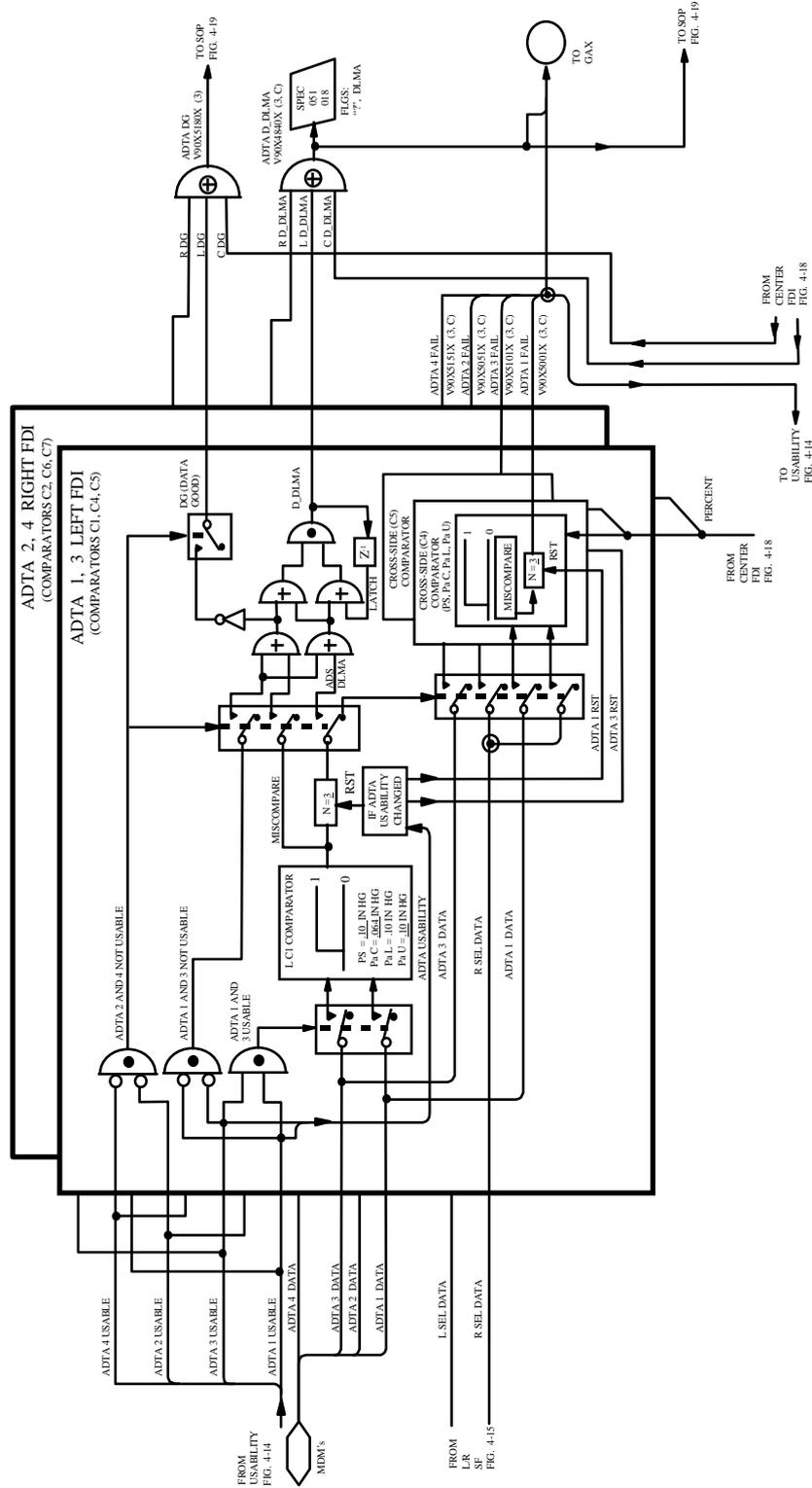


Figure 4-17. - Left/right FDI.

Figure 4 - 17 Left/Right FDI

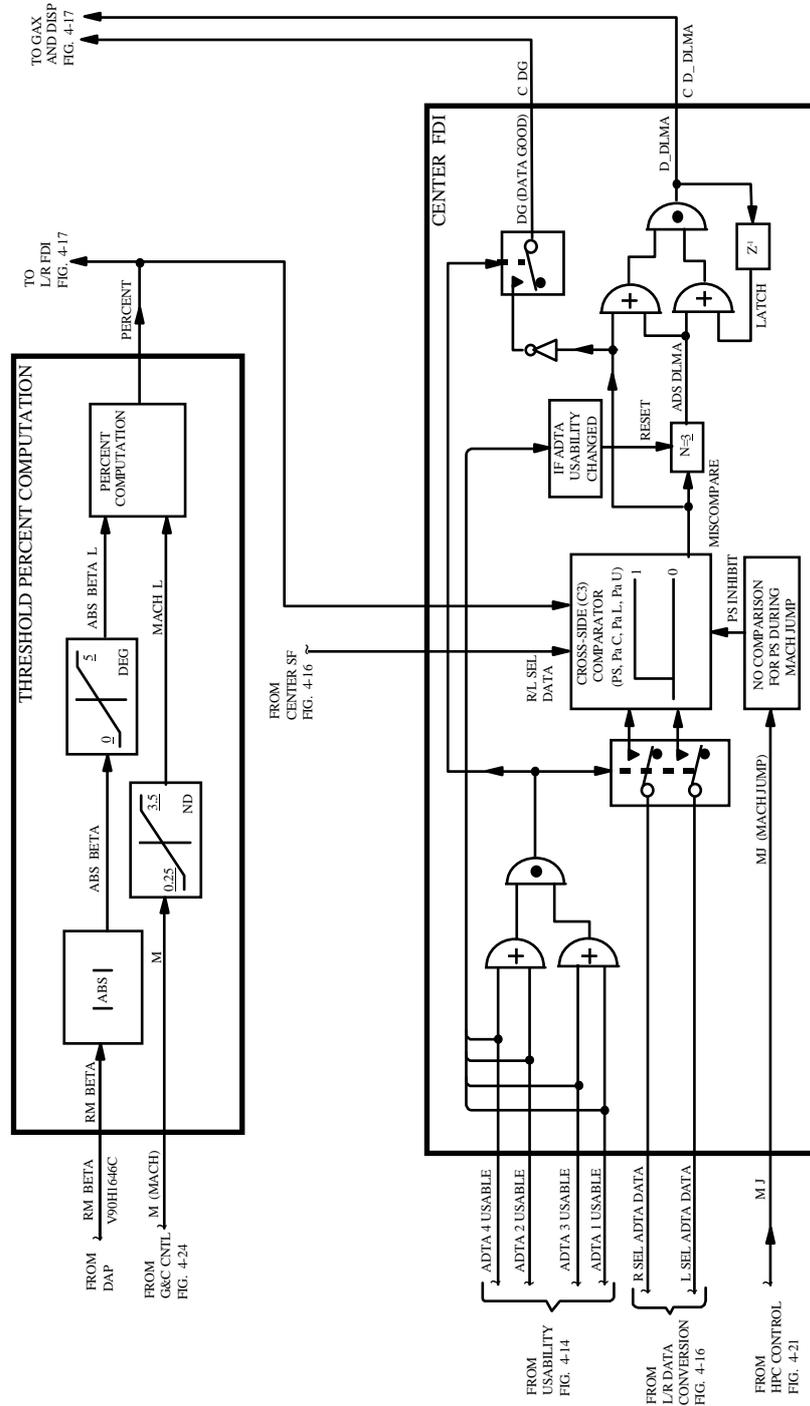


Figure 4-18. - Center FDI and threshold percent computation.

Figure 4 - 18 Center FDI and Threshold Percent Computation

Set DG_PRIME flag to bad state.

ELSE: Reset the C1 counter for the ith parameter.

If, for any parameter, the C1 comparison test mismatches N=3 consecutive times, then the C1 test shall be deemed failed. Once failed, the C1 test cannot be reset to the not-failed state until one or more of the ADTA's has its usability state changed (USE_j).

c. C2 Right-Side Comparator

The C2 comparator operates on the right side (ADTA 2 and ADTA 4) in exactly the same manner as C1 operates on the left side.

d. C4 and C5 Cross-Side Comparators

The C4 and C5 comparison tests are performed only when C1 has detected a parameter failure (after N=3 count) and at least one ADTA is usable on the right side. The C4 and C5 comparison tests are performed as follows.

IF: $|(ADTA\ 1\ Data)_i - (Right\ Selected\ Data)_i| \geq (Left\ Threshold)_i$

THEN: Deem C4 test mismatched.

IF: $|(ADTA\ 3\ Data)_i - (Right\ Selected\ Data)_i| \geq (Left\ Threshold)_i$

THEN: Deem C5 test mismatched.

IF: neither or both comparator test \geq threshold, test will be repeated on subsequent cycles.

e. C6 and C7 Cross-Side Comparators

The C6 and C7 comparison tests are performed only when C2 has detected a parameter failure (after N=3 count) and at least one ADTA is usable on the left side. The C6 and C7 comparison tests are performed as follows.

IF: $|(ADTA\ 2\ Data)_i - (Left\ Selected\ Data)_i| \geq (Right\ Threshold)_i$

THEN: Deem C6 test mismatched.

IF: $|(ADTA\ 4\ Data)_i - (Left\ Selected\ Data)_i| \geq (Right\ Threshold)_i$

THEN: Deem C7 test mismatched.

IF: Neither or both comparator test \geq threshold, test will be repeated on subsequent cycles.

f. Comparison Test Counter Reset

Any time an ADTA's usability status (USE_j) has changed, then all comparison test counters (C1, C2, C3) shall be reset to zero.

g. Dilemma

A dilemma is defined as a detected fault that has no possibility of being identified because only two sources of data are available and they disagree. The conditions for which a dilemma will be declared are summarized as follows.

- (1) ADTA's are usable on both sides.
Detected failure exists between sides.
- (2) No ADTA's are usable on one side.
Detected failure exists between ADTA's on the remaining side.
- (3) No ADTA's are usable on either side.
Automatically declare a dilemma and set DG false.

The logic is given below.

Usable ADTA's

| <u>Left side</u> | <u>Right side</u> | <u>Dilemma logic</u> |
|------------------|-------------------|---------------------------------|
| Both ADTA's | Both ADTA's | DILEMMA = C3 test failed |
| Both ADTA's | Only one ADTA | DILEMMA = C3 test failed |
| Only one ADTA | Both ADTA's | DILEMMA = C3 test failed |
| Only one ADTA | Only one ADTA | DILEMMA = C3 test failed |
| Both ADTA's | No ADTA's | DILEMMA = C1 test failed |
| No ADTA's | Both ADTA's | DILEMMA = C2 test failed |
| No ADTA's | No ADTA's | Declare a dilemma; set DG false |

Exception: If in the Mach-jump (MJ) region, do not declare a dilemma if static pressure fails the C3 test.

A dilemma will always be treated the same no matter if detected by left-side, right-side, or cross-side comparators.

If a dilemma is declared, the following shall occur.

- (1) Set the delayed dilemma flag D_DILEMMA to true. This will cause ADS SOP to invoke the DEFAULT ADS or NAV-derived system as long as the D_DILEMMA flag is true. This will also be used to alert the crew, via CRT display and C&W, that an ADS dilemma exists.
- (2) Set the ADS_DILEMMA flag to true.

Once a dilemma has been declared and the dilemma flags set, they shall stay set until the following occurs. When any ADTA is SELECTED or DESELECTED by the crew, CF cleared, or ADTA declared failed, then the ADS_DILEMMA flag shall be reset to the no-dilemma state. In addition, if the probe deploy has changed states, then the ADS_DILEMMA flag shall be reset to the no-dilemma state. The D_DILEMMA flag reset shall be delayed until both the ADS_DILEMMA is reset and the appropriate transient test has been performed and the ADTA_DG flag set to the good state.

The comparison test threshold computations are shown in table 4-IV.

4.3 ADTA SOP

The purpose of the ADTA SOP is to calculate or throughput air data parameters for GNC, dedicated displays, ADTA RM, and the OVERRIDE display. A functional block diagram of the ADTA SOP and its related interfaces is shown in figure 4-19. The ADTA SOP is active in MM's 304, 305, 602, and 603. The SOP receives left, right, and center SF pressures in addition to a status indicating probe deployed from RM. RM also supplies DG and dilemma indications which are utilized by the SOP to mode its final set of air data parameters for GNC. Pressures associated with each ADTA are processed by the SOP to generate H, a, and M for display on the OVERRIDE display. During periods when probe-sensed data cannot be utilized, the SOP throughputs air data parameters based on NAV data. Crew control of the SOP is provided via item entries from the horizontal situation display (HSD). The HYDRAULIC SOP provides an indication of when the nose gear is down; while GNC SW RM provides indications of switch requests for dedicated displays. Finally, it is important to note that the SOP provides Mach number (M) and an indication of when the probes are in the MJ region to ADTA RM. These are utilized by RM in performing various FDIR functions for the ADTA's.

4.3.1 ADS CALCULATION INITIATION

Calculation of ADTA ADS air data parameters are initiated whenever the following conditions or flags exist.

- a. Either probe deploy flag set (fig. 4-14)
- b. ADTA delay dilemma flag not set (fig. 4-17)
- c. ADTA DG flag set (fig. 4-17)

Table 4 - 1 ADTA FDI Comparator Thresholds

| Comparator | Equations for comparator thresholds |
|---|---|
| C1 = C2 Same side pressure comparisons | C1 = C2 PS = COUNTS = 0.10 in. Hg PaC = COUNTS = 0.064 in. Hg PaL = COUNTS = 0.10 in. Hg PaU = COUNTS = 0.10 in. Hg |
| C3 (In. Hg) (Left side vs. Right side) | C3_THRESH_PS = (PERCENT _{PS}) (center selected PS) C3_THRESH_PAC = (PERCENT _{PAC}) (center selected PAC) C3_THRESH_PAL = (PERCENT _{PAL}) (center selected PAL) C3_THRESH_PAU = (PERCENT _{PAU}) (center selected PAU) |
| C4 AND C5 (counts) (Left ADTA vs. Right side) | LT_THRESH_PS = (PERCENT _{PS}) (right selected PS) LT_THRESH_PAC = (PERCENT _{PAC}) (right selected PAC) LT_THRESH_PAL = (PERCENT _{PAL}) (right selected PAL) LT_THRESH_PAU = (PERCENT _{PAU}) (right selected PAU) |
| C6 AND C7 (counts) (Right ADTA vs. Left side) | RT_THRESH_PS = (PERCENT _{PS}) (left selected PS) RT_THRESH_PAC = (PERCENT _{PAC}) (left selected PAC) RT_THRESH_PAL = (PERCENT _{PAL}) (left selected PAL) RT_THRESH_PAU = (PERCENT _{PAU}) (left selected PAU) |

Note: Threshold percent computations (fig. 4-18) vary with Mach and estimated sideslip (RM beta). The percent also varies with the parameter being tested. No C3 comparison test is performed on PS during the MJ region from 1.6 to 1.1 Mach, due to unstable air pressure at that time.

Transition from NAVDAD to ADTA ADS air data is performed via the HSD (SPEC 50). The HSD auto/inhibit/force function is configured in the inhibit mode (INH) until the crew deploys the ADP's and evaluates the ADTA ADS system operation. When in AUT position, the auto transition to ADTA ADS occurs when either or both probes are deployed, no data transients (DG set) or dilemma conditions exist, and relative velocity is less than 2.5K ft/s (V_{ADS}, I-LOAD). The ADS USE FLG is set to 1 when ADTA ADS is to be used by Guidance and Control (G&C), and set to 0 when NAVDAD data are to be used. Once the velocity limit (<2.5K ft/s) has been achieved, this velocity check is bypassed and ADS USE FLG latched on, thus preventing subsequent velocity glitches from transitioning from ADTA ADS back to NAVDAD while in auto transition.

In OPS 6 only, an additional test is made before automatically transitioning to ADS from NAVDAD. If, in a contingency abort, the corrected pressure altitude (HPC) in the ADS calculations has been limited at the upper end of its range (105K ft), then the vehicle is above the region of air data calibration, and ADTA ADS outputs are assumed to have degraded. While this excess altitude condition is true, the ADS air data are automatically downmoded to NAVDAD.

Subsequently, the crew may inhibit ADS air data and recall NAVDAD via the keyboard entry (INH). Also, with the keyboard entry (FOR), he may force the data in lieu of NAVDAD. In the event of an ADS RM dilemma condition in either the AUTO or FORCE states, the ADS air data are automatically downmoded (ADS USE FLG set false) to NAVDAD data, or downmoded to default (a, Q) and NAVDAD data if relative velocity (V_r) is less than 1.5K ft/s. This allows G&C to run on NAVDAD while crew resolves the ADTA dilemma.

For relative velocity (V_r) less than 1.5K ft/s (V-FREEZE, I-load), and ADTA DG false with D_DLMA flag not set, the ADS CALC flag will be set false and ADS USE flag will remain set on (fig. 4-20). Under these conditions, G&C will continue to use the last valid data from the ADTA ADS calculations (fig. 4-21).

4.3.2 ADTA ADS CALCULATIONS

The ADS air data parameters refer to those parameters which are calculated based on the center SF (fig. 4-16) outputs (usually reflecting the average of the left and right selected ADTA values). The pressure inputs to ADS calculations are SEL R/L ADTA PS, PaC, PaL, and PaU.

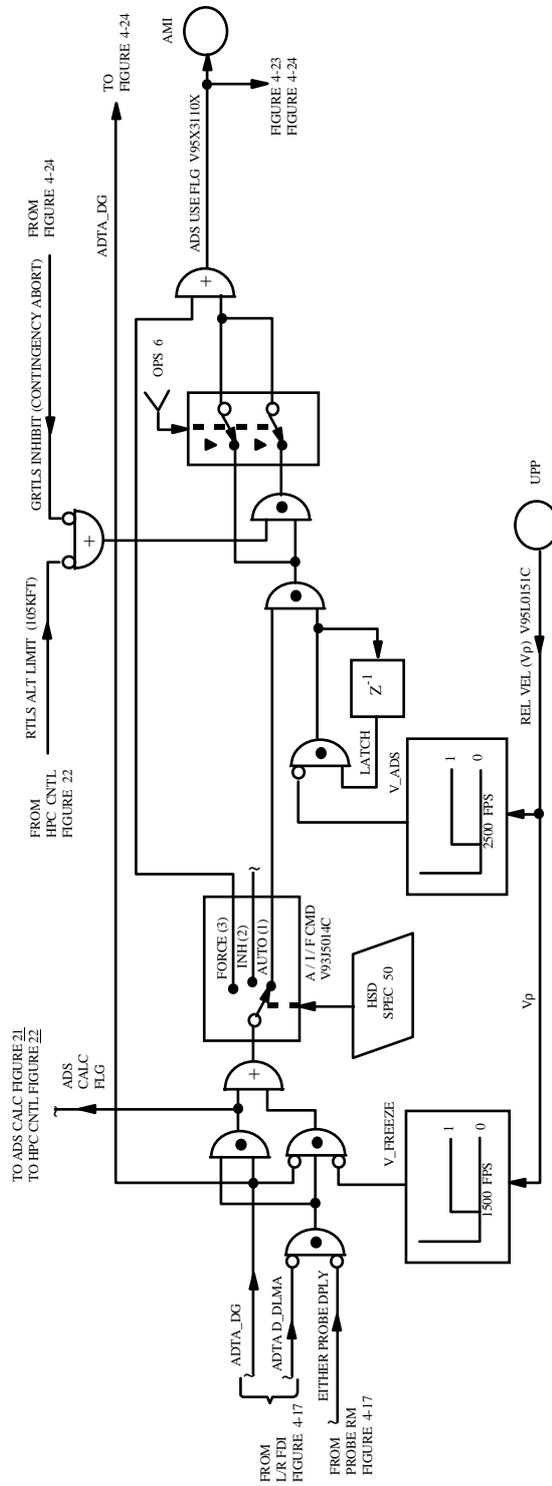


Figure 4-20. - ADS calculation and use control

Figure 4 - 20 ADS Calculation and Use Control

Calculation of ADS air data parameters is initiated whenever ADS CALC flag is set. This flag is set if either probe deploy flag is set, ADTA D_DLMA flag not set, and ADTA_DG flag is set (fig. 4-20). The ADS calculations require about four computational cycles (4 sec) to converge to steady state values. The ADS calibration data is not accurate during rollout but has been included for crew convenience.

The ADTA ADS calculations generate the following parameters.

- a. Angle of attack (α_A) - Guidance & control
- b. Corrected pressure altitude (HPC) - Navigation
- c. Mach number (M_A) - Guidance & control
- d. Dynamic pressure (Q_A) - Guidance & control
- e. Equivalent airspeed (EAS_A) - Guidance
- f. True airspeed (TAS_A) - Guidance & control
- g. ADS MJ flag - Navigation
- h. ADTA ADS calculation time (THPC) - Navigation

The ADTA ADS calculation functional block diagram depicting the relationships between the subfunction modules is shown in figure 4-21. A brief operational description of each subfunction is tabulated in table 4-V.

4.3.3 ADS HPC CONTROL

If ADTA ADS calculations are not being performed (ADS CALC flag off), the corrected pressure altitude (HPC) and time (THPC) will be the last valid data in COMPOOL and ALT DG, and ADS MJ flags will be set off. The following operating states can exist with ADS CALC flag on. Nominal operation is with THPC and HPC updated to NAV every sec, ALT DG flag on, and ADS MJ flag off. During MJ (1.6 to 1.1 Mach), ADS MJ flag will be sent to NAV to inhibit the use of HPC in updating the NAV-derived altitude. The ADS MJ flag is sent to ADTA RM (fig. 4-18) to disable C3 tracking test on static pressure (PS).

If in OPS 6 (RTLS) and the ADTA calculated HPC is equal to its limit of 105,000 ft, the ALT DG flag will be set false and NAV will cease using HPC. In RTLS, the MACH and ALT could be such that the ADP is deployed above 105,000 ft.

4.3.4 NAVDAD SOP OVERVIEW

The purpose of the NAVDAD SOP is to calculate certain air data parameters based on NAV-supplied data. The NAVDAD parameters provided by the NAVDAD SOP are as follows.

- Dynamic pressure (Q_{BAR} , \bar{Q}_N)
- Equivalent airspeed (EAS_N)
- Mach number (M_N)

These parameters are sent to the ADTA SOP to provide a central location for transitioning between NAVDAD and ADS air data parameters. Figure 4-23 depicts a functional block diagram of the NAVDAD SOP.

It should be noted that the NAV-derived angle of attack (a_N) from attitude processing, and ground relative velocity magnitude (V_r) from the user parameter processing (UPP) functions are sent directly to the ADTA SOP for transitioning. The remaining NAVDAD parameters (altitude-H, altitude rate-H, drag acceleration- A_D , and sideslip angle-b) are calculated by navigation, attitude processing, or UPP.

During periods when NAV-derived data set is invoked for G&C, NAV could be updated with HPC based on probe-sensed pressures. This can occur if the probe is deployed prior to $M = 2.5$, and valid ADTA data exist (ADS CALC flag on).

4.3.4.1 NAVDAD SOP DETAILED DESCRIPTION

The NAVDAD are needed during entry to drive G&C and the dedicated displays. After the crew has manually transferred to ADS air data, the NAVDAD are used as a backup source. The details required to compute the NAVDAD parameters are discussed in the subsequent paragraphs and shown in figure 4-23.

a. DYPRS_N - NAV Derived Dynamic Pressure - Q_N output

For relative velocity (V_r) magnitude equal to or greater than V_{NAVDAD} (1400 ft/s), the Q_N output is computed as follows.

Drag coefficient (C_D) will be computed as a function of V_r , NAV-derived angle of attack (a_{ND}), and constants KCD1 through KCD6.

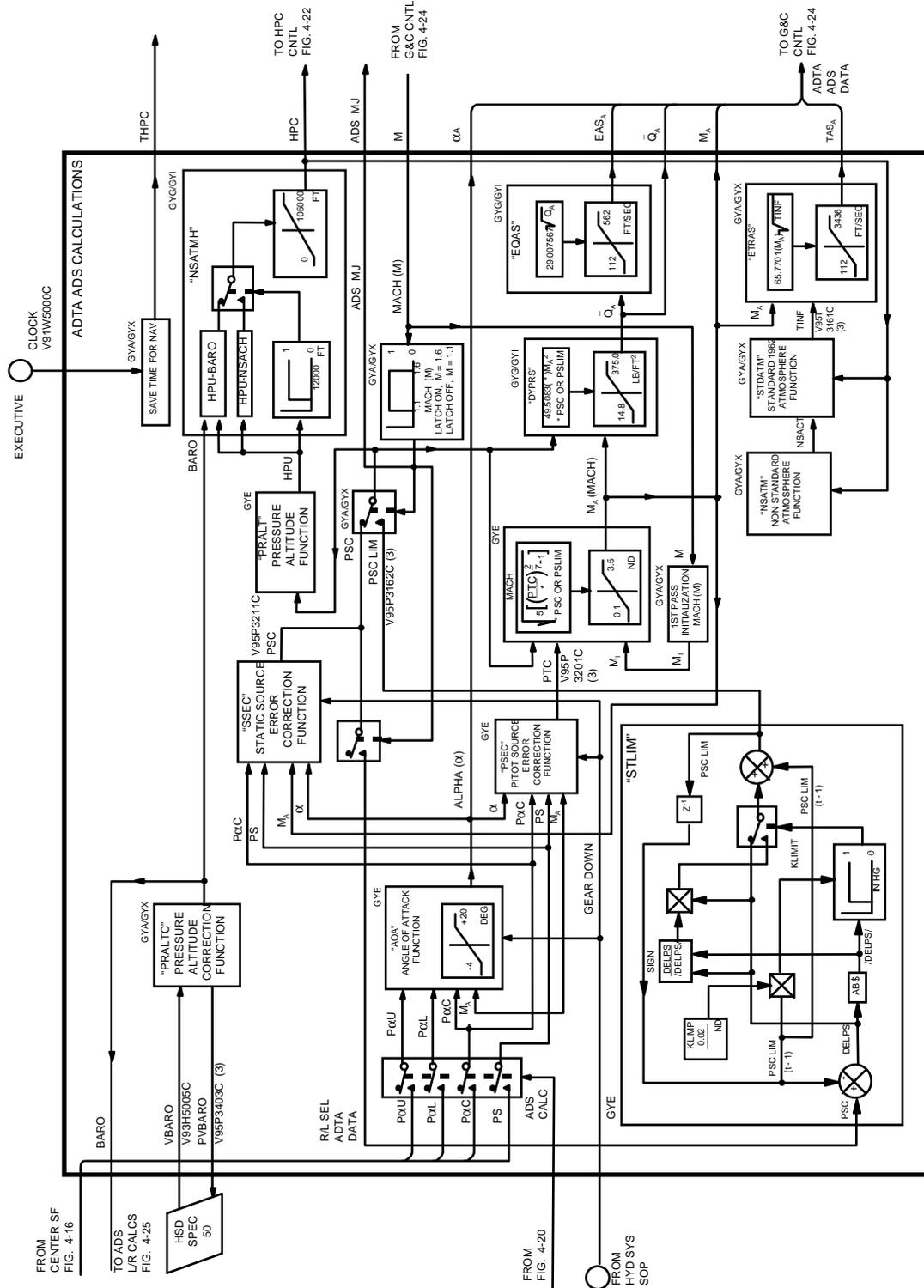


Figure 4-21.- ADTA ADS calculation.

Figure 4 - 21 ADTA ADS Calculation

TABLE 4-V.- ADS CALCULATION SUBFUNCTION MODULES

AOA - Angle of Attack - Alpha (α_A) output - Mach
PaC,
interpolate
is

Determine Mach range index and Mach ratio based on Mach number and boundaries. Compute alpha pressure ratio using angle of attack pressure PaL, and PaU. Determine alpha at Mach boundaries and linearly the alpha basic term. If required, adjust alpha for nose gear down air disturbance corrections. Alpha output is limited from -4° to $+20^\circ$. Module required for ADS, left/right, and ADTA RM air data calculations. Alpha is initialized at 12° .

SSEC Static Source Error Correction - PSC output - Mach
Mach
using the
0.2

Determine Mach range index and Mach ratio based on Mach number and boundaries. With alpha, determine the static pressure coefficients at boundaries and linearly interpolate the static pressure basic coefficient. If required, adjust static pressure basic coefficient for nose gear down air disturbances. The corrected static pressure (PSC) is then calculated static pressure coefficient, PS, and PaC. PSC output is then limited from to 36 in. Hg. Module is required for ADS, left/right, and ADTA RM air data calculations. PSC is initialized at 36 in. Hg.

PSEC - Pitot Source Error Correction - PTC output - Mach
the
0.2 to

Determine Mach range index and Mach ratio based on Mach number and boundaries. With alpha, determine the total pressure coefficient at Mach boundaries and linearly interpolate the total pressure basic coefficient. If required, adjust total pressure basic coefficient for nose gear down air disturbances. The corrected total pressure (PTC) is then calculated using total pressure coefficient, PS, and PaC. PTC output is then limited from 52 in. Hg. Module is required for ADS, left/right, and ADTA RM air data calculations. PTC is initialized at 52 in. Hg.

MACH - Mach Number - MA output - as a
pressure
gas) is

The purpose of this subfunction is to calculate free-stream Mach number function of the PTC and either limited or corrected free-stream static depending on whether or not the vehicle is in the Mach jump region. The implicit equation for Mach (applicable for the isentropic flow of a perfect gas) is shown below.

$$M = \sqrt{5 \left(\frac{PTC}{*} \right)^{\frac{2}{7}} - 1} \quad * = \text{PSC or PSLIM}$$

(M_A)

Module is required for ADS, left/right, and ADTA RM air data calculations. Mach is initialized at NAV value of 25 for OPS 3, and 6 for OPS-6. Mach output is limited from 0.1 to 3.5.

PRALT - Pressure Altitude - HPU output -

This subfunction calculates uncorrected pressure altitude (HPU) using the polynomial form of equations. Three polynomial equations (using PSC or PSLIM and constants) cover the following ranges of pressure.
TABLE 4-V.- Continued

36 \geq PSC or PSLIM \geq 6.4480 in. Hg
1.5850 \leq PSC or PSLIM $<$ 6.4480 in. Hg
0.2 \leq PSC or PSLIM $<$ 1.5850 in. Hg

Module is required for ADS, left/right, and ADTA RM air data calculations.

PRALTC - Pressure Altitude Correction -
using
BARO output -

A barometric correction term (VBARO, IN HG) is entered by the keyboard

HSD SPEC 51. VBARO parameter is initialized at 29.92 in. Hg. The baro
correction term (BARO) accounts for variations in the base level pressure

at

mean sea level and is computed only if the correction input has changed.

ADS

computed baro is limited between -1000 and +8000 ft. Module is required

for

ADS and left/right air data calculations.

TABLE 4-V.- Continued

| | |
|--|--|
| <p>NSATMH - Nonstandard Atmosphere Correction for adjustment Altitude - HPC output - altitudes subtracted HPU to HPC is</p> | <p>If altitude (HPU) is above 12,000 ft, the correction term NSACH (an to 1962 standard atmosphere for changes in atmosphere for different at the landing site and for seasonal changes) shall be computed and from HPU to generate a corrected pressure altitude (HPC). NSACH is calculated for altitudes from 0 to 105,000 ft.</p> <p>If altitude (HPU) is below 12,000 ft, the correction term BARO (the actual measured landing site pressure) shall be computed and subtracted from generate HPC.</p> <p>The ADS computed HPC output is then limited from 0 to 105,000 feet. initialized at 90,000 ft. Module is required for ADS and left/right air data calculations.</p> <p>The ADS HPC is time tagged (THPC) for use by navigation.</p> |
| <p>NSATM - Nonstandard Atmosphere Corrections for of Temperature - NSACT output -</p> | <p>This subfunction computes the correction term for improving the estimate free-stream static air temperature (NSACT). The term compensates for expected temperature changes at different altitudes for a specific mission launch time (seasonal) and landing site. The correction term is in six continuous altitude segments in the following form.</p> $\text{NSACT} = \text{TSLOPE (HPC)} + \text{TBIAS}$ <p>Module is required for ADS air data calculations.</p> |
| <p>STDATM - U.S. Standard Atmosphere 1962 Model - function TINF output - for</p> | <p>This subfunction computes free-stream static temperature (TINF) as a of HPC, correction term NSACT, and constants. Three sets of equations TINF are established with the following pressure ranges.</p> $\begin{aligned} 105,000 &\geq \text{HPC} \geq 65,617 \text{ ft} \\ 36,089 &\leq \text{HPC} < 65,617 \text{ ft} \\ 0 &\leq \text{HPC} < 36,089 \text{ ft} \end{aligned}$ <p>Module is required for ADS air data calculations.</p> |
| <p>ETRAS - Estimated True Airspeed - TAS_A output - Mach ÖTINF. required for</p> | <p>This subfunction computes an estimate of true airspeed as a function of and estimated static air temperature.</p> <p>The equation for estimated true airspeed shall be $\text{TAS} = 65.7701 (M)$</p> <p>The computed TAS is limited from 112 to 3436 ft/s. The module is ADS air data calculations. TAS is initialized at 2800 ft/s.</p> |
| <p>DYPRS - Dynamic Pressure - Q_A(QBAR) output - number to 1.1</p> | <p>This subfunction computes dynamic pressure as a function of Mach and either limited (PSLIM) or corrected (HPC) free-stream static pressure depending on whether or not the vehicle is in the Mach jump region (1.6 Mach). The equation for calculating dynamic pressure (QBAR) is:</p> $49.5083 (*) M^2 * = \text{PSC or PSLIM}$ |

The computed $Q_{BAR}(Q_A)$ is limited from 14.8 to 375.0 lb/ft². Q_{BAR} is initialized to .01 for OPS 3 and to 6 for OPS 6. The module is required for ADS and left/ right air data calculations.

TABLE 4-V.- Concluded

| | |
|--|--|
| <p>EQAS - Equivalent Airspeed - EAS_A output - data</p> | <p>This subfunction computes equivalent airspeed as a function of dynamic pressure.</p> $EAS = 29.007567 \sqrt{\bar{Q}} \text{ QBAR}$ <p>The computed EAS is limited from 112 to 562 ft/s. The EAS parameter is initialized to 430 ft/s. This module is required for ADS and left/right air calculations.</p> |
| <p>MJ - ADTA ADS Mach Jump - MJ flag output - the in the sensitive output to that eliminate</p> | <p>The Mach jump region is predicted to occur during $1.6 \geq M \geq 1.1$. When ADP's approach sonic conditions (Mach jump region) large errors result sensed static pressure (PS). Alpha calculation (module AOA) is not to errors in PS. During Mach jump time, corrected static pressure (PSC) is replaced with corrected static pressure limiter output (PSCLIM) modules PRALT (HPU), MACH (M), and DYPRS (Q) as shown in figure 4-21. The MJ flag is sent to NAV (fig. 4-22) and ADTA RM-FDI users the MJ state exists.</p> <p>The MJ flag is latched on at 1.6 Mach, and latched off at 1.1 Mach to any cycling during the transition.</p> |
| <p>STLIM - Static Pressure Limiter - PSCLIM output - during output module is</p> | <p>The purpose of this subfunction module is to limit the change in PSC the Mach jump region of 1.6 to 1.1 Mach. On the first pass through the module, the PSCLIM and PSCLIM (t-1) are set equal to PSC. On all subsequent passes, the STLIM module never allows any change in value greater than .02 of the previous PSCLIM output data. This required for ADS and left/right air data calculations.</p> |
| <p>PARAT - Pressure Altitude Rate - H_L, H_R output - AIR</p> | <p>The pressure altitude rate (HDOT, H) is computed by differentiating HPU using a cascade of two first order lag filters. This subfunction module is processed by the ADTA ADS LEFT/RIGHT CALC whenever the crew's DATA source switch is in the LEFT or RIGHT position. The ADTA calculation of H_L and H_R is initialized to NAV Altitude rate (H_N).</p> |

Next, dynamic pressure (\bar{Q}_N) is computed as a function of drag coefficient (C_D), drag acceleration (A_D), vehicle mass (M_V), and vehicle reference area (REF-AREA).

$$\bar{Q}_N = \frac{(A_D)(M_V)}{C_D (REF-AREA)} = \frac{(A_D)(M_{10})}{C_D (2690) \text{ ft}^2}$$

MV: OPS-3 = VARIABLE FROM GUID
OPS-6 = Permission value

For relative velocity (V_r) magnitude less than 1400 ft/s, the Q_N output is computed as follows.

The air density (r_N) computation is as follows.

After the altitude layer is determined by vehicle altitude (H_N), the altitude lapse rate and temperature constants are determined. Air density is then calculated using altitude lapse rate constants, temperature constants, gravity divided by universal gas constant of air, altitude (H_N), and static air temperature (T_{SN}).

The static air temperature (T_{SN}) computation is as follows.

The atmosphere is modeled into two altitude layers with the boundary for OPS-6 at 50,199 ft and for OPS-3 at 36,089 ft. After the layer is determined by the vehicle altitude (H_N), the constants (altitude lapse rate and temperature) describing the layer are used to compute static air temperature (T_{SN}).

Then \bar{Q}_N is computed as follows.

$$\bar{Q}_N = 0.5 \rho_N (V \rho^2) \frac{\text{lb}}{\text{ft}^2}$$

b. MACH_N - NAV-derived Mach Number - M_N output

The MACH_N subfunction computes the Mach number (M_N) as a function of NAV supplied altitude (H_N) and relative velocity (V_r).

For relative velocity (V_r) equal to or greater than V_{NAVDAD} (1400 ft/s),

$$M_N = 0.001 (V_r)$$

For (V_r) less than 1400 ft/s, M_N is computed as a function of static air temperature (T_{SN}) and ground relative velocity (V_r).

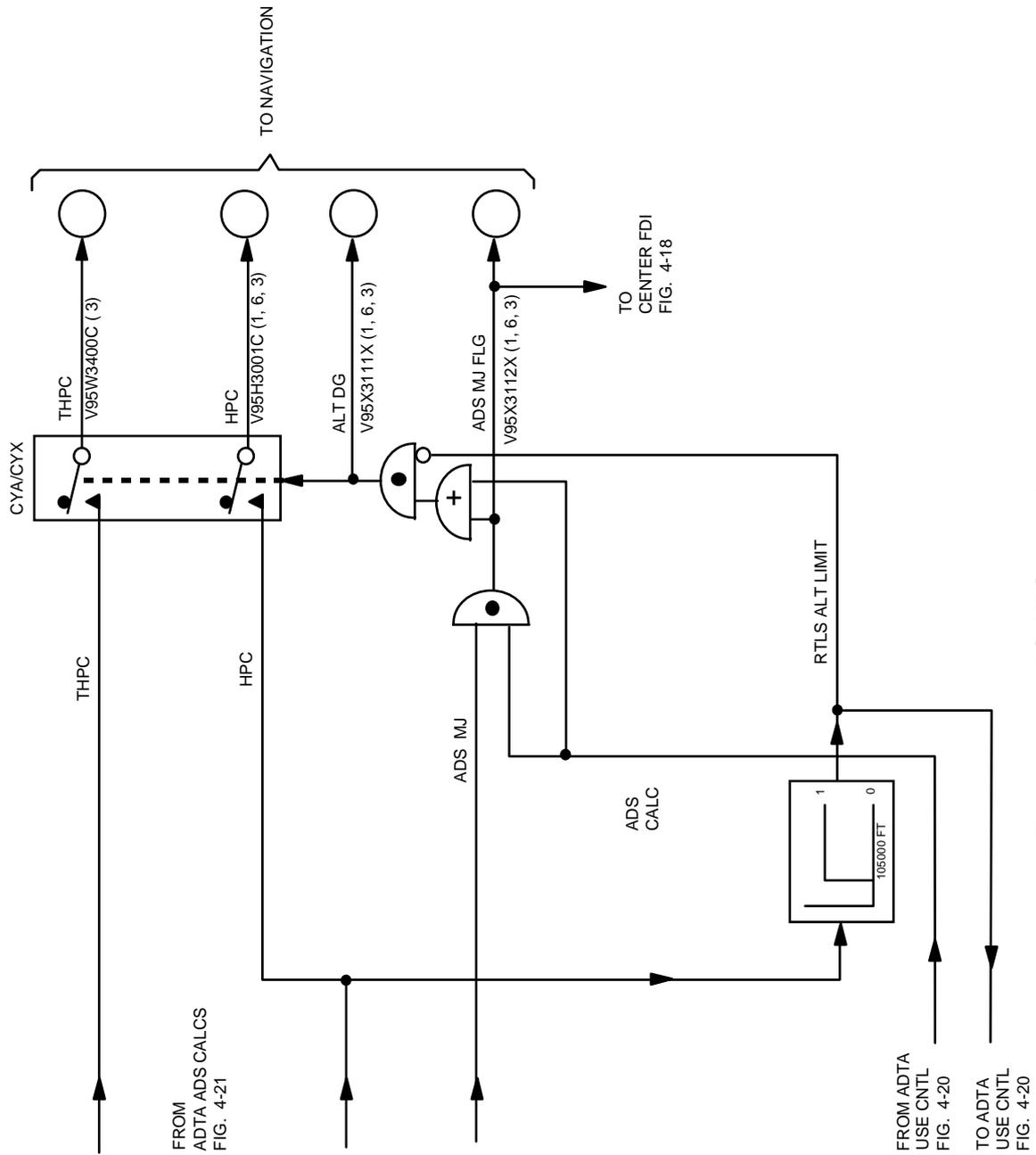


Figure 4 - 22 ADS HPC Control

Figure 4-22.- ADS HPC control.

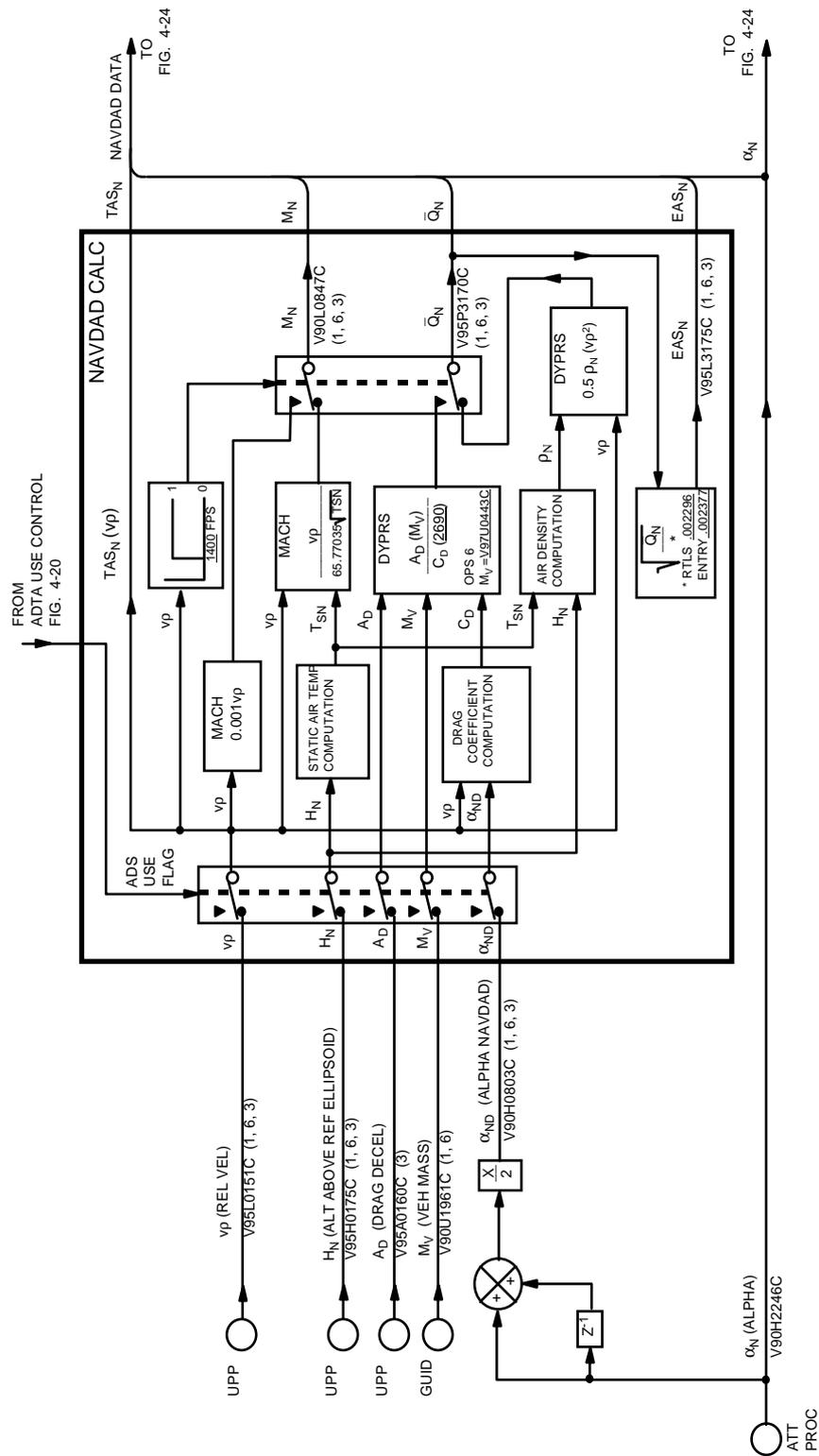


Figure 4-23.- NAVDAD calculations.

Figure 4 - 23 NAVDAD Calculations

$$M_N = \frac{V\rho}{SPD_SND_CON \text{ rootTSN}}$$

$$SPD_SND_CON = 65.77035$$

- c. EQAS_N-NAV Derived Equivalent Airspeed - EAS_N output

The EQAS_N subfunction computes the equivalent airspeed (EAS_N) as a function of NAV-derived dynamic pressure (Q_N). The EAS_N is computed as follows.

- d. TAS - true airspeed - TAS_N output

The true airspeed (TAS_N) is set equal to the ground relative velocity (V_r) magnitude from the UPP and is sent to the ADTA SOP for transitioning to G&C.

- e. ALPHA - Angle of Attack - a_N output

The NAV-derived angle of attack (a_N) from attitude processing is sent directly to the ADTA SOP for transitioning to G&C.

4.3.4.2 NAVAIDS/DEFAULT AIR DATA TRANSITION

The transitions between NAVDAD, ADS, and DEFAULT parameters are shown in figure 4-24. Transitions between NAVDAD and ADS air data are controlled by the crew from the HSD. The ADS USE flag set true will cause ADS (ADTA) data to be sent to G&C. With ADS USE flag on and ADTA DG flag off below V_r of 1500 ft/s, the last valid ADTA data will be used by G&C. In the event of an RM dilemma condition (ADS USE FLG false) in either the auto or force state, the ADS air data is downmoded to NAVDAD parameters above 1500 ft/s, and NAVDAD and default Q (Q_{DF}) and alpha (a_{DF}) when below 1500 ft/s. In OPS 6 only, automatic transition from NAVDAD to default parameters (Q_{DF} and a_{DF}) is inhibited for contingency abort. When angle of attack exceeds ADS alpha (a_A) upper limit of 20° with ADTA DG flag set, the angle of attack will transition to NAV-derived alpha (a_N).

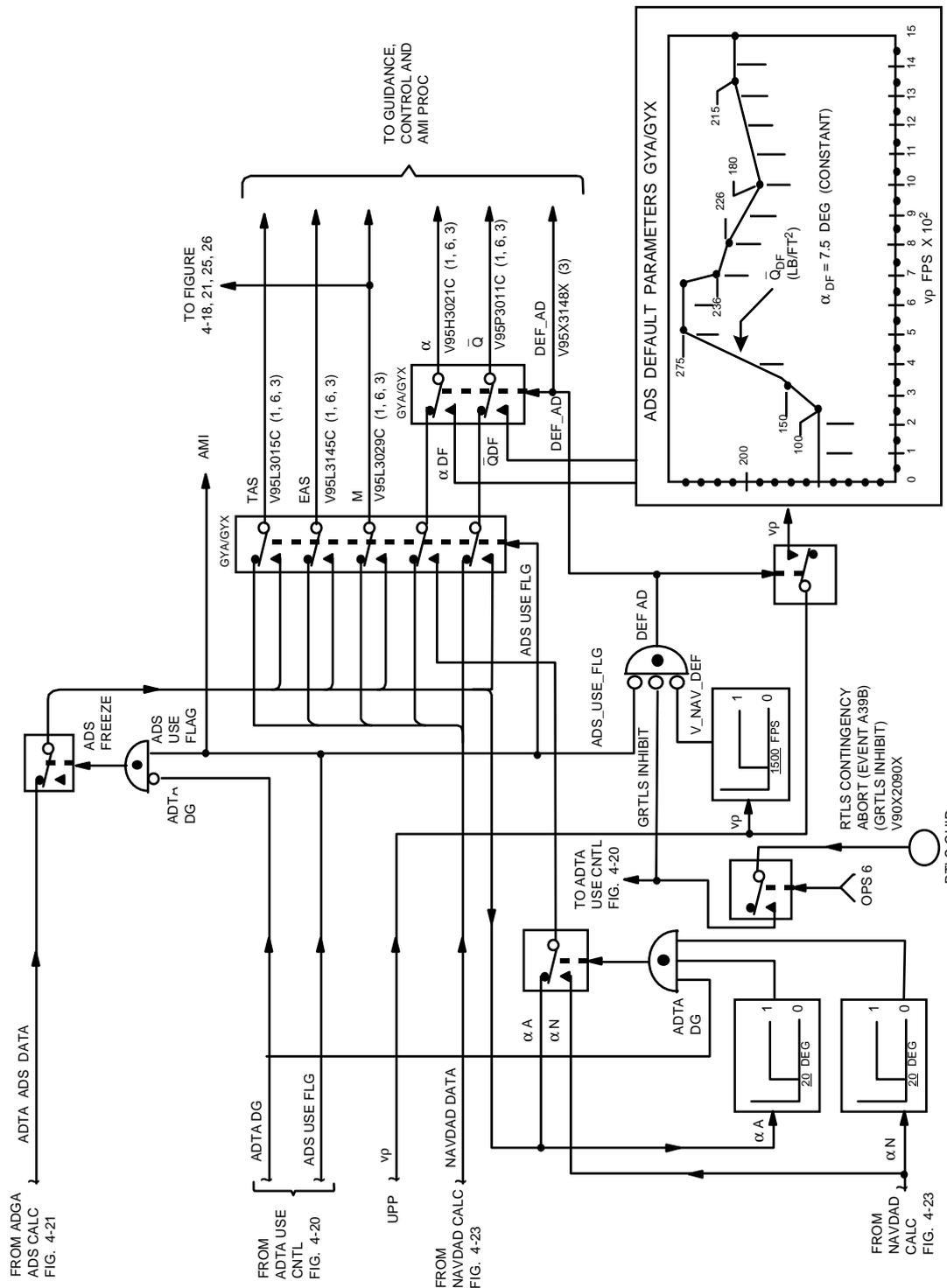


Figure 4-24.- G&C ADS transition and control

Figure 4 - 24 G&C ADS Transition and Control

4.3.4.3 LEFT/RIGHT AIR DATA CALCULATIONS FOR AVVI/AMI

The calculations are initiated by the CDR or PLT selecting the LEFT or RIGHT AIR DATA on panel F6 or F8. If ADP's are deployed and ADTA's not COMM faulted, the following data will be calculated from the left or right selected pressures and displayed on the AMI and AVVI.

| | |
|---|------------|
| Air data probe valid flag (ADP VALID _{L/R}) | AMI & AVVI |
| Angle of attack-alpha (a_L/a_R) | AMI |
| Mach number (M_L/M_R) | AMI |
| Equivalent airspeed (EAS _L /EAS _R) | AMI |
| Corrected pressure altitude (HPC _L /HPC _R) | AVVI |
| Pressure altitude rate (H_L/H_R) | AVVI |

Flags will be displayed on the AVVI and AMI if the ADP is not deployed, or ADTA's are COMM faulted for the LEFT or RIGHT side selected by the AIR DATA switch. The ADP VALID flag contains the validity status for the AVVI and AMI.

The ADTA ADS LEFT/RIGHT calculation functional block diagram is shown in figure 4-25. A brief operational description of each subfunction module is tabulated in table 4-V.

4.3.4.4 NAV SELECTED AIR DATA FOR AVVI AND AMI

With the AIR DATA switch in the NAV position on panels F6 and/or F8, the corresponding AVVI will display NAV altitude (H_N) and NAV altitude rate (HDOT) derived from navigation. If the ALT-DG flag is true, the NAV-derived altitude (H_N) will be updated with the ADTA ADS calculated altitude (HPC).

The AMI, with AIR DATA switch in NAV position, will display the air data (Alpha, Mach, and EAS) selected for G&C by the ADTA SOP. If data to G&C switches to default parameters (ALPHA) and (QBAR), the AMI will display NAV-derived alpha.

If the AIR DATA switch fails (COMM faulted, no output, or more than one output), the software will default to NAV position, and an SM ALERT and fault message SW DEFAULT L(R) will be generated.

With the AIR DATA switch in NAV and ADS USE FLG true (ADS data to G&C), the equivalent airspeed (EAS) on the AMI is replaced by relative velocity magnitude (V_r) when weight on the nose gear is indicated. This is required to provide the crew with an indication of ground speed for manual braking purposes.

4.3.4.5 ADTA OVERRIDE DISPLAY AIR DATA CALCULATIONS

The RM sensor calculations compute the following air data parameters for each ADTA output pressure whenever the OVERRIDE SPEC 51 display function is executed and the ADTA is not COMM faulted.

| | |
|-------------------------------|----------|
| Uncorrected pressure altitude | HPU_RM |
| Mach number | M_RM |
| Angle of attack | ALPHA_RM |

The pressures PS, PaU, PaL, and PaC are converted from counts to engineering units (in. Hg) and used to calculate the output parameters as shown in figure 4-26. These parameters are used by the crew,

along with NAV-derived data, to assess an ADTA pressure dilemma condition. A brief operational description of each subfunction module is tabulated in table 4-V.

4.3.5 ADTA SENSOR SELF-TEST

There are two sensor self-test SPEC's that support ADTA sensor self-test function. The on-orbit (OPS-8) SPEC O41 display shown in figure 4-9 will provide the crew with a means to verify sensor test data comparison during sensor self-testing. The display will also control the SF and FDIR status for the ADTA's. The preflight (OPS-9) SPEC 102 display shown in figure 4-27 provides the means to verify ADTA sensor test data comparison during KSC OPS-9 self-testing.

The purpose of ADTA self-test processing (1 Hz rate) is to compare ADTA hardware stored pressures (PaC, PaL, PaU, and PS) and total temperature (T) to a set of onboard limits contained in vehicle utility (VU) software. Additionally, VU software receives each ADTA's mode status word which contains the results of various tests performed within the hardware (BITE).

4.3.5.1 ON-ORBIT (OPS-8) ADTA SENSOR SELF-TEST

When SPEC 41 display is requested by the crew keyboard input, the cyclic processing is initiated and the status and data columns are driven dynamically. The STATUS column will display BITE if the first bit of the mode status word is set to false (ADTA GOOD bit = 0). The ADTA internal status checks and monitoring (mode status bits 2 thru 16) which set data good bit false, are noted in section 4.2.1.2 and table 4-II of the hardware section. It should be noted that the pressures displayed will be zero due to the probes residing in a compartment which is vented to the atmosphere. The temperature will be a function of thermal conditioning of the Orbiter.

The flight crew will conduct self-test of the ADTA's utilizing the keyboard item entries from the RGA/ADTA/RCS SPEC 41 display. DES items will display an asterisk (*) next to the item number when ADTA is removed as an RM candidate. A deselected ADTA will still be self-tested. HI TEST item entry displays an asterisk when high self-test is requested. The asterisk will blank, and the data will be frozen approximately 10 sec after execution of the item number. LOW TEST item entry operates the same as HIGH TEST. STOP TEST item entry removes self-test stimulus to ADTA's, returns display to cyclic processing of probe-sensed pressures and temperature, and displays asterisk after stop item number. After LOW or HIGH TEST, the STOP TEST must be executed to properly reset the test logic. Crew exiting OPS-8 or dropping SPEC 41 will terminate and reset the self-test processing. The SF state determined in OPS-8 shall be maintained across OPS transitions.

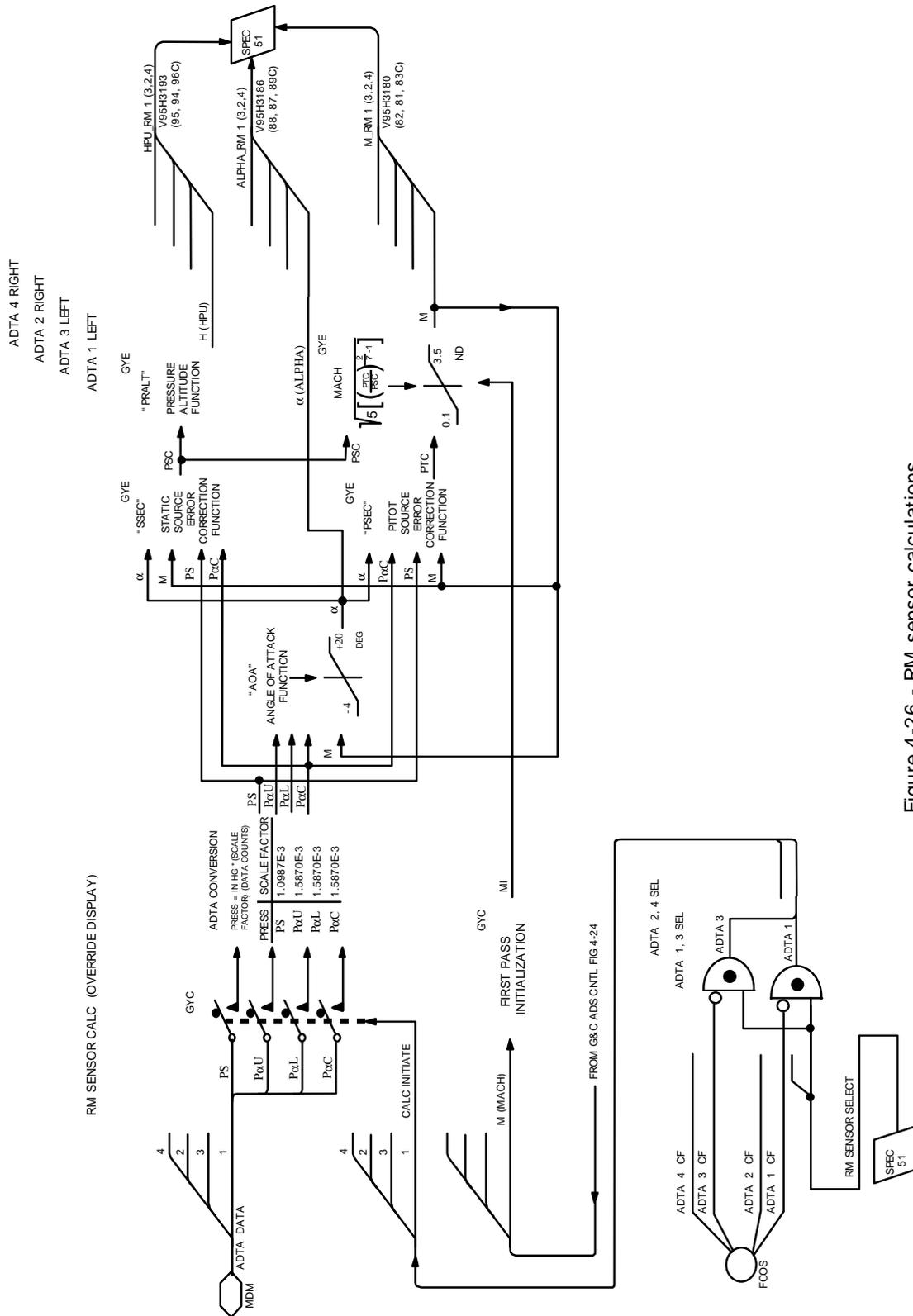


Figure 4-26.- RM sensor calculations.

Figure 4 - 26 RM Sensor Calculations

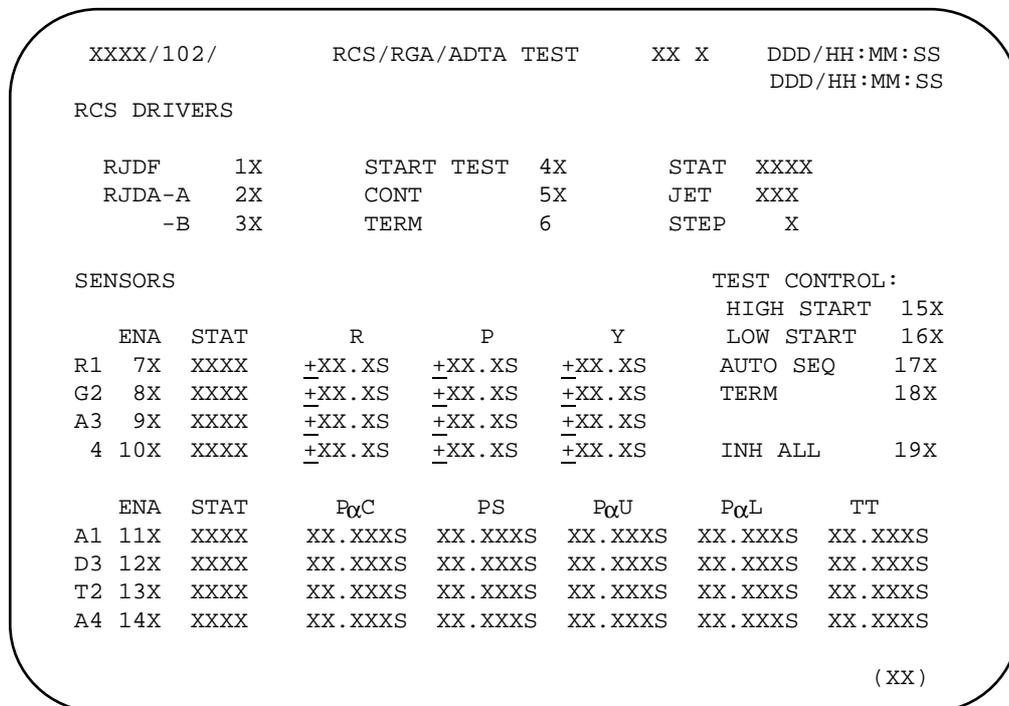


Figure 4 - 27 Prelaunch RCS/RGA/ADTA Test Display

When a valid self-test request is received by the ADTA's, the ADTA processor outputs a prestored and data path from the ADTA to MPM to GPC set of constants for pressures and temperature (the ADTA computer processor is the only hardware tested by this portion of self-test). There is one set for high self-test and one set for low self-test. The GNC software reads these constants approximately 10 sec after the request for self-test is made. The software compares the ADTA processor outputs against a prestored set of high and low limits. The limits are shown in table 4-VI. A threshold is given about each limit to allow for a tolerance for the binary to decimal conversion. In the event of an out-of-tolerance condition, a for low and for high will be displayed to the right of the affected parameters. Secondly, if software receives a BITE out-of-tolerance condition (ADTA GOOD bit set false), the status column will display BITE.

Included in BITE for self-test is a cross-comparison check of the four pressure transducers. The comparison is performed as follows:

$$\begin{aligned}
 |PaL - PaU| &> Tol \\
 |PaU - PaC| &> Tol \\
 |PaC - PS| &> Tol \\
 |PS - PaL| &> Tol
 \end{aligned}$$

where the tolerance (Tol) is 0.080 in Hg. If any two of the comparisons exceed the tolerance then the corresponding pressure good bit for the common transducer is set bad (refer to table 4-II, bits 5, 6, 7, and 8). If ADTA BITE is displayed, perform memory read to determine if ADTA should be deselected. During ADTA self-test, deselect ADTA if up or down arrows are present after data (except after temperature T).

Table 4 - 2 ADTA Software Test Limits

| Parameter | Units | High test limits | Low test limits |
|-----------------|--------|------------------|------------------|
| P _S | in. Hg | .813 to .815 | 24.887 to 24.889 |
| P _{aC} | in. Hg | 5.473 to 5.477 | 29.527 to 29.531 |
| P _{aU} | in. Hg | 2.740 to 2.744 | 21.708 to 21.712 |
| P _{aL} | in. Hg | 1.708 to 1.712 | 14.713 to 14.717 |
| T | Deg. C | 173.56 to 173.60 | 18.50 to 18.54 |

4.3.5.2 PREFLIGHT (OPS-9) ADTA SENSOR SELF-TEST

Preflight self-test utilizes SPEC 102 and operates in a manner similar to that used for Orbit self-test. In preflight self-test, the sensors can be enabled (ENA*) or disabled for self-test, but cannot be deselected or selected as a candidate for RM. HIGH START and LOW START item entry will cause self-test to be performed on only the ADTA's enabled (ENA*). INH ALL item entry will disable all sensors from self-test. TERM item entry has the same function as STOP for OPS-8 self-test. AUTO SEQ item entry initiates self-testing for all sensors listed on the display. The rest of the self-test processing and control are the same in OPS-8 and OPS-9 self-test.

4.4 BACKUP FLIGHT SYSTEM ADS (ADTA HIP)

4.4.1 BACKUP FLIGHT SYSTEM ADS OVERVIEW

The function of the air data hardware interface program (HIP) (fig. 4-28) is to control and monitor the operation of the backup flight system (BFS) ADS (pre- and postengaged) and to process air data information at 1.04 Hz in MM's 304, 305, 602, 603 for utilization by GNC.

The BFS software (GPC-5) receives ADTA serial digital data (pressures and mode status word) over the flight-critical forward MDM buses. Since the ADTA's are digital devices, the BFS software will declare a CF (BCE STRG 1 (2,3,4) ADTA) if it does not receive data from an ADTA. Preengaged, the BFS receives and utilizes the ALTM input from the primary HORIZ SIT SPEC 50 display (fig. 4-10). Pre-engaged, as well as post-engaged, the BFS contains a GNC SYS SUMM 1 SPEC 18 display which provides BFS ADTA status (= CF or RM fail). The BFS HORIZ SIT display provides crew control of the ADTA HIP software outputs to G&C as well as NAV filter controls. Postengaged, the BFS ADTA HIP software (GPC-5) provides all of the air data parameters for GNC and dedicated displays.

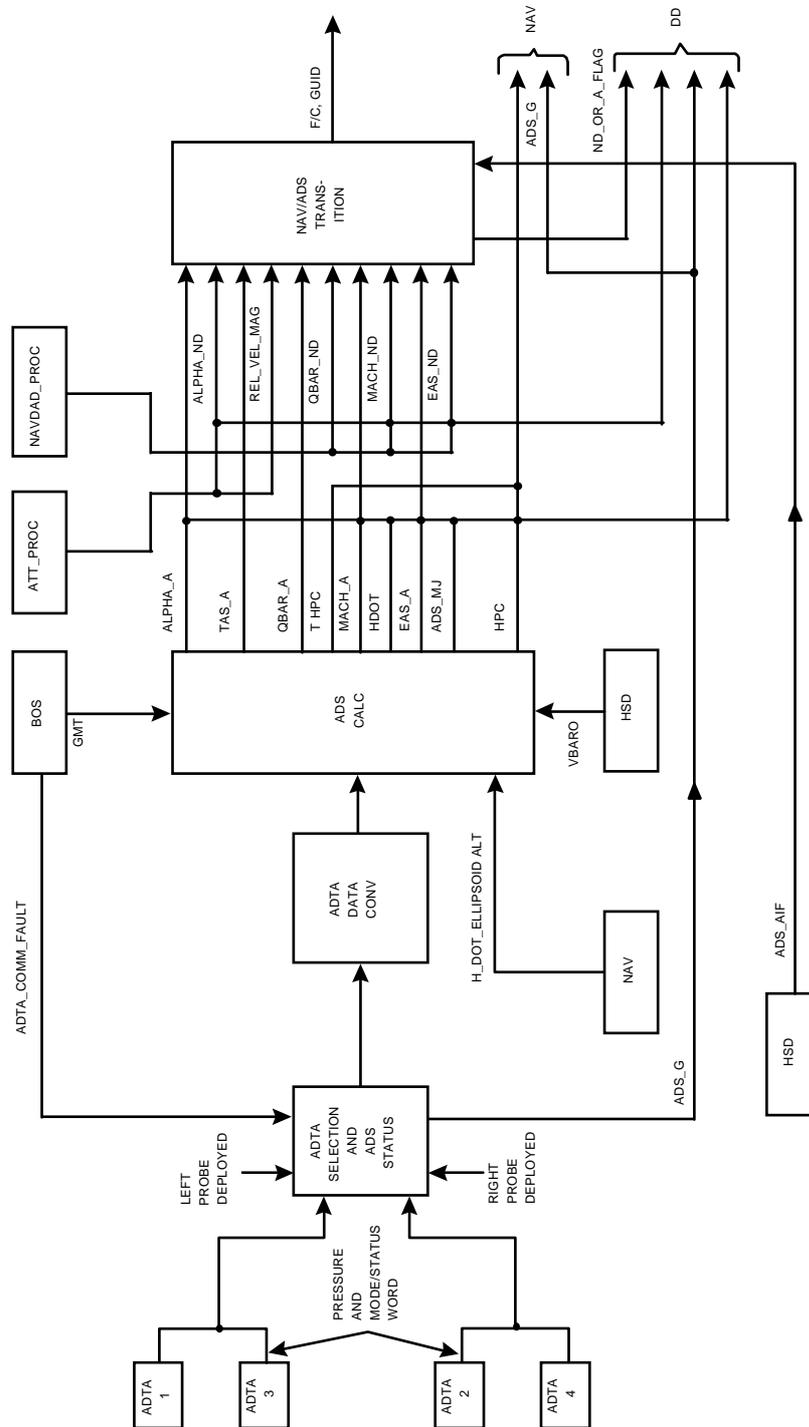


Figure 4-28.- ADTA_HIP functional block diagram.

Figure 4 - 28 ADTA HIP Functional Block Diagram

The BFS ADTA_HIP provides the following functions.

- a. Monitor probe deployment for initiation of ADS processing.

- b. Determine ADTA usability, select single set of pressure data, and convert pressure to engineering units.
- c. Calculate air data parameters for GNC.
- d. Provide transition between NAVDAD and ADTA ADS derived air data to G&C.
- e. Calculate and provide parameters for the AMI and AVVI dedicated displays.
- f. Provide validity flags to NAV and dedicated displays AMI and AVVI.

4.4.2 BFS ADTA STATUS AND USABILITY

The BFS HIP does not contain an ADTA probe RM function like the pass software. An ADTA probe is considered deployed if at least one of the two deploy discretes is ON from a given side. The BFS does not require a "not STOW" but rather, looks only at the deploy discretes. In this manner, it is possible to have a stow discrete failed high (and the other one commfaulted, for example) when the probe is deployed, and unlike the PASS, the BFS will still use the data. There is also no RM associated with the ADTA pressure selection. The ADTA probe deploy discretes, CF's, and certain bits of the ADTA mode/status word are all that are used to determine the usability of the ADTA pressures. The mode/status word bits are those associated with pressure and hardware reasonableness tests (BITE) performed and output by the ADTA internal processor. The ADTA mode/status word structure and generic description for each bit is described in section 4.2.1.2 and table 4-II. The ADTA BITE GD flag is set true if the following states exist for the bits monitored in the ADTA mode/status word:

| | |
|--------------------------------------|-----------|
| Radiate Circuit good * | Bit 2 |
| Not high test mode | Bit 3 (a) |
| Not low test mode | Bit 4 (a) |
| Alpha 1 (PaL) transducer good | Bit 5 (b) |
| Alpha 2 (PaU) transducer good | Bit 6 (b) |
| Total pressure (PaC) transducer good | Bit 7 (b) |
| Static pressure (PS) transducer good | Bit 8 (b) |
| A/D converter good * | Bit 9 |
| Electronics good | Bit 10 |
| Code Memory CRC good | Bit 12 |
| Power supply good | Bit 13 |
| RAM good * | Bit 14 |

* Beginning with OI-29, the Radiate Good and RAM Good BITE will set ADTA BITE GD to false, and the A/D converter good will not. (BFS CR 92421).

where Bit count is Bit 1 through Bit 16.

- a. Self-Test mode only.
- b. During prelaunch, bit toggling acceptable if windy conditions.

Note: Toggling of the ADTA Good bit (not shown here) is acceptable as long as no other bits listed are changing.

If the ADTA is not usable because of a probe not deployed, CF present, or selected BITE, the BFS software will set the PROBE_SELECT flag false and remove the ADTA pressures from the RM function. In addition, if a CF is present or a BITE GD is set false, the software will generate an SM alert, display the fault message RM FAIL ADTA, and place a down arrow after the ADTA on the GNC SYS SUMM 1 display.

4.4.3 BFS DATA SELECTION AND CONVERSION

The BFS HIP software performs a selection process which prime selects ADTA 1 on the left side and ADTA 2 on the right side. These pressure outputs are then averaged to obtain a single set of pressures for air data parameter calculations. In the event of a failure of ADTA 1 or 2, the standby ADTA 3 or 4 is utilized and the averaging process continues. If data are only usable from the left side ADTA (1 or 3) or the right side ADTA (2 or 4), the single ADTA pressures shall be used. The logic in figure 4-30 shows the selection process and resultant outputs.

The selected ADTA pressures are converted from data bit counts to engineering units of in. Hg by scaling the data.

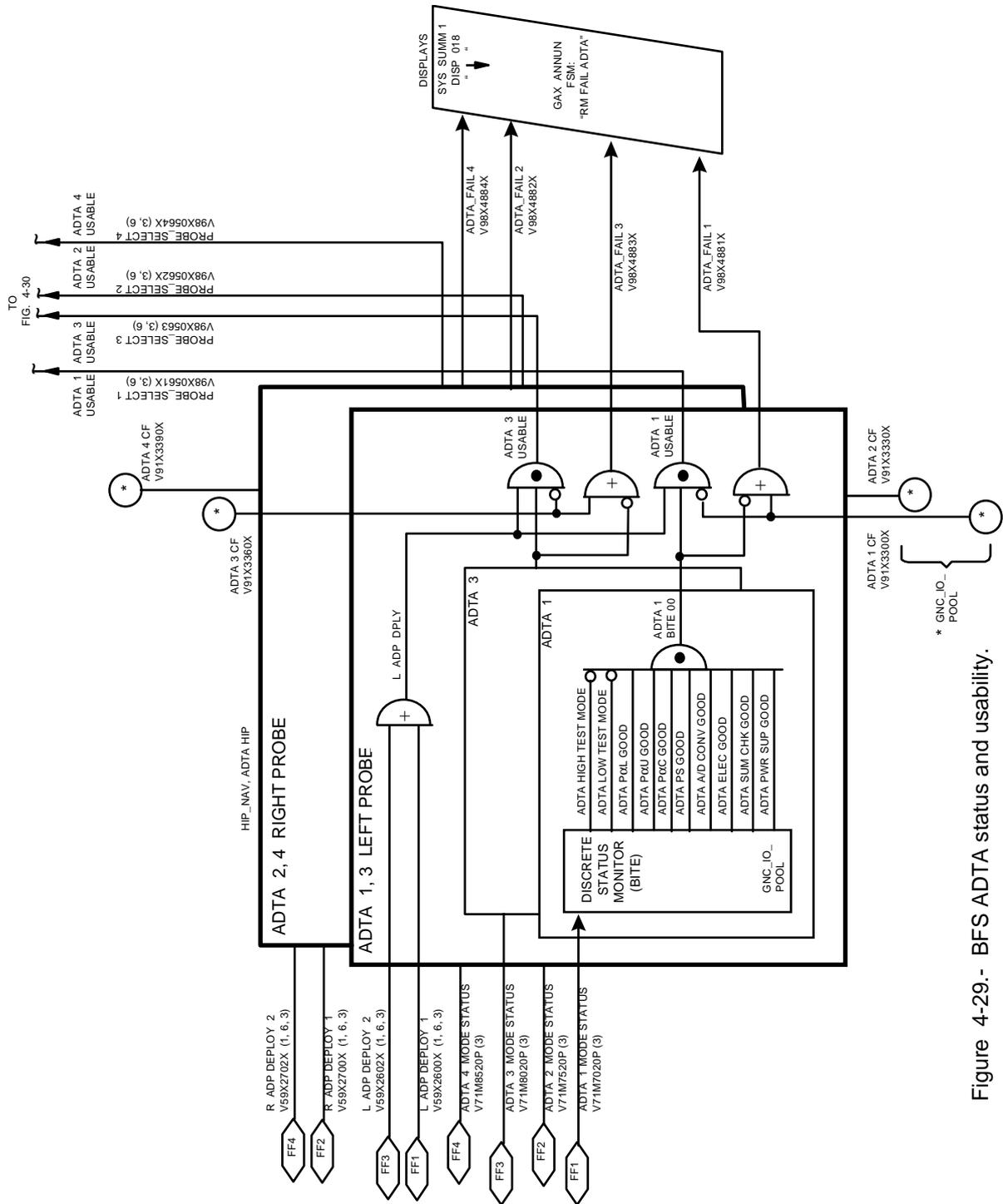


Figure 4 - 29 BFS ADTA Status and Usability

Figure 4-29.- BFS ADTA status and usability.

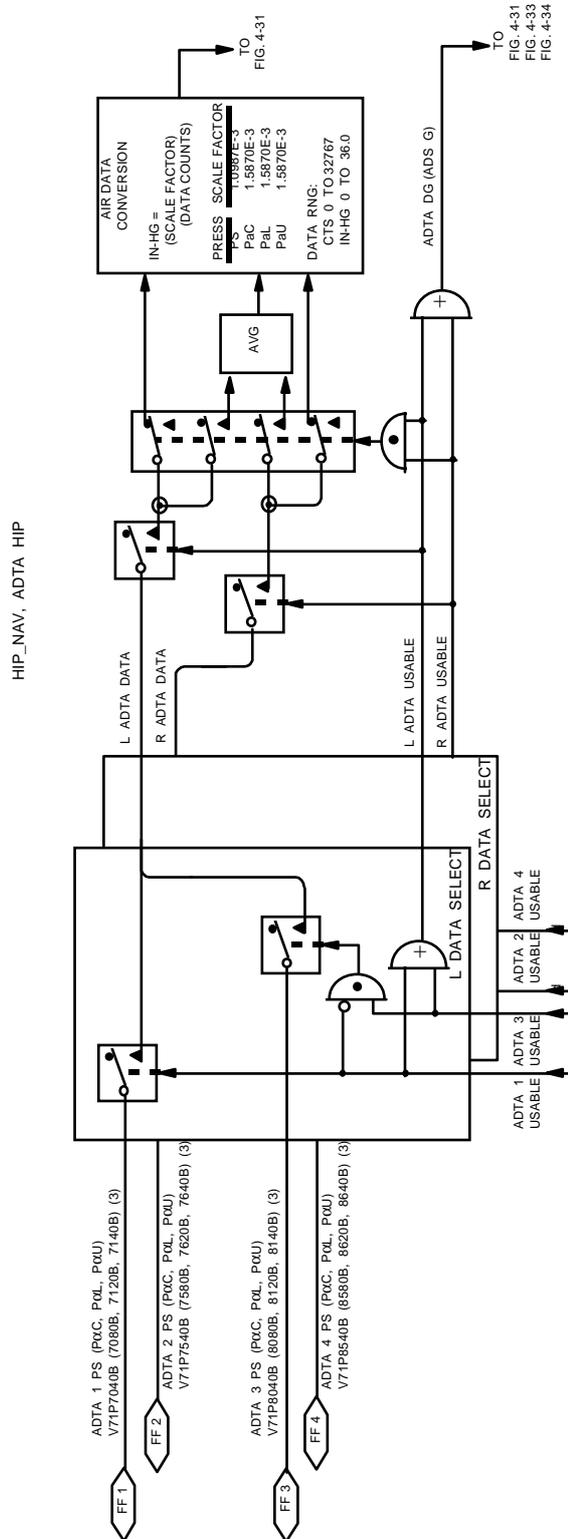


Figure 4-30.- BFS data selection and conversion.

Figure 4 - 30 BFS Data Selection and Conversion

The ADTA data good flag (ADS_G) is set false if no ADTA's are considered usable. The functions of the ADS_G flag are as follows.

- a. Initiate calculations of the ADS air data parameters.
- b. Enable transition from NAV- to ADS-derived air data upon crew request from the HORIZ SIT, SPEC 50, display.
- c. Provide a validity discrete to NAV and the dedicated displays AMI and AVVI.

4.4.4 BFS ADTA ADS CALCULATIONS

The BFS ADTA HIP software performs the same basic processing for air data parameters (HPC, HDOT, THPC, ADS_MJ flag, MACH, ALPHA, EAS, QBAR, and TAS) as the ADTA SOP software does in the primary system. The BFS ADS calculation block diagram is shown in figure 4-31 and is similar to figure 4-21, ADTA ADS calculations in pass software. As long as the ADS_G flag is set true, the selected pressures from figure 4-30 are used in calculating the ADS parameters. With the relationship and operation of the subfunctions the same as the pass software, a brief operational description of each subfunction module can be found in table 4-5.

4.4.5 BFS NAV-DERIVED ADS CALCULATIONS (NAVDAD)

Prior to ADTA probe deployment or in the event of an ADS system failure (ADS__G flag set false), the BFS HIP software will output air data parameters from NAVDAD, UPP, and attitude processing. The BFS NAVDAD processing of air data parameters (TAS = Vr, MACH, QBAR, EAS, and ALPHA) is shown in figure 4-32. The BFS NAVDAD processing is the same as the pass software described in section 4.7.1, Navigationally Derived Air Data (NAVDAD).

4.4.6 BFS NAV/ADS AIR DATA TRANSITION

The purpose of this function is to provide transitioning between certain NAV-derived and ADS air data parameters as shown in figure 4-34. The transition shall be performed (reference fig. 4-33) as a function of ADTA usability (ADS_G), Rel_Vel_Mag (Vr), and crew control (ADS_AIF) from the SPEC 50 HORIZ SIT display.

The three control modes available to the crew from the HORIZ SIT display are as follows.

AUTO

IF: ADS_G is true and Vr less than 2500 ft/s, set ADS USE flag true and select ADS data for G&C. Note that in the BFS, the HSD display is initialized with item 28 (AUT) selected.

ELSE: Set ADS USE flag false and select NAVDAD data for G&C.

INH

Set ADS USE flag false and select NAVDAD data for G&C.

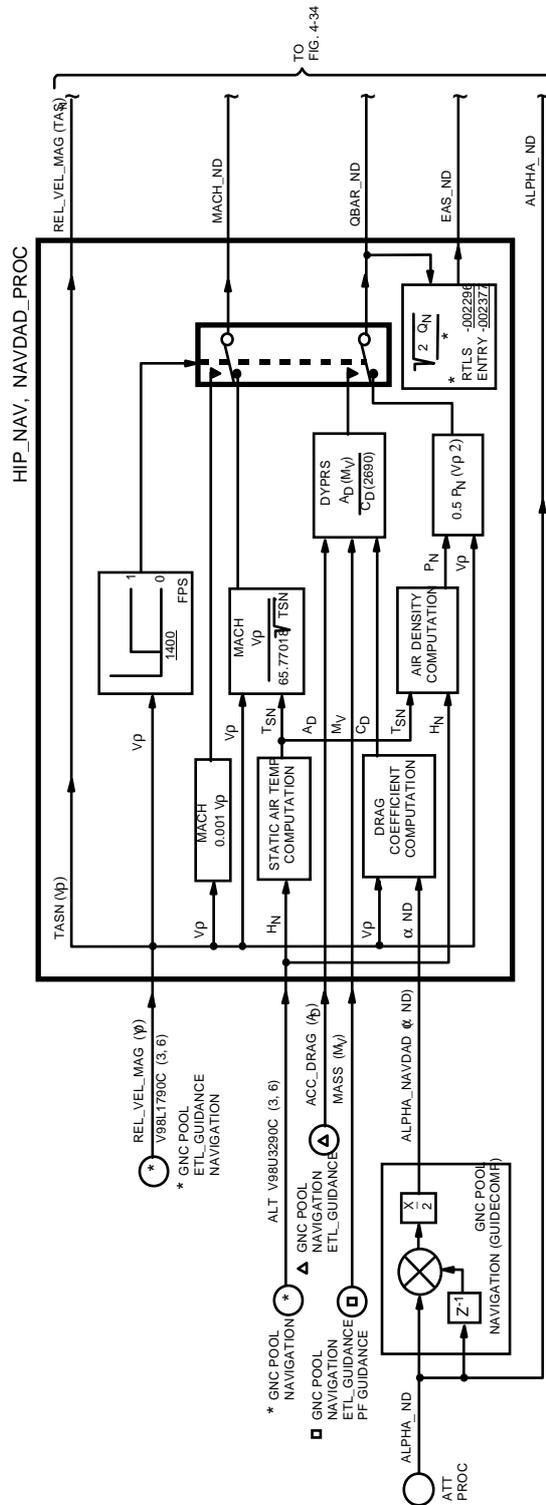


Figure 4-32.- NAV-derived ADS calculation (NAVDAD).

Figure 4 - 32 NAV-Derived ADS Calculation (NAVDAD)

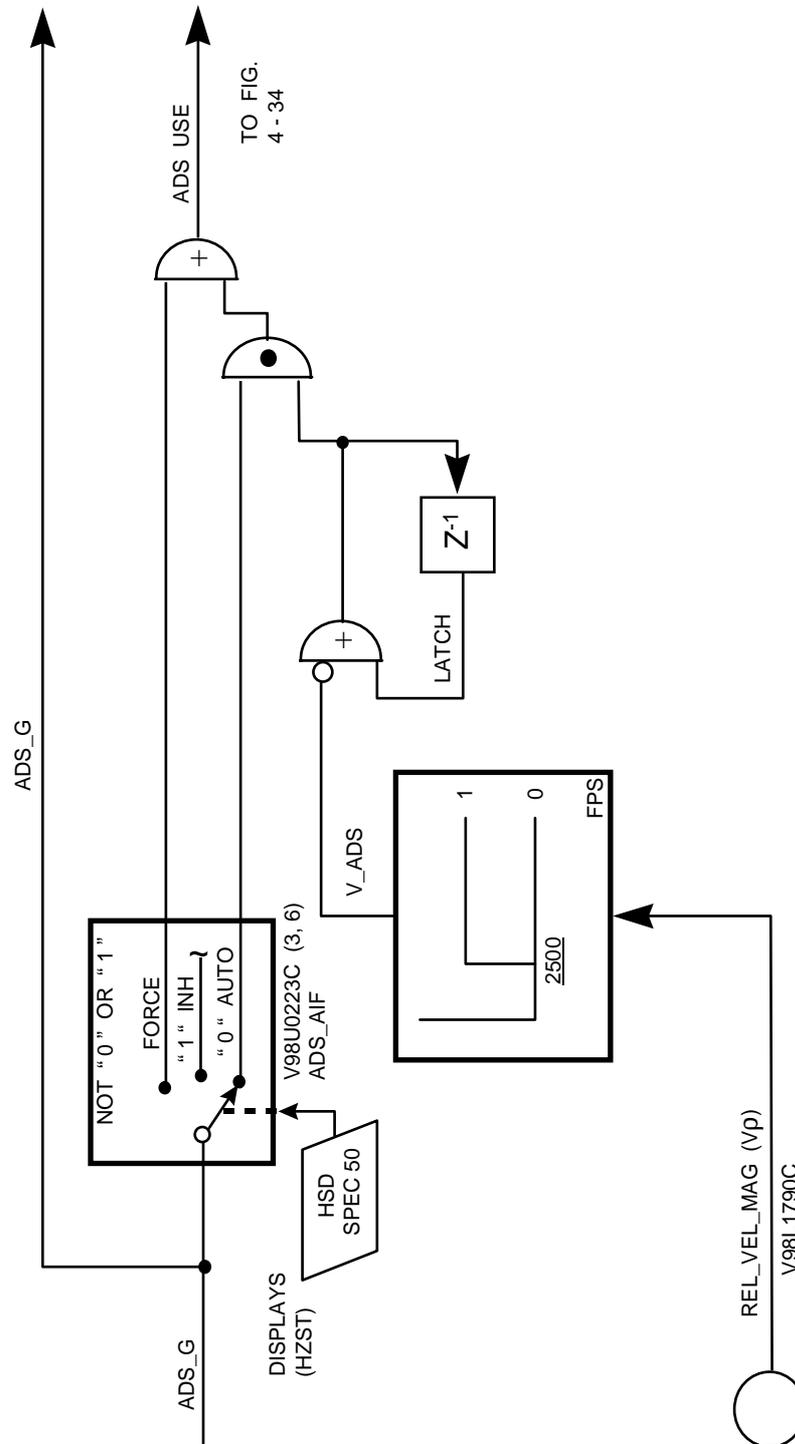


Figure 4 - 33 BFS NAV/ADS Air Data Transition Control

Figure 4-33. - BFS NAV/ADS air data transition control.

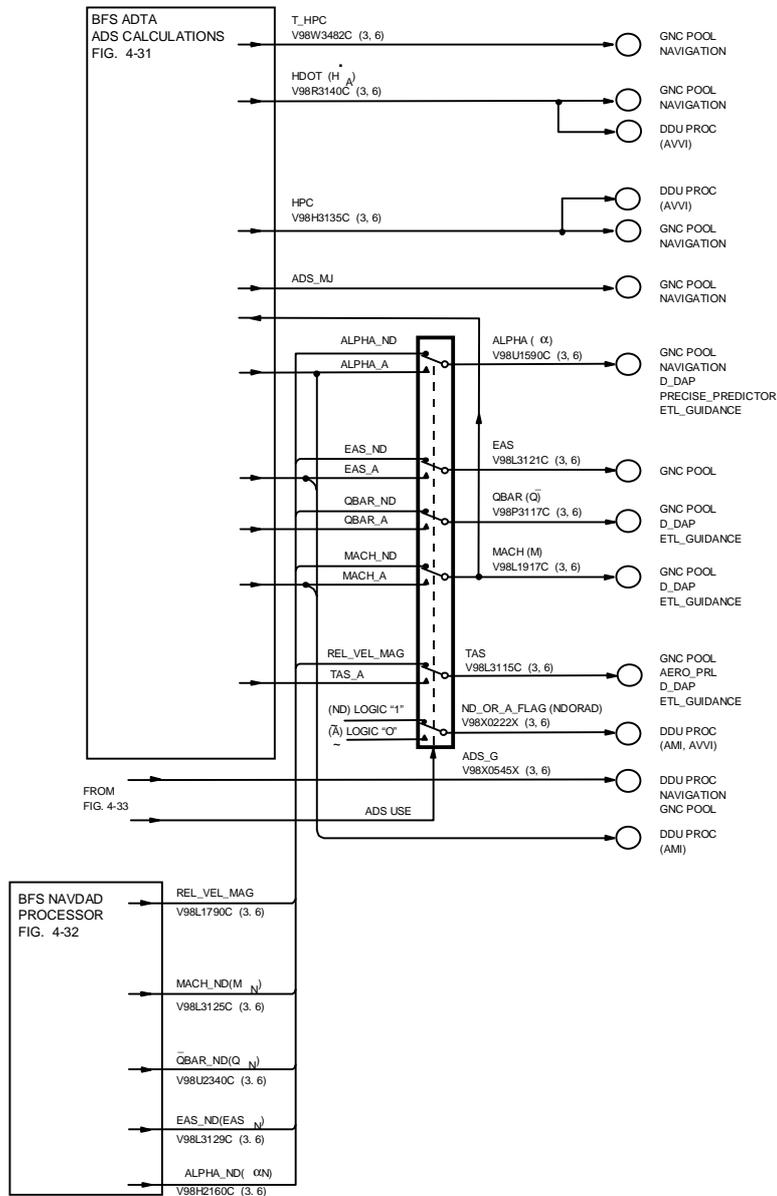


Figure 4 - 34 BFS NAV/ADS Air Data Transition

FORCE

IF: ADS_G is true, set ADS USE flag true and select ADS data for G&C.

ELSE: Set ADS USE flag false and select NAVDAD data for G&C.

The BFS HIP software does not generate default QBAR and ALPHA for NAV-derived air data below Mach 1.5, but continues to use the NAVDAD parameters. Unlike the pass software, the BFS ADS air data does not downmode to NAVDAD in return to launch site (RTLS) for contingency abort with ADS HPC parameter limited at its upper limit of 105,000 ft. Also, during RTLS, ADS HPC is not marked invalid when ADS calculated altitude is limited at the upper limit of 105,000 ft. Another difference in the BFS software is that ALPHA does not transition from ADS to NAVDAD while the ADS ALPHA calculation is limited at the upper limit of 20°.

4.4.7 BFS AIR DATA PARAMETER FOR DEDICATED DISPLAYS

The air data parameters provided for display processing are as follows:

- | | | |
|----|--|------------------|
| a. | HPC, displayed as | ALT on AVVI |
| b. | HDOT, displayed as | ALT RATE on AVVI |
| c. | ALPHA, displayed as | ALPHA on AMI |
| d. | MACH, displayed as | M/VEL on AMI |
| e. | EAS, displayed as | EAS on AMI |
| f. | ADS_G, ADS data valid flag to | AVVI and AMI |
| g. | ND or A_FLAG, ADS or NAVDAD to G&C for | AMI |

4.4.8 BFS AMI DATA PROCESSING

The air data displayed on the AMI is determined by the following software logic.

IF: ADS_G false, or ND_OR_A_FLAG true

THEN: Display NAVDAD (ALPHA, MACH, EAS) data.

ELSE: Display ADS (ALPHA, MACH, EAS) data.

ADS G is set false if no ADTA is usable for BFS processing (fig. 4-29). ADTA is usable only if its probe is deployed, not CF, and hardware BITE not indicating a failure.

The status of the ND or A_FLAG is determined by the crew selecting auto (AUT), inhibit (INH), or force (FOR) on the HSD, and is shown in figure 4-33 and figure 4-34. For ND or A_FLAG to be false (ADS USE flag on), the following conditions must exist:

- HSD in AUT, ADS_G on, Vr < 2500 ft/s
- HSD in FOR, ADS_G on
- HSD not in INH

In brief, the AMI will always display the air data selected for G&C, regardless of where the AIR DATA select switch on panel F6 or F8 is positioned.

4.4.9 BFS AVVI DATA PROCESSING

The air data displayed on the AVVI is determined by the following software logic.

IF: ADS_G false, or NAV_SW true

THEN: Display NAVDAD (HPC, HDOT) data

ELSE: Display ADS (HPC, H) data

How ADS_G is set false or true was previously described in the AMI data processing section.

The NAV_SW flag is set true only if the AIR DATA switch on the commander's (CDR's) panel F6 is in the NAV position. The BFS software does not look at the PLT's AIR DATA switch on panel F8, and does not contain default logic for the AIR DATA switch.

In brief, the AVVI will display NAV-derived altitude (H) and altitude rate (HDOT) with CDR's AIR DATA switch in NAV position. If ADS_G flag true and ADS_MJ flag false, the H will be updated with the ADS-generated HPC.

With the CDR's AIR DATA switch in the LEFT or RIGHT position, and ADS_G true, the AVVI will display ADS altitude (HPC) and HDOT as calculated by the selected ADTA pressures. If ADS_G is false, Nav-derived data will be displayed in the LEFT or RIGHT position as well. If ADS_G is true, the NAVDAD and ADS display data for NAV and LEFT/RIGHT switch position will essentially be the same because of the NAV system using ADS HPC to update the Nav-derived altitude.

4.5 Notes of Interest

NOTE OF INTEREST #73
STS-4 & Subs

CF62 (82-51)

TO: Distribution
FROM: CF62/Robert A. O'Donnell
SUBJECT: STS BFS AMI/AVVI Air Data Processing

1. The ADS data (HPC, HDOT, ALPHA, MACH, EAS) for the BFS dedicated displays AVVI/AMI is the same as the parameters calculated for GNC. Air data switch requests for LEFT/NAV/RIGHT are honored from panel F6 only (CDR's side). Switch requests from LEFT or RIGHT will result in identical outputs. There is no default logic for the AIR DATA switch; however, the dedicated displays will always receive valid data. In case of CF's, the last valid data in compool is used.

2. AVVI Air Data Parameters (HPC, HDOT) - BFS

- a. If AIR DATA switch in NAV and ADTA DATA NOT GOOD, display NAV derived air data.
- b. If AIR DATA switch in NAV and ADTA DATA GOOD, display NAV data updated with ADS (ADTA) HPC.
- c. If AIR DATA switch in L/R and ADTA DATA NOT GOOD, display NAV derived air data.
- d. If AIR DATA switch in L/R and ADTA DATA GOOD, display ADTA derived data. Data in (b) and (d) will normally reflect the same value after ADTA HPC updates the NAV derived altitude.

3. AMI Air Data Parameters (ALPHA, MACH, EAS) - BFS

The AMI will always display the air data parameters (ADS or NAVDAD) selected for G&C, regardless of where the AIR DATA select switch is positioned.

4. Instrumentation - BFS

| | <u>ENTRY DOWNLIST</u> | <u>MRL</u> | <u>MCC</u> |
|------------------------|-----------------------|------------|------------|
| V98X0222X ND_OR_A_FLAG | YES | NO | |
| V98U0223C ADS AIF IND | YES | NO | |
| V98X0545X ADS_G | YES | YES | MSK1413 |
| V98U3431C HPC AIF IND | YES | YES | MSK1413 |

SECTION 5 RATE GYRO/ACCELEROMETER ASSEMBLIES

5.1 ORBITER RATE GYRO ASSEMBLIES

5.1.1 Introduction

The Orbiter is equipped with four rate gyro assemblies (RGA's) located at the aft bulkhead. RGA's 1, 2, 3, and 4 are located side by side at the bottom of the bulkhead in the wing well facing the payload bay (fig. 5-1). Each assembly consists of three single degree-of-freedom rate gyros positioned to sense rates about the roll, pitch, and yaw body axes. During the ascent and entry flight phases, the Orbiter rate gyros provide the sensed roll, pitch, and yaw body rates to flight control, where the sensed rates act as damping feedback which provides stability augmentation of the vehicle, and to the attitude director indicator (ADI) rate needles. The RGA's are powered off during the on-orbit phase and reactivated for the sensor self-test prior to deorbit.

5.1.2 Hardware Description

Power and data flow for each RGA are depicted in figure 5-2. All RGA's receive main bus power through the use of switches and remote power controllers (RPC's). RGA 1 and RGA 4 receive redundant main bus power.

The rate gyros within each assembly are identical with respect to operation and hardware components. The only difference between the roll, pitch, and yaw gyros is their positioning within the assembly. The following description covers one single degree-of-freedom rate gyro.

The major elements of a rate gyro consist of the gyro wheel, gimbal assembly, signal generator, and spin motor run detector (SMRD) circuit.

The rate gyro is a floated, single degree-of-freedom rate gyro. When the gyro wheel is running at synchronous speed (24,000 rpm), movement of the vehicle about the input axis triggers movement of the gimbal about the output axis (gimbal precession) (fig. 5-3). As the gyro wheel precesses, torque is developed. The torquer cancels gimbal precession and thus measures input rates accurately without permitting excessive gimbal displacement. The precession rate of the gimbal assembly is directly proportional to the rotational rate of the gyro case. The signal generator provides an output voltage proportional to the angle of gimbal rotation. The torque generator provides the capability to torque the gimbal about the output axis in proportion to the magnitude of an externally applied current, a self-test command.

The Orbiter rate gyros are equipped with a SMRD circuit per axis which detects an under speed or over speed of the spin motor. The SMRD circuit will indicate an invalid signal at 98 percent or 102 percent of the spin motor synchronous speed. The SMRD circuit is the rate gyro built-in test equipment (BITE).

The SMRD is to protect against allowing power off transients from being processed by flight control. When power is removed from an RGA (orbiter or SRB) there is a transient response that lasts for approximately 20 seconds and reaches a peak value of approximately 50% (absolute value) of the output range (± 10 deg/sec in pitch and yaw, ± 20 deg/sec in roll). When two RGA's are unpowered

simultaneously, the SMRD logic in RM will prevent processing this bad data. Details of SMRD implementation will be covered in detail in the Redundancy Management (RM) section.

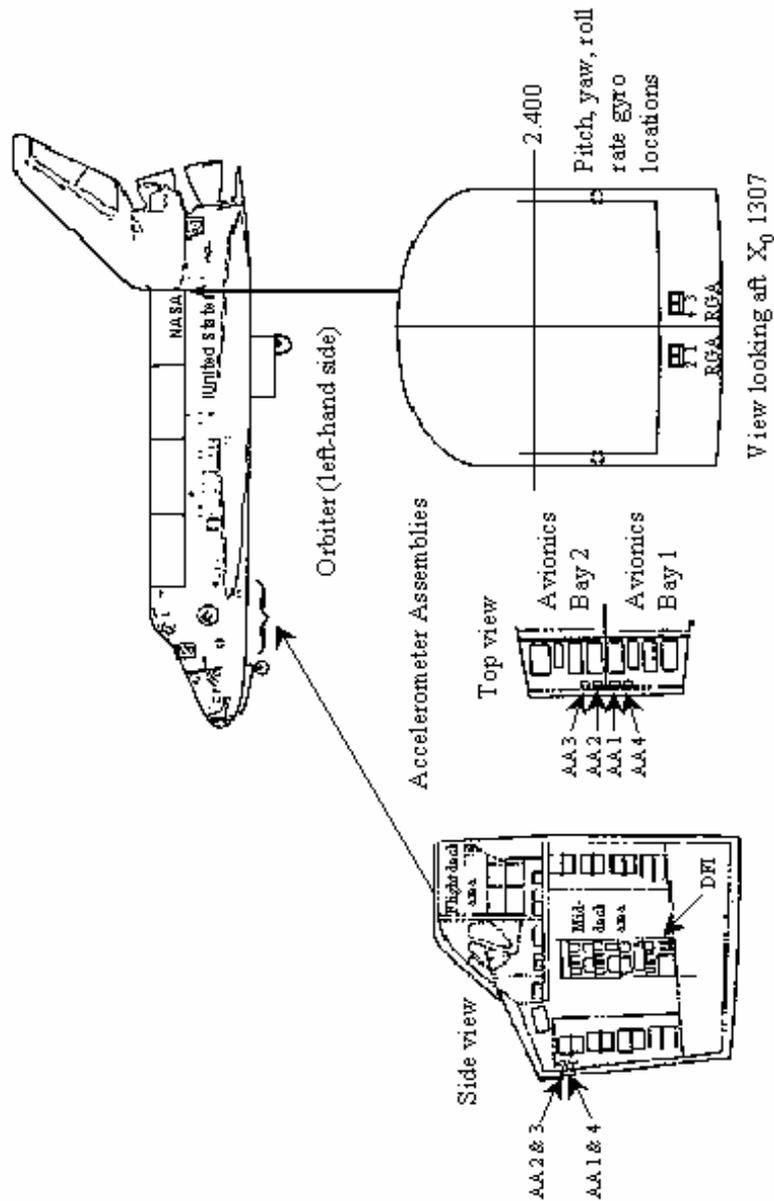


Figure 5-1 - The GNC hardware locations

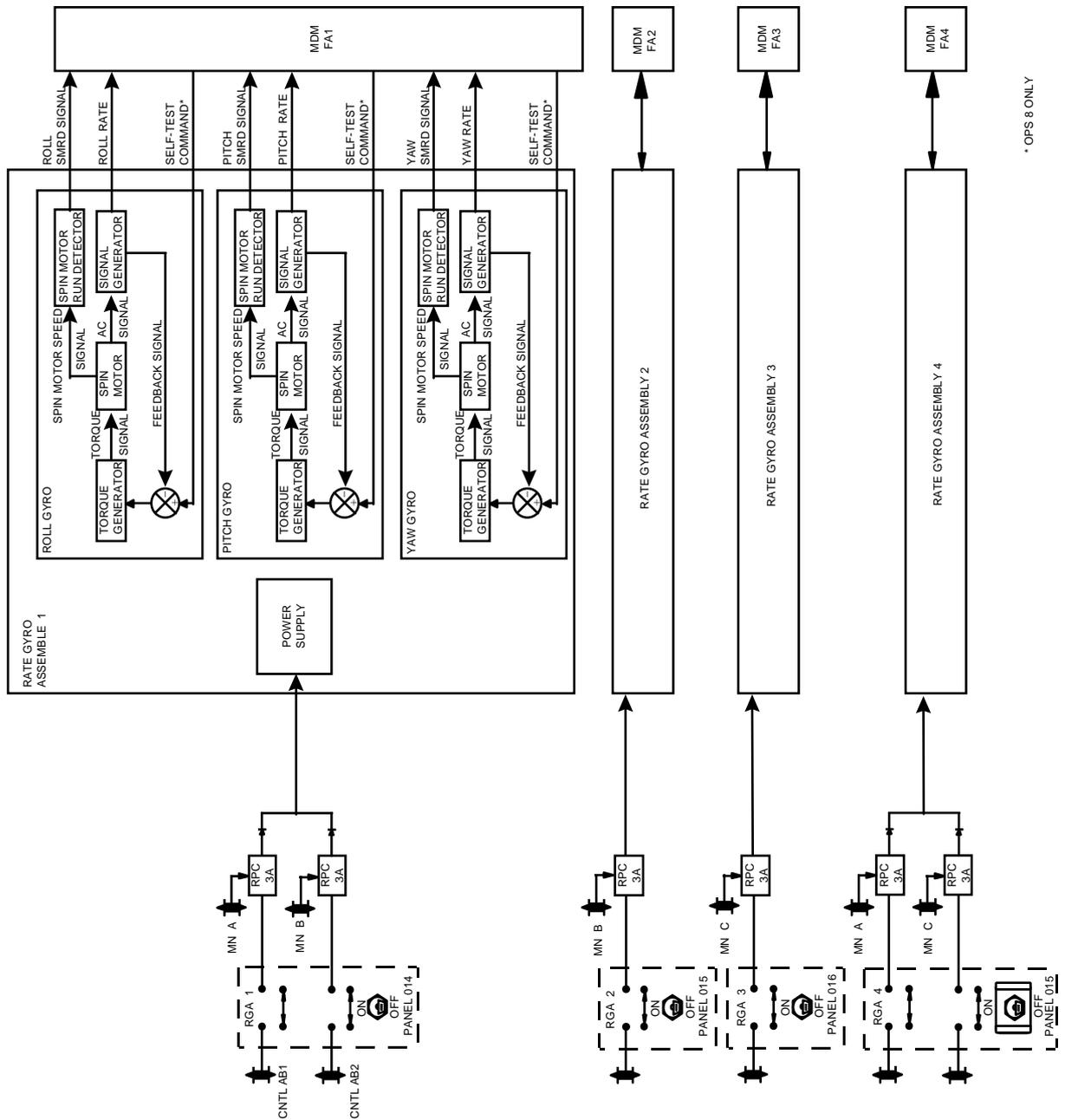


Figure 5-2 - Orbiter RGA diagram

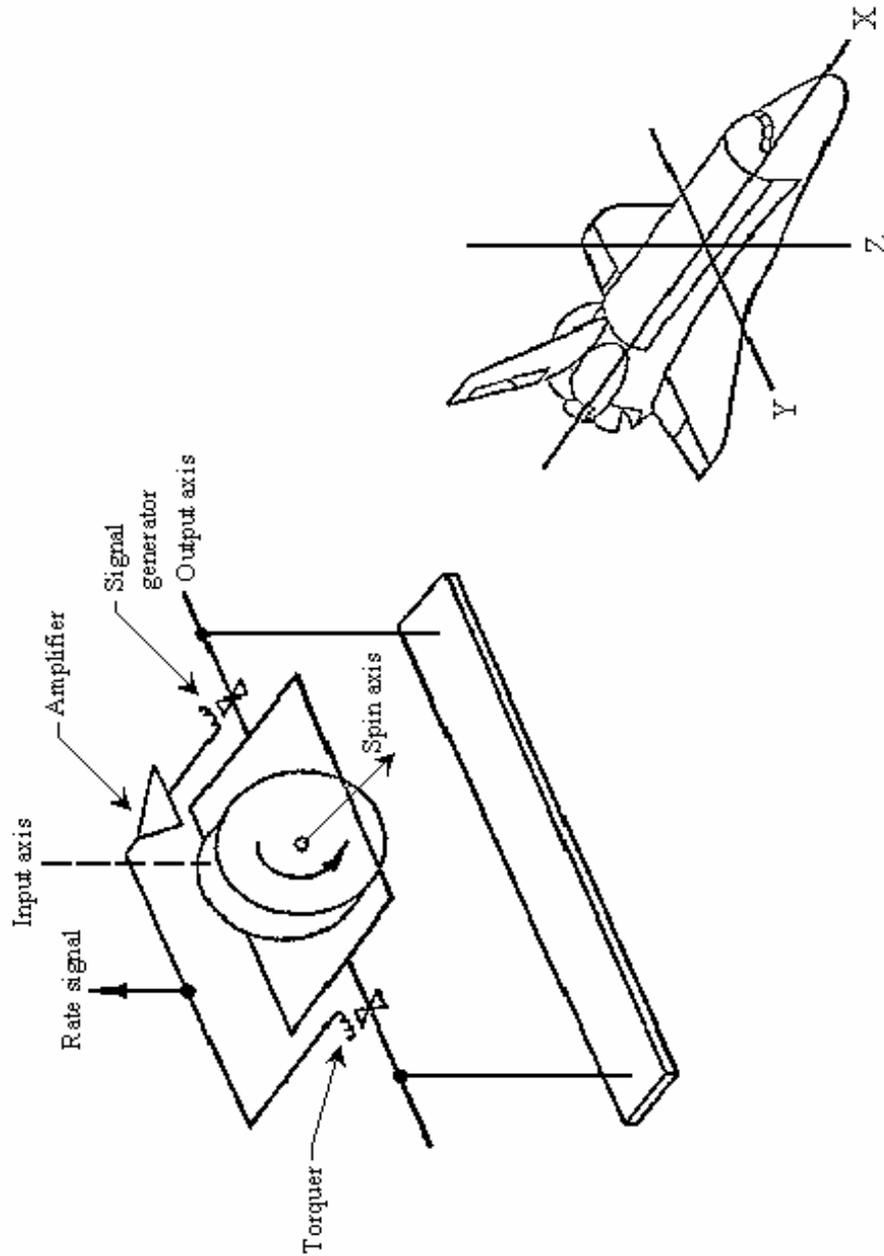


Figure 5-3 Rate Gyro, in Yaw Orientation

5.1.3 Performance Data/Limitations

| | |
|---------------------|--|
| Maximum gyro output | ±20 deg/s pitch, yaw rate ±40 deg/s roll rate |
| Warm up time | 2 min |
| Power requirements | 25 W at sync speed 36 W during spin-up |

5.2 SOLID ROCKET BOOSTER RATE GYRO ASSEMBLIES

5.2.1 Introduction

The two solid rocket boosters (SRB's) are each equipped with two RGA's (fig. 5-5). Each SRB RGA contains two single degree-of-freedom rate gyros; one is positioned to sense pitch rates, and the other is positioned to sense yaw rates. The SRB RGA pitch and yaw rates are used in flight control feedback loops to assist in SRB thrust vector control and to provide stability augmentation during the ascent flight phase until SRB separation. The SRB RGA's are mounted on the forward ring within the forward skirt of the SRB attach point.

5.2.2 Hardware Description

Power flow and data flow for the SRB RGA's are depicted in figure 5-4. Two RGA's on each SRB are powered from the separate SRB buses which receive their power from two Orbiter main buses.

The single degree-of-freedom rate gyros contained in the SRB RGA's have the same functional components as those contained in the rate gyros that comprise the Orbiter RGA's (ref. section 5.1). The only hardware difference between the Orbiter RGA's and the SRB RGA's is the additional roll gyro within the Orbiter RGA's.

5.2.3 Performance Data/Limitations

| | |
|---------------------|--|
| Maximum gyro output | ±10 deg/s pitch rate ±10 deg/s yaw rate |
| Warmup time | 5 min |
| Power requirements | 18 W at sync speed 28 W during spin-up |

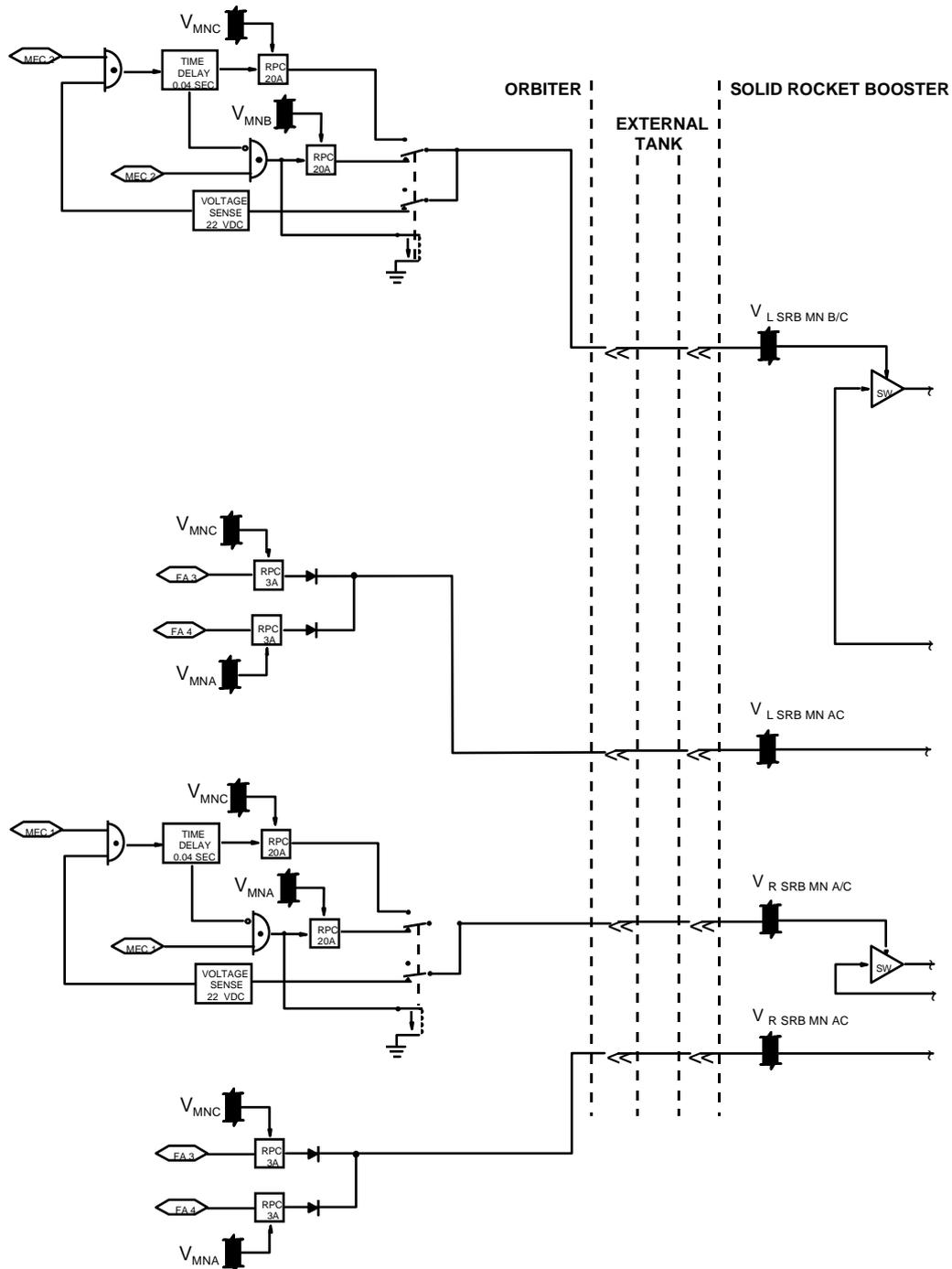


Figure 5-4 - Solid rocket booster rate gyro assembly (sheet 1 of 2)

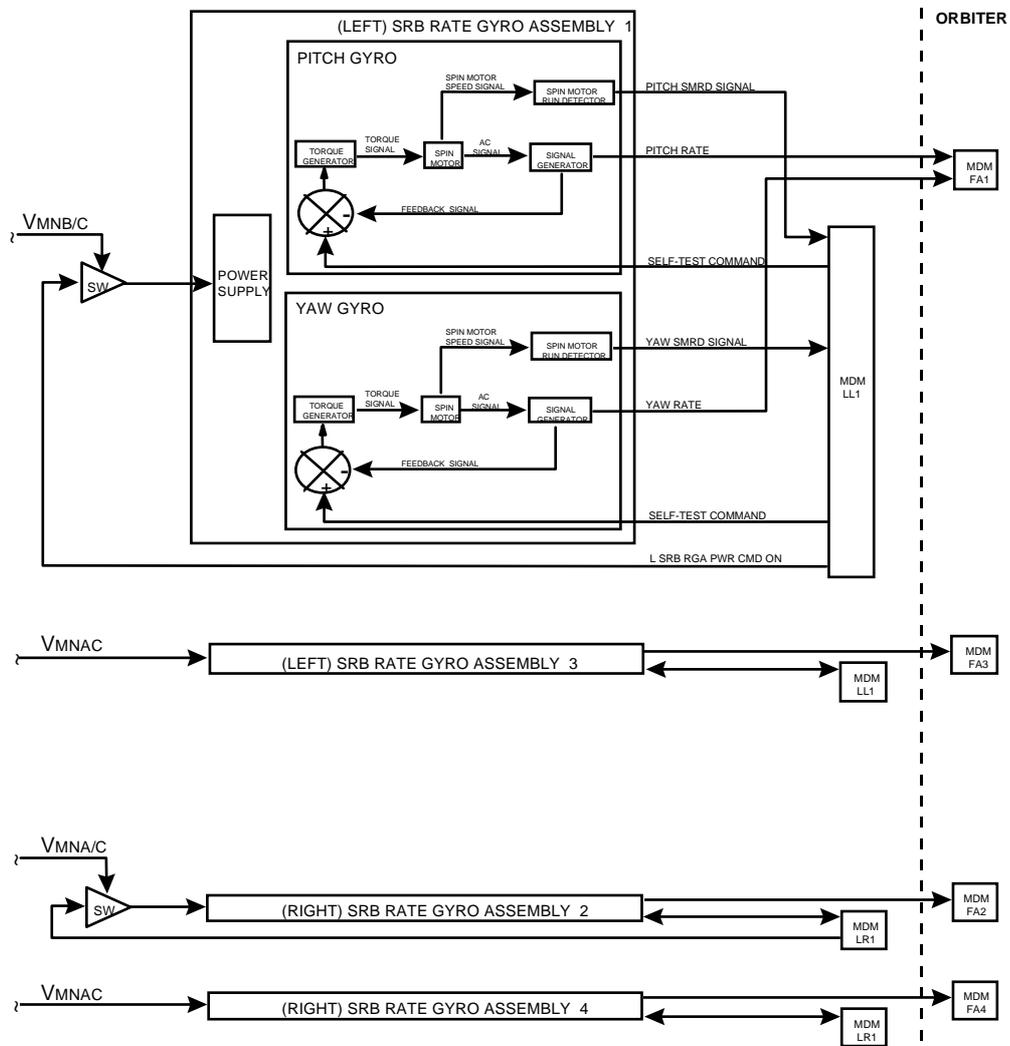


Figure 5-4 - Solid rocket booster rate gyro assembly (sheet 2 of 2)

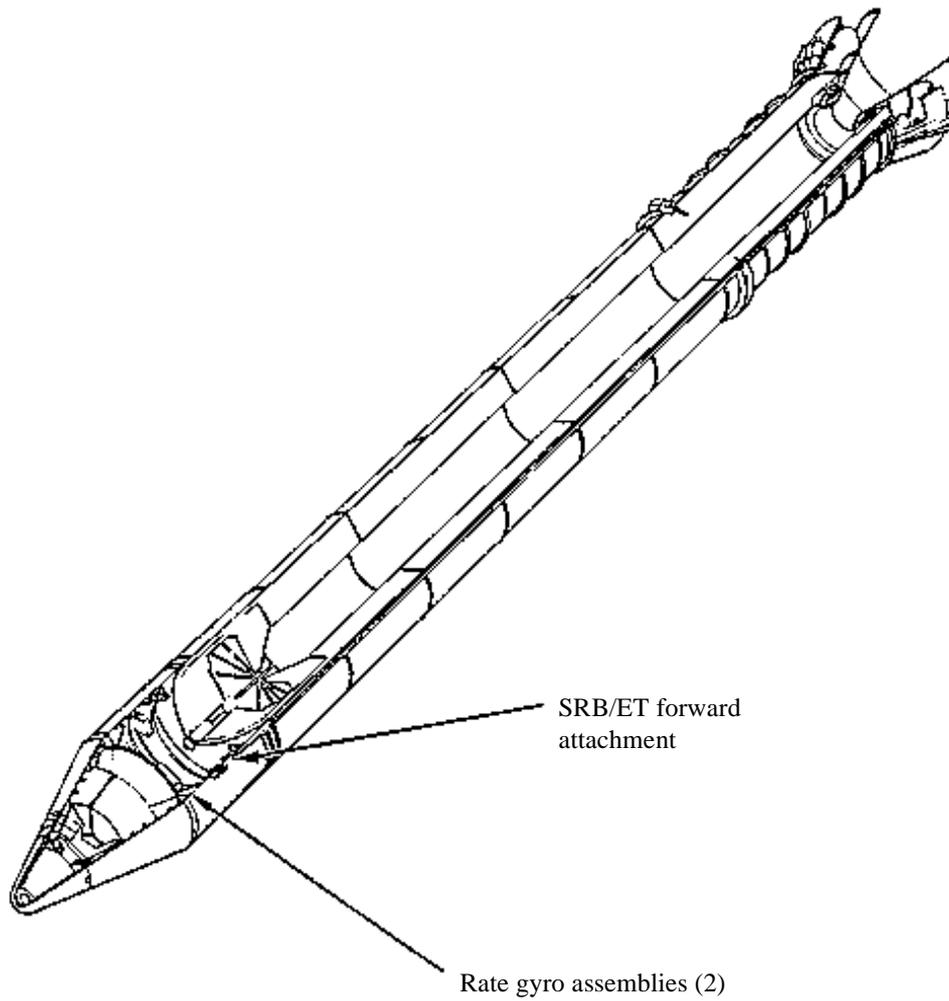


Figure 5-5 - The SRB RGA location.

5.3 ACCELEROMETER ASSEMBLIES

5.3.1 Introduction

There are four body-mounted accelerometer assemblies (AA's) on the Orbiter. The AA's are mounted in the forward avionics bays (fig. 5-1). Each AA contains two single-axis Honeywell accelerometers positioned such that one senses lateral accelerations along the Y-axis and one senses normal accelerations along the Z-axis. Lateral and normal accelerations from the AA's are used in flight control feedback loops to provide stability augmentation during ascent and entry. Currently, the AA's are powered off during the on-orbit phase and reactivated for the sensor self-test prior to deorbit.

5.3.2 Hardware Description

Power flow and data flow for each AA are depicted in figure 5-6. AA 1 and AA 2 receive main bus power through circuit breakers, and AA 3 and AA 4 receive main bus power through the use of switches and RPC's.

The single-axis accelerometers within an AA function identically and have identical components; therefore, the functional description of the elements and interaction of those elements for one single-axis accelerometer will be given. The only difference between a lateral and a normal accelerometer is their positioning within the assembly.

A Honeywell single-axis accelerometer consists of a quartz fiber suspension and pendulum, a photo diode pickoff, a one-turn torque generator and a dc amplifier (fig. 5-7). A thin coat of silver is vapor-deposited over the length of the suspension and pendulum, providing a conducting surface. The base of the pendulum is positioned in a permanent magnetic field such that current flowing in the pendulum circuit acts as a one-turn torque generator. The optical pickoff consists of a light emitting diode (LED) lamp and a silicon junction photo diode.

The pendulum assembly is mounted in a hermetically sealed aluminum housing. The housing is filled with a low viscosity silicon oil to control the dynamic characteristics of the accelerometer.

When zero acceleration exists, the null position is defined such that the base of the pendulum coincides with the photo diode pickoff slot and zero differential voltage exists at the output of the photo diodes. The pendulum deflects from the null position when acceleration is applied. Light unbalance on the two halves of the photo diode results in a differential voltage output to the dc amplifier. Polarity of voltage depends upon which direction the pendulum moves. The signal from the amplifier is fed back to the pendulum one-turn torque generator, in the proper polarity, to rebalance the pendulum at an offset position. When the acceleration is removed, the amplifier and pendulum are returned to the null position.

The torque generator also provides the capability to generate an output signal in proportion to the magnitude of an external signal, a self-test command.

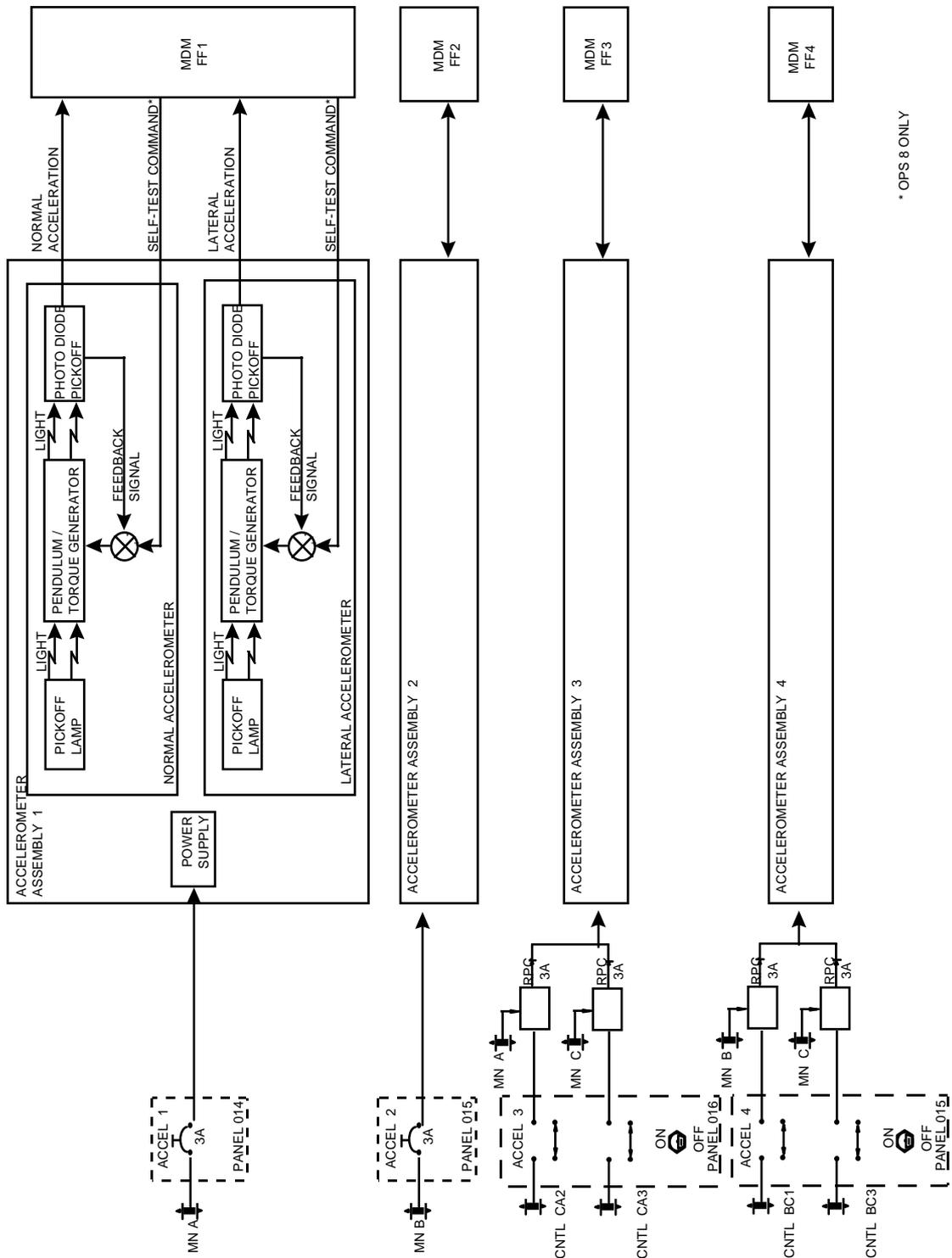


Figure 5-6 - Accelerometer assembly diagram

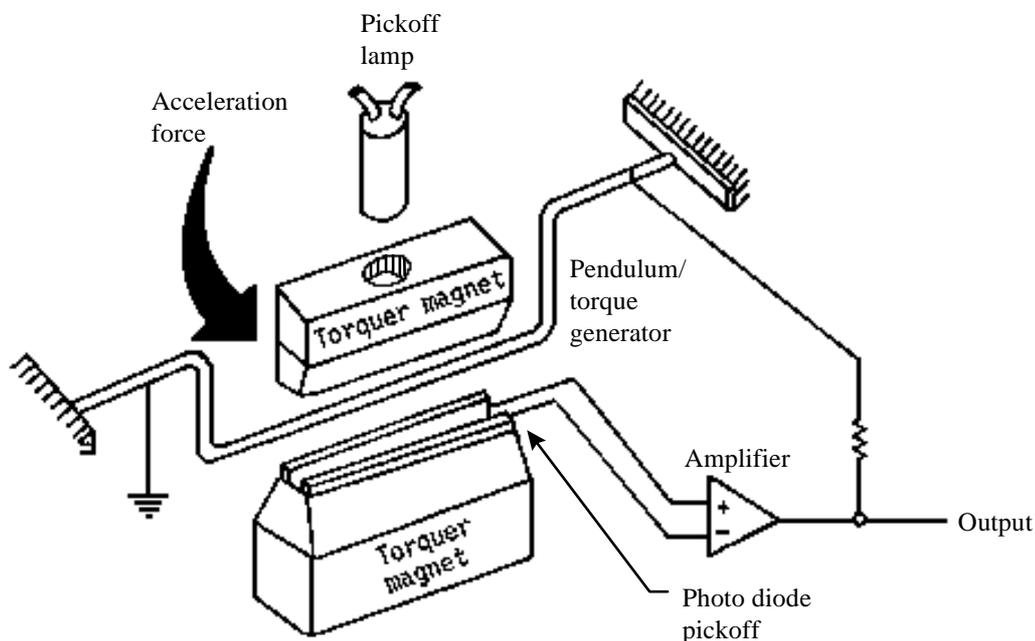


Figure 5-7 - Accelerometer assembly

5.3.3 Performance Data/Limitations

| | | | |
|----------------|--------|--------------------|-------|
| Maximum output | | Power requirement | |
| Normal | ±4 g's | Low (quiescent) | 2.2 W |
| Lateral | ±1 g | High (operational) | 2.4 W |
| Warmup time | 5 min | | |

5.4 GNC/MDM/BUS LOSS MATRIX

a. Purpose

To provide a quick-look reference to determine what guidance, navigation, and control (GNC) equipment or redundancy is lost for given power bus or multiplexer/demultiplexer (MDM) failures (table 5-1).

b. Procedures

The matrices are set up with all GNC redundant components listed along the top in vertical columns. Along the side in horizontal rows are listed all of the power buses and the flight-critical MDM's. An x in a given column and row indicates that a failure of the power bus or MDM labeled in the left column for that row will result in the loss of the equipment or redundancy labeled at the top of that column.

Table 5-I, - RGA/AA MDM/Bus Loss Matrix

| | RGA 1 | | RGA 2 | RGA 3 | RGA 4 | | AA 1 | AA 2 | AA 3 | | AA 4 | SRB | RGA 1 | | SRB | RGA 2 | SRB | RGA 3 | | SRB | RGA 4 | | |
|-------------|-------|-------|-------|-------|-------|-------|------|------|-------|-------|------|-----|-------|-------|-----|-------|-----|-------|-------|-----|-------|-------|-------|
| | RPC A | RPC B | | | RPC A | RPC C | | | RPC A | RPC C | | | RPC B | RPC C | | | | RPC B | RPC C | | RPC A | RPC C | RPC A |
| MNADA1 | | | | | | | | | | | | | | | | | | | | | | | |
| FPC1 | | | | | | | | | | | | | | | | | | | | | | | |
| FLC1 | | | | | | | | | | | | | | | | | | | | | | | |
| FMC1 | | | | | | | | | | | | | | | | | | | | | | | |
| AC1 pC | | | | | | | | | | | | | | | | | | | | | | | |
| AC1 3p FMC1 | | | | | | | | | | | | | | | | | | | | | | | |
| MPC1 | | | | | | | | | | | | | | | | | | | | | | | |
| APC4 | | | | | | | | | | | | | | | | | | | | | | | |
| ALC1 | | | | | | | | | | | | | | | | | | | | | | | |
| APC1 | | | | | | | | | | | | | | | | | | | | | | | |
| O14 | | | | | | | | | | | | | | | | | | | | | | | |
| O14 & A8 | | | | | | | | | | | | | | | | | | | | | | | |
| MNADA2 | | | | | | | | | | | | | | | | | | | | | | | |
| FPC2 | | | | | | | | | | | | | | | | | | | | | | | |
| FLC2 | | | | | | | | | | | | | | | | | | | | | | | |
| FMC2 | | | | | | | | | | | | | | | | | | | | | | | |
| AC2 pC | | | | | | | | | | | | | | | | | | | | | | | |
| AC2 3p FMC2 | | | | | | | | | | | | | | | | | | | | | | | |
| APC5 | | | | | | | | | | | | | | | | | | | | | | | |
| ALC2 | | | | | | | | | | | | | | | | | | | | | | | |
| APC2 | | | | | | | | | | | | | | | | | | | | | | | |
| O15 | | | | | | | | | | | | | | | | | | | | | | | |
| O15 & A8 | | | | | | | | | | | | | | | | | | | | | | | |
| MNADA3 | | | | | | | | | | | | | | | | | | | | | | | |
| FPC3 | | | | | | | | | | | | | | | | | | | | | | | |
| FLC3 | | | | | | | | | | | | | | | | | | | | | | | |
| FMC3 | | | | | | | | | | | | | | | | | | | | | | | |
| AC3 pA | | | | | | | | | | | | | | | | | | | | | | | |
| AC3 pC | | | | | | | | | | | | | | | | | | | | | | | |
| AC3 3p FMC3 | | | | | | | | | | | | | | | | | | | | | | | |
| MPC3 | | | | | | | | | | | | | | | | | | | | | | | |
| APC6 | | | | | | | | | | | | | | | | | | | | | | | |
| ALC3 | | | | | | | | | | | | | | | | | | | | | | | |
| APC3 | | | | | | | | | | | | | | | | | | | | | | | |
| O16 | | | | | | | | | | | | | | | | | | | | | | | |
| O16RJD | | | | | | | | | | | | | | | | | | | | | | | |
| CNTL AB1 | | | | | | | | | | | | | | | | | | | | | | | |
| AB2 | | | | | | | | | | | | | | | | | | | | | | | |
| AB3 | | | | | | | | | | | | | | | | | | | | | | | |
| CNTL BC1 | | | | | | | | | | | | | | | | | | | | | | | |
| BC2 | | | | | | | | | | | | | | | | | | | | | | | |
| BC3 | | | | | | | | | | | | | | | | | | | | | | | |
| CNTL CA1 | | | | | | | | | | | | | | | | | | | | | | | |
| CA2 | | | | | | | | | | | | | | | | | | | | | | | |
| CA3 | | | | | | | | | | | | | | | | | | | | | | | |
| ESS 1BC | | | | | | | | | | | | | | | | | | | | | | | |
| 2CA | | | | | | | | | | | | | | | | | | | | | | | |
| 3AB | | | | | | | | | | | | | | | | | | | | | | | |
| FF 1 | | | | | | | | | | | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | | | | | | | | | | | |
| FA 1 | | | | | | | | | | | | | | | | | | | | | | | |
| 2 | | | | | | | | | | | | | | | | | | | | | | | |
| 3 | | | | | | | | | | | | | | | | | | | | | | | |
| 4 | | | | | | | | | | | | | | | | | | | | | | | |

5.5 ORBITER RATE GYRO ASSEMBLIES SOFTWARE OPERATIONAL DESCRIPTION

5.5.1 General Software Description

The four Orbiter RGA's provide body rates for flight control stability augmentation and ADI rate needle signals. The RGA subsystem operating program (SOP) functions are unit conversion from voltages to deg/s and rate compensation. RGA self-test is carried out in operational sequence (OPS) 9 (prelaunch) and OPS 8 (on orbit). The RGA SOP sends RGA 1, 2, 3, 4 roll, pitch, and yaw rates to redundancy management (RM) for selection filtering (SF) and fault detection, isolation, and reconfiguration (FDIR) (fig. 5-8).

A significant difference between Orbiter RGA software and either SRB RGA or AA software is the addition of SMRD RM. The outputs of SMRD RM will be passed on to the Orbiter RGA FDIR, TLM, crew display, and GAX (fault message).

5.5.2 Orbiter Rate Gyro SMRD RM

The Orbiter RGA SMRD RM's principal purpose is to monitor the Orbiter Spin Motor Rotation Detector Indication (SRMD IND) discrettes and to determine the power fail status of the associated RGAs. If the SMRD RM detects a RGA failure it deselects the failed RGA and removes it from the RM set. SMRD RM terminates if only one RGA is available.

SMRD RM processing will utilize the pitch, yaw, and roll SMRD IND discrettes from each Orbiter RGA. The SMRD determines if the gyro is spinning outside of the specified spin speed by $\pm 2\%$. SMRD FDIR and SF will be performed on each set of RGA discrettes. A SMRD IND discrete of '0' indicates a fail condition.

A significant failure mode that SMRD RM protects against is a dual RGA failure due to loss of power. When a RGA losses power, a rate spike is generated in each of the three axis. This response lasts about 20 seconds and reaches a peak value of approximately 50% (absolute value) of the output range (± 10 deg/sec in pitch and yaw, ± 20 deg/sec in roll). Without SMRD RM the Orbiter RGA RM will select erroneous rates for use by flight control for this dual fail case.

5.5.2.1 Orbiter RGA SMRD FDIR

The basic function of Orbiter RGA SMRD FDIR is to detect a single axis SMRD_IND failure. For each axis, a miscompare counter is incremented when that axis does not agree with the other two. If this miscompare exists for ten cycles then SMRD FDIR declares that SMRD axis failed and sets RGA_SMRD_FAIL to off (0) for that RGA axis. For SMRD_FAIL, '1' indicates that an axis has not failed and is available, and '0' indicates that an axis has failed. SMRD_FAIL can take on any of the following values shown in table 5-II.

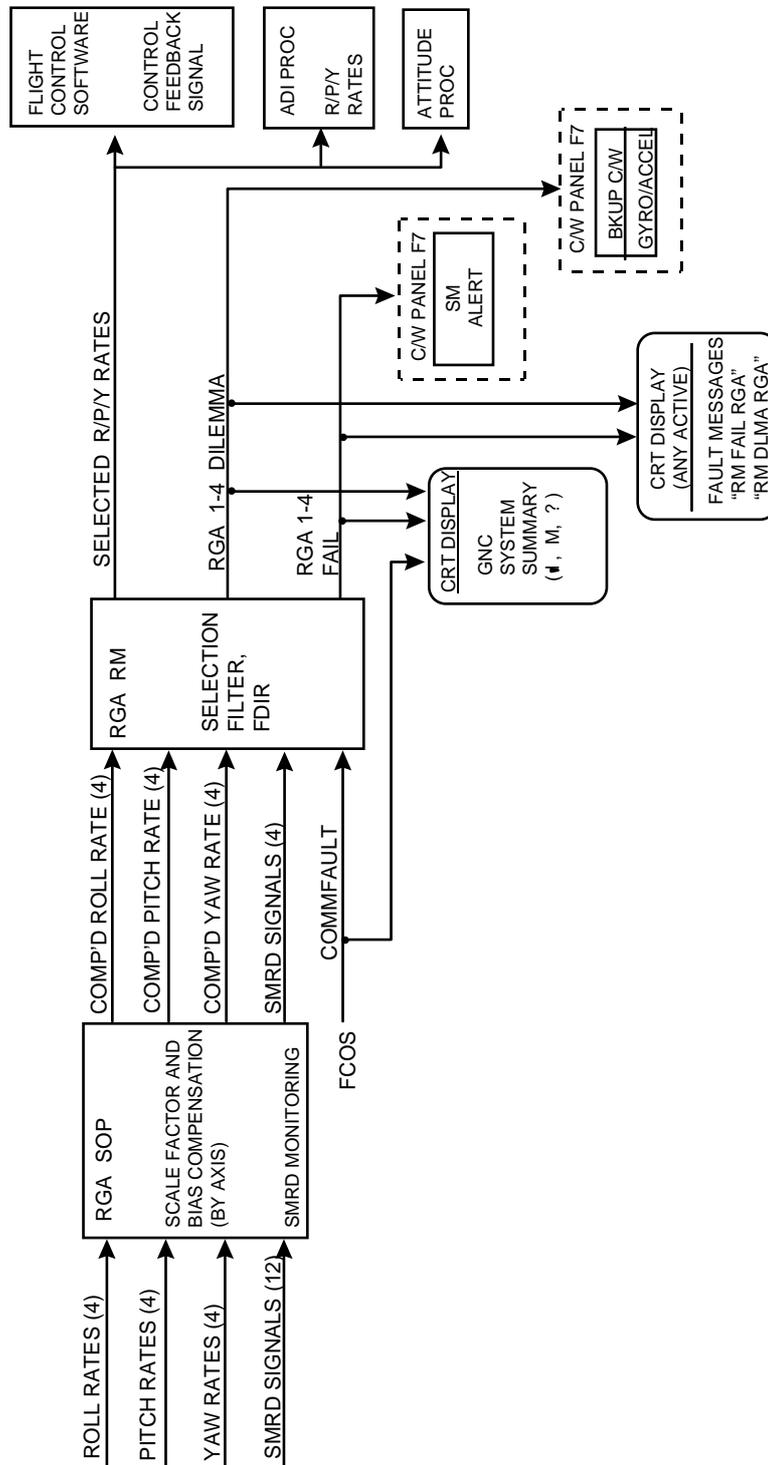


Figure 5-8 - Orbiter rate gyro data flow.

Table 5-II, Possible Orbiter RGA SMRD FAIL combinations

| SMRD_IND (RPY) | Indication |
|------------------------|--|
| '111' | All three SMRD axis are indicating a non-failed status and are available for use in the SMRD SF. |
| '011', '101', or '110' | S single SMRD axis indicates a fail status. Only two SMRD axes are available for use by SMRD SF. |

Once SMRD FDIR has failed a single SMRD axis for a RGA, SMRD FDIR is no longer executed for that RGA. The actions of SMRD FDIR are summarized in Table 5-III.

Table 5-III, Orbiter RGA SMRD FDIR actions

| Available SMRDs | Action |
|-----------------|--|
| 3 | SMRD axis declared failed if it disagrees with the majority vote for ten consecutive cycles. |
| 2, 1, or 0 | RGA SMRD FDIR is terminated. |

5.5.2.2 Orbiter RGA SMRD SF

Orbiter RGA SMRD SF is performed if there are at least two RGAs available, otherwise SRMD RM terminates. At the three level of available SMRDs, SMRD_SF_OUT ('1' = not failed, '0' = failed) is set to the majority vote of the SMRD axes. If a dual (or triple) axis SMRD failure exists for this cycle, the SMRD SF will indicate a power failure since the majority vote is '0'. SMRD FDIR will not be allowed to cycle and determine if this is a transient failure - SMRD RM will immediately fail this RGA.

If Orbiter RGA SMRD FDIR has failed a SMRD axis then SMRD SF is performed at the two level. If either of the remaining two axes are good (SMRD_IND = '1'), then the RGA is considered operational. When both of the remaining SMRD axes have failed, SMRD_SF_OUT is set to fail for the RGA. The output of the SMRD SF is summarized in Table 5-IV.

Table 5-IV, Orbiter RGA SMRD SF output

| Available SMRDs | Output |
|-----------------|--|
| 3 | Majority vote of three SMRDs If SMRD_IND = '111', '011', '101', or '110' then: SMRD_SF_OUT = good ('1') Else: SMRD_SF_OUT = fail ('0') |
| 2 | If either available SMRD_IND is good ('1') then: SMRD_SF_OUT = good ('1') Else: SMRD_SF_OUT = fail ('0') |
| 1 or 0 | Not possible (FDIR termination after first fail) |

It should be noted that SMRD SF is not processed at the one or zero level since FDIR terminates after declaring one SMRD axis failed.

If SMRD SF has identified a power fail, then the RGA is deselected to avoid the associated transient spike in sensed rates (reference Flight Rule A8.1.3-3). The power fail is then annunciated for GAX, TLM, and crew displays.

5.5.2.3 Functional description of Orbiter RGA SMRD RM Module

The following is a summary of the software module GQR_ORB_RGA_SMRD_RM. Refer to figures 5-9.a through 5-9.c with the following text.

On the first pass, SMRD RM is initialized to the not failed case. On the first and each subsequent pass, the number of RGAs available is determined.

A DO loop is set up to process each of the four RGA separately. For each non-comfaulted (non-CF) RGA, the following steps are performed:

1. The SMRD discrettes for this RGA are copied into a local bit string.
2. [Fault Detection, Isolation and Reconfiguration] If none of the SMRD RM parameters for a RGA are failed, then SMRD FDIR is performed. FDIR identifies a faulty SMRD and removes it from the SMRD SF. A counter is incremented for each cycle that the SMRD disagrees with the other SMRDs, otherwise the counter is reset to zero. The faulty SMRD is failed when the counter reaches ten.
3. If at least two RGAs are not failed or deselected, then continue, otherwise SMRD RM terminates for the current cycle and the following step is not performed.
4. [Selection Filter] The SMRD SF outputs a majority vote of three SMRDs, or a logical 'OR' if only two are available. If the SMRD SF indicates that the RGA has suffered a power failure (BIN = '0'), then the RGA is immediately failed and deselected, display and GAX are notified of the power fail, and the associated POWER_FAIL flag is set.

It is possible for some cases to immediately fail a RGA. (Power or SMRD IND transients.) In the event of such a failure, SMRD RM does not latch the power fail/deselect command for a RGA, but continues to monitor the SMRD IND discrettes after it has detected a power fail. Latching occurs only when SMRD RM sends a command to the Orbiter RGA SF to latch the RGA as failed. The fail status may be removed (reselected, unlatched) via an item entry on SPEC 53. The only variable that is latched in SMRD RM is the SMRD FAIL discrete which is latched in SMRD FDIR. Therefore, if a RGA is failed by SMRD RM for a transient problem, it may be reselected.

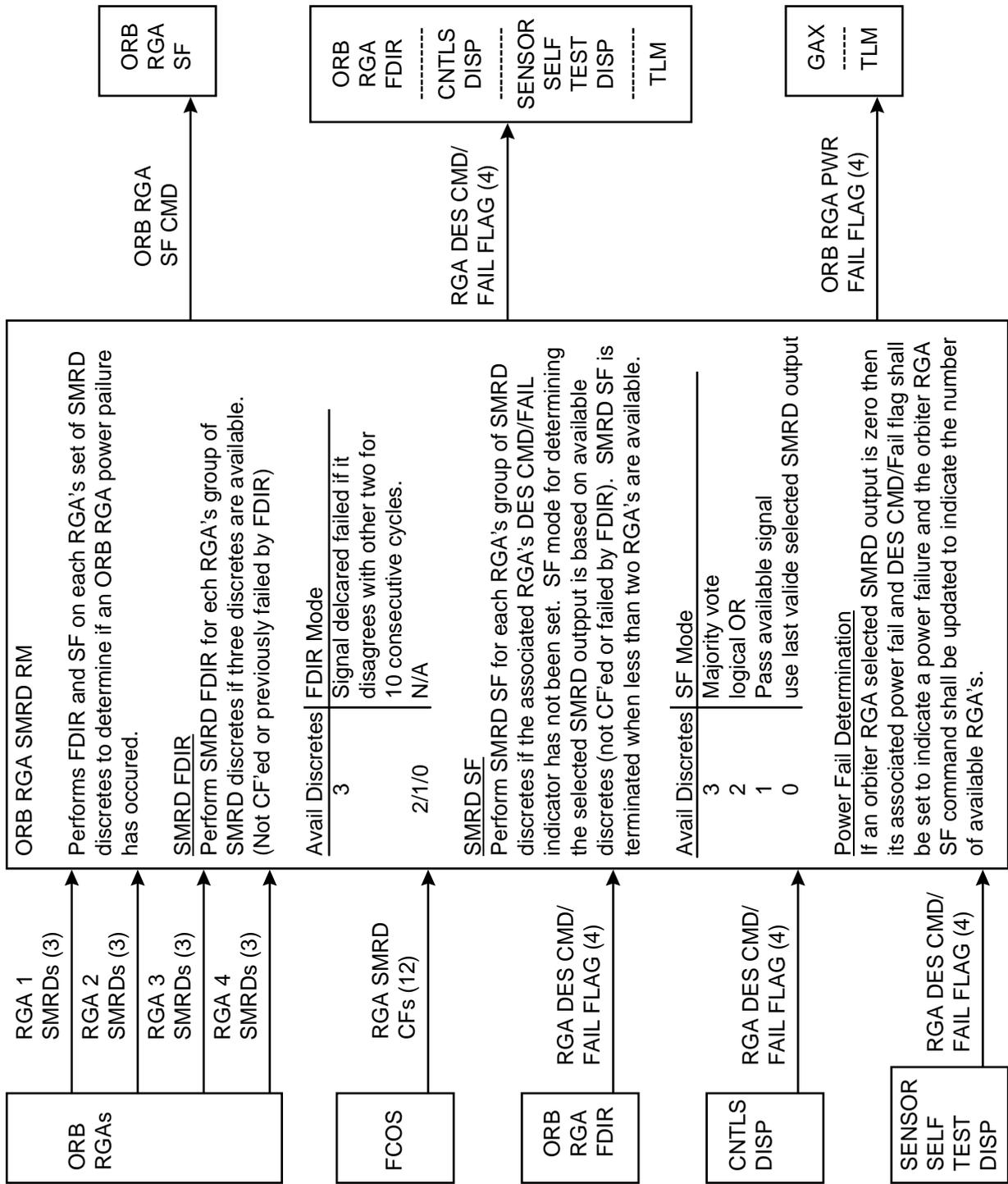


Figure 5-9.a - Orbiter RGA SMRD RM module overview

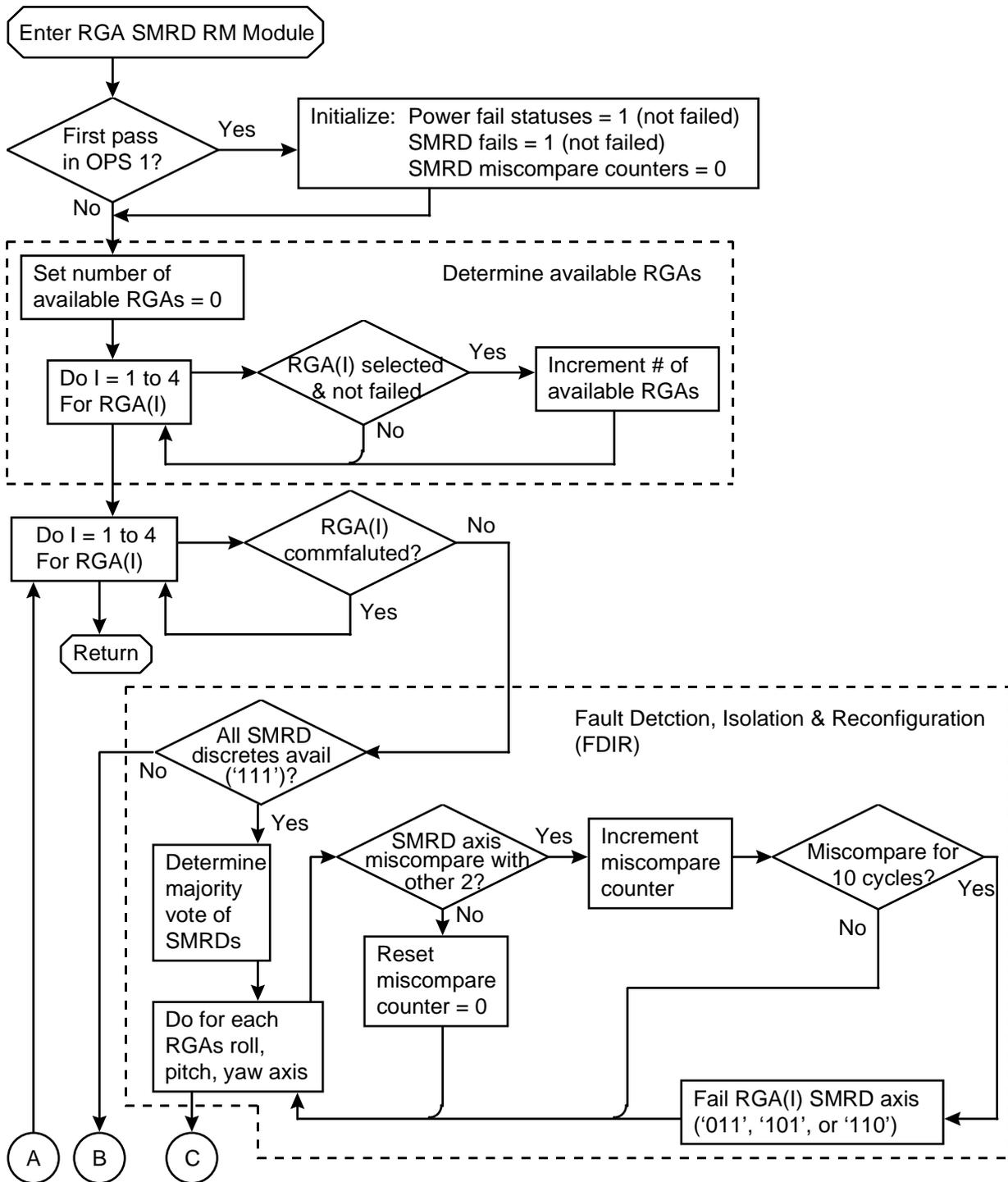


Figure 5-9.b - Orbiter RGA SMRD RM module initialization, RGA availability, and FDIR

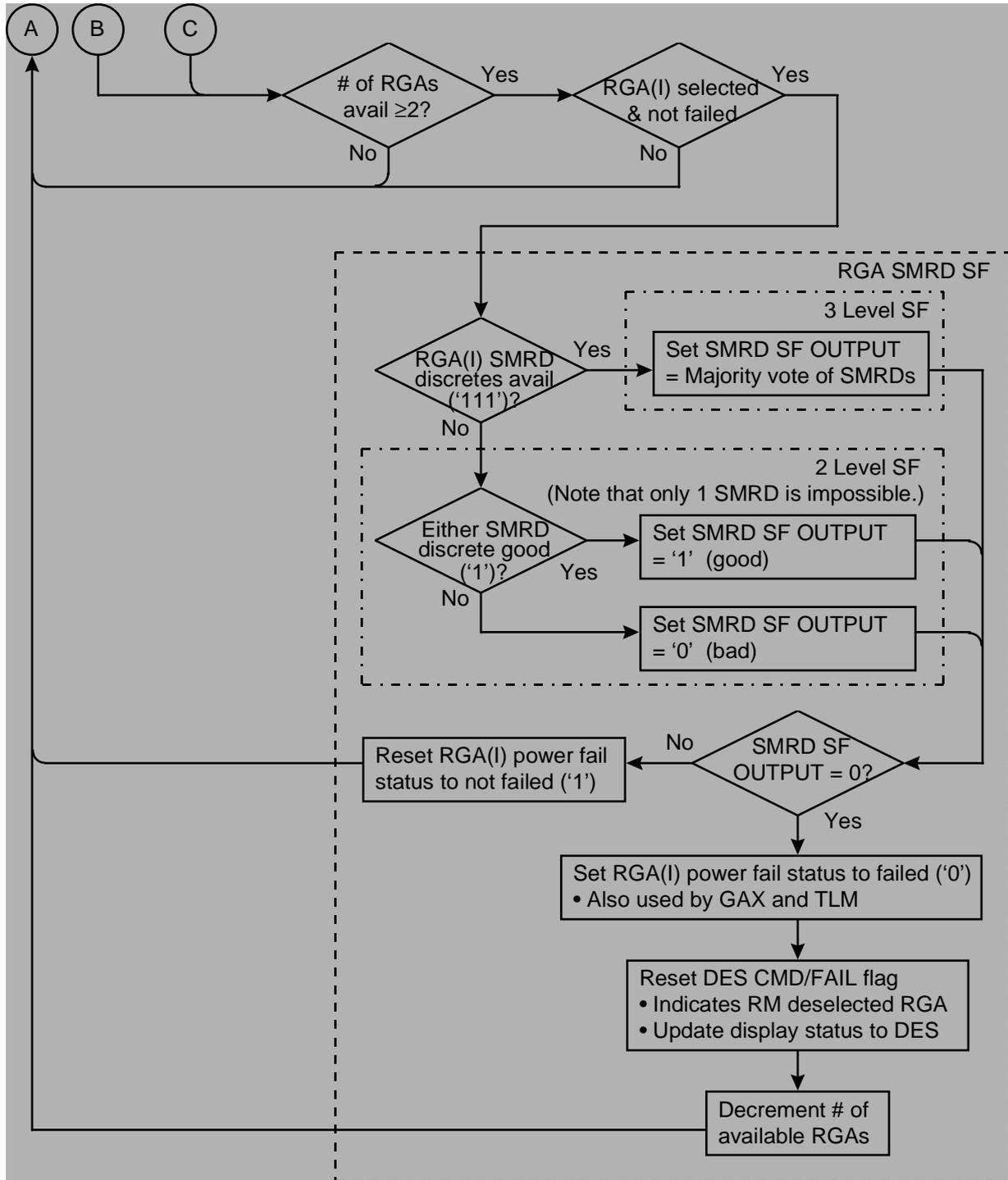


Figure 5-9.c - Orbiter RGA SMRD SF and RM outputs

5.5.3 Orbiter Rate Gyro SOP

The principle function of the Orbiter rate gyro (ORB RG) SOP is to perform data compensation of the measured Orbiter rate gyro roll, pitch, and yaw signals and send the four redundant signals to RM. The compensation function is executed at the 25 Hz rate during MM's 101, 102, 103 (until main engine zero thrust), 304, 305, and 601 through 603; and at the 12.5 Hz rate after main engine zero thrust through MM 106, and MM's 301 through 303. The ORB RG SOP is processed prior to the SF (fig. 5-10).

The ORB RG compensation function shall consist of applying bias and scale factor terms to convert the measured rate data into engineering units (deg/ sec) in floating point format. Provisions shall be made for unique scale factor and bias values for each measured signal. The compensation equation shall be as follows.

$$YC = YM \cdot C + K$$

where:

YC = Compensated RG signal (deg/s)

YM = Measured RG signal (V dc)

C = Compensation scale factor (deg/s/V dc)

K = Compensation bias (deg/s)

Four redundant signals are sent by the rate gyros (roll, pitch, and yaw) to the Orbiter RGA RM. If an out-of-range signal is received, this signal is not used in determining the final selected data.

If an orbiter RGA bias, or scale factor, is observed during operations in flight, an uplink is available to regain function of the LRU. This is implemented per RGA to correct either the C or K terms in the above equation in the PASS MM 301. There is no uplink compensation provided in the BFS. Once a RGA has been corrected and after evaluation of the compensated signal, it may be reselected in the selection filter. For details on implementation of the uplink compensation, see the Sensor AA/RGA Calibration procedure in the GNC Console Handbook.

The ORB RG SOP interfaces are indicated on the functional block diagram in figure 5-10. The input and output interface parameters are presented in tables 5-V and 5-VI. The parameter initialization requirements are shown in table 5-VII.

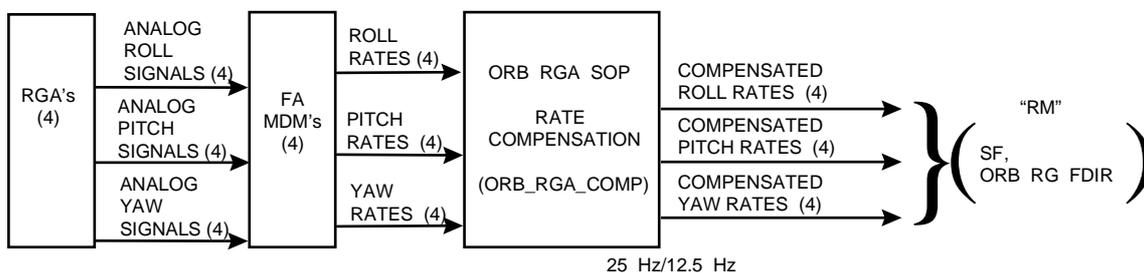


Figure 5-10, Orbiter RG SOP functional block diagram

Table 5-V - ORB RGA SOP Input Parameters

| FSSR Name | MSID | Description | Input source | Signal type | Range (See note A) | Units | Rate (Hz) |
|-----------|-----------|-------------------|--------------|-------------|------------------------------|-----------------|------------|
| PM(1) | V79R1830C | Gyro 1 roll rate | RGA 1 | Fixed pt | ± 5.00 (+40.0, -40.0) | V dc (deg/s) | see note B |
| PM(2) | V79R1835C | Gyro 2 roll rate | RGA 2 | Fixed pt | ± 5.00 (+40.0, -40.0) | V dc (deg/s) | see note B |
| PM(3) | V79R1840C | Gyro 3 roll rate | RGA 3 | Fixed pt | ± 5.00 (+40.0, -40.0) | V dc (deg/s) | see note B |
| PM(4) | V79R1845C | Gyro 4 roll rate | RGA 4 | Fixed pt | ± 5.00 (+40.0, -40.0) | V dc (deg/s) | see note B |
| QM(1) | V79R1831C | Gyro 1 pitch rate | RGA 1 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| QM(2) | V79R1836C | Gyro 2 pitch rate | RGA 2 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| QM(3) | V79R1841C | Gyro 3 pitch rate | RGA 3 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| QM(4) | V79R1846C | Gyro 4 pitch rate | RGA 4 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| RM(1) | V79R1832C | Gyro 1 yaw rate | RGA 1 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| RM(2) | V79R1837C | Gyro 2 yaw rate | RGA 2 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| RM(3) | V79R1842C | Gyro 3 yaw rate | RGA 3 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |
| RM(4) | V79R1847C | Gyro 4 yaw rate | RGA 4 | Fixed pt | ± 5.00 (+20.0, -20.0) | V dc (deg/s) | see note B |

Notes: A) Values in parentheses indicate equivalent engineering units.
 B) 25 Hz in MM 101, 102, 103, 304, 305, 601, 602, 603
 12.5 Hz after zero thrust, MM 104, 105, 106, 301, 302, 303

Table 5-VI - ORB RGA SOP Output Parameters

| FSSR Name | MSID | Description | Output destination | Signal type | Range | Units | Rate (Hz) |
|-----------|-----------|--------------------------|--------------------------|-------------|------------|-------|------------|
| PC(1) | V95R4001C | Compensated roll rate 1 | SF ORB RG FDIR TLM | Float pt | ± 40.0 | Deg/s | see note A |
| PC(2) | V95R4002C | Compensated roll rate 2 | SF ORB RG FDIR TLM | Float pt | ± 40.0 | Deg/s | see note A |
| PC(3) | V95R4003C | Compensated roll rate 3 | SF ORB RG FDIR TLM | Float pt | ± 40.0 | Deg/s | see note A |
| PC(4) | V95R4004C | Compensated roll rate 4 | SF ORB RG FDIR TLM | Float pt | ± 40.0 | Deg/s | see note A |
| QC(1) | V95R4031C | Compensated pitch rate 1 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| QC(2) | V95R4032C | Compensated pitch rate 2 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| QC(3) | V95R4033C | Compensated pitch rate 3 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| QC(4) | V95R4034C | Compensated pitch rate 4 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| RC(1) | V95R4061C | Compensated yaw rate 1 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| RC(2) | V95R4062C | Compensated yaw rate 2 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| RC(3) | V95R4063C | Compensated yaw rate 3 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |
| RC(4) | V95R4064C | Compensated yaw rate 4 | SF ORB RG FDIR TLM | Float pt | ± 20.0 | Deg/s | see note A |

Notes: A) 25 Hz in MM 101, 102, 103, 304, 305, 601, 602, 603
12.5 Hz after zero thrust, MM 104, 105, 106, 301, 302, 303

Table 5-VII - ORB RGA SOP Initialization Parameters

| FSSR Name | MSID | Nomenclature | MC | Value | RI units |
|-----------|-----------|----------------------------------|----------|--------------------------------|------------|
| CP(1) | V97U3418C | Comp scale factor - roll rate 1 | 01 03 | +8.000000E+00 +8.000000E+00 | Deg/s/V dc |
| CP(2) | V97U3419C | Comp scale factor - roll rate 2 | 01 03 | +8.000000E+00 +8.000000E+00 | Deg/s/V dc |
| CP(3) | V97U3420C | Comp scale factor - roll rate 3 | 01 03 | +8.000000E+00 +8.000000E+00 | Deg/s/V dc |
| CP(4) | V97U3421C | Comp scale factor - roll rate 4 | 01 03 | +8.000000E+00 +8.000000E+00 | Deg/s/V dc |
| CQ(1) | V97U3422C | Comp scale factor - pitch rate 1 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CQ(2) | V97U3423C | Comp scale factor - pitch rate 2 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CQ(3) | V97U3424C | Comp scale factor - pitch rate 3 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CQ(4) | V97U3425C | Comp scale factor - pitch rate 4 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CR(1) | V97U3426C | Comp scale factor - yaw rate 1 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CR(2) | V97U3427C | Comp scale factor - yaw rate 2 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CR(3) | V97U3428C | Comp scale factor - yaw rate 3 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |
| CR(4) | V97U3429C | Comp scale factor - yaw rate 4 | 01 03 | +4.000000E+00 +4.000000E+00 | Deg/s/V dc |

Table 5-VII - Concluded

| FSSR Name | MSID | Nomenclature | MC | Value | RI units |
|-----------|-----------|--------------------------|----------|----------------------|----------|
| KPORB(1) | V97U3532C | Comp bias - roll rate 1 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KPORB(2) | V97U3533C | Comp bias - roll rate 2 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KPORB(3) | V97U3534C | Comp bias - roll rate 3 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KPORB(4) | V97U3535C | Comp bias - roll rate 4 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KQ(1) | V97U3583C | Comp bias - pitch rate 1 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KQ(2) | V97U3584C | Comp bias - pitch rate 2 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KQ(3) | V97U3585C | Comp bias - pitch rate 3 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KQ(4) | V97U3586C | Comp bias - pitch rate 4 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KRORB(1) | V97U3589C | Comp bias - yaw rate 1 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KRORB(2) | V97U3590C | Comp bias - yaw rate 2 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KRORB(3) | V97U3591C | Comp bias - yaw rate 3 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |
| KRORB(4) | V97U3592C | Comp bias - yaw rate 4 | 01 03 | +0.0E+00 +0.0E+00 | Deg/s |

5.5.4 Orbiter Rate Gyro RM

The status of each RGA can be found on the GNC SYS SUMM 1 display (fig. 5-11) which receives failure (↓) and COMM fault (CF) (M) indications from RM FDIR. A general block diagram is shown in figure 5-12.

A significant difference between Orbiter RGA RM and either SRB RGA RM or AA RM is the addition of SMRD RM. The orbiter RGA SMRD RM monitors the Orbiter Spin Motor Rotation Indication (SMRD IND) discretes to determine the power fail status of the associated RGA.

5.5.4.1 Orbiter RGA FDIR

The purpose of RGA FDIR is to determine which of the RGA parameters (roll rate, pitch rate, or yaw rate) should be declared failed (fig. 5-12).

Candidates for FDIR are roll, pitch, and yaw rates from RGA 1, 2, 3, and 4. FDIR will detect the first failure only. Once the failed RGA is identified, the failure will be annunciated through a fault summary message (FSM) RM FAIL RGA, an SM alert light and tone, and status indication on GNC SYS SUMM 1 display of a ↓ to indicate which RGA has failed. Since FDIR is terminated after the first failure, no RM DLMA RGA fault message will be generated. In the early design of the space shuttle flight software, three- and two-level FD was included. In order to limit the authority of RM, the decision was made to not implement it. A GAX message, "RM DLMA RGA" still exists, but it can never be activated. CF's from the flight computer operating system (FCOS) will also downmode the RM to the three level and display M on SYS SUMM 1 display. Crew deselect of RGA in OPS-8 will also downmode RM to the three level and display a ↓ on SYS SUMM 1 display. See table 5-VIII, ORB RGA GAX Annunciation.

For each RGA parameter (roll rate, pitch rate, and yaw rate), the following comparisons are made.

| | |
|---------------------|---|
| $ P_1 - P_S > LIM$ | where: |
| | $P_{1(2,3,4)}$ = each RGA rate |
| $ P_2 - P_S > LIM$ | P_S = selected rate from SF |
| | --- = absolute value |
| $ P_3 - P_S > LIM$ | LIM = (selected rate) (scale factor) + bias |
| $ P_4 - P_S > LIM$ | |

An RGA is declared failed if any parameter of that RGA miscompares with the selected rate for 10 consecutive times (10 sec period) in OPS 1 and 6, and 3 times (3 sec period) in OPS 3. FDIR downmodes the failed line replaceable unit (LRU) and then FDIR terminates. No subsequent failures will be detected, only CF's and crew deselections will downmode the SF scheme.

A parameter is downmoded (removed from FDIR, but not declared failed) for a CF, and the CF is annunciated for the particular RGA. If the CF disappears, the parameter would become a selection filter candidate again. Should a CF occur before any failures, the CF would be treated as a first failure by FDIR, such that FDIR would terminate upon recognition of that CF. Should that CF be resolved, FDIR would resume processing.

Table 5-VIII - ORB RGA GAX Annunciation

| MML no. | Nomenclature | Source | OPS avail | N F | ALRM | | MSG ID | Message text | |
|-----------|-----------------------------|----------|-----------|-----|------|----|--------|--------------|-----|
| | | | | | CL | ID | | Major | Min |
| V90X5401X | ORB RGA 1 fail | RGA FDIR | 1,3,6 | | | | | | |
| V90X5501X | ORB RGA 2 fail | RGA FDIR | 1,3,6 | | | | | | |
| V90X5601X | ORB RGA 3 fail | RGA FDIR | 1,3,6 | | | | | | |
| V90X5651X | ORB RGA 4 fail | RGA FDIR | 1,3,6 | | | | | | |
| V94X4733X | RGA 1/2/3/4 FL MSG | | 1,3,6 | | 3 | | 13654 | RM FAIL | RGA |
| V90X5429X | ORB RGA roll rate disagree | RGA FDIR | 1,3,6 | | | | | | |
| V90X5430X | ORB RGA pitch rate disagree | RGA FDIR | 1,3,6 | | | | | | |
| V90X5431X | ORB RGA yaw rate disagree | RGA FDIR | 1,3,6 | | | | | | |
| V94X4791X | ORB RGA DLMA MSG* | RGA FDIR | 1,3,6 | | 2 | 3 | 13854 | RM DLMA | RGA |

| XXXX / XXX / 018 | | | GNC SYS SUMM 1 | | | XX X | | DDD / HH : MM : SS | | | | |
|------------------|-------------|-------------|----------------|------------|------------|------------|----------|--------------------|----------|----------|----------|--|
| | | | | | | | | DDD / HH : MM : SS | | | | |
| <u>RCS</u> | <u>JET</u> | <u>ISOL</u> | <u>SURF</u> | <u>POS</u> | <u>MOM</u> | <u>DPS</u> | <u>1</u> | <u>2</u> | <u>3</u> | <u>4</u> | <u>5</u> | |
| <u>MANF</u> | <u>FAIL</u> | <u>VLV</u> | | | | | | | | | | |
| F1 | XXX | XXS | L OB | XXX.XS | XXS | GPC | S | S | S | S | S | |
| 2 | XXX | XXS | IB | XXX.XS | XXS | MDM FF | S | S | S | S | | |
| 3 | XXX | XXS | R IB | XXX.XS | XXS | FA | S | S | S | S | | |
| 4 | XXX | XXS | OB | XXX.XS | XXS | | | | | | | |
| 5 | XXX | XXS | AIL | XXX.X | | | | | | | | |
| L1 | XXX | XXS | RUD | XXX.X | | FCS CH | 1 | 2 | 3 | 4 | | |
| 2 | XXX | XXS | SPD BRK | XXX.X | | | S | S | S | S | | |
| 3 | XXX | XXS | BDY FLP | XXX.X | | | | | | | | |
| 4 | XXX | XXS | | XXXXXX | | NAV | 1 | 2 | 3 | 4 | | |
| 5 | XXX | XXS | | | | IMU | S | S | S | | | |
| R1 | XXX | XXS | CNTRLR | 1 2 3 | | ACC | S | S | S | S | | |
| 2 | XXX | XXS | RHC L | S S S | | RGA | S | S | S | S | | |
| 3 | XXX | XXS | R | S S S | | TAC | S | S | S | | | |
| 4 | XXX | XXS | A | S S S | | MLS | S | S | S | | | |
| 5 | XXX | XXS | THC L | S S S | | ADTA | S | S | S | S | | |
| | | | A | S S S | | | | | | | | |
| | | | SBTC L | S S S | | | | | | | | |
| | | | R | S S S | | | | | | | | |

Figure 5-11, GNC SYS SUMM 1 display (PFS)

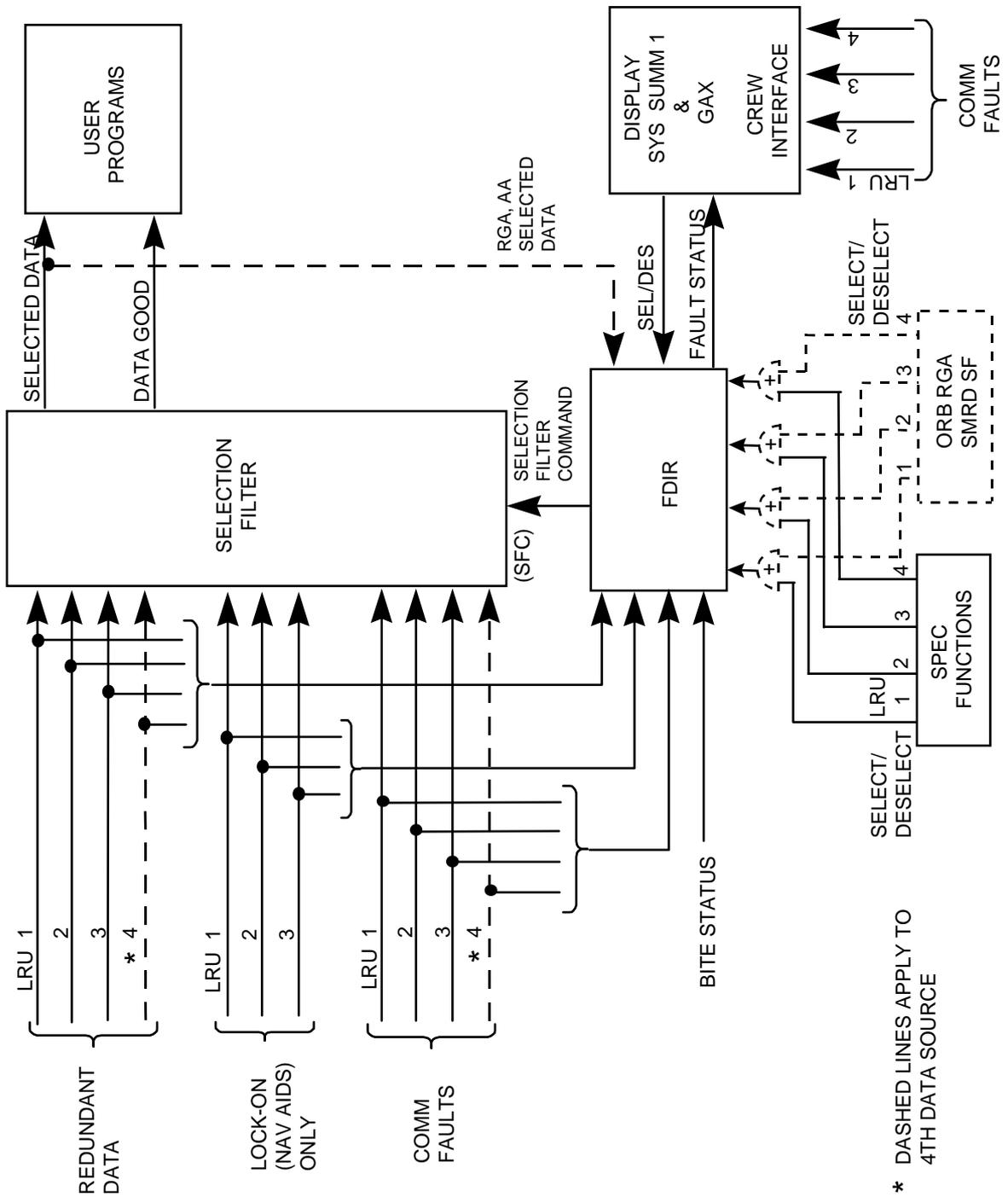


Figure 5-12, General RM block diagram

5.5.4.2 SF

The purpose of the RGA SF is to select single values of roll, pitch, and yaw rates to provide to the users. Candidates are the rates from RGA 1, 2, 3, and 4.

An RGA is downmoded in the SF if a failure is indicated by the SF command (SFC) from FDIR. SF is also downmoded for CF's. If the CF disappears, the SF will upmode.

There are five different SF functional modes depending on the usable level of input data redundancy. The interchangeable midvalue select (IMVS) filter mode shall provide signal selection from four valid input signals and shall provide protection from any single failure and from dual null failures. Whenever the true input value is greater than the I-load null threshold value, a non-null signal will be selected. When the true input cycles above and below the null threshold level, the switching of the output between the null and true values will be minimized. See figure 5-13, IMVS flow diagram. The midvalue select (MVS) mode provides a datum output value equal to the input value that lies between the other two values when three valid inputs are available. See figure 5-14 MVS routine flow diagram. The average (AVE) mode provides an output datum value that is the arithmetical average of two values when two valid inputs are available. See figure 5-15, AVE and tracking test flow diagram. The primary avionics system software (PASS) mode provides a datum output value equal to a single datum input value. The no-available-sensor (NAS) mode does not provide a new datum output value; i.e., the output value shall remain unchanged from its previous valid value. (This action results from the fact that the SF output common data pool (COMPOOL) location is fixed until updated. Failure to output a new value leaves the old value in the COMPOOL location.)

The ORB RGA FDIR input and output interface parameters are presented in tables 5-IX and 5-X. The FDIR parameter initialization requirements are shown in table 5-XI. The ORB RGA SF input and output interface parameters are shown in table 5-XII.

5.5.5 Self-Test Operations

a. Ops 8 On-Orbit Self-Test Procedures

The self-test procedure for the Orbiter RGA's are executed from specialist function (SPEC) 041, RGA/ADTA/RCS. The SPEC display and its functional operations are shown in section 5.13, Sensor Self-Test. The vehicle rates are nulled (<0.02 deg/s) using the RCS JETS VERN, and minimum vehicle accelerations are established by placing the vehicle in free drift (DAP: FREE).

TABLE 5-IX - ORB RGA FDIR INPUT PARAMETERS

| MSID | Level B no. | FSSR name | Nomenclature | Source | Range | Units |
|------------|-------------|----------------|--------------------------|------------|-------|-------|
| V95R4001C | G06907 | PC(1) | Compensated roll rate 1 | ORB RG SOP | ±40 | Deg/s |
| V95R4002C | G06908 | PC(2) | Compensated roll rate 2 | ORB RG SOP | ±40 | Deg/s |
| V95R4003C | G06909 | PC(3) | Compensated roll rate 3 | ORB RG SOP | ±40 | Deg/s |
| V95R4004C | G47120 | PC(4) | Compensated roll rate 4 | ORB RG SOP | ±40 | Deg/s |
| V95R4031C | G06904 | QC(1) | Compensated pitch rate 1 | ORB RG SOP | ±20 | Deg/s |
| V95R4032C | G06905 | QC(2) | Compensated pitch rate 2 | ORB RG SOP | ±20 | Deg/s |
| V95R4033C | G06906 | QC(3) | Compensated pitch rate 3 | ORB RG SOP | ±20 | Deg/s |
| V95R4034C | G47119 | QC(4) | Compensated pitch rate 4 | ORB RG SOP | ±20 | Deg/s |
| V95R4061C | G06910 | RC(1) | Compensated yaw rate 1 | ORB RG SOP | ±20 | Deg/s |
| V95R4062C | G06911 | RC(2) | Compensated yaw rate 2 | ORB RG SOP | ±20 | Deg/s |
| V95R4063C | G06912 | RC(3) | Compensated yaw rate 3 | ORB RG SOP | ±20 | Deg/s |
| V95R4064C | G47121 | RC(4) | Compensated yaw rate 4 | ORB RG SOP | ±20 | Deg/s |
| V90R5301CA | G14443 | P_ORB | Selected RGA roll rate | SF | ±40 | Deg/s |
| V90R5321CA | G14442 | Q_ORB | Selected RGA pitch rate | SF | ±20 | Deg/s |
| V90R5341CA | G14444 | R_ORB | Selected RGA yaw rate | SF | ±20 | Deg/s |
| V91X3700X | G46223 | RGA_ROLL_CF(1) | RGA 1 roll COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3701X | G46220 | RGA_PIT_CF(1) | RGA 1 pitch COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3702X | G46225 | RGA_YAW_CF(1) | RGA 1 yaw COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3730X | G46224 | RGA_ROLL_CF(2) | RGA 2 roll COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3731X | G46221 | RGA_PIT_CF(2) | RGA 2 pitch COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3732X | G46227 | RGA_YAW_CF(2) | RGA 2 yaw COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3760X | G46225 | RGA_ROLL_CF(3) | RGA 3 roll COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3761X | G46222 | RGA_PIT_CF(3) | RGA 3 pitch COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3762X | G46284 | RGA_YAW_CF(3) | RGA 3 yaw COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3790X | G15610 | RGA_ROLL_CF(4) | RGA 4 roll COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3791X | G15609 | RGA_PIT_CF(4) | RGA 4 pitch COMM fault | FCOS | 0, 1 | Deg/s |
| V91X3792X | G15613 | RGA_YAW_CF(4) | RGA 4 yaw COMM fault | FCOS | 0, 1 | Deg/s |
| V93X4516X | C03028 | RGA | 1 DES CMD | V5.1.4 | 0, 1 | Deg/s |
| V93X4518X | C03029 | RGA | 2 DES CMD | V5.1.4 | 0, 1 | Deg/s |
| V93X4520X | C03030 | RGA | 3 DES CMD | V5.1.4 | 0, 1 | Deg/s |
| V93X4522X | C03051 | RGA | 4 DES CMD | V5.1.4 | 0, 1 | Deg/s |

TABLE 5-X - ORB RGA FDIR Output Parameters

| MSID | Nomenclature | Destination | Range | Units |
|-----------|-----------------------------|-------------|-------|-------|
| V90X5401X | ORB RGA 1 fail | GAX,TLM | 0, 1 | |
| V90X5501X | ORB RGA 2 fail | GAX,TLM | 0, 1 | |
| V90X5601X | ORB RGA 3 fail | GAX,TLM | 0, 1 | |
| V90X5651X | ORB RGA 4 fail | GAX,TLM | 0, 1 | |
| V90U5660X | ORB RGA SF command | SF | 1, 4 | |
| V90X5429X | ORB RGA roll rate disagree | GAX,TLM | 0, 1 | |
| V90X5430X | ORB RGA pitch rate disagree | GAX,TLM | 0, 1 | |
| V90X5431X | ORB RGA yaw rate disagree | GAX,TLM | 0, 1 | |
| V90X5417X | ORB RGA 1 pitch fail | TLM | | |
| V90X5418X | ORB RGA 1 yaw fail | JLM | | |
| V90X5419X | ORB RGA 1 roll fail | TLM | | |
| V90X5420X | ORB RGA 2 pitch fail | TLM | | |
| V90X5421X | ORB RGA 2 yaw fail | TLM | | |
| V90X5422X | ORB RGA 2 roll fail | TLM | | |
| V90X5423X | ORB RGA 3 pitch fail | TLM | | |
| V90X5424X | ORB RGA 3 yaw fail | TLM | | |
| V90X5425X | ORB RGA 3 roll fail | TLM | | |
| V90X5426X | ORB RGA 4 pitch fail | TLM | | |
| V90X5427X | ORB RGA 4 yaw fail | TLM | | |
| V90X5428X | ORB RGA 4 roll fail | TLM | | |
| V93X4516X | RGA 1 DES CMD | 6.46 | 0, 1 | |
| V93X4518X | RGA 2 DES CND | 6.46 | 0, 1 | |
| V93X4520X | RGA 3 DES CMD | 6.46 | 0, 1 | |
| V93X4522X | RGA 4 DES CMD | 6.46 | 0, 1 | |

TABLE 5-XI- ORB RGA FDIR Initialization Parameters

| MSID | FSSR name | Nomenclature | MC | Value | RI units |
|-----------|-----------|------------------------------------|----------|-----------------------|----------|
| V97U3021C | CON(01) | ORG RGA roll rate limit constant | 01 03 | +8.0E-01 +1.19E-00 | Deg/s |
| V97U3022C | CON(02) | ORG RGA pitch rate limit constant | 01 03 | +5.5E-01 +6.5E-01 | Deg/s |
| V97U3023C | CON(03) | ORG RGA yaw rate limit constant | 01 03 | +6.3E-01 +6.5E-01 | Deg/s |
| V97U3053C | LSF(01) | ORG RGA roll rate limit SC factor | 01 03 | +8.0E+00 +8.9E+00 | PCT |
| V97U3054C | LSF(02) | ORG RGA pitch rate limit SC factor | 01 03 | +6.5E+00 +6.0E+00 | PCT |
| V97U3055C | LSF(03) | ORG RGA yaw rate limit SC factor | 01 03 | +6.5E+00 +6.0E+00 | PCT |
| V97U3079C | N_ORB_RGA | ORG RGA N counter | 01 03 | +10 +3 | ND |
| V96U8591C | PQC | ORG RGA roll QMVS constant | 01 03 | +4.0E-01 +5.95E-01 | Deg/s |
| V96U8592C | QQC | ORG RGA pitch QMVS constant | 01 03 | +2.2E-01 +3.22E-01 | Deg/s |
| V96U8593C | RQC | ORG RGA yaw QMVS constant | 01 03 | +2.2E-01 +2.2E-01 | Deg/s |

TABLE 5-XII.- ORB RGA SF Input/Output Parameters

(a) Input functional parameters for the selection filtering module

| MSID | FSSR Nomenclature | Nomenclature | Source | Range | Units |
|-----------|-------------------|--------------------------|-------------|-------|-------|
| V90U5660C | | ORB RGA SF command | ORB RG FDIR | 1,4 | |
| V91X3700X | RGA_ROLL_CF(1) | RGA 1 roll COMM fault | FCOS | 0,1 | |
| V91X3701X | RGA_PIT_CF(1) | RGA 1 pitch COMM fault | FCOS | 0,1 | |
| V91X3702X | RGA_YAW_CF(1) | RGA 1 yaw COMM fault | FCOS | 0,1 | |
| V91X3730X | RGA_ROLL_CF(2) | RGA 2 roll COMM fault | FCOS | 0,1 | |
| V91X3731X | RGA_PIT_CF(2) | RGA 2 pitch COMM fault | FCOS | 0,1 | |
| V91X3732X | RGA_YAW_CF(2) | RGA 2 yaw COMM fault | FCOS | 0,1 | |
| V91X3760X | RGA_ROLL_CF(3) | RGA 3 roll COMM fault | FCOS | 0,1 | |
| V91X3761X | RGA_PIT_CF(3) | RGA 3 pitch COMM fault | FCOS | 0,1 | |
| V91X3762X | RGA_YAW_CF(3) | RGA 3 yaw COMM fault | FCOS | 0,1 | |
| V91X3790X | RGA_ROLL_CF(4) | RGA 4 roll COMM fault | FCOS | 0,1 | |
| V91X3791X | RGA_PIT_CF(4) | RGA 4 pitch COMM fault | FCOS | 0,1 | |
| V91X3792X | RGA_YAW_CF(4) | RGA 4 yaw COMM fault | FCOS | 0,1 | |
| V95R4001C | PC(1) | Compensated roll rate 1 | ORB RG SOP | ±40 | Deg/s |
| V95R4002C | PC(2) | Compensated roll rate 2 | ORB RG SOP | ±40 | Deg/s |
| V95R4003C | PC(3) | Compensated roll rate 3 | ORB RG SOP | ±40 | Deg/s |
| V95R4004C | PC(4) | Compensated roll rate 4 | ORB RG SOP | ±40 | Deg/s |
| V95R4031C | QC(1) | Compensated pitch rate 1 | ORB RG SOP | ±20 | Deg/s |
| V95R4032C | QC(2) | Compensated pitch rate 2 | ORB RG SOP | ±20 | Deg/s |
| V95R4033C | QC(3) | Compensated pitch rate 3 | ORB RG SOP | ±20 | Deg/s |
| V95R4034C | QC(4) | Compensated pitch rate 4 | ORB RG SOP | ±20 | Deg/s |
| V95R4061C | RC(1) | Compensated yaw rate 1 | ORB RG SOP | ±20 | Deg/s |
| V95R4062C | RC(2) | Compensated yaw rate 2 | ORB RG SOP | ±20 | Deg/s |
| V95R4063C | RC(3) | Compensated yaw rate 3 | ORB RG SOP | ±20 | Deg/s |
| V95R4064C | RC(4) | Compensated yaw rate 4 | ORB RG SOP | ±20 | Deg/s |

(b) Output functional parameters form the selection filtering module

| MSID | FSSR Nomenclature | Nomenclature | Destination | Range | Units |
|------------|-------------------|-------------------------|--|-------|-------|
| V90R5301CA | P_ORB | Selected RGA roll rate | G/C STEER, A/E ATT PROC, E/L ADI PROC, AEROJET DAP, ASC ADI PROC, ASC DAP, TRANS DAP, GRTLS DAP, ET SEP SEQ, SRB SEP SEQ, STAR TRCK SOP, GN&C/SM-PL IF, ORB RG FDIR, TLM | ±40 | Deg/s |
| V90R5321CA | Q_ORB | Selected RGA pitch rate | AEROJET DAP,A/E ATT PROC, ASC DAP,TRANS DAP,TAEM UPP, A/L UPP,ASC ADI PROC,RTLS UPP,GRTLS DAP, GRTLS TAEM GUID, E/L ADI PROC, ET SEP SEQ, SRB SEP SEQ, STAR TRCK SOP,G/C STEER,GN&C/SM-PL IF, ORB RG DFIR, TLM | ±20 | Deg/s |
| V90R5341CA | R_ORB | Selected RGA yaw rate | G/C STEER, ASC ADI PROC, E/L ADI PROC, TRANS DAP, A/E ATT PROC, GRTLS DAP, ET SEP SEQ,SRB SEP SEQ, ASC DAP, AEROJET DAP, STAR TRCK SOP,GN&C/SM-PL IF, ORB RG FDIR, TLM | ±20 | Deg/s |

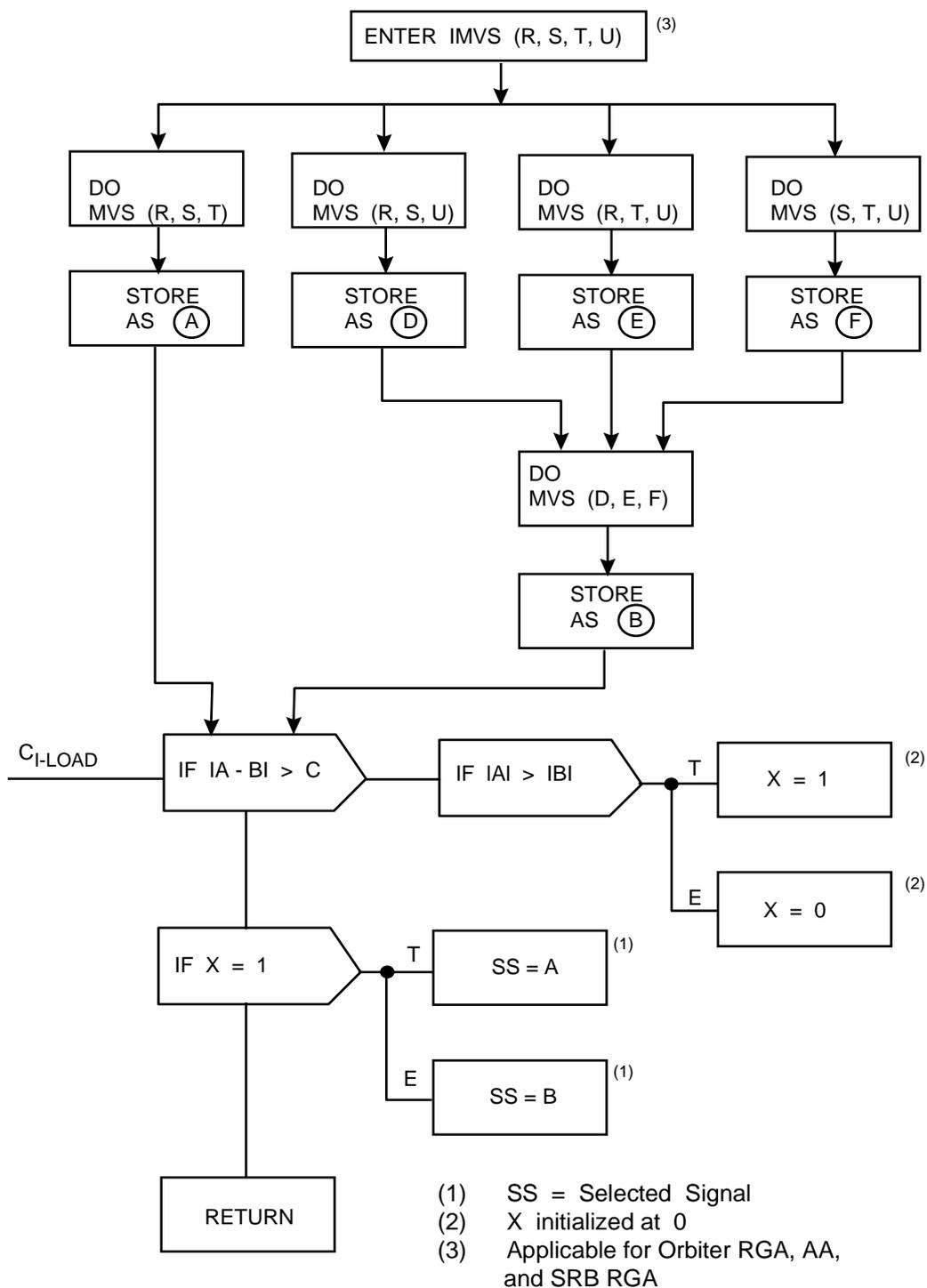


Figure 5-13 - IMVS flow diagram

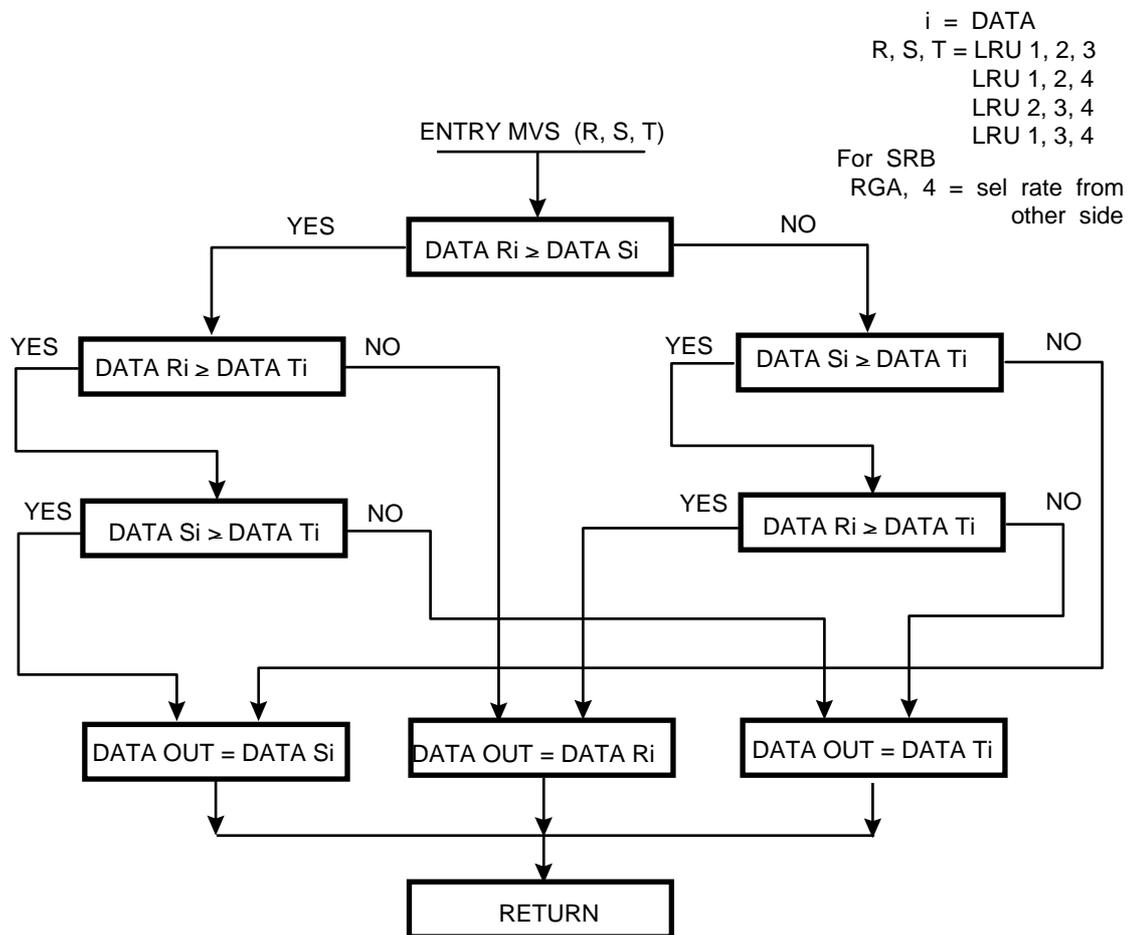


Figure 5-14 - MVS routine flow diagram

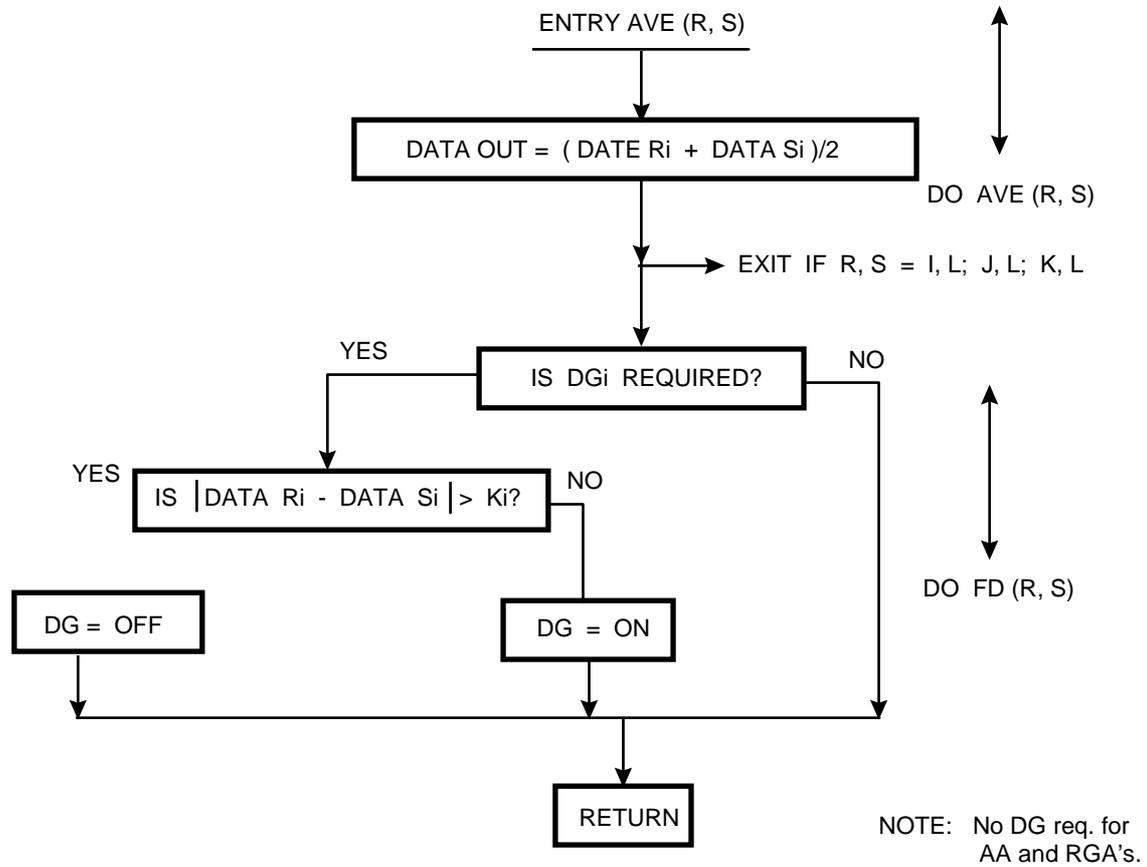


Figure 5-15 - AVE and tracking test flow diagram

Next, the RGA bias data are verified by examining the sensed rates shown on the display SPEC 41. The bias should be less than the values listed below.

| | |
|-----------|------------|
| Roll (R) | ±0.4 deg/s |
| Pitch (P) | ±0.2 deg/s |
| Yaw (Y) | ±0.2 deg/s |

If bias data are outside these limits, the option exists to deselect the appropriate RGA unit from the RM.

After the bias test has been accomplished, the high and low test is performed from SPEC 41. The rate data are checked and verified within the limits listed below.

| | <u>High test</u> | <u>Low test</u> |
|-----------|------------------|------------------|
| Roll (R) | 18.88 to 21.12 | -21.12 to -18.88 |
| Pitch (P) | 9.44 to 10.56 | -10.56 to -9.44 |
| Yaw (Y) | 9.44 to 10.56 | -10.56 to -9.44 |

If the data are not within the proper limits, deselection of the appropriate RGA is recommended.

The following parameter status indicators (PSI's) displayed beside the appropriate data indicates the following.

- ↑ Data above limit
- ↓ Data below limit
- M Data missing (COMM fault)

SMRD indication will be displayed in STAT column if RGA spin motor is not at sync speed.

b. OPS 9 Prelaunch Self-Test Procedure

The self-test procedure for Orbiter RGA's, prelaunch, are executed from SPEC 102, RCS/RGA/ADTA TEST. The SPEC display and its functional operations are shown in section 5.13.4. RGA's cannot be deselected from RM, but can be enabled/disabled from self-test. The high and low self-test start initiates self-test on only those LRU's enabled. The auto sequence initiates both high and low test on all sensors. The self-test limits and operation are the same as in OPS 8.

5.5.6 BAD SELECTED RGA RATE

This section addresses the problem of bad selected RGA rate feedback in the TRANS DAP, and in particular as a result of failures during powered flight. The ATT PROC software module and the hip-pocket procedures which have been developed for dealing with these failure scenarios are discussed.

5.5.6.1 Attitude Processor (ATT PROC) Wbias

The ATT PROC module (ATT_PROC_INNER) processes selected RGA rate data at 6.25 Hz. These rates are integrated to form an attitude quaternion, which is compared to the IMU derived attitude, or the Quaternion-to-Body Inertial (QBI) attitude quaternion. The difference between these two quaternions defines an error quaternion (Qincr). If Qincr is nonzero, then the rate correction term, Wbias (OMEGA BIAS) builds up. This Wbias term builds up slowly (at a rate of .01 deg/sec/sec) during this time if a bias remains in the system causing Qincr to be nonzero. (It should be noted that ATT_PROC_INNER does not reside within the DAP, but rather the various DAPs are users of this module.) The attitude processor

was designed to detect rate gyro assembly (RGA) biases using IMU data and to generate Wbias, which is automatically applied to the selected RGA rate data for use only by the TRANS DAP and to calculate QBI between the IMU once-per-second updates. The Wbias is used to calculate a corrected rate, RG_CORR, which is what the TRANS DAP uses for rate feedback ($RG_CORR = RATE_GYRO_READ - Wbias$). In this equation, RATE_GYRO_READ is the selected RGA rate feedback. Once the corrected rate data from RG_CORR agrees with the IMU data, Wbias will stop incrementing. Any changes that cause RG_CORR to disagree with the IMU data (IMU problems and/or RGA problems) will cause Wbias to change in order to make RG_CORR consistent with the IMU data. Aside from any additional sensor failures, if the vehicle has a real rate, then that will be reflected in the IMU data and in the RATE_GYRO_READ term. Wbias will change only to make RG_CORR agree with the rates from the IMU data. Thus, with bad selected RGA rate, RATE_GYRO_READ will deliver an incorrect vehicle rate, and Wbias will bias it to compensate for the rate discrepancy.

5.5.6.2 Example Case

Assume that two RGAs are failed and a bias is induced on one of the two remaining RGA's (three failure case). The IMU derived QBI will disagree with the RGA derived attitude quaternion, and Wbias begins building at .01 deg/sec/sec. When RC_CORR (the sum of Wbias and the selected RGA rate feedback) agrees with the IMU data, the system has been compensated and Wbias remains constant. Because the rate at which the correction term can be generated is limited to .01 deg/sec/sec, its effectiveness in dealing with large RGA biases is dependent upon the amount of time there is for Wbias to build prior to being used in the TRANS DAP (or, prior to MECO). This complicates the crew procedures for dealing with large biases (once in the TRANS DAP post MECO) in cases where the bad RGA cannot be identified, or when it cannot be immediately deselected.

For a specific case let us assume that only RGA's 1 and 4 are available, and that the selected RGA Roll rate is biased by about 4.5 deg/sec (RGA 4 biased +9.0 deg/sec, RGA 1 good), for about 8 minutes of powered flight. (Assume that the crew is unable to deselect RGA 4 due to other systems failures and/or crew workload.) Ascent control is maintained because the IMU data updates the indicated vehicle rate data at 1.04 Hz, and because the bias is in the Roll axis. Attitude and rate oscillations are seen throughout powered flight (plots show a ratcheting signature), but the ASCENT DAP does not allow the apparent rate to manifest itself because of the correcting IMU data every second. The calculation of Wbias is a function of the magnitude of the rate bias (RGA Roll bias in this case) for bias magnitudes up to 0.1 deg/sec. For larger biases (as in this case), Wbias simply builds at .01 deg/sec/sec. For this case Wbias increases to a value of 4.5 deg/sec (0.01 deg/sec/sec x 480 sec maximum) before MECO. The actual vehicle rate at MECO is about 0 deg/sec, so with RGA 4 still selected, the selected RGA Roll rate is about 4.5 deg/sec (average of RGA 4 and RGA 1). The TRANS DAP compares the corrected RGA rate (RG_CORR) with the commanded rate to form a rate error command. Immediately after MECO, RG_CORR is about zero, since $RG_CORR = RATE_GYRO_READ - Wbias$. This is because there is sufficient time for Wbias to become equal to the average RGA bias prior to MECO, and fully compensate the RATE_GYRO_READ which was biased. There is no rate buildup when first entering the TRANS DAP (even if the DAP is not in INRTL PULSE in Roll), because the system has compensated for the RGA bias. But, if RGA 4 is deselected, a rate will build up since Wbias is nonzero.

In this case where RGA 4 is not deselected (for whatever reason) during powered flight, the system is compensated and there is no problem. Recall that for this case there is no rate build up immediately post MECO is because Wbias had sufficient time to compensate for the RGA bias. This will not always be the case. Any time the RGA bias goes into the system late in powered flight, Wbias will have less time to build up. If the RGA bias is large enough, then RG_CORR will be nonzero when the TRANS DAP is entered at MECO, and without going free drift (or INRTL PULSE), the vehicle rate will build up.

5.5.6.3 Suggested Hip-Pocket Procedures

There are two basic cases of interest here. Because this is a three failure scenario, it is prudent to keep the procedures as simple as possible so that they may be applied to variations of the two scenarios. One scenario is that of a fairly nominal ascent with respect to performance and trajectory, so that an OMS-1 burn is not required. In this case, there will be more time post ET SEP before AUTO DAP (AUTO or INRTL DISC) is required, and thus we can wait for Wbias to go to zero. The other case is where either an OMS-1 is required, or there is other pressing reason to get back to AUTO DAP after ET sep. These scenarios are discussed separately here. It should be noted that the ADI rate pointers are driven with selected RGA data, not RG_CORR data. This means that with a good RGA and an RGA failed to null for example, the rate pointers will still indicate when the actual vehicle rates are zero.

I) Procedure for No OMS-I Required

NOTE: Always note the total time that the failure is in the system, causing a bad selected rate.

Pre-MECO

Bad RGA - DES (SPEC 53)

ET SEP – MAN

Post-MECO, No OMS-1 Required

ET SEP - MAN (pre-MECO if possible)

DAP - INRTL PULSE in affected axis

Bad RGA - DES (SPEC 53, pre-MECO if possible)

NULL vehicle rates

- * Use rate pointer on the ADI as a NULL indicator. When *
- * actual vehicle rate is zero, the pointer will be zeroed. *

ET SEP - AUTO, when rates equal zero (or, MCC confirms less than 0.7 all axes)

DAP - AUTO, after waiting an elapsed time equal to the amount of time that the RGA bias was in the system.

Once the actual vehicle rate is zero and the bad RGA is deselected, Wbias will decrease to zero at .01 deg/sec/sec, to make RG_CORR agree with the IMU data. It will take the same amount of time for Wbias to decrease to zero that it took to build to its previous value. Once Wbias is zeroed, it is safe go back to AUTO DAP without building rates. If possible, the configuration back to AUTO should be delayed a little longer to ensure that Wbias is zero. Also, note that this procedure assumes that good IMU data was available throughout powered flight. If this is not the case, Wbias will be reflecting RGA biases and IMU problems. In any case, if a rate builds on the vehicle when going to back AUTO, go back to INRTL PULSE and assume that Wbias is not zero.

II) Procedure for OMS-I Required

NOTE: Always note the total time that the failure is in the system, causing a bad selected rate.

Pre-MECO

Bad RGA - DES (SPEC 53)

ET SEP - MAN

Post-MECO, OHS-1 Required

ET SEP - MAN (pre-MECO if possible)

DAP - INRTL PULSE in affected axis

Bad RGA - DES (SPEC 53, pre-MECO if possible)

NULL vehicle rates

- * Use rate pointer on the ADI as a NULL indicator. When *
- * actual vehicle rate is zero, the pointer will be zeroed. *

IMU 2,3 - DES (SPEC 51)

IMU 1 – DES

- * Wait for IMU DG flag to go BAD, indicated *
- * by "RGA" on SPEC 51, and ADI OFF flags. *

IMU 1,2,3 – RES

DAP - AUTO

ET SEP - AUTO, when rates equal zero (or, MCC confirms less than 0.7 all axes)

ATT_PROC_INNER monitors the IMU data good (DG) flag, and when the DG goes BAD, Wbias is reset to zero. The assumption is that AUTO DAP is required ASAP for OMS-1. Resetting the IMU DG flag resets Wbias to zero and allows AUTO DAP immediately. Note that if the RGA bias went in very late in powered flight, then Wbias will be much smaller. The time required for Wbias to go to zero will also be smaller. If that relatively short time period can be afforded before selecting AUTO DAP for OMS-1, then Wbias can be allowed to go to zero and the IMU DES/RES is not necessary. Note that this assumes no IMU failures, and if there are IMU failures, then the DES/RES process should apply only to the good IMU's. The bad IMU's should remain deselected. Also, note that the IMU DES/RES is done to reset the IMU DG flag and make Wbias zero, in order to get back to AUTO DAP sooner. This could be done after ET sep, since the sep could be done in INRTL PULSE as in the previous procedure. However, the assumption here is that it takes the same amount of time to perform the DES/RES whether done before or after ET sep. By doing it before ET sep, the DAP can be in AUTO during, and immediately following the ET sep.

As a final note, while Wbias is used to compensate rate data by the TRANS DAP only, there are some concerns which apply to the aerojet DAP. The aerojet DAP does not use Wbias for rate data, but the attitude quaternion (QBI) will be ratchety (updated by the once-per-second IMU data), until Wbias compensates the RGA bias in the system.

5.6 BFS ORBITER RGA SOFTWARE

The main functional module for the Orbiter RGA backup flight system (BFS) is the Orbiter RGA Hardware Interface Program (ORB RGA HIP). The ORB RGA HIP consists of two submodules that perform SF

and rate compensation. In the ORB RGA HIP, SF is performed before rate compensation (fig. 5-16). In the primary system, rate compensation is performed before SF (fig. 5-10).

SF in the ORB RGA HIP is performed using the measured rate signals. One roll, one pitch, and one yaw rate are selected and sent for rate compensation. The SF parallels the PASS SF. At the four level the rates are processed by IMVS, at the three level by MVS, at the two level by averaging, and passing the sole rates at the one level. There is no FDIR contained in the ORB RGA HIP. If failures or bad rates occur, selection is performed using the bad rates; for CF's, the last value that was seen in COMPOOL will be used as the SF input for the COMM faulted RGA.

Selected rates are then processed through a rate compensation module which multiplies the selected middle value rate for each axis by a scale factor and then adds a bias to obtain compensated rates. There is only one scale factor and bias term for each selected axis output (roll, pitch, and yaw), not for each RGA axis input as in the PASS. The scale factor and bias values are I-loaded. The compensated rates are then output to the appropriate users.

5.7 SRB RGA SOFTWARE

The four SRB RGA's provide pitch and yaw body rates for flight control stability augmentation in addition to the Orbiter roll rate gyros during ascent until SRB separation. The SRB RGA SOP performs rate compensation and unit conversion and sends the compensated rates to RM for SF and fault identification (fig. 5-17). The ascent thrust vector control module (ASC-TVC) has the ability to blend the selected SRB yaw rate with the Orbiter yaw selected rate to provide the needed yaw flight stability with filament wound case (FWC) SRB's. Because the FWC SRB's currently don't exist, however, the ASC-TVC software module is I-loaded to use only the selected SRB yaw rate until SRB separation.

5.7.1 SRB Rate Gyro SOP

The purpose of the SRB rate gyro (SRB RG) SOP principal function shall be to perform data compensation of the measured left and right SRB RG pitch and yaw signals. The compensation function shall be executed at the 25 Hz rate prior to the SF principal function.

The SRB RG compensation function shall consist of applying bias and scale factor terms to convert the measured rate data into engineering units (deg/ sec) in floating point format. Provisions shall be made for unique scale factor and bias values for each measured signal. The compensation equation shall be as follows.

$$YC = YM \cdot C + K$$

where:

YC = Compensated signal (deg/s)

YM = Measured signal (V dc)

C = Compensation scale factor (deg/s/V dc)

K = Compensation bias (deg/s)

The SRB RG SOP interfaces are indicated on the functional block diagram (fig. 5-18). The input and output interface parameters are presented in tables 5-XIII and 5-XIV. The parameter initialization requirements are shown in table 5-XV.

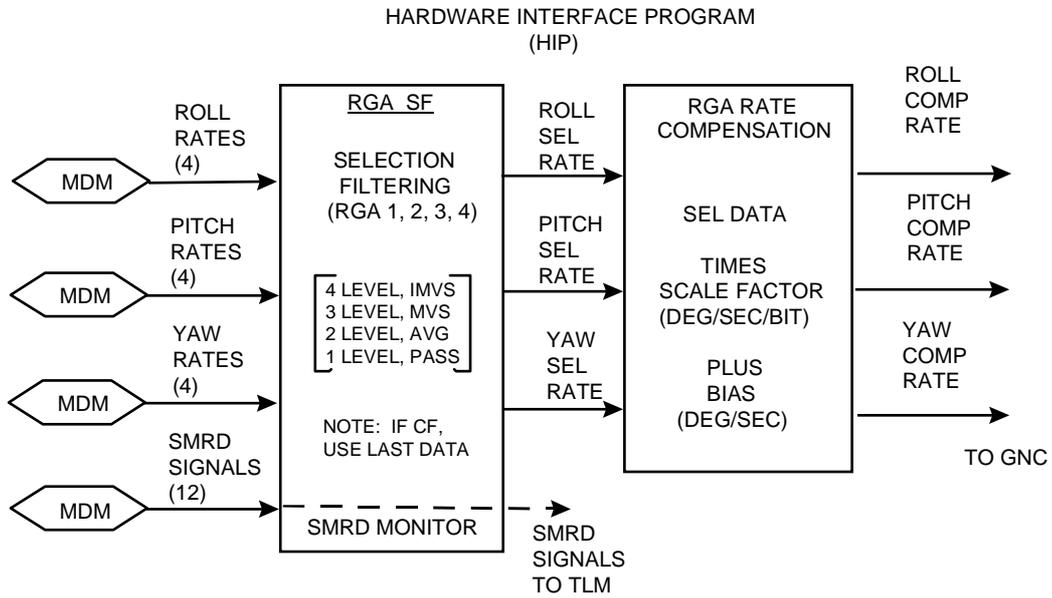


Figure 5-16 - BFS ORB RGA data flow

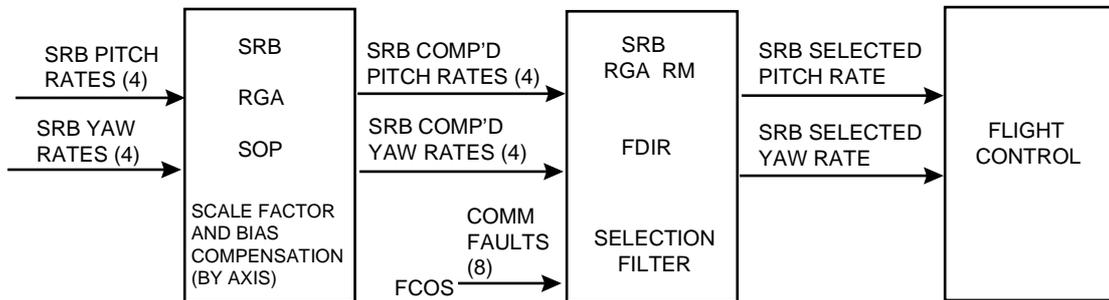


Figure 5-17 - SRB RGA data flow

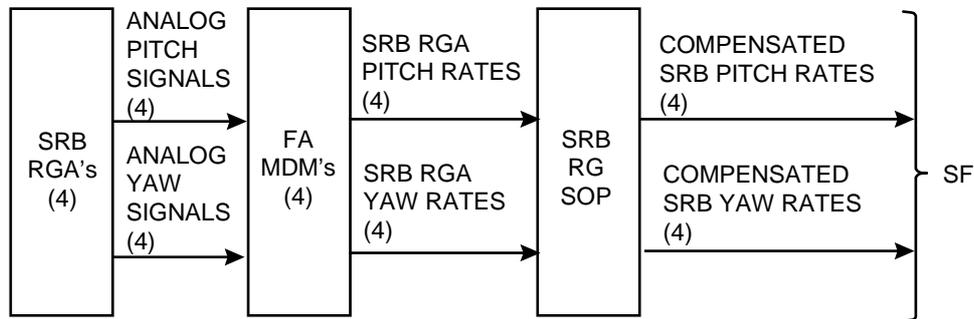


Figure 5-18 - SRB RG SOP functional block diagram

Table 5-XIII- SRB RG SOP Input Parameters

| Name | MSID | Description | Input source | Signal type | Range | Units | Rate (Hz) |
|---------|-----------|--------------------------|--------------|-------------|-------------|--------------|-----------|
| RGQM(1) | B79R1410C | SRGA uncomp pitch rate 1 | SRGA 1 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGQM(2) | B79R2410C | SRGA uncomp pitch rate 2 | SRGA 2 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGQM(3) | B79R1412C | SRGA uncomp pitch rate 3 | SRGA 3 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGQM(4) | B79R2412C | SRGA uncomp pitch rate 4 | SRGA 4 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGRM(1) | B79R1411C | SRGA uncomp yaw rate 1 | SRGA 1 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGRM(2) | B79R2411C | SRGA uncomp yaw rate 2 | SRGA 2 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGRM(3) | B79R1413C | SRGA uncomp yaw rate 3 | SRGA 3 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |
| RGRM(4) | B79R2413C | SRGA uncomp yaw rate 4 | SRGA 4 | Fixed pt | ±5.00 (±10) | V dc (Deg/s) | 25.0 |

Note: Values in parentheses indicate equivalent engineering units.

Table 5-XIV.- SRB RG SOP Output Parameters

| Name | MSID | Description | Output destination | Signal type | Range | Units | Rate (Hz) |
|----------|-----------|-----------------------|---------------------------|-------------|-------|-------|-----------|
| SRBQC(1) | V95R4260C | SRB comp pitch rate 1 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBQC(2) | V95R4262C | SRB comp pitch rate 2 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBQC(3) | V95R4270C | SRB comp pitch rate 3 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBQC(4) | V95R4272C | SRB comp pitch rate 4 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBRC(1) | V95R4261C | SRB comp yaw rate 1 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBRC(2) | V95R4263C | SRB comp yaw rate 2 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBRC(3) | V95R4271C | SRB comp yaw rate 3 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |
| SRBRC(4) | V95R4273C | SRB comp yaw rate 4 | SF SRB RGA FDIR TLM | Float pt | ±10 | Deg/s | 25.0 |

Table 5-XV.- SRB RGA SOP Initialization Parameters

| MSID | FSSR name | Nomenclature | Value | RI units |
|-----------|-----------|--------------------------------------|-------|------------|
| V97U3408C | CSRBQ(1) | Comp scale factor - SRB pitch rate 1 | +2.0 | Deg/s/V dc |
| V97U3469C | CSRBQ(2) | Comp scale factor - SRB pitch rate 2 | +2.0 | Deg/s/V dc |
| V97U3410C | CSRBQ(3) | Comp scale factor - SRB pitch rate 3 | +2.0 | Deg/s/V dc |
| V97U3411C | CSRBQ(4) | Comp scale factor - SRB pitch rate 4 | +2.0 | Deg/s/V dc |
| V97U3441C | CSRBR(1) | Comp scale factor - SRB yaw rate 1 | +2.0 | Deg/s/V dc |
| V97U3442C | CSRBR(2) | Comp scale factor - SRB yaw rate 2 | +2.0 | Deg/s/V dc |
| V97U3443C | CSRBR(3) | Comp scale factor - SRB yaw rate 3 | +2.0 | Deg/s/V dc |
| V97U3444C | CSRBR(4) | Comp scale factor - SRB yaw rate 4 | +2.0 | Deg/s/V dc |
| V97U3520C | KSRBQ(1) | Comp bias - SRB pitch rate 1 | +0.0 | Deg/s |
| V97U3521C | KSRBQ(2) | Comp bias - SRB pitch rate 2 | +0.0 | Deg/s |
| V97U3522C | KSRBQ(3) | Comp bias - SRB pitch rate 3 | +0.0 | Deg/s |
| V97U3523C | KSRBQ(4) | Comp bias - SRB pitch rate 4 | +0.0 | Deg/s |
| V97U3608C | KSRBR(1) | Comp bias - SRB yaw rate 1 | +0.0 | Deg/s |
| V97U3609C | KSRBR(2) | Comp bias - SRB yaw rate 2 | +0.0 | Deg/s |
| V97U3610C | KSRBR(3) | Comp bias - SRB yaw rate 3 | +0.0 | Deg/s |
| V97U3611C | KSRBR(4) | Comp bias - SRB yaw rate 4 | +0.0 | Deg/s |

5.7.2 SRB RGA RM

The SRB RGA RM consists of FDIR and an SF; however, there is no provision for fault annunciation to the crew.

5.7.3 FDIR

The FDIR operates only until the first RM failure or CF. After the first RGA failure, FDIR is not available for that SRB. SMRD signals are available, but are not used and data good (DG) flags are not generated. Candidate elements for FDIR shall be those RGA signals currently selected for the SF. For each parameter type (pitch rate and yaw rate), the following comparisons are made:

$$\begin{aligned}
 |P_1 - P_S| > LIM & \quad \text{where:} \\
 |P_2 - P_S| > LIM & \quad P_{1(2,3,4)} = \text{SRB RGA rate} \\
 |P_3 - P_S| > LIM & \quad P_S = \text{previous selected rate} \\
 & \quad | \quad = \text{absolute value} \\
 & \quad LIM = (\text{Selected SRB RGA rate}) \cdot (\text{Scale Factor}) + (\text{Limit Constant})
 \end{aligned}$$

An SRB RGA is declared failed if any parameter of that RGA miscompares with its previous selected rate for five consecutive times (10 sec). FDIR downmodes the failed LRU, and then FDIR terminates. No subsequent failures will be detected; only CFs will further downmode the selection filtering scheme. A CF before the first failure will terminate FDIR. The FDIR will be reactivated if the CF should be resolved. No fault messages or deselect/select capability exists.

5.7.4 SF

The four SF functional modes (IMVS, MVS, AVE, PASS) are determined by the usable level of input data redundancy. The IMVS SF mode provides signal selection from the four valid input signals and shall

provide protection from any single failure and dual null failures. The SF modes are described in detail for Orbiter rate gyros in section 5.5.3.2 and in figures 5-12, 5-13, and 5-14.

The SRB RGA FDIR input and output interface parameters are presented in tables 5-XIII and 5-XIV. The FDIR parameter initialization requirements are shown in table 5-XV. The SRB SF input and output interface parameters are shown in table 5-XVI.

5.8 BFS SRB RGA SOFTWARE

The main functional module for the SRB RGA BFS is the SRB RGA HIP. The SRB RGA HIP consists of two submodules that perform SF and rate compensation (fig. 5-18). In the SRB RGA HIP, SF is performed before rate compensation. In the primary system, rate compensation is performed before SF.

The SRB RGA HIP performs midvalue selection only on the measured pitch and yaw rates from SRB RGAs 1, 2, and 3. Midvalue selection is the only mode of SF available to the SRB RGA HIP as there is no FDIR available to the SRB RGA HIP. If failures or bad rates occur, midvalue selection would be performed using the bad rates; for a CF, the last value that was seen in COMPOOL will be used as the SF input for the COMM faulted RGA.

Selected rates are then processed through rate compensation which multiplies the selected midvalue for each axis for each SRB by a scale factor, and then adds a bias to obtain compensated rates. The scale factor and bias values are I-loaded. The compensated rates are then output to the appropriate users.

5.9 AA SOFTWARE OPERATIONAL DESCRIPTION

The four AA's provide normal and lateral body accelerations for load relief in ascent first stage, flight control stability augmentation in entry and return to launch site (RTLS), and for ADI error needle signals during entry major mode (MM) 305. The AA SOP performs acceleration compensation and unit conversion. AA self-test is accomplished during OPS 9 (prelaunch) and OPS 8 (on-orbit). The SOP sends compensated accelerations from AA 1, 2, 3, and 4 to RM (fig. 5-19).

5.9.1 AA SOP

The purpose of the AA SOP principal function shall be to perform data compensation of the measured AA lateral and normal acceleration signals. The compensation function shall be executed at the 25 Hz rate prior to the SF principal function except for MM's 301, 302, 303 where it shall be executed at the 0.52 Hz rate.

Table 5-XVI - SRB RGA FDIR Input Parameters

| MSID | FSSR name | Nomenclature | Source | Range | Units |
|-----------|-----------|--------------------------|------------|-------|-------|
| V95R4260C | SRBQC(1) | SRGA comp pitch rate 1 | SRB RG SOP | ±10 | Deg/s |
| V95R4262C | SRBQC(2) | SRGA comp pitch rate 2 | SRB RG SOP | ±10 | Deg/s |
| V95R4270C | SRBQC(3) | SRGA comp pitch rate 3 | SRB RG SOP | ±10 | Deg/s |
| V95R4272C | SRBQC(4) | SRGA comp pitch rate 4 | SRB RG SOP | ±10 | Deg/s |
| V95R4261C | SRBRC(1) | SRGA comp yaw rate 1 | SRB RG SOP | ±10 | Deg/s |
| V95R4263C | SRBRC(2) | SRGA comp yaw rate 2 | SRB RG SOP | ±10 | Deg/s |
| V95R4271C | SRBRC(3) | SRGA comp yaw rate 3 | SRB RG SOP | ±10 | Deg/s |
| V95R4273C | SRBRC(4) | SRGA comp yaw rate 4 | SRB RG SOP | ±10 | Deg/s |
| V90R2511C | Q-SRB | SRGA selected pitch rate | SF | ±10 | Deg/s |
| V90R2512C | R-SRB | SRGA selected yaw rate | SF | ±10 | Deg/s |
| V91X2780X | RGQCF(1) | SRGA pitch rate CF 1 | FCOS | | |
| V91X2792X | RGQCF(2) | SRGA pitch rate CF 2 | FCOS | | |
| V91X2782X | RGQCF(3) | SRGA pitch rate CF 3 | FCOS | | |
| V91X2794X | RGQCF(4) | SRGA pitch rate CF 4 | FCOS | | |
| V91X2781X | RGRCF(1) | SRGA yaw rate CF 1 | FCOS | | |
| V91X2793X | RGRCF(2) | SRGA yaw rate CF 2 | FCOS | | |
| V91X2783X | RGRCF(3) | SRGA yaw rate CF 3 | FCOS | | |
| V91X2795X | RGRCF(4) | SRGA yaw rate CF 4 | FCOS | | |

Table 5-XVII - SRB RGA FDIR Output Parameters

| MSID | FSSR name | Nomenclature | Destination | Range | Units |
|-----------|-----------|--------------------|-------------|-------|-------|
| V90X6748X | | String 1 SRGA fail | TLM | | |
| V90X6749X | | String 2 SRGA fail | TLM | | |
| V90X6750X | | String 3 SRGA fail | TLM | | |
| V90X6751X | | String 4 SRGA fail | TLM | | |
| V90R6752C | | SRGA command | SF | | |

Table 5-XVIII- SRB RGA FDIR Initialization Parameters

| MSID | FSSR name | Nomenclature | Value | RI units |
|-----------|---------------|---------------------------------------|----------|----------|
| V98U7956C | CON(04) | SRB RGA pitch rate limit | +4.5E-01 | Deg/s |
| V98U7957C | CON(05) | SRB RGA yaw rate limit | +4.5E-01 | Deg/s |
| V98U7958C | LSF(04) | SRB RGA pitch rate limit scale factor | +5.0E+00 | Pct |
| V98U7959C | LSF(05) | SRB RGA yaw rate limit scale factor | +5.0E+00 | Pct |
| V98U7960C | SRB_RGA_N_CNT | SRB RGA N counter | +5 | ND |

Table 5-XIX - SRB RGA SF Input/Output Parameters

(a) Input functional parameters for the selection filtering module

| MSID | FSSR Name | Nomenclature | Source | Range | Units |
|-----------|-----------|-------------------------|-------------|-------|-------|
| V90R2511C | Q_SRB | Selected SRB pitch rate | SF | ±10 | |
| V90R2512C | R_SRB | Selected SRB yaw rate | SF | ±10 | |
| V90R6752C | n/a | SRB RG SF command | SRB RG FDIR | | |
| V91X2780X | RGQCF(1) | SRB pitch rate 1 CF | FCOS | | |
| V91X2792X | RGQCF(2) | SRB pitch rate 2 CF | FCOS | | |
| V91X2782X | RGQCF(3) | SRB pitch rate 3 CF | FCOS | | |
| V91X2794X | RGQCF(4) | SRB pitch rate 4 CF | FCOS | | |
| V91X2781X | RGRCF(1) | SRB yaw rate 1 CF | FCOS | | |
| V91X2793X | RGRCF(2) | SRB yaw rate 2 CF | FCOS | | |
| V91X2783X | RGRCF(3) | SRB yaw rate 3 CF | FCOS | | |
| V91X2795X | RGRCF(4) | SRB yaw rate 4 CF | FCOS | | |
| V95R4260C | SRBQC(1) | SRB comp pitch rate 1 | SRB RG SOP | ±10 | Deg/s |
| V95R4262C | SRBQC(2) | SRB comp pitch rate 2 | SRB RG SOP | ±10 | Deg/s |
| V95R4270C | SRBQC(3) | SRB comp pitch rate 3 | SRB RG SOP | ±10 | Deg/s |
| V95R4272C | SRBQC(4) | SRB comp pitch rate 4 | SRB RG SOP | ±10 | Deg/s |
| V95R4261C | SRBRC(1) | SRB comp yaw rate 1 | SRB RG SOP | ±10 | Deg/s |
| V95R4263C | SRBRC(2) | SRB comp yaw rate 2 | SRB RG SOP | ±10 | Deg/s |
| V95R4271C | SRBRC(3) | SRB comp yaw rate 3 | SRB RG SOP | ±10 | Deg/s |
| V95R4273C | SRBRC(4) | SRB comp yaw rate 4 | SRB RG SOP | ±10 | Deg/s |

(b) Output functional parameters form the selection filtering module

| MSID | FSSR name | Nomenclature | Destination | Range | Units |
|-----------|-----------|-------------------------|--------------------------------------|-------|-------|
| V90R2511C | R_SRB | Selected SRB pitch rate | ASC DAP, A/E ATT PROC, SF, TLM, FDIR | ±10 | |
| V90R2512C | Q_SRB | Selected SRB yaw rate | ASC DAP, A/E ATT PROC, SF, TLM, FDIR | ±10 | |

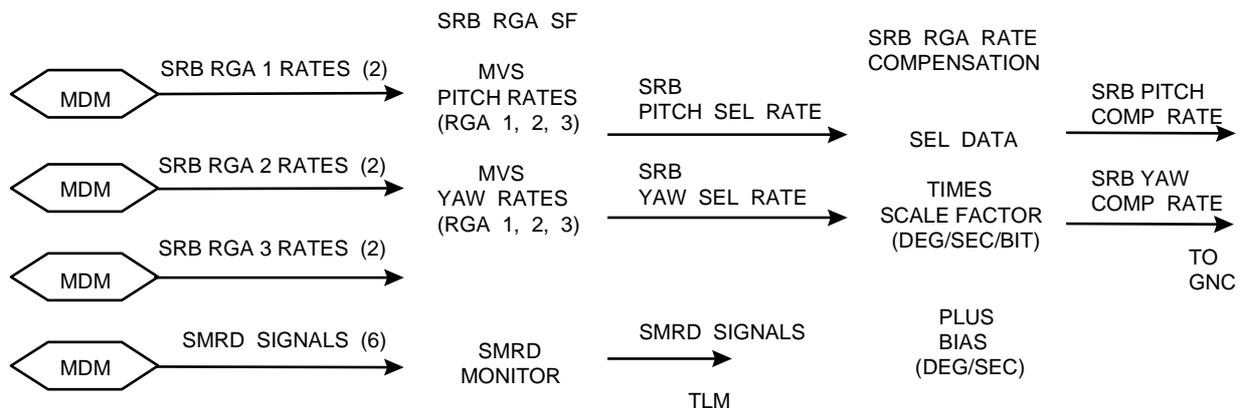


Figure 5-19 - BFS SRB RGA data flow

The AA SOP compensation function shall consist of applying biases and scale factors to convert the measured accelerometer data into engineering units (G's) in floating point format. Unique bias and scale factor values for each measured signal shall be provided. The general compensation equation shall be as follows.

$$NC = (NM) (CN) + KN$$

$$LC = (LM) (CL) + KL + (NC) (S)$$

where

NC/LC = Compensated normal/lateral AA signal (G's)

NM/LM = Measured normal/lateral AA signal (V dc)

CN/CL = Compensation scale factor (G's/V dc)

KN/KL = Compensation bias (G's)

S = Lateral AA misalignment angle (radians)

The AA SOP interfaces are indicated on the functional block diagram in figure 5-21. The input and output interface parameters are presented in tables 5-XX and 5-XXI. The parameter initialization requirements are shown in table 5-XXII.

5.9.2 AA RM

The status of each AA can be found on the GNC SYS SUMM 1 display (fig. 5-11) which receives failure (↓) and CF (M) indications from RM FDIR.

5.9.2.1 FDIR

The purpose of AA FDIR is to determine which of the AA parameters (lateral or normal acceleration) should be declared failed. See figure 5-12, General RM block diagram.

Candidates for FDIR are lateral and normal acceleration from AA 1, 2, 3, and 4. FDIR will detect the first failure only. Once the failed AA is identified, the failure will be annunciated through an FSM RM FAIL AA, a systems management (SM) alert light and tone, status indication on GNC SYS SUMM 1 display of a ↓ to indicate which AA has failed, and FDIR will shut down. Since FDIR is terminated after the first failure, no RM DLMA AA fault message will be generated for subsequent failures. In the early design of the Space Shuttle flight software, three- and two-level FD was included. In order to limit the authority of RM, the decision was made to not implement it. A GAX message, "RM DLMA AA" still exists, but it can never be activated with the current software design. CF's from FCOS will also downmode the RM to the 3-level and display M on SYS SUMM 1 display. Crew deselection of AA in OPS 8 will also downmode RM to the 3 level and display a ↓ on SYS SUMM 1 display. FDIR is terminated below the RM 4 level. See table 5-XXIII, AA GAX Annunciation.

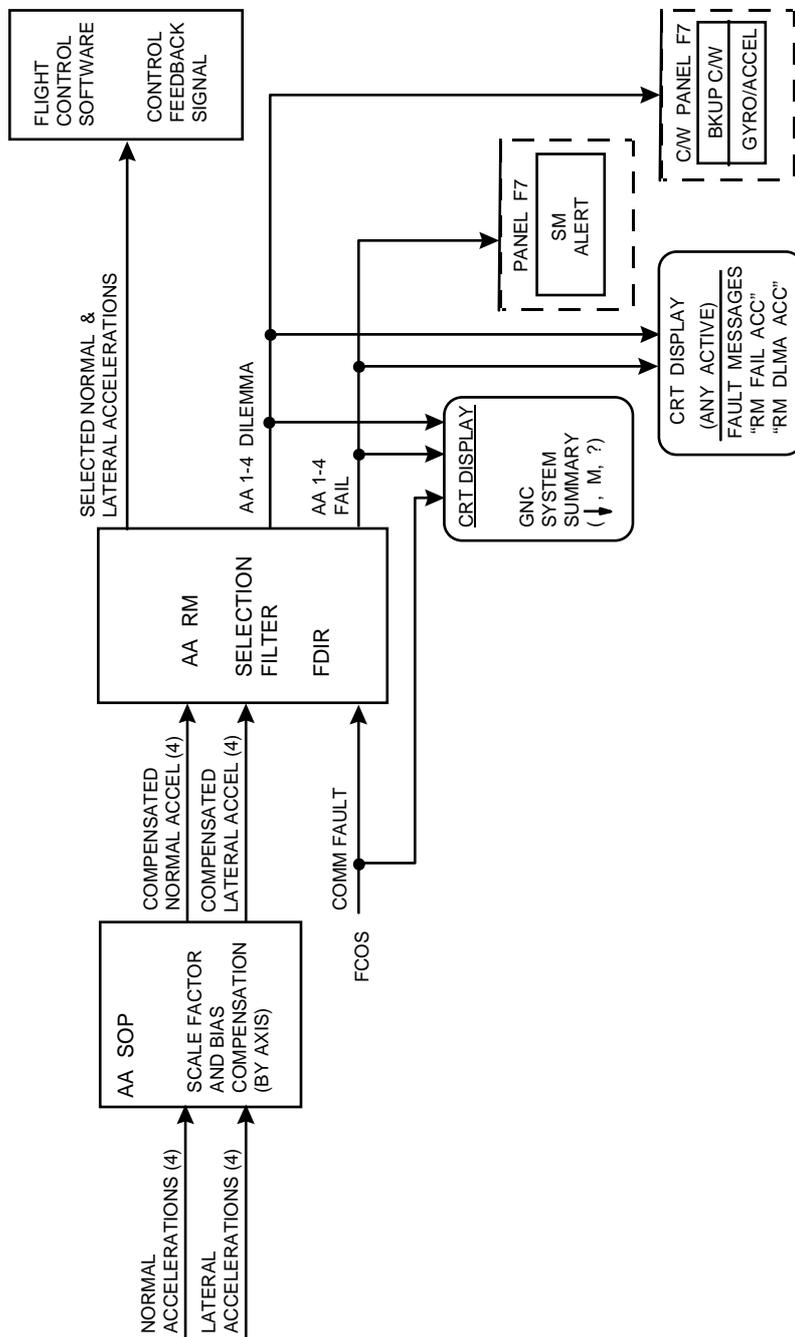


Figure 5-20 - Accelerometer assembly data flow

Table 5-XX - AA SOP Input Parameters

| Name | MSID | Description | Input source | Signal type | Range | Units | Rate (Hz) |
|-------|-----------|--------------------------|--------------|-------------|---------------|---------------|-----------|
| LM(1) | V79A2040C | Lateral accel 1 signal | AA 1 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| LM(2) | V79A2043C | Lateral accel 2 signal | AA 2 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| LM(3) | V79A2046C | Lateral accel 3 signal | AA 3 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| LM(4) | V79A2048C | Lateral accel 4 signal | AA 4 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| NM(1) | V79A2041C | Normal accel 1 signal | AA 1 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| NM(2) | V79A2044C | Normal accel 2 signal | AA 2 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| NM(3) | V79A2047C | Normal accel 3 signal | AA 3 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| NM(4) | V79A2049C | Normal accel 4 signal | AA 4 | Fixed pt | ±5.00 (±1) | V dc (G's) | 25.0 |
| KL(1) | V97U3515C | Comp bias - lat accel 1 | uplink | | 0.0 | G's | |
| KL(2) | V97U3516C | Comp bias - lat accel 2 | uplink | | 0.0 | G's | |
| KL(3) | V97U3517C | Comp bias - lat accel 3 | uplink | | 0.0 | G's | |
| KL(4) | V97U3518C | Comp bias - lat accel 4 | uplink | | 0.0 | G's | |
| KN(1) | V97U3528C | Comp bias - norm accel 1 | uplink | | 0.0 | G's | |
| KN(2) | V97U3529C | Comp bias - norm accel 2 | uplink | | 0.0 | G's | |
| KN(3) | V97U3530C | Comp bias - norm accel 3 | uplink | | 0.0 | G's | |
| KN(4) | V97U3531C | Comp bias - norm accel 4 | uplink | | 0.0 | G's | |

Note: Values in parentheses indicate equivalent engineering units.

Table 5-XXI - AA SOP Output Parameters

| Name | MSID | Description | Output destination | Signal type | Range | Units | Rate (Hz) |
|-------|-----------|--------------------------|--------------------|-------------|-------|-------|-----------|
| LC(1) | V95A4101C | Compensated lat accel 1 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| LC(2) | V95A4102C | Compensated lat accel 2 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| LC(3) | V95A4103C | Compensated lat accel 3 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| LC(4) | V95A4104C | Compensated lat accel 4 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| NC(1) | V95A4151C | Compensated norm accel 1 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| NC(2) | V95A4152C | Compensated norm accel 2 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| NC(3) | V95A4153C | Compensated norm accel 3 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |
| NC(4) | V95A4154C | Compensated norm accel 4 | SF, AA FDIR, TLM | Float pt | ±1 | G's | 25.0* |

*0.52 Hz in MM 301, 302, 303

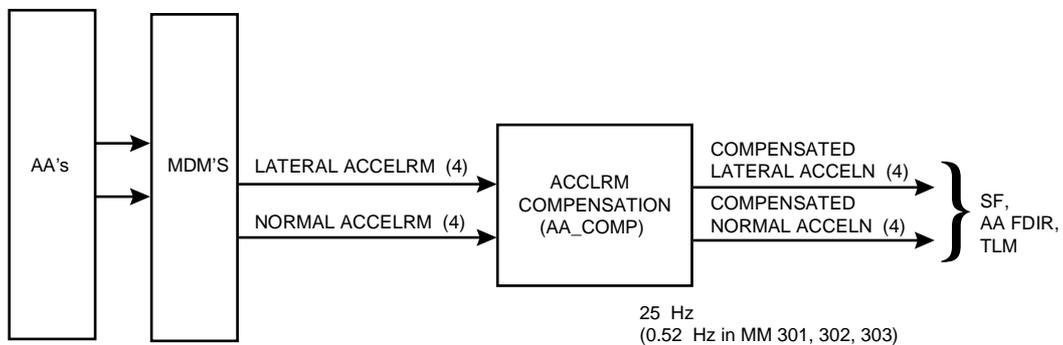


Figure 5-21, AA SOP functional block diagram

Table 5-XXII - AA SOP Initialization Parameters

| MSID | FSSR name | Nomenclature | MC | Value | RI units |
|-----------|-----------|----------------------------------|----------|----------------------|----------|
| V97U3404C | CL(1) | Comp scale factor - lat accel 1 | 01 03 | +2.0E-01 +2.0E-01 | G/V dc |
| V97U3405C | CL(2) | Comp scale factor - lat accel 2 | 01 03 | +2.0E-01 +2.0E-01 | G/V dc |
| V97U3406C | CL(3) | Comp scale factor - lat accel 3 | 01 03 | +2.0E-01 +2.0E-01 | G/V dc |
| V97U3407C | CL(4) | Comp scale factor - lat accel 4 | 01 03 | +2.0E-01 +2.0E-01 | G/V dc |
| V97U3414C | CN(1) | Comp scale factor - norm accel 1 | 01 03 | +8.0E-01 +8.0E-01 | G/V dc |
| V97U3415C | CN(2) | Comp scale factor - norm accel 2 | 01 03 | +8.0E-01 +8.0E-01 | G/V dc |
| V97U3416C | CN(3) | Comp scale factor - norm accel 3 | 01 03 | +8.0E-01 +8.0E-01 | G/V dc |
| V97U3417C | CN(4) | Comp scale factor - norm accel 4 | 01 03 | +8.0E-01 +8.0E-01 | G/V dc |
| V97U3515C | KL(1) | Comp bias - lat accel 1 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3516C | KL(2) | Comp bias - lat accel 2 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3517C | KL(3) | Comp bias - lat accel 3 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3518C | KL(4) | Comp bias - lat accel 4 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3528C | KN(1) | Comp bias - norm accel 1 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3529C | KN(2) | Comp bias - norm accel 2 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3530C | KN(3) | Comp bias - norm accel 3 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U3531C | KN(4) | Comp bias - norm accel 4 | 01 03 | +0.0E-01 +0.0E-01 | G |
| V97U8739C | S(1) | Misalignment - lat accel 1 | 01 03 | +0.0E-01 +0.0E-01 | RAD |
| V97U8740C | S(2) | Misalignment - lat accel 2 | 01 03 | +0.0E-01 +0.0E-01 | RAD |
| V97U8741C | S(3) | Misalignment - lat accel 3 | 01 03 | +0.0E-01 +0.0E-01 | RAD |
| V97U8742C | S(4) | Misalignment - lat accel 4 | 01 03 | +0.0E-01 +0.0E-01 | RAD |

Table 5-XXIII- AA GAX Annunciation

| MSID | Nomenclature | Source | OPS avail | NF | Alarm | | MSG ID | Message text | |
|-----------|------------------------|---------|-----------|----|-------|----|--------|--------------|----------------|
| | | | | | Class | ID | | Major | Minor |
| V90X5701X | AA 1 fail | AA FDIR | 1/3/6 | | | | | | |
| V90X5801X | AA 2 fail | AA FDIR | 1/3/6 | | | | | | |
| V90X5901X | AA 3 fail | AA FDIR | 1/3/6 | | | | | | |
| V90X5951X | AA 4 fail | AA FDIR | 1/3/6 | | | | | | |
| V94X4729X | AA 1/2/3/4 FL MSG | AA FDIR | 1/3/6 | | 3 | | 1365 | 5 | RM FAIL ACC |
| V90X5725X | AA norm accel disagree | AA FDIR | 1/3/6 | | | | | | |
| V90X5726X | AA lat accel disagree | AA FDIR | 1/3/6 | | | | | | |
| V94X4790X | AA DLMA MSG* | AA FDIR | 1/3/6 | | 2 | 3 | 1385 | 5 | RM DLMA ACC |

* See section 5.9.2.1

For each AA parameter (lateral and normal acceleration), the following comparisons are made.

$$|P_1 - P_s| > \text{CONSTANT LIM}$$

$$|P_2 - P_s| > \text{CONSTANT LIM}$$

$$|P_3 - P_s| > \text{CONSTANT LIM}$$

$$|P_4 - P_s| > \text{CONSTANT LIM}$$

where: $P_{1,2,3,4}$ = each AA acceleration
 P_s = selected rate from SF
|----| = absolute value

An AA is declared failed if any parameter of that AA miscompares with the selected accelerations for five consecutive times (10 sec period). FDIR downmodes the failed LRU and then FDIR terminates. No subsequent failures will be detected, only CF's and crew deselects will downmode the SF scheme.

A parameter is downmoded (removed from FDIR, and not declared failed) for a CF, and the CF is annunciated for the particular AA. If the CF disappears, the parameter will become an SF candidate again. Should a CF occur before any failures, the CF will be treated as a first failure by the FDIR, such that FDIR will terminate upon recognition of that CF. Should that CF be resolved, FDIR will resume processing, if at the four level.

5.9.2.2 SF

The purpose of the AA SF is to select single values of lateral and normal acceleration to provide to the users. Candidates are the accelerations from AA 1, 2, 3, and 4.

An AA is downmoded in the SF if a failure is indicated by the selection filter control (SFC) from FDIR. SF is also downmoded for CF's, and crew deselect via SPEC 40 and 53. If the CF disappears or the deselect is removed, the SF will upmode.

There are five different SF functional modes depending on the usable level of input data redundancy. The IMVS filter mode shall provide signal selection from four valid input signals and shall provide protection from any single failure and from dual null failures. Whenever the true input value is greater than the I-load null threshold value, a non-null signal will be selected. When the true input cycles above and below the null threshold level, the switching of the output between the null and true values will be minimized. See figure 5-12, IMVS flow diagram. The MVS mode provides a datum output value equal to the input value that lies between the other two values when three valid inputs are available (fig. 5-13). The AVE mode provides an output datum value that is the arithmetical average of two values when two valid inputs are available (fig. 5-14). The PASS mode provides a datum output value equal to a single datum input value. The NAS mode does not provide a new datum output value; i.e., the output value shall remain unchanged from its previous valid value. (This action results from the fact that the SF output COMPOOL location is fixed until updated. Failure to output a new value leaves the old value in the COMPOOL location.)

The AA FDIR input and output interface parameters are presented in tables 5-XXI and 5-XXII. The FDIR parameter initialization requirements are shown in table 5-XXIII. The AA SF input and output interface

parameters are shown in table 5-XXIV. The RGA/AA RM and data usability summary are shown in tables 5-XXV and 5-XXVI.

5.10 SELF-TEST OPERATIONS

a. OPS 8 On-Orbit Self-Test Procedures

The self-test procedure for the AA's is executed from SPEC 040, SENSOR TEST. The SPEC display and its functional operations are shown in section 5.13, Sensor Self-Test. The vehicle rates are nulled (<0.02 deg/s) using the RCS JETS VERN, and minimum vehicle accelerations are established by placing the vehicle in free drift (DAP ROT: PULSE/PULSE/PULSE, DAP: A1/MAN/VERN).

Next, the AA bias data is verified by examining the sensed rates shown on the display SPEC 40. The bias should be less than the values listed below.

| | |
|-------------|-----------------------------|
| Lateral (Y) | ± 0.5 ft/s ² |
| Normal (Z) | ± 1.3 ft/s ² |

If bias data are outside these limits, the option exists to deselect the appropriate AA unit from the RM.

After the bias test has been accomplished, the data limit test is performed from SPEC 40. The acceleration data are checked and verified within the limits listed below.

| | |
|-------------|--------------------------------|
| Lateral (Y) | 14.4 to 17.8 ft/s ² |
| Normal (Z) | 57.7 to 71.1 ft/s ² |

If the data are not within the proper limits, deselection of the appropriate AA is recommended.

The following PSI's displayed beside the appropriate data indicate the following.

| | |
|---|-------------------|
| ↑ | Data above limit |
| ↓ | Data below limit |
| M | Data missing (CF) |

The OPS 8 on-orbit self-test procedure is shown in section 5.13, Sensor Self-Test.

b. OPS 9 Prelaunch Self-Test Procedures

The self-test procedure for Orbiter AA's, prelaunch, are executed from SPEC 101, SENSOR SELF-TEST. The SPEC display and its functional operations are shown in section 5.13, Sensor Self-Test.

AA's cannot be deselected from RM, but can be enabled/disabled from self-test. The data limit test control start initiates self-test on only those LRU's enabled. The auto sequence initiates both high and low test on all sensors. The self-test limits and operations are the same as in OPS 8.

Table 5-XXIV- AA FDIR Input Parameters

| MSID | FSSR name | Nomenclature | Source | Range | Units |
|-----------|--------------|-------------------------------|---------|-------|-------|
| V95A4101C | LC(1) | Compensated lat accel 1 | AA SOP | ±1 | G |
| V95A4102C | LC(2) | Compensated lat accel 2 | AA SOP | ±1 | G |
| V95A4103C | LC(3) | Compensated lat accel 3 | AA SOP | ±1 | G |
| V95A4104C | LC(4) | Compensated lat accel 4 | AA SOP | ±1 | G |
| V95A4151C | NC(l) | Compensated norm accel 1 | AA SOP | ±4 | G |
| V95A4152C | NC(2) | Compensated norm accel 2 | AA. SOP | ±4 | G |
| V95A4153C | NC(3) | Compensated norm accel 3 | AA SOP | ±4 | G |
| V95A4154C | NC(4) | Compensated norm accel 4 | AA SOP | ±4 | G |
| V90A5361C | NY | Selected AA lateral accel | SF | ±1 | G |
| V90A5381C | NZ | Selected AA normal accel | SF | ±4 | G |
| V91X3650X | AA_LAT_CF(1) | AA 1 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3651X | AA_NOR_CF(1) | AA 1 normal accel COMM fault | FCOS | 0,1 | |
| V91X3660X | AA_LAT_CF(2) | AA 2 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3661X | AA_NOR_CF(2) | AA 2 normal accel COMM fault | FCOS | 0,1 | |
| V91X3670X | AA_LAT_CF(3) | AA 3 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3671X | AA_NOR_CF(3) | AA 3 normal accel COMM fault | FCOS | 0,1 | |
| V91X3680X | AA_LAT_CF(4) | AA 4 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3681X | AA_NOR_CF(4) | AA 4 normal accel COMM fault | FCOS | 0,1 | |
| V93X4435X | n/a | AA 1 DES CMD | V5.1.4 | 0,1 | |
| V93X4436X | n/a | AA 2 DES CMD | V5.1.4 | 0,1 | |
| V93X4437X | n/a | AA 3 DES CMD | V5.1.4 | 0,1 | |
| V93X4438X | n/a | AA 4 DES CMD | V5.1.4 | 0,1 | |

Table 5-XXV - AA FDIR Output Parameters

| MSID | Nomenclature | Destination | Range | Units |
|-----------|---------------------------|-------------|-------|-------|
| V90U5960C | AA SF command | SF | 1,4 | |
| V90X5701X | AA 1 fail | GAX,TLM | 0,1 | |
| V90X5801X | AA 2 fail | GAX,TLM | 0,1 | |
| V90X5901X | AA 3 fail | GAX,TLM | 0,1 | |
| V90X5951X | AA 4 fail | GAX,TLM | 0,1 | |
| V90X5717X | AA 1 normal accel fail | TLM | | |
| V90X5718X | AA 1 lateral accel fail | TLM | | |
| V90X5719X | AA 2 normal accel fail | TLM | | |
| V90X5720X | AA 2 lateral accel fail | TLM | | |
| V90X5721X | AA 3 normal accel fail | TLM | | |
| V90X5722X | AA 3 lateral accel fail | TLM | | |
| V90X5723X | AA 4 normal accel fail | TLM | | |
| V90X5724X | AA 4 lateral accel fail | TLM | | |
| V90X5725X | AA normal accel disagree | GAX,TLM | 0,1 | |
| V90X5726X | AA lateral accel disagree | GAX,TLM | 0,1 | |
| V93X4435X | AA 1 DES CMD | 6.46 | 0,1 | |
| V93X4436X | AA 2 DES CMD | 6.46 | 0,1 | |
| V93X4437X | AA 3 DES CMD | 6.46 | 0,1 | |
| V93X4438X | AA 4 DES CMD | 6.46 | 0,1 | |

Table 5-XXVI- AA FDIR Initialization Parameters

| MSID | FSSR name | Nomenclature | MC | Value | RI units |
|-----------|-----------|-----------------------------|----|----------|----------|
| V97U3030C | Limit(04) | AA normal accel limit | 01 | +6.3E-02 | G |
| | | | 03 | +2.0E-01 | |
| V97U3031C | Limit(05) | AA lateral accel limit | 01 | +8.0E-02 | G |
| | | | 03 | +1.3E-01 | |
| V97U3000C | N_AA | AA N counter | SV | +5 | ND |
| V96U8594C | NYQC | AA lateral accel IMVS const | 01 | +3.0E-02 | G |
| | | | 03 | +3.0E-02 | |
| V96U8595C | NZQC | AA normal accel IMVS const | 01 | +0.0E-00 | G |
| | | | 03 | +0.0E-00 | |

Table 5-XXVII- AA SF Input/Output Parameters

(a) Input functional parameters for the selection filtering module

| MSID | FSSR name | Nomenclature | Source | Range | Units |
|-----------|--------------|-------------------------------|---------|-------|-------|
| V90U5960C | n/a | AA SF command | AA FOIR | 1,4 | |
| V91X3650X | AA_LAT_CF(1) | AA 1 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3651X | AA_NOR_CF(1) | AA 1 normal accel COMM fault | FCOS | 0,1 | |
| V91X3660X | AA_LAT_CF(2) | AA 2 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3661X | AA_NOR_CF(2) | AA 2 normal accel COMM fault | FCOS | 0,1 | |
| V91X3670X | AA_LAT_CF(3) | AA 3 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3671X | AA_NOR_CF(3) | AA 3 normal accel COMM fault | FCOS | 0,1 | |
| V91X3680X | AA_LAT_CF(4) | AA 4 lateral accel COMM fault | FCOS | 0,1 | |
| V91X3681X | AA_NOR_CF(4) | AA 4 normal accel COMM fault | FCOS | 0,1 | |
| V95A4101C | LC(1) | Compensated lat accel 1 | AA SOP | +1 | G |
| V95A4102C | LC(2) | Compensated lat accel 2 | AA SOP | +1 | G |
| V95A4103C | LC(3) | Compensated lat accel 3 | AA SOP | +1 | G |
| V95A4104C | LC(4) | Compensated lat accel 4 | AA SOP | +1 | G |
| V95A4151C | NC(1) | Compensated norm accel 1 | AA SOP | +4 | G |
| V95A4152C | NC(Z) | Compensated norm accel 2 | AA SOP | +4 | G |
| V95A4153C | NC(3) | Compensated norm accel 3 | AA SOP | +4 | G |
| V95A4154C | NC(4) | Compensated norm accel 4 | AA SOP | +4 | G |

(b) Output functional parameters for the selection filtering module

| MSID | FSSR name | Nomenclature | Destination |
|------------|-----------|---------------------------|---|
| V90A5361C | NY | Selected AA lateral accel | AEROJET DAP, ASC DAP, GRTLS DAP, |
| | ±1 | G | 6.15, 6.16, 6.17, 6.18, 6.19, 6.20, 6.40, AA FOIR, TLM |
| V90A5381C | NZ | Selected AA normal accel | AEROJET DAP, GRTLS TAEM GUID, |
| | ±4 | G | AMI PROC, ASC DAP, GRTLS DAP, ABT CNTL SEQ, E/L EV AN PROC, AA FDIR, 5.22, 5.7, TLM |
| V93A8201CB | | Displayed normal accel | 6.21 |

Table 5-XXVIII - RM Summary
(a) RGA/AA FDIR/SOP major mode applicability

| Functional module Hz rate | Major Mode | | | | | | | | | | | | | | | |
|---------------------------|------------|------|------|------|------|------|-----|-----|------|------|------|------|------|------|------|------|
| | 101 | 102 | 103 | 104 | 105 | 106 | 201 | 201 | 301 | 302 | 303 | 304 | 305 | 601 | 602 | 603 |
| ORB RGA FDIR | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 | | | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 | 1.04 |
| AA FDIR | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | | | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 |
| SRB RGA FDIR | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | | | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 | 0.52 |
| ORB RGA SOP/SF | 25.0 | 25.0 | 25.0 | 12.5 | 12.5 | 12.5 | | | 12.5 | 12.5 | 12.5 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 |
| AA SOP/SF | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | | | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 |
| SRB RGA SOP/SF | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | | | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 | 25.0 |

(b) RGA/AA FDIR parameter reference and process requirements

| Parameter | Redundancy level | Crew sel/des | COMM fault | Lock-on | BITE used | BITE used | Notes |
|---|------------------|--------------|------------|---------|-----------|-----------|-------|
| Orb pitch rate roll rate yaw rate | 4 | Yes, OPS 8 | Yes | N/A | Yes | No | 1 |
| | 4 | Yes, OPS 8 | Yes | N/A | Yes | No | 1 |
| | 4 | Yes, OPS 8 | Yes | N/A | Yes | No | 1 |
| AA normal accel lateral accel | 4 | Yes, OPS 8 | Yes | N/A | No | N/A | 1 |
| | 4 | Yes, OPS 8 | Yes | N/A | No | N/A | 1 |
| SRB pitch rate yaw rate | 4 | No | Yes | N/A | Yes | No | 2 |
| | 4 | No | Yes | N/A | Yes | No | 2 |

Notes: 1. FDIR terminated if any LRU not usable (FAIL, CF, DES)
2. FDIR terminated if any LRU not usable (FAIL, CF)

TABLE 5-XXIX - RM DATA USABILITY SUMMARY

(a) RGA/AA SF usable input parameter determination

| SF Command (SFC) (FDIR FL/DES) | COMM fault (CF) | Lock-on (data valid) | SF input usability |
|-----------------------------------|--------------------|-------------------------|-----------------------|
| Disallow | Invalid | N/A | Do not use |
| Disallow | Valid | N/A | Do not use |
| Allow | Invalid | N/A | Do not use |
| Allow | Valid | N/A | Use |

FDIR will not change the SFC flag below the 4-level RM.

(b) RGA/AA SF modes

| Usability status | | | | SF | Output | DG |
|------------------|------------|------------|------------|------|--------------------------------------|------|
| Input 1 | Input 2 | Input 3 | Input 4 | mode | data | flag |
| DNU | DNU | DNU | DNU | NAS | No update of output | N/A |
| DNU | DNU | DNU | U | PASS | Input data 4 | N/A |
| DNU | DNU | U | DNU | PASS | Input data 3 | N/A |
| DNU | U | DNU | DNU | PASS | Input data 2 | N/A |
| U | DNU | DNU | DNU | PASS | Input data 1 | N/A |
| DNU | DNU | U | U | AVE | Input data (3 + 4)/2 | N/A |
| DNU | U | DNU | U | AVE | (2 + 4)/2 | N/A |
| U | DNU | DNU | U | AVE | (1 + 4)/2 | N/A |
| DNU | U | U | DNU | AVE | (2 + 3)/2 | N/A |
| U | DNU | U | DNU | AVE | (1 + 3)/2 | N/A |
| U | U | DNU | DNU | AVE | (1 + 2)/2 | N/A |
| DNU | U | U | U | MVS | Midvalue input data 2, 3, 4 | N/A |
| U | DNU | U | U | MVS | 1, 3, 4 | N/A |
| U | U | DNU | U | MVS | 1, 2, 4 | N/A |
| U | U | U | DNU | MVS | 1, 2, 3 | N/A |
| U | U | U | U | IMVS | Will not select dual null input data | N/A |

SRB RGA fourth parameter always usable.
DNU = Do not use, U = Use, NAS = No available sensor

5.11 BFS AA SOFTWARE

The main functional module for the AA BFS is the ORB AA HIP. The ORB AA HIP consists of two submodules that perform SF and acceleration compensation. In the ORB AA HIP, SF is performed before acceleration compensation; in the primary system, acceleration compensation is performed before SF (fig. 5-21).

SF in the ORB AA HIP is performed using the measured accelerations. One lateral and one normal acceleration are selected and sent for acceleration compensation. The SF parallels the PASS SF. At the four level the rates are processed by IMVS, at the three level by MVS, at the two level by averaging, and passing the sole rates at the one level. There is no FDIR contained in the ORB AA HIP. If failures or bad accelerations occur, selection is performed using the bad accelerations; for CF's, the last value that was seen in COMPOOL will be used as the SF input for the COMM faulted AA.

Selected accelerations are then processed through the acceleration compensation module which multiplies the selected middle value acceleration for each axis by a scale factor and then adds a bias to obtain compensated accelerations. The scale factor and bias values are I-loaded. The compensated rates are then output to the appropriate users.

5.12 GNC SYSTEMS SUMMARY 1 (PFS DISPLAY)

TITLE: GNC SYS SUMM 1
TYPE: DISP/O18 OR SYS SUMM KEY
AVAILABILITY: OPS 1, 2, 3, 6 and 8

5.12.1 Purpose, Operation, and Organization

The GNC SYS SUMM 1 display (fig. 5-10) shall be the primary format for monitoring of the Orbiter GNC systems operational status during the prelaunch, ascent, orbit, entry, and RTLS phases. The parameters common to these phases shall be displayed on a single display page. The displayed parameters were selected to support the caution and warning (C&W) function. The status information presented provides sufficient detail for a crewmember to respond to a GN&C C&W alarm. The display shall present the status for the navigation (NAV) sensors.

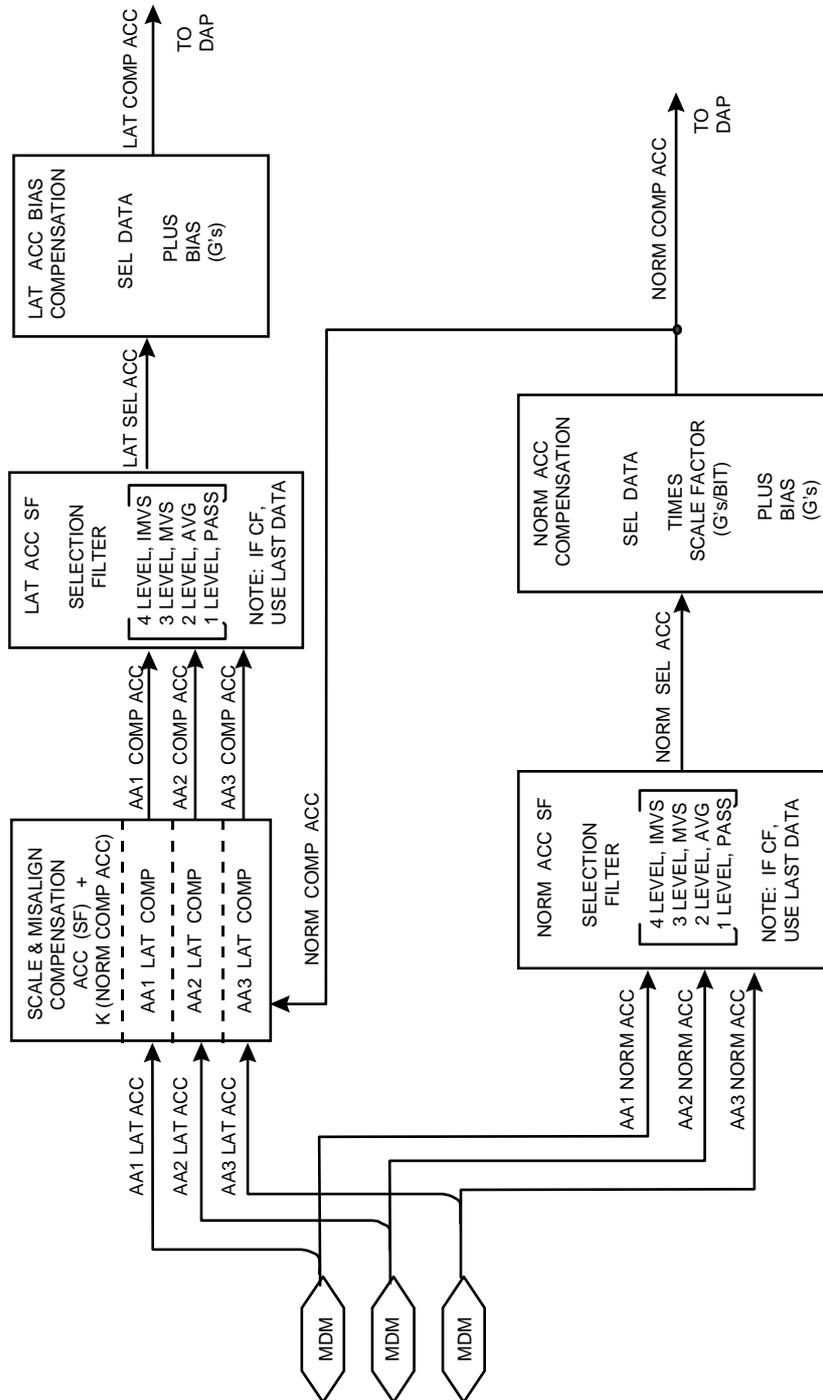


Figure 5-22, - BFS AA data flow

5.12.1.1 DISP/O18, GNC SYS SUMM 1, Functional Requirements

The status for the inertial measurement unit (IMU), accelerometer RGA, tactical air navigation (TAC), microwave landing system (MLS), and air data transducer assembly (ADTA) NAV sensors are shown on figure 5-10 and shall be supported as follows.

- (1) During OPS 1: IMU 1, 2, 3
 ACC 1, 2, 3, 4
 RGA 1, 2, 3, 4

- (2) During OPS 2 and 8: IMU 1, 2, 3

- (3) During OPS 3 and 6: IMU 1, 2, 3
 ACC 1, 2, 3, 4
 RGA 1, 2, 3, 4
 TAC 1, 2, 3
 MLS 1, 2, 3
 ADTA 1, 2, 3, 4

For each LRU, input/output (I/O) errors, failure, manual deselection, and dilemma status indicators shall be displayed. For the ACC, normal and lateral acceleration discretetes shall be OR'ed for I/O errors and disagrees. For the RGA, roll, pitch, and yaw rate parameters shall be OR'ed for I/O errors and disagrees. For all LRU's, failure and manual deselects shall be OR'ed. The status output for each LRU shall normally be blank. For failures detected by RM or manual deselects, down arrows (↓) shall be output. Question marks (?) shall be output for each LRU in the voting logic if a dilemma (disagree and/or fault detect) is declared. If the data for any sensor are missing (I/O error and/or loss of lock), the CF shall be reflected by the output of an M status indicator.

The display shall be called via dedicated key (SYS SUMM) or by the SPEC key sequence. The display page shall be available in OPS 1, 2, 3, 6, and 8 with those data which are not supported in a particular OPS being blanked. The data presented are monitored in conjunction with the C&W matrix, fault message line, and other dedicated displays.

5.13 SENSOR SELF-TEST

There are four sensor self-test SPEC's that support the sensor self-test function. The two on-orbit SPEC displays shown in SENSOR TEST and RGA/ADTA/RCS shall be displayed upon keyboard request in GNC OPS 8 and shall provide the crew with a means to verify sensor test data comparison during sensor self-testing and monitor and control the SF's for the AA's, radar altimeters (RA's), RGA's, microwave scanning beam landing system (MSBLS's), ADTA's, and TACAN's. To deselect a sensor from being an SF candidate, the proper item number is selected, followed by an execute command. An input of the same item number shall reselect the LRU as a SF candidate. The two preflight SPEC displays shown in SENSOR SELF-TEST and RCS/RGA/ADTA TEST shall be displayed upon keyboard request when in GNC OPS 9 or in SM OPS 9 and shall provide the ground with a means to verify sensor test data comparison. The SPEC displays shall be memory-resident with the sensor self-test function. Test item selection shall be supported as shown in SENSOR TEST and RGA/ADTA/RCS (GNC OPS 8) and in SENSOR SELF-TEST and RCS/RGA/ADTA TEST (SM OPS 9 or GNC OPS 9). When the SPEC displays are called, the scaled sensor data shall be cyclically displayed at 1 Hz and the STAT column shall be driven to indicate (as applicable) sensor power status, SMRD, or BITE status.

5.13.1 On-Orbit Sensor Test (PFS)

TITLE: SENSOR TEST
TYPE: SPEC/040
AVAILABILITY: GN&C OPS 8

5.13.1.1 Purpose, Operation, and Organization

- a. SENSOR TEST - This SPEC permits self-testing of the MLS, TACAN, RA, and AA sensors while on orbit. Figure 5-22 is the on-orbit SENSOR TEST display. Tests are not individually selectable, but are all started and stopped simultaneously. For each of the test parameters generated, the value is displayed, compared with expected results, and a status indication is driven beside the displayed parameter if it is outside predetermined limits; up arrow (↑) indicating test value too high, or down arrow (↓) indicating test value too low. An M will be displayed beside the last valid data value if data are missing (COMM faulted).
- b. AA - For the AA sensors, compensated lateral acceleration (Y) and compensated normal acceleration (Z) in feet per second per second are displayed. There is no STAT column associated with AA.

Whether or not testing is in progress, items 1 through 12 may be used to reconfigure the corresponding sensors as candidates for the SF's during the entry phase. Execution of one of these items reverses the selection/deselection for the corresponding sensor. An asterisk (*) beside the item number indicates the corresponding sensor is deselected.

- c. START/STOP - Execution of item 13 initiates parallel testing of all the sensors listed on the display. The self-tests will run to completion regardless of an earlier LRU self-test failure. An asterisk (*) appears next to item 13 while the test is in progress and is blanked upon test completion. Execution of item 14 terminates testing, removes all self-test stimuli from the LRU's, clears the PSI's on the display, blanks the asterisk (*) beside item 13, blanks the RA status column, enables cyclic update of sensor parametric data, and displays the asterisk (*) beside item 14. Items 13 and 14 are initialized blank anytime the display is called up.

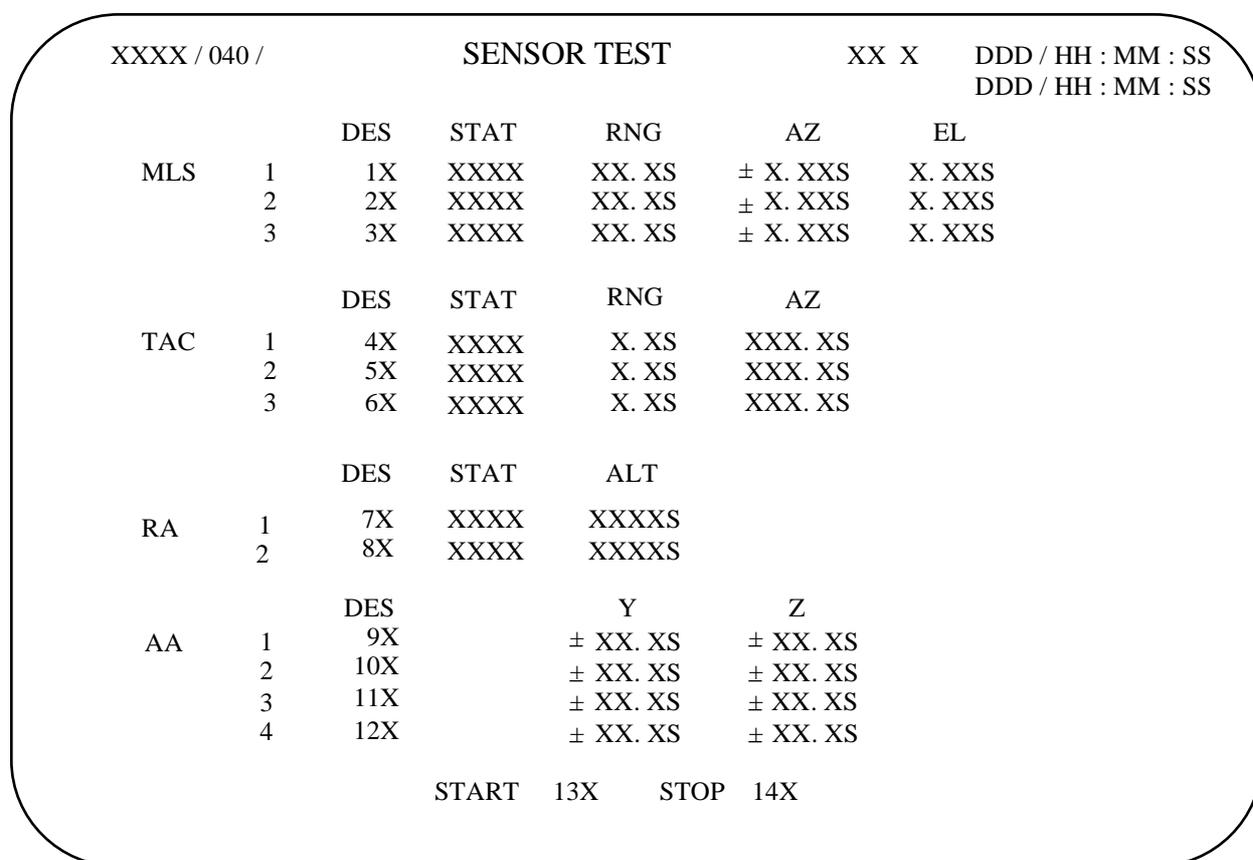


Figure 5-23, - On-orbit SENSOR TEST display (PFS)

5.13.2 On-Orbit RGA/ADTA/RCS Test (PFS)

TITLE: RGA/ADTA/RCS
TYPE: SPEC/O41
AVAILABILITY: GNC OPS 8

5.13.2.1 Purpose, Operation, and Organization

- a. RGA/ADTA/RCS - This SPEC permits testing of the RGA's, ADTA's, and the reaction control system (RCS) while on orbit. Figure 5-23 is the on-orbit RGA/ADTA/RCS TEST display preflight sensor self test (PFS).
- b. RGA's - For the RGA's, compensated roll rate (R), pitch rate (P), and yaw rate (Y) are displayed in deg/sec while testing is in progress. A status parameter by each value will be driven with an (M) to indicate missing data, an up arrow (↑) to indicate test data too high, a down arrow (↓) to indicate test data too low, or blank if data are normal. RGA status (STAT) shows SMRD if SMRD detects a malfunction on any axis. If the SMRD indicator is set, the SMRD display will be reset if the error is corrected, or upon execution of item 11. Whether or not testing is in progress, items 1 through 4 may be used to deselect or select the corresponding sensor as a candidate for the selection filter during

ENTRY (OPS 3). An asterisk (*) beside the item number indicates that the corresponding RGA is deselected.

- c. TEST - Execution of item 9 (legal only if there is no asterisk beside item 9 or item 10) causes the RGA's and ADTA's to be tested with a set of values called the high test. Execution of item 10 (legal only if there is no asterisk (*) beside item 9 or item 10) causes the RGA's and ADTA's to be tested with a set of values called the low test. The self-tests will run to completion regardless of an earlier LRU self-test failure. An asterisk (*) appears next to item 9 or 10 when the test is in progress. Execution of item 11 causes testing of the RGA's and ADTA's to be terminated, removes all self-test stimuli from the LRU's, clears the PSI's on the display, blanks the asterisk (*) beside item 9 or 10, enables cyclic update of sensor parameter data, and displays an asterisk (*) beside item 11. Items 9 and 10 are initialized blank.

| | | | | | | | | |
|-------------|------|--------------|---------|---------|---------|---------|--------------|--------------|
| XXXX/041/ | | RGA/ADTA/RCS | | | | XX | X | DDD/HH:MM:SS |
| | | X | | | X | | DDD/HH:MM:SS | |
| DES | STAT | R | P | Y | | | | |
| R1 | 1X | XXXX | +XX.XS | +XX.XS | +XX.XS | | | |
| G2 | 2X | XXXX | +XX.XS | +XX.XS | +XX.XS | | | |
| A3 | 3X | XXXX | +XX.XS | +XX.XS | +XX.XS | | | |
| 4 | 4X | XXXX | +XX.XS | +XX.XS | +XX.XS | | | |
| DES | STAT | PS | PαC | PαU | PαL | T | | |
| A1 | 5X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS | |
| D3 | 6X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS | |
| T2 | 7X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS | |
| A4 | 8X | XXXX | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXXS | XX.XXS | |
| TEST | HI | 9X | LOW | 10X | STOP | 11X | | |
| RCS DRIVERS | | | | | | | | |
| RJDF | 12X | TEST EXEC | 15X | TEST | XXXX | | | |
| RJDA-A | 13X | CONTINUE | 16X | JET | XXX | | | |
| -B | 14X | STOP | 17 | | | | | |

(XX)

Figure 5-24, On-orbit RGA/ADTA/RCS test display (PFS)

5.13.3 PFS

TITLE: SENSOR SELF-TEST

TYPE: SPEC/101

AVAILABILITY: SM OPS 9, GNC OPS 9

5.13.3.1 Purpose, Operation, and Organization

- a. SENSOR SELF-TEST - Figure 5-24 is the PFS display. This display provides the means to enable testing of the LRU's listed in table 5-XXVII. This table also lists, for each LRU tested, the parameters for which resultant test data are displayed. In addition to the listed results, STAT is displayed as follows for all LRU's except AA.

| | |
|-------|---------------------------|
| blank | Normal |
| OFF | Power off |
| BITE | Internal hardware failure |

- b. ENABLE (ENA) - Execution of items 1 through 12 enables the respective LRU's for testing. Any number of these sensors may be in the enabled state at once. An asterisk (*) appears beside each item number as it is selected. It is illegal to execute an item from 1 through 12 while testing is in progress (asterisk beside item 13 or 14).
- c. START - Execution of item 13 initiates self-testing of all the LRU's selected for testing (indicated by asterisks beside the respective item numbers). When item 13 is selected, an asterisk appears beside the item number. The asterisk blanks when testing is complete. If no LRU is selected or if testing is in progress (item 13 or 14 selected), or after a test has been completed unless TERM item is entered first, an attempt to execute item 13 is illegal.
- d. AUTO SEQ - Execution of item 14 initiates self-testing of all the LRU's listed on the display. When item 14 is selected, an asterisk appears beside the item number. If testing is already in progress (item 13 or 14 selected), or after a test has been completed unless TERM item is entered first, an attempt to execute item 14 is illegal.
- e. TERM - After test results have been displayed for all sensors undergoing test, selection of item 15 will deselect (inhibit) testing of all LRU's, blank all test results, display an asterisk beside item 15 and enable cyclic output of dynamic parameter data. If testing is still in progress when item 15 is executed, an ILLEGAL ENTRY message is displayed.
- f. INH ALL - When testing is not in progress (asterisks absent from beside items 13 and 14), item 16 may be selected. The effect is to inhibit all sensors, to remove all asterisks from beside items 1 through 12, and to cause an asterisk to appear beside item 16. Execution of item 16 when item 13 or 14 is selected is illegal.

5.13.4 Preflight RCS/RGA/ADTA Test PFS

TITLE: RCS/RGA/ADTA TEST
TYPE: SPEC/102
AVAILABILITY: SM OPS 9, GNC OPS 9

5.13.4.1 Purpose, Operation, and Organization

- a. RCS/RGA/ADTA TEST - Figure 5-25 is the preflight RCS/RGA/ADTA TEST display preflight sensor self-test (PFS). This display provides the means to enable testing of the LRU's listed in table 5-XXVIII. The table also lists, for each sensor tested, the parameters for which resultant test data are displayed.

In addition to the listed results for sensors (RGA's and ADTA's), test STAT is displayed as follows:

Blank Normal
BITE Internal hardware failure
SMRD Spin-motor run detector error

For the RGA's, the STAT values are blank and SMRD. For the ADTA's, the STAT values are blank and BITE.

TABLE 5-XXXI - LRU'S available for testing with preflight RCS/RGA/ADTA test

| LRU | Resultant test data displayed |
|---|---|
| Sensors | |
| Rate gyro assemblies (RGA's) | Roll rate (R) Pitch rate (P) Yaw rate (Y) |
| Air data transducer assemblies (ADTA's) | Center alpha pressure (PaC) Static pressure (PS) Upper alpha pressure (PaU) Lower alpha pressure (PaL) Total temperature (TT) |
| Forward reaction jet drivers (RJDF) Aft reaction jet drivers, group A (RJDF-A) Aft reaction jet drivers, group B (RJDF-B) | |

- b. SENSORS - Execution of items 7 through 14 enables the respective LRU's for testing. Any number of these sensors may be in the enabled state at once. An asterisk (*) appears beside each item number as it is selected. It is illegal to execute an item from 7 through 14 while testing is selected (asterisk beside item 15, 16, or 17).
- c. TEST CONTROL - Execution of item 15 or 16 initiates the high test or the low test, respectively, of all the sensors selected for testing (indicated by asterisks beside the respective item numbers). When item 15 or 16 is selected, an asterisk appears beside the item number. If no sensor is selected, if testing is in progress (items 15, 16, or 17 selected), or if a test has been completed unless TERM has been entered first, an attempt to execute item 15 or 16 is illegal.

- d. AUTO SEQ - Execution of item 17 initiates self-testing of all the sensors listed on the display. When item 17 is selected, an asterisk appears beside the item number. If testing is already in progress (item 15, 16, or 17 selected), or if a test has been completed unless TERM has been entered first, an attempt to execute item 17 is illegal.
- e. TERM - After test results have been displayed for all sensors under-going test, selection of item 18 will deselect (inhibit) testing of all sensors, blank all test results, display an asterisk beside item 18 and cyclically display the dynamic parameter data associated with the RGA's and ADTA's. If testing is still in progress, execution of item 18 is illegal.
- f. INH ALL - When testing is not in progress (asterisks absent from beside items 15, 16 and 17), item 19 may be selected. The effect is to inhibit all sensors, remove all asterisks from beside items 7 through 14, and cause an asterisk to appear beside item 19. Execution of item 19 when item 15, 16, or 17 is selected is illegal.

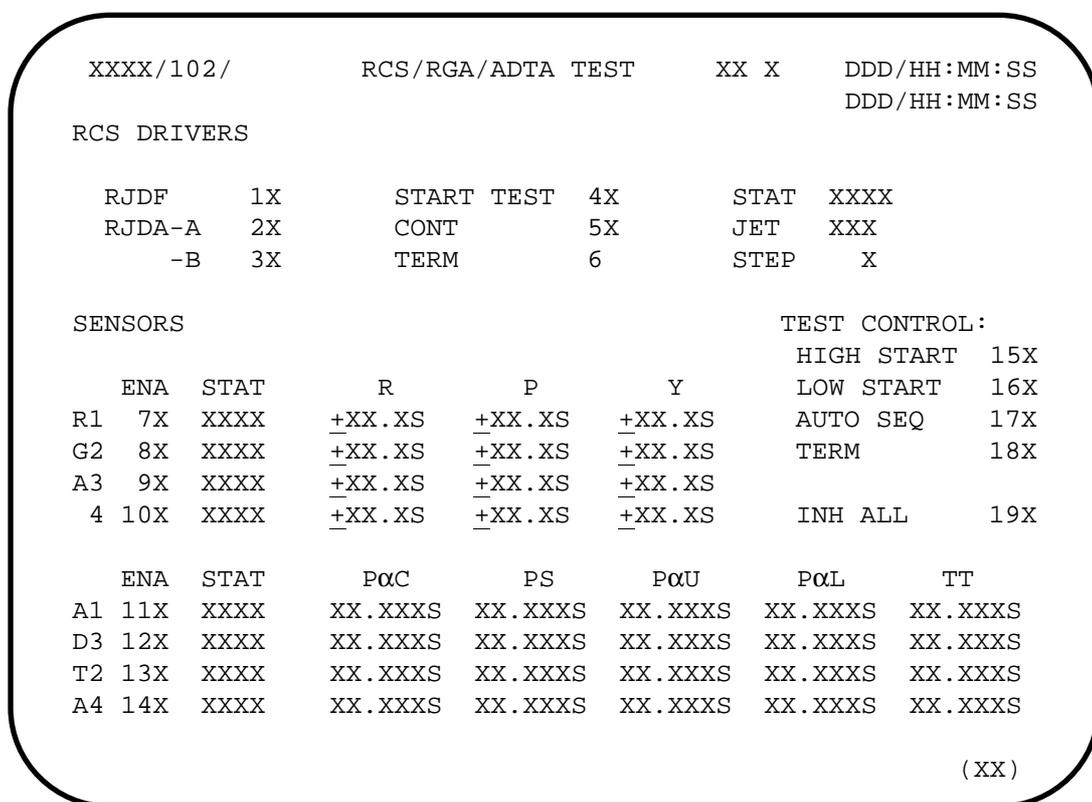


Figure 5-26, - Preflight RCS/RGA/ADTA test display (PFS)

6.1 INTRODUCTION 6
6.1.1 Physical Characteristics 8
6.1.1.1 Weight and Size 8
6.2 HARDWARE 8
6.2.1 Attitude Channels 8
6.2.1.1 Gimbal Description 11
6.2.1.1.1 Gimbal Angle Outputs 25
6.2.1.2 Resolver Description 25
6.2.1.3 Gyro Description 31
6.2.1.3.1 Slewing and Pulse Torquing Loops 34
6.2.1.3.1.1 Slew Capability 34
6.2.1.3.1.2 Pulse Torquing Capability 36
6.2.1.4 Redundant Gyro Axis Rate Channel Description 38
6.2.1.4.1 Performance/Design 40
6.2.1.5 Attitude Software Interface 43
6.2.1.5.1 Resolver processing (module name: GMD_RES_PROC) 43
6.2.1.5.2 Gyro compensation (GMK_GYO_COMP) 45
6.2.1.5.3 Gyro torque processing (GMF_GYO_TORQ) 46
6.2.1.5.4 Large angle torquing (GMM_LAT_FUNC) 46
6.2.1.5.5 Redundant gyro monitor comp (GMY_RGM_COMP) 46
6.2.2 Velocity Channels 47
6.2.2.1 Accelerometer Description 47
6.2.2.1.1 Performance Design 48
6.2.2.1.2 Acceleration Measurement Performance 50
6.2.2.1.3 Definition of Acceleration Terms 51
6.2.2.2 Velocity Software Interfaces 51
6.2.2.2.1 Accelerometer processing (GMC_ACP_ACUM) 52
6.2.2.2.2 Accelerometer Pulse Compensation (GMH_ACP_COMP) 52
6.2.2.2.3 Accelerometer pulse transformation (GML_ACP_TRSF) 53
6.2.3 Power and Grounding 53
6.2.3.1 Input power 53
6.2.3.2 Generated dc Voltage 54
6.2.3.3 Power Consumption 55
6.2.3.4 Grounding Requirements 55
6.2.4 System Cooling and Temperature Control 56
6.2.4.1 Cooling Air 56
6.2.4.2 Environmental Conditions 61
6.2.4.3 Temperature Control Circuitry 61
6.2.5 Electronics and Clocks 63
6.2.5.1 Electrical Interface 63
6.2.5.2 Platform Electronics 63
6.2.5.2.1 Preamplifiers 65
6.2.5.2.2 Spring Rate Compensation 65
6.2.5.2.3 GYPTO Torquing Electronics 65
6.2.5.3 Clocks 67
6.2.5.3.1 System Clocks 67
6.2.6 H/W Physical Alignment 67
6.2.6.1 Mounting 68
6.2.6.2 Mounting Alignment Repeatability 68
6.2.6.3 Hardware Alignments 68
6.3 OPS MODES 72
6.3.1 Off Mode 72
6.3.2 WARM-UP/STANDBY Mode 72
6.3.3 Start-up Sequence 74

- 6.3.4 Operate/Calibrate Mode..... 78
 - 6.3.4.1 HAINS Calibration and Acceptance Test 78
 - 6.3.4.1.1 Component Level Testing..... 79
 - 6.3.4.1.2 Integration Tests..... 79
 - 6.3.4.1.3 Acceptance Test Procedure (ATP) Tests..... 79
 - 6.3.4.1.3.1 Operational and Up Mode Test 79
 - 6.3.4.1.3.2 Multical Test 79
 - 6.3.4.1.3.3 Inner Roll Tracking Calibration Test 80
 - 6.3.4.1.3.4 Attitude Calibration & Accuracy Test (ACAT) . 80
 - 6.3.4.1.3.5 Short Term Stability Test 80
 - 6.3.4.1.3.6 Gimbal Flip Test 81
 - 6.3.4.1.3.7 Environmental Tests 81
 - 6.3.4.1.3.8 Heading Sensitivity Test 82
 - 6.3.4.1.3.9 Launchpad Verification Sequence 82
 - 6.3.4.1.4 Attitude Determination..... 82
 - 6.3.4.1.5 Ground Calibrations..... 83
 - 6.3.4.1.5.1 Factory CAL 85
 - 6.3.4.1.5.2 Hangar CAL 85
 - 6.3.4.1.5.3 Preflight CAL 88
 - 6.3.4.1.6 Alignments..... 92
 - 6.3.4.1.6.1 Preflight Alignment (GC_ALIGN & GC_ALIGN1) .. 92
 - 6.3.4.1.6.2 Inertial 96
 - 6.3.4.1.6.3 On-Orbit Alignments 97
- 6.4 HAINS INPUT/OUTPUT (I/O) CONTROL 104
 - 6.4.1 Discription..... 105
 - 6.4.1.1 Input Data 105
 - 6.4.1.2 Input Data Control 106
 - 6.4.1.3 Output Data 110
 - 6.4.1.4 Output Data Control 110
- 6.5 HAINS SOFTWARE 121
 - 6.5.1 Overview..... 121
 - 6.5.2 Executive..... 123
 - 6.5.2.1 Executive 123
 - 6.5.2.2 Real-Time Executive 124
 - 6.5.2.3 Cycle Executive 125
 - 6.5.3 HAINS Interface Function..... 127
 - 6.5.3.1 Input/Output 127
 - 6.5.3.1.1 Level I Interrupt Handler (RLEV1)..... 127
 - 6.5.3.1.2 Gyro Torquing (GYPTO)..... 127
 - 6.5.3.1.3 Gyro Compensation (GYRO_COMP)..... 127
 - 6.5.3.1.4 5 Hz Gyro Compensation (GY5COMP)..... 128
 - 6.5.3.1.5 Accelerometer Pulse Accumulation..... 128
 - 6.5.3.1.6 Long Term Accelerometer Compensation (AC1_COMP) 128
 - 6.5.3.1.7 Accelerometer 25 Hz Compensation (A25_COMP).... 129
 - 6.5.3.1.8 Accelerometer Pulse Compensation (ACP_COMP).... 129
 - 6.5.3.1.9 Redundant Gyro Processing (RGM_COMP)..... 130
 - 6.5.3.1.9 Redundant Gyro Processing (RGM_COMP)..... 130
 - 6.5.4 HAINS Input/Output Function..... 130
 - 6.5.4.1 Analog I/O 130
 - 6.5.4.1.1 Input A/D Data..... 130
 - 6.5.4.1.2 Resolver Processing at 25Hz..... 130
 - 6.5.4.1.3 8X Resolver Processing..... 131
 - 6.5.4.1.4 1X Resolver Processing..... 131
 - 6.5.4.2 MUX Function 131
 - 6.5.4.2.1 Input GPC Word..... 131

| | | |
|--------------|--|-----|
| 6.5.4.2.2 | <u>Input Words for PROVE Configuration</u> | 132 |
| 6.5.4.2.3 | <u>Output Words for PROVE Configuration</u> | 132 |
| 6.5.5 | HAINS BITES..... | 132 |
| 6.5.5.1 | Hardware BITES | 132 |
| 6.5.5.1.1 | <u>BITE Output</u> | 133 |
| 6.5.5.1.2 | <u>Description</u> | 133 |
| 6.5.5.1.2.1 | <i>Communication Interfaces</i> | 133 |
| 6.5.5.1.2.2 | <i>System Excitations</i> | 134 |
| 6.5.5.1.2.3 | <i>Velocity Channel</i> | 134 |
| 6.5.5.1.2.4 | <i>Attitude Channel</i> | 134 |
| 6.5.5.1.3 | <u>MUX Fail</u> | 134 |
| 6.5.5.1.3.1 | <i>MUX Start-up Test</i> | 134 |
| 6.5.5.1.3.2 | <i>DMA Controller BITE (DMA Request & Grant)</i> .. | 135 |
| 6.5.5.1.3.3 | <i>DMA Parallel Test</i> | 135 |
| 6.5.5.1.3.4 | <i>End-Around Transmission Test</i> | 136 |
| 6.5.5.1.4 | <u>A/D Fail</u> | 137 |
| 6.5.5.1.4.1 | <i>AC Channel Accuracy Check</i> | 137 |
| 6.5.5.1.4.2 | <i>DC Channel Accuracy Check</i> | 137 |
| 6.5.5.1.4.3 | <i>Gimbal Angles Checksum</i> | 137 |
| 6.5.5.1.4.4 | <i>A/D Converter Interrupt Goodness Check</i> | 137 |
| 6.5.5.1.4.5 | <i>A/D Performed BITE</i> | 138 |
| 6.5.5.1.4.6 | <i>Recurring A/D Fail BITES</i> | 139 |
| 6.5.5.1.5 | <u>Platform Fail</u> | 139 |
| 6.5.5.1.5.1 | <i>Gimbal Loop</i> | 140 |
| 6.5.5.1.5.2 | <i>19.2 kHz Excitation Current</i> | 140 |
| 6.5.5.1.5.3 | <i>Gyro Wheel Supply (GWS)</i> | 141 |
| 6.5.5.1.5.4 | <i>Sequencer Test</i> | 141 |
| 6.5.5.1.5.5 | <i>Pulse Accumulator Overflow Test</i> | 141 |
| 6.5.5.1.5.6 | <i>DAL Regulated Voltages</i> | 141 |
| 6.5.5.1.5.7 | <i>Inner Roll Reasonableness</i> | 142 |
| 6.5.5.1.5.8 | <i>1X/8X Gimbal Angle Comparison</i> | 142 |
| 6.5.5.1.5.9 | <i>Missing Resolver Inputs</i> | 142 |
| 6.5.5.1.6 | <u>Digital Subsystem Fail</u> | 142 |
| 6.5.5.1.6.1 | <i>Pulse Accumulator Test (SPULTST)</i> | 143 |
| 6.5.5.1.6.2 | <i>DOR Wraparound BITE</i> | 143 |
| 6.5.5.1.6.3 | <i>MUX Tag Timer Goodness</i> | 144 |
| 6.5.5.1.6.4 | <i>CPU Instruction Test</i> | 144 |
| 6.5.5.1.6.5 | <i>CPU Software Traps</i> | 144 |
| 6.5.5.1.6.6 | <i>RAM Pattern Test (RSTART)</i> | 144 |
| 6.5.5.1.6.7 | <i>EEPROM Checksum Test (XINIT)</i> | 145 |
| 6.5.5.1.6.8 | <i>PROM Checksum</i> | 145 |
| 6.5.5.1.6.9 | <i>Multiply Bit Error</i> | 145 |
| 6.5.5.1.6.10 | <i>Memory Error Detection and Correction Self-Test</i> | 145 |
| 6.5.5.1.7 | <u>DC/DC Fail</u> | 146 |
| 6.5.5.1.7.1 | <i>DC/DC</i> | 146 |
| 6.5.5.1.8 | <u>Discrete Power Supply Fail</u> | 146 |
| 6.5.5.1.8.1 | <i>Discrete Power Supply</i> | 146 |
| 6.5.5.1.9 | <u>Resolver Fail</u> | 146 |
| 6.5.5.1.9.1 | <i>Resolver Excitation Fail</i> | 147 |
| 6.5.5.1.10 | <u>Circuit Card Overtemp</u> | 147 |
| 6.5.5.1.10.1 | <i>Circuit Card Over-temperature</i> | 147 |
| 6.5.5.1.11 | <u>Watchdog Timer Fail</u> | 147 |

| | | |
|--------------|--|-----|
| 6.5.5.1.11.1 | <i>CPU Watchdog Timer</i> | 148 |
| 6.5.5.1.12 | <u>Transmission Word 1 Fail</u> | 148 |
| 6.5.5.1.13 | <u>Transmission Word 2 Fail</u> | 149 |
| 6.5.5.1.14 | <u>Communication Good</u> | 149 |
| 6.5.5.1.15 | <u>Platform Temp Ready Fail</u> | 149 |
| 6.5.5.1.16 | <u>Platform Temp Safe Fail</u> | 150 |
| 6.5.5.1.17 | <u>Temp Ready Fail</u> | 150 |
| 6.5.5.1.18 | <u>Temp Safe Fail</u> | 150 |
| 6.5.5.2 | Software BITEs | 150 |
| 6.5.5.2.1 | <u>Redundant Rate Fail</u> | 151 |
| 6.5.5.2.2 | <u>Inner Roll Null Fail</u> | 151 |
| 6.5.5.2.3 | <u>Resolver Limit Fail</u> | 151 |
| 6.5.5.2.4 | <u>Velocity Limit Fail</u> | 151 |
| 6.5.5.2.5 | <u>Echo 1 Fail</u> | 152 |
| 6.5.5.2.6 | <u>Echo 2 Fail</u> | 152 |
| 6.5.5.2.7 | <u>Accelerometer Scale Factor Fail</u> | 152 |
| 6.5.5.2.8 | <u>Failure Indications</u> | 152 |
| 6.5.5.2.8.1 | <i>Discrete Inputs</i> | 152 |
| 6.5.5.2.8.1 | <i>Discrete Outputs</i> | 153 |
| 6.5.5.2.8.2 | <i>Serial Output BITE/Status Word</i> | 155 |
| 6.5.5.2.9 | <i>Performance/Design</i> | 159 |
| 6.5.5.2.9.1 | <i>BITE Thresholds</i> | 159 |
| 6.5.5.2.9.2 | <i>BITE Summary Words (not used in-flight)</i> | 162 |
| 6.5.5.2.9.3 | <i>Turn-On Count (STONCNT)</i> | 162 |
| 6.5.5.2.9.4 | <i>Post-Flight Failure Storage</i> | 162 |
| 6.5.6 | HAINS Special Mode | 163 |
| 6.5.7 | HAINS Utility Function | 163 |
| 6.6 | IMU RM | 164 |
| 6.6.1 | Attitude RM | 166 |
| 6.6.1.1 | Selection Filter | 166 |
| 6.6.1.2 | Fault Detection | 168 |
| 6.6.1.3 | Fault Identification | 169 |
| 6.6.2 | Velocity RM | 171 |
| 6.6.2.1 | Selection Filter | 171 |
| 6.6.2.2 | Fault Detection/Identification | 175 |
| 6.6.3 | BFS RM | 176 |
| 6.7 | IMU IN-FLIGHT FAILURE HISTORY | 177 |
| 6.7.1 | KT-70 Flight Failures | 178 |
| 6.7.1.1 | STS-3 | 178 |
| 6.7.1.2 | STS-6 | 178 |
| 6.7.1.3 | STS-7 | 179 |
| 6.7.1.4 | STS-8 | 179 |
| 6.7.1.5 | STS-9 | 180 |
| 6.7.1.6 | STS-32 | 180 |
| 6.7.1.7 | STS-40 (Prelaunch) | 180 |
| 6.7.1.8 | STS-41 | 181 |
| 6.7.1.9 | STS-44 | 182 |
| 6.7.1.10 | STS-55 | 186 |
| 6.7.2 | HAINS In-Flight Failures | 187 |
| 6.7.2.1 | STS-63 (Prelaunch) | 187 |
| 6.7.2.2 | STS-65 | 187 |
| 6.7.2.3 | STS-75 | 189 |
| 6.7.2.4 | STS-80 | 189 |
| 6.7.2.5 | STS-81 | 190 |

SECTION 6
INERTIAL MEASUREMENT UNIT

6.1 INTRODUCTION

This system brief describes the Space Shuttle Orbiter High Accuracy Inertial Navigation System Unit (HAINS) overall performance, physical characteristics, interface with the Orbiter, modes of operation, and data and control channels. Details of the internal operational Flight Program (OFP) software (including BITEs) and calibration are also discussed. Lastly, an outline of the IMU redundancy management (RM) scheme is provided.

Briefly, the HAINS Space Shuttle Orbiter unit was designed to be used interchangeably with the old Space Shuttle KT-70 IMU's. The IMU is an all-attitude, four-gimbal, inertially stabilized platform with associated electronics. The HAINS unit provides three velocity components (Vx, Vy, Vz) along mutually orthogonal axes and four axes of attitude data (Outer Roll, Pitch, Inner Roll, and Azimuth Angles) to the Orbiter General Purpose Computer. These outputs are proportional to Orbiter's attitude and inertially derived velocity and are used to control the vehicle's flight. Each Orbiter carries three IMU's (for redundancy) which are mounted on the flight deck on a common navigation base.

The Orbiter general purpose computers (GPC's) receive extrapolated velocity and attitude data (except Inner Roll) from the IMU's through the flight-critical forward multiplexer/demultiplexers (MDM's FF) 1, 2, and 3. Data from the GPC's are also transmitted to the IMU's through the same MDM's. In short, the IMU's are under the constant control of the GPC's.

Figure 6-1 is a functional block diagram of the HAINS. This diagram depicts the functional interface of each major subassembly or component. The HAINS consists of an Inertial Platform, 12 Electronic Circuit Cards, and a Wiring Harness Assembly installed in a structural chassis.

Under the control of the Space Shuttle Orbiter's General Purpose Computers, the HAINS can be independently calibrated and aligned prior to launch. During flight, the HAINS provides attitude and velocity data to the Orbiter's computers. During orbit, the HAINS can be independently aligned by the Orbiter's star trackers.

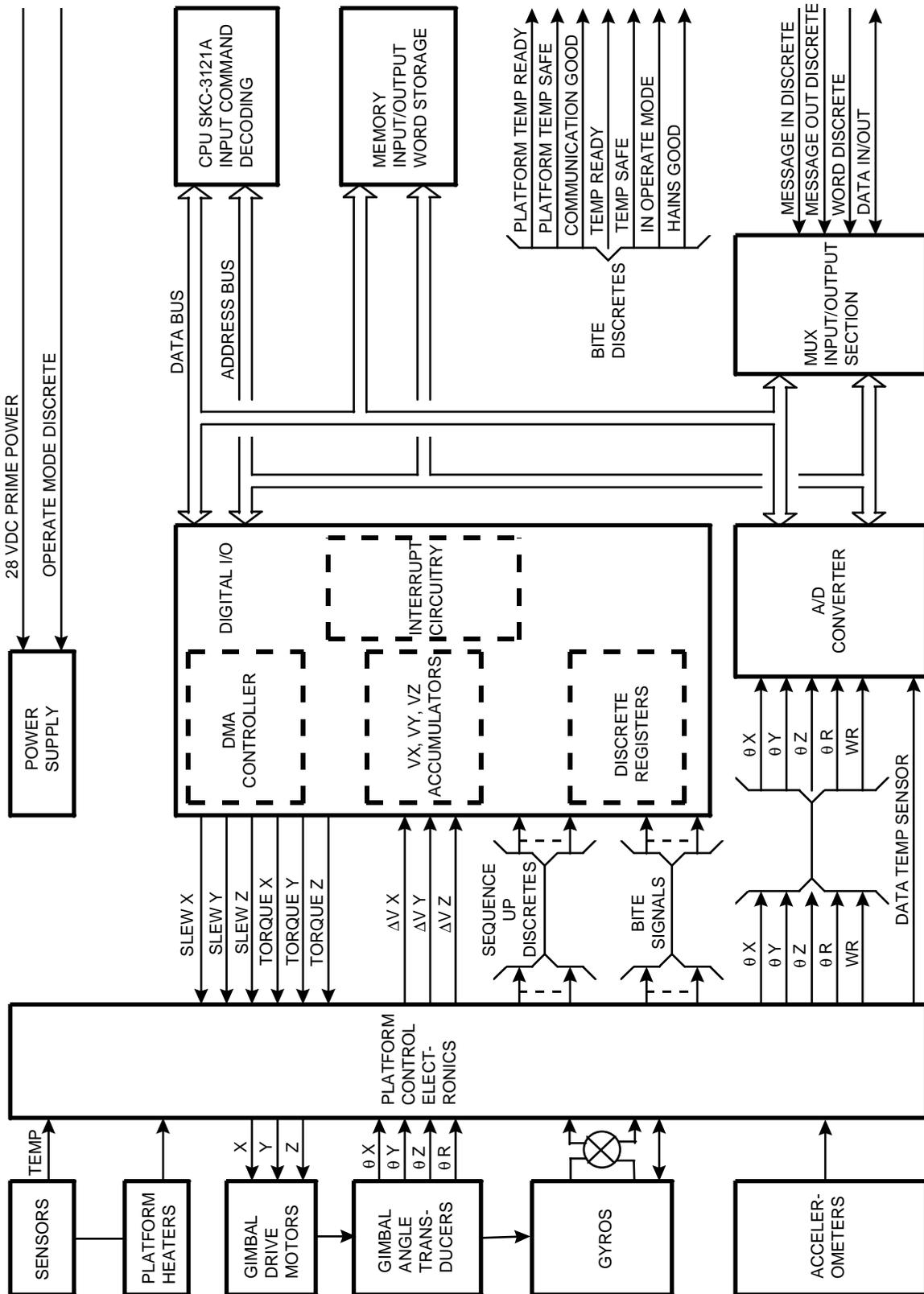


FIGURE 6-1, HAINS BLOCK DIAGRAM

6.1.1 Physical Characteristics

6.1.1.1 Weight and Size

Table 6-I summarizes the physical characteristics of the HAINS:

| CHARACTERISTIC | VALUE |
|----------------|----------------|
| Weight | 43.5 lbs (max) |
| Size: | |
| Height | 9.24 inches |
| Width | 10.88 inches |
| Length | 22.00 inches |

TABLE 6-I, HAINS PHYSICAL CHARACTERISTICS

6.2 HARDWARE

The HAINS consists of the following assemblies:

ASSEMBLY

- Chassis and Harness
- Inertial Platform
- Input Power Conditioner
- Multiplex Terminal Unit
- Input/Output Electronics
- Computer Processor Unit
- Support Circuit Card
- Memory
- Analog/Digital Converter
- DC/DC Power Supply
- Gyro Wheel Supply
- Platform Electronics No. 2
- Platform Electronics No. 1
- Ancillary Board

A detailed description of these assemblies and their functions are given in section three of the Orbiter High Accuracy Inertial Navigation System (HAINS) Technical Description Volume I of II document (document #Y256C392).

6.2.1 Attitude Channels

The Platform Assembly of the IMU contains four gimbals to enable inertial stabilization of the innermost gimbal to which the gyros and accelerometers are mounted (detailed descriptions of the gyros and accelerometers are presented later). The gimbal order from outermost to innermost is as follows (see Figure 6-2):

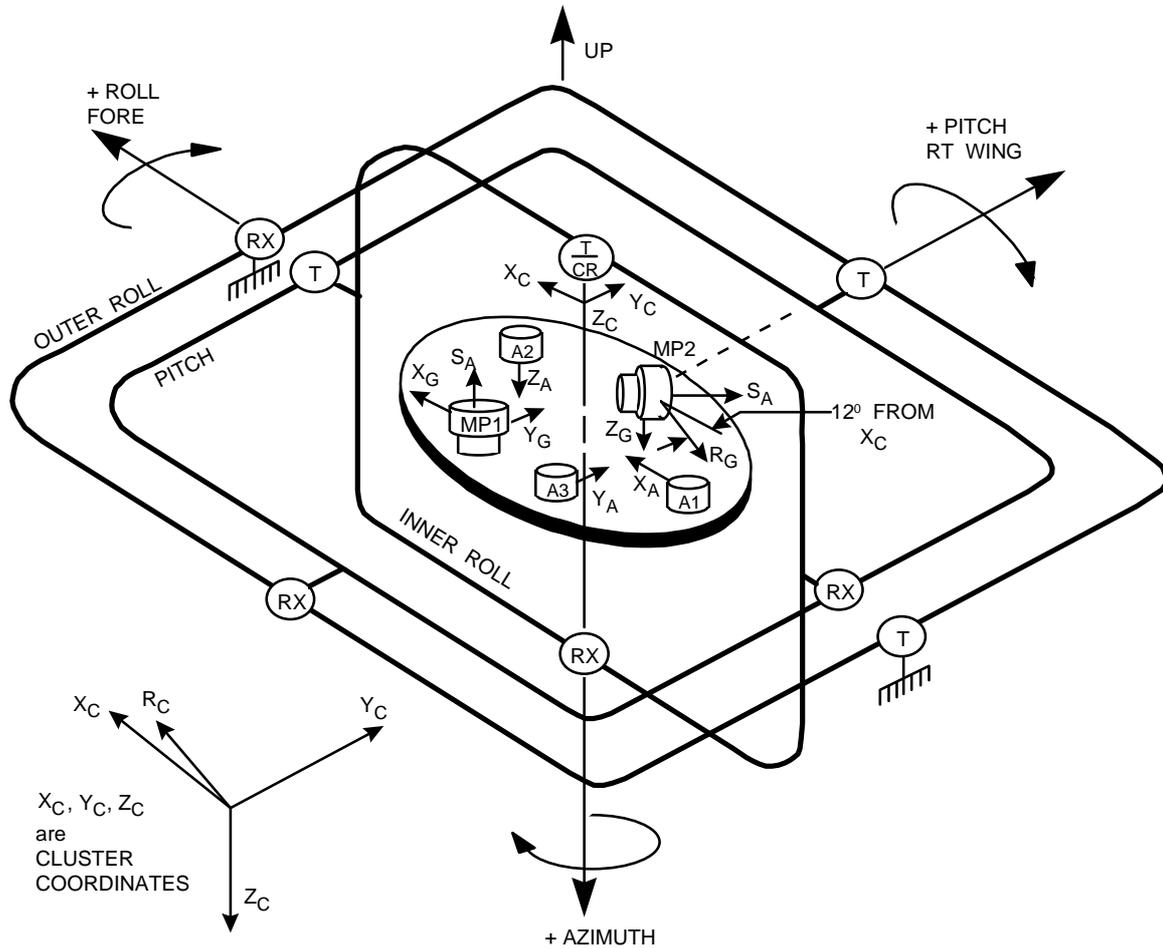
- a. Outer roll (X-axis) gimbal
- b. Pitch (Y-axis) gimbal

c. Inner roll (redundant axis) gimbal

d. Azimuth (Z-axis) gimbal

The Platform's gimbals are gyro-stabilized using signals derived from 2 two-axis GYROFLEX gyros mounted on the innermost gimbal. The gyros sense cluster motion; their pickoff signals are amplified by the gimbal loop electronics and the appropriate gimbals are torqued via direct drive DC torquers to null the motion sensed by the gyros. This operation is done with such high gain electronics that the platform does not really move before the gimbals respond to the gyro signal. In this fashion, the cluster maintains its inertial orientation while the gimbals move in response to vehicle motion. Gimbal position can be controlled by torquing the gyros through the high rate torquers (slew) or through the precise, fine torquers on the gyro under computer control.

Precision gimbal angle readouts are provided by dual speed resolvers , which can be thought of as potentiometers having 1-speed (1X) and 8-speed (8X) outputs on each of the 3 gimbals, Azimuth, Pitch, and Outer Roll. A more detailed description of resolvers is given in a following section. The resolver outputs are sent to the A/D converter where they are converted into a digital form directly representing the gimbal angles that the GPC's uses to determine vehicle attitude. The 4th gimbal, Inner Roll, has a linear pickoff excited by 19.2 kHz source which is conditioned by the PE No. 1 card for a coarse A/D input and by the PE No. 2 card for a fine A/D input. The A/D converter converts these signals into a digital form representing the Inner Roll gimbal angles.



- NOTES:
- $X_G = X$ GYRO I. A. } VERTICAL GYRO
 - $Y_G = Y$ GYRO I. A. }
 - $Z_G = Z$ GYRO I. A. } AZIMUTH GYRO
 - $R_G = REDUNDANT$ GYRO I. A. }
 - $Z_A = Z$ ACCEL SENS. AXIS → SINGLE AXIS ACCEL.
 - $X_A = X$ ACCEL SENS. AXIS → SINGLE AXIS ACCEL.
 - $Y_A = Y$ ACCEL SENS. AXIS → SINGLE AXIS ACCEL.
 - (T) = GIMBAL TORQUER
 - (RX) = GIMBAL ANGLE RESOLVER

Figure 6 - 2, Cluster axes.

6.2.1.1 Gimbal Description

This section describes the means used to position the gimbals of the HAINS platform and describes how the gimbals are controlled to inertially stabilize the stable element. The gimbal angle data channels provide accurate attitude angle information to the Orbiter General Purpose Computer (GPC), via the MDM, for each of the four gimbals (Outer Roll, Pitch, Inner Roll, and Azimuth).

The gimbal isolation loops are used to maintain the orientation of the stable element (cluster) at any desired Orbiter attitude. These loops isolate the stable element from the effects of motion of the Orbiter which would otherwise disturb the stable element's inertial orientation. The X-, Y-, and Z-gyros are used to sense any rotation of the stable element about their respective X-, Y-, and Z-sensing axes. When the vehicle has rotational motion, the gyro or gyros with sensing axes oriented in the direction of the axis of rotation sense the motion and generate 19.2 kHz signals at their pickoff outputs. These signals are preamplified then sent to the isolation loop electronics where they are amplified, demodulated, and servo-compensated. The signals are then used to excite the proper gimbal torquers which cause the gimbals to rotate in a direction to move the gyros back to the null positions of the gyro pickoffs, thereby closing the isolation loops. The gain of this loop is so high and the dynamics are so fast that the isolation operations are practically instantaneous and the stable element remains essentially undisturbed (inertial).

The four gimbal loops consist of the sensor, electrical demodulator, integrator (except for outer roll) used in the high-gain isolation mode, a low-gain voltage amplifier used in the low-gain isolation mode (during sequencing up the platform), a notch filter (except for outer roll), summing amplifier and power output amplifiers. Mode switching within the loop circuits is accomplished by solid state electronic switching.

The configurations of the isolation loops are shown in Figure 6-3. The outputs of the X- and Y-gyros are resolved, as a function of the azimuth gimbal angle, into components along the inner roll and pitch gimbal axes such that they cause the required rotation of these gimbals to maintain the X- and Y-gyro pickoffs at their null position. The inner roll gimbal is a redundant (fourth) gimbal that is used to provide an all-attitude IMU while preventing the possibility of gimbal lock (a condition which will cause the inertial platform to lose its reference). A resolver that is driven by the inner roll gimbal is used to excite the outer roll electronics thereby causing the outer roll gimbal to absorb the roll motion and leaving the inner roll gimbal at its null position. This also causes the inner roll gimbal to remain orthogonal to the pitch gimbal.

When the pitch gimbal angle is 0° , the inner roll and outer roll axes are colinear and therefore have maximum kinematic coupling to one another. At exactly $\pm 90^\circ$ of pitch, however, the inner roll and outer roll axes are orthogonal and no coupling exists. This condition is known as the gimbal flip condition. When this condition exists, any motion other than pure pitch will cause rapid movements of the outer roll and azimuth gimbals (hence the name gimbal flip) and further pitch movements will cause the pitch gimbal to move back toward null. The movement of the outer roll and azimuth gimbals results in the IMU attitude's being described by a different Euler sequence. Since any attitude can be described by a number of different Euler sequences, the gimbal flip has no effect on the inertial reference or the inertial attitude processing software. If, however, the IMU is in the gimbal flip region and pure pitch motion (to within an arc-minute or so) is continued, the gimbals may not flip and the IMU will lose its reference (commonly called platform tumble).

However, this condition is minimized by the addition of a Pitch switch, on the Platform Electronic No. 2 card, which increases the gain of the outer roll loop when the pitch angle is 92° degrees. The switch is controlled by the OFP software. The gain of the loop is decreased, following gimbal flip, when the pitch angle decreases to 88° degrees. As little as 1 arc-minute Orbiter offset from a pure pitch travel through $\pm 90^\circ$ degree pitch will insure that gimbal flip occurs.

The probability of a tumble condition occurring is very small. Furthermore, the three HAINS are initially aligned so that their cluster orientations are skewed relative to one another. Hence only one of the above HAINS would be exposed to the possibility of a tumble condition. In general, during the gimbal flip itself, the inertial orientation of the stable element is maintained.

The final axis that has not been addressed is the Z isolation loop. Unlike the X- and Y-loops, the Z-axis is straightforward. The Z-gyro causes closed loop gimbal action only via the azimuth torquer which drives the Z-gimbal. Figures 6-4, 6-5, and 6-6 are schematic diagrams of the isolation loops.

The gimbal system and its servo loops are capable of following vehicle motions with maximum angular rates of 35 deg/sec and maximum angular accelerations of 35 deg/sec² (Orbiter linear acceleration of 6 g's) while maintaining the inertial reference within specification accuracy requirements. For sinusoidal-type motions, whether caused by maneuvers or vehicle flexing, the IMU performs well within the limits shown in Figure 6-7.

Although the outer roll, pitch, and azimuth gimbals have unlimited freedom of rotation, the inner roll gimbal is mechanically limited to a maximum travel of $\pm 16^\circ$.

The gimbals each have two single-speed (1X) and eight-speed (8X) resolver outputs, with the exception of the inner roll gimbal.

Since the inner roll gimbal has limited rotational freedom, the 1X outputs are not needed. The resolver outputs are proportional to the sine and cosine of the actual gimbal angle and are 400-Hz outputs.

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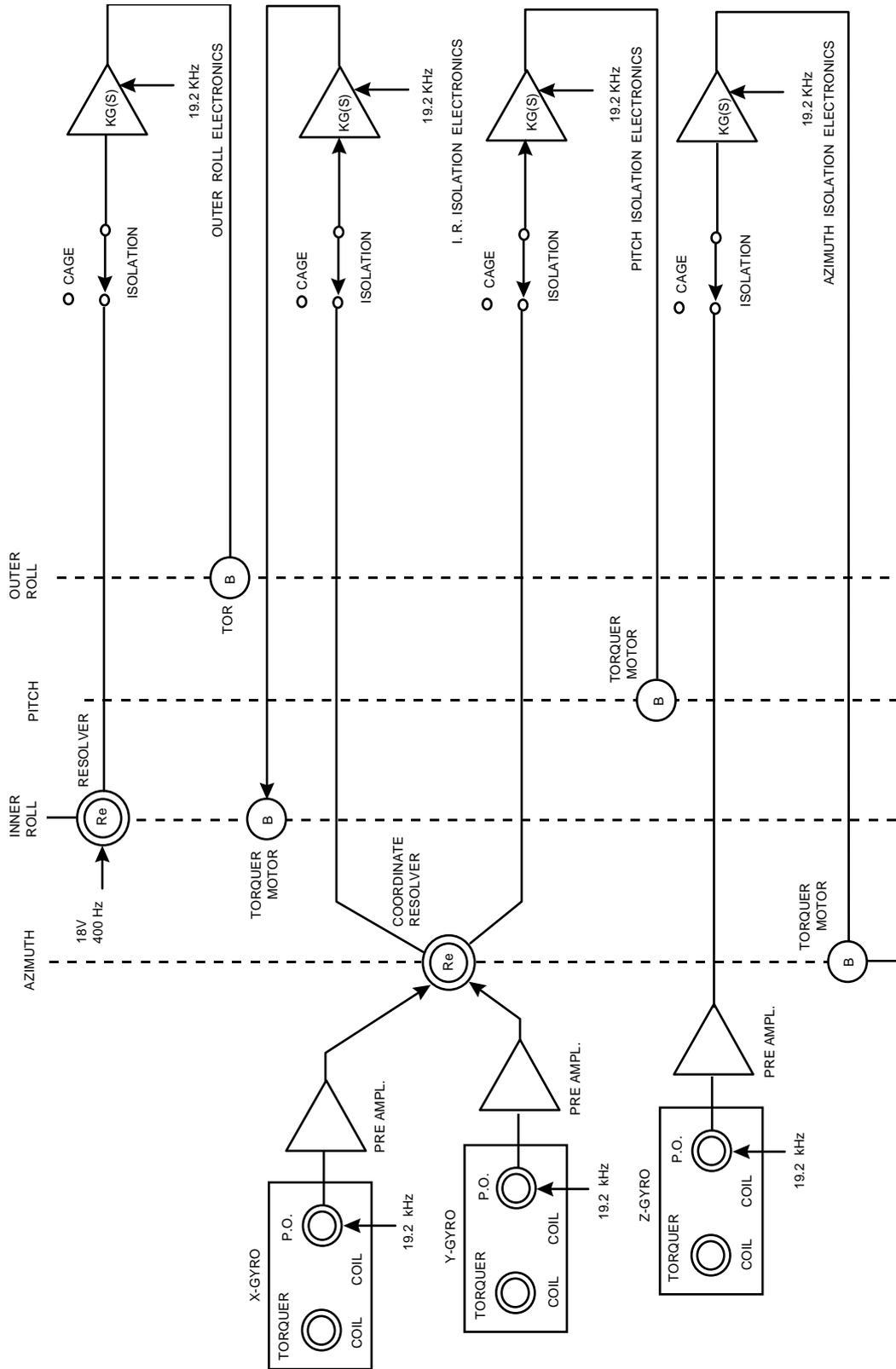


Figure 6 - 3, Isolation loops (simplified).

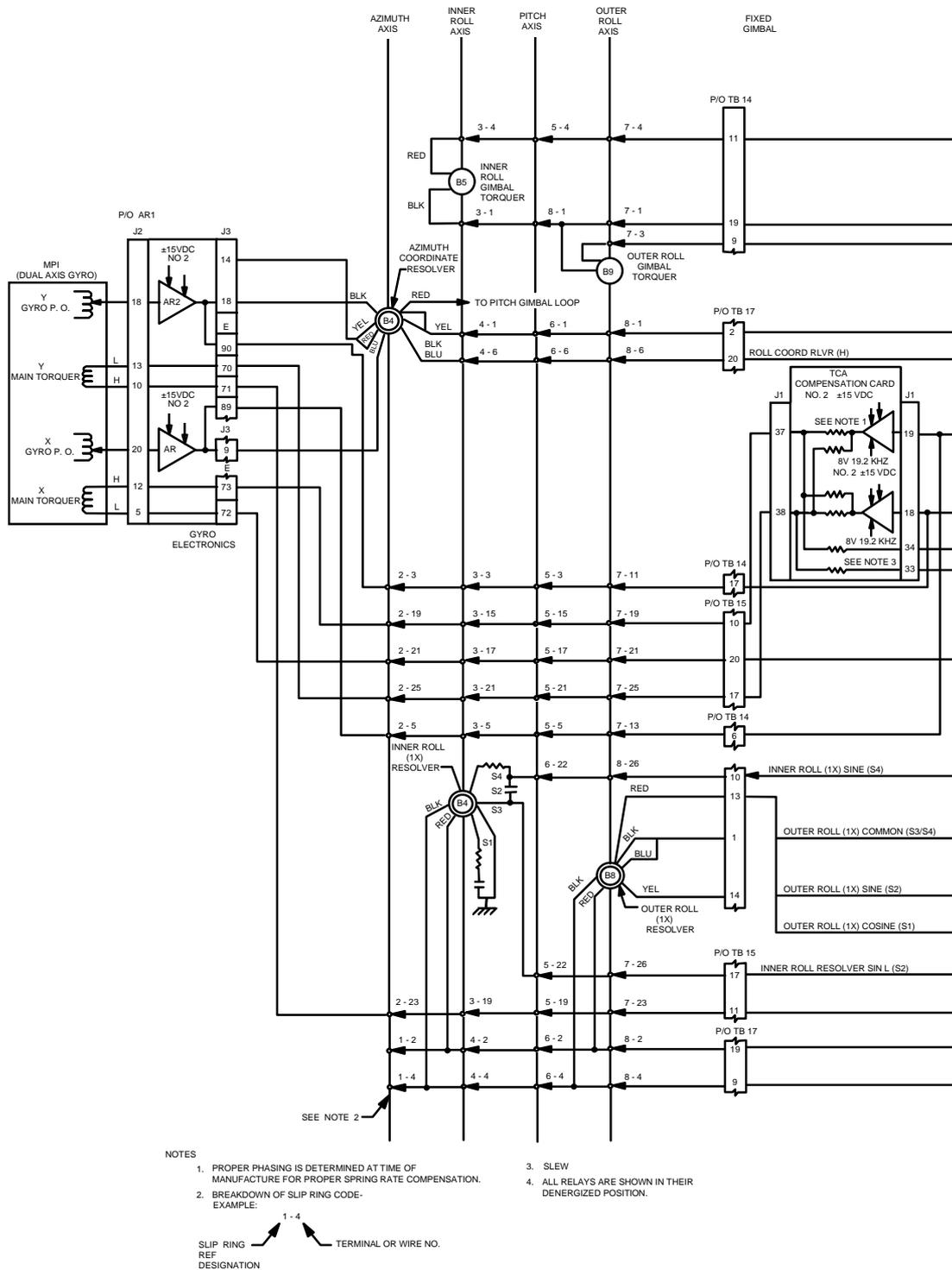


Figure 6 - 4 (sheet 1 of 2). - Schematic diagram, inner and outer roll cage isolation loop.

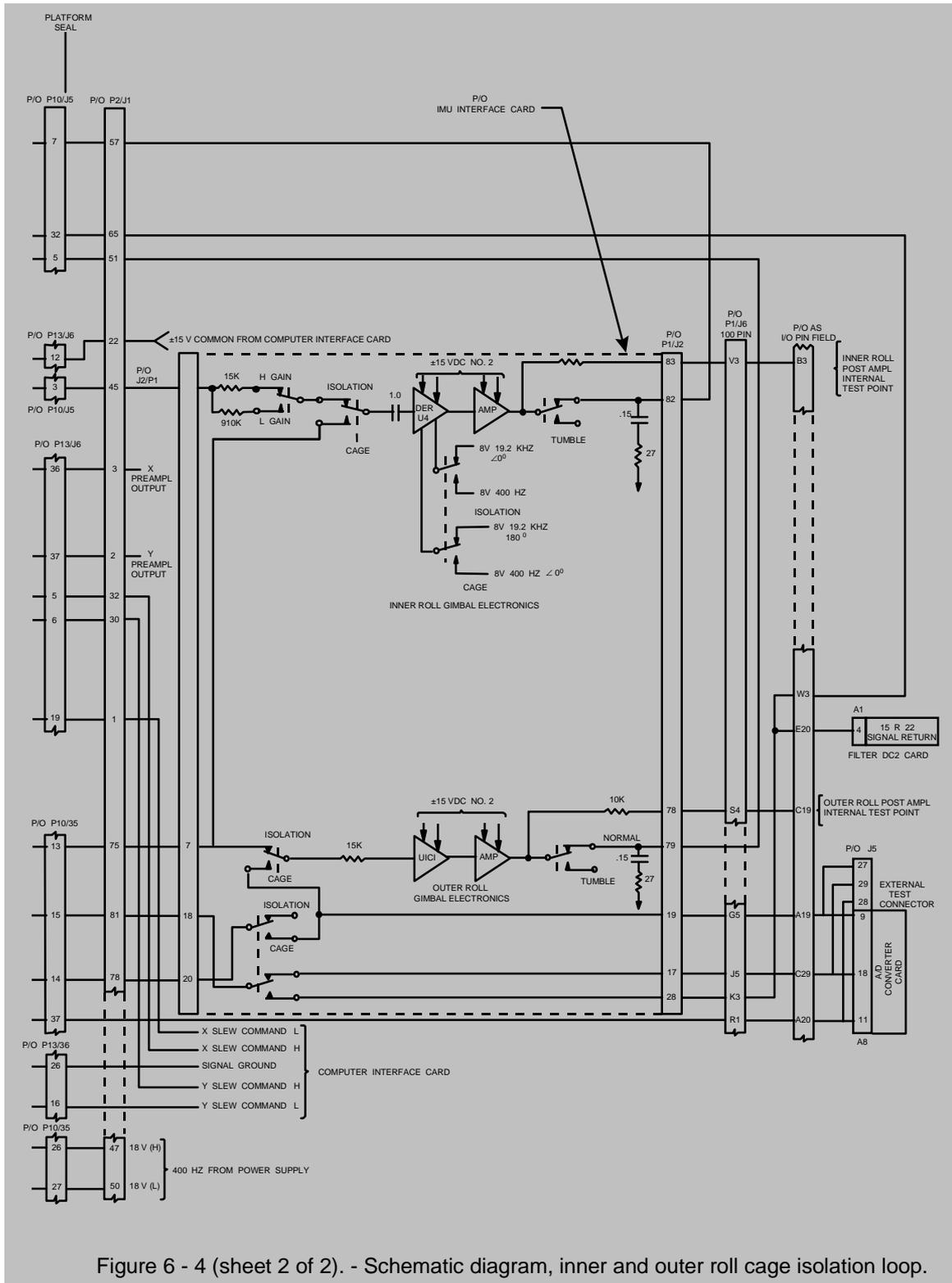


Figure 6 - 4 (sheet 2 of 2). - Schematic diagram, inner and outer roll cage isolation loop.

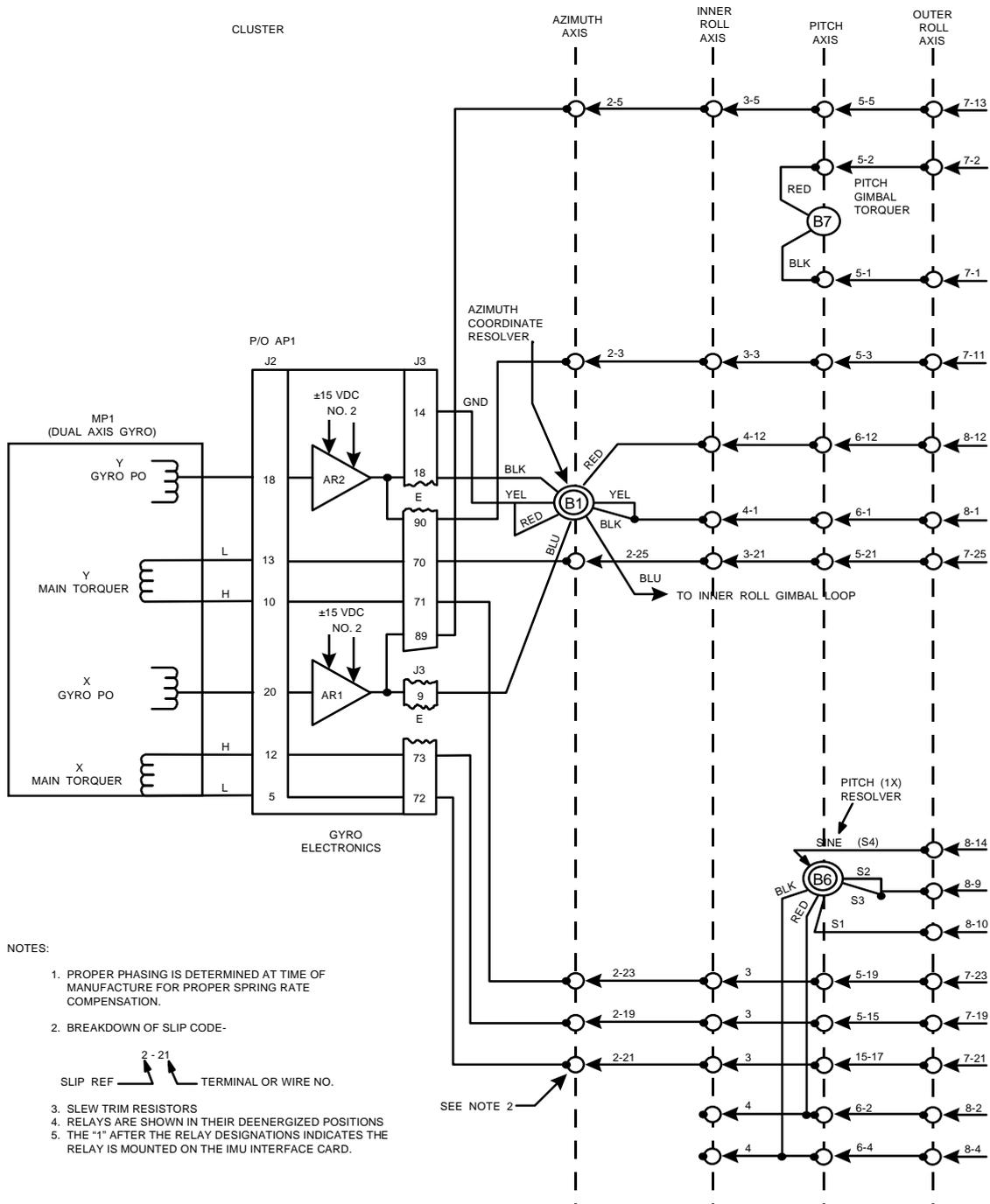


Figure 6 - 5 (Sheet 1 of 3) - Schematic diagram, pitch cage and isolation loop.

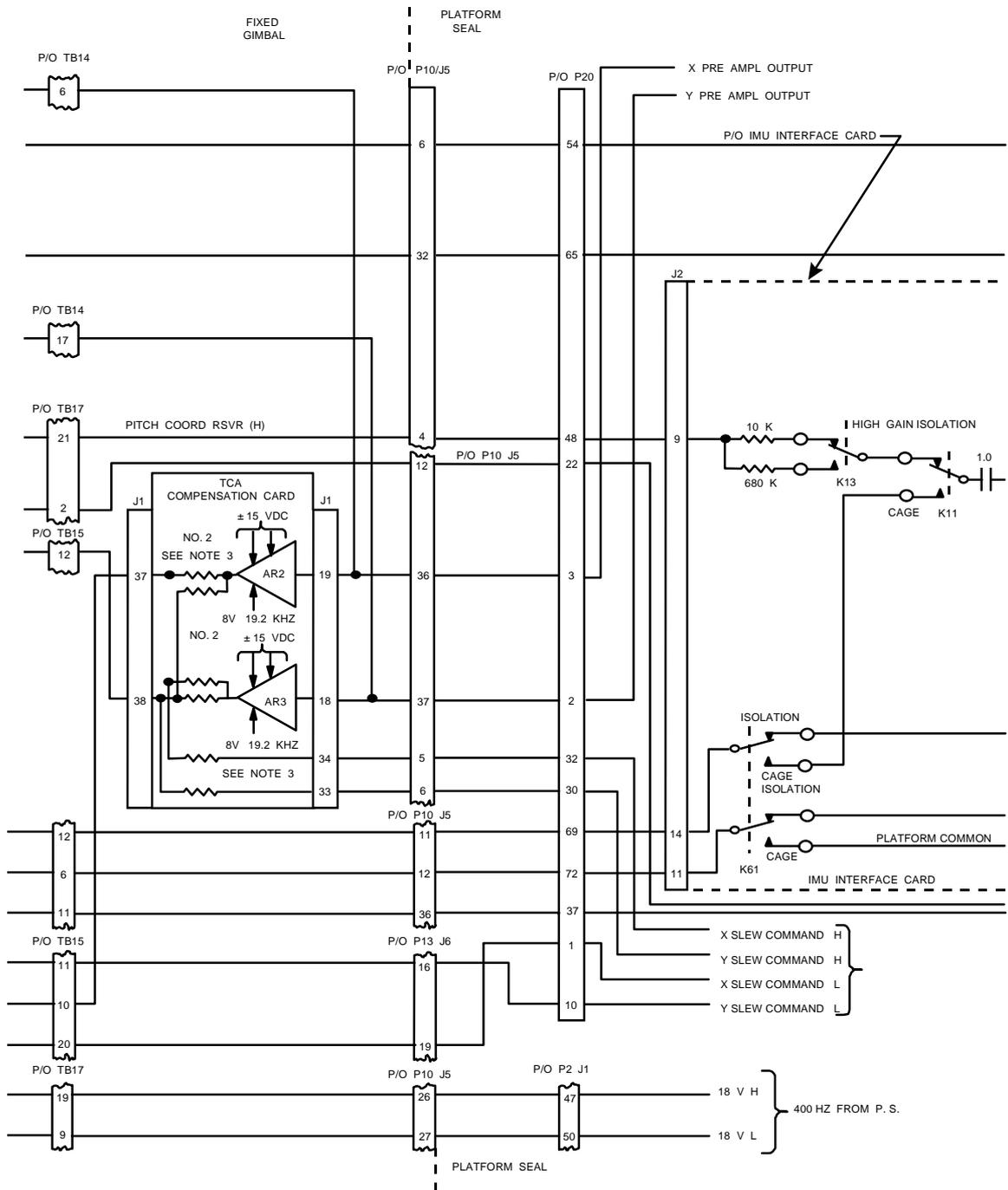


Figure 6 - 5 (Sheet 2 of 3) - Schematic diagram, pitch cage and isolation loop.

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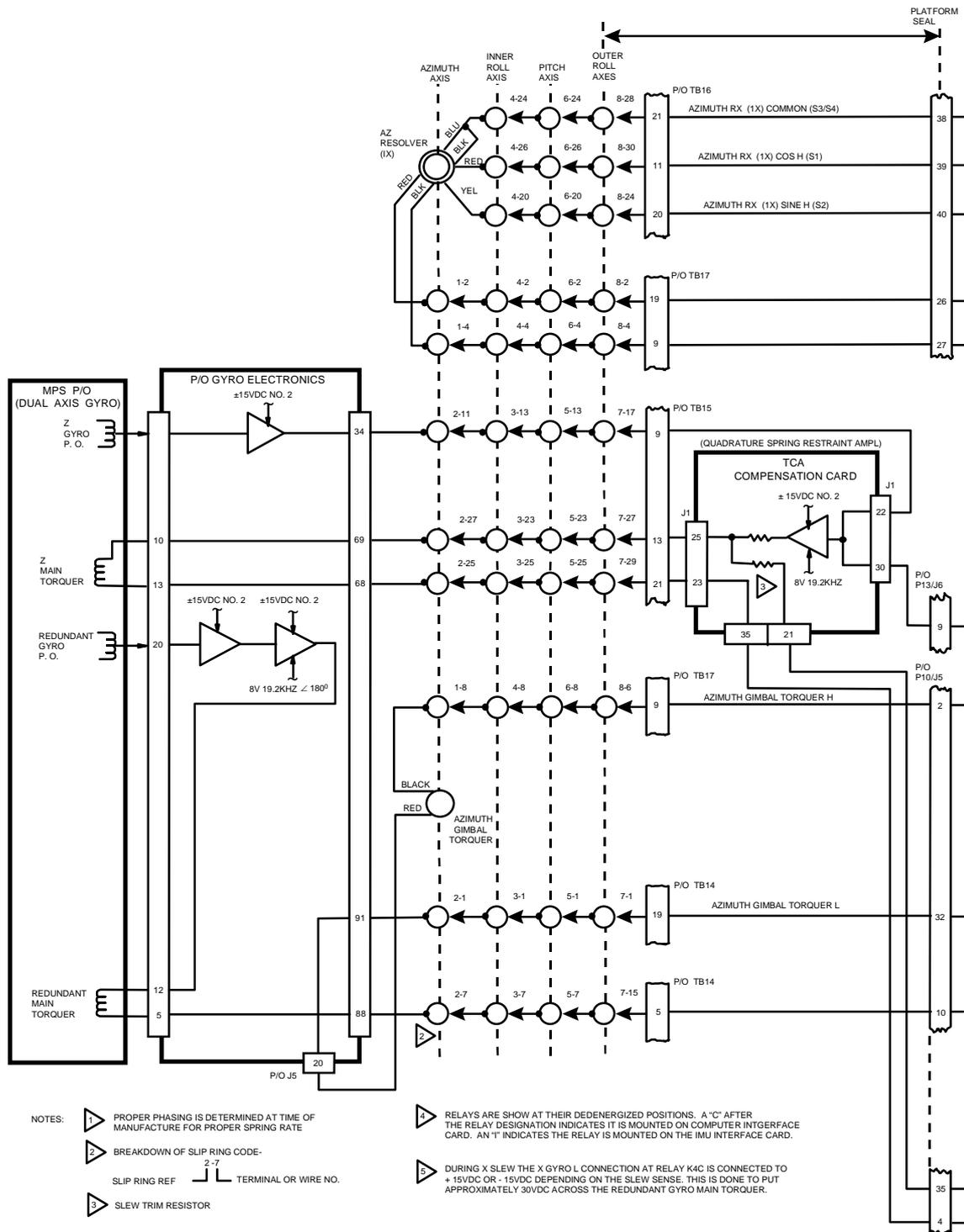


Fig. 6-6 (Sheet 1 of 2) - Schematic diagram, azimuth cage and isolation loops.

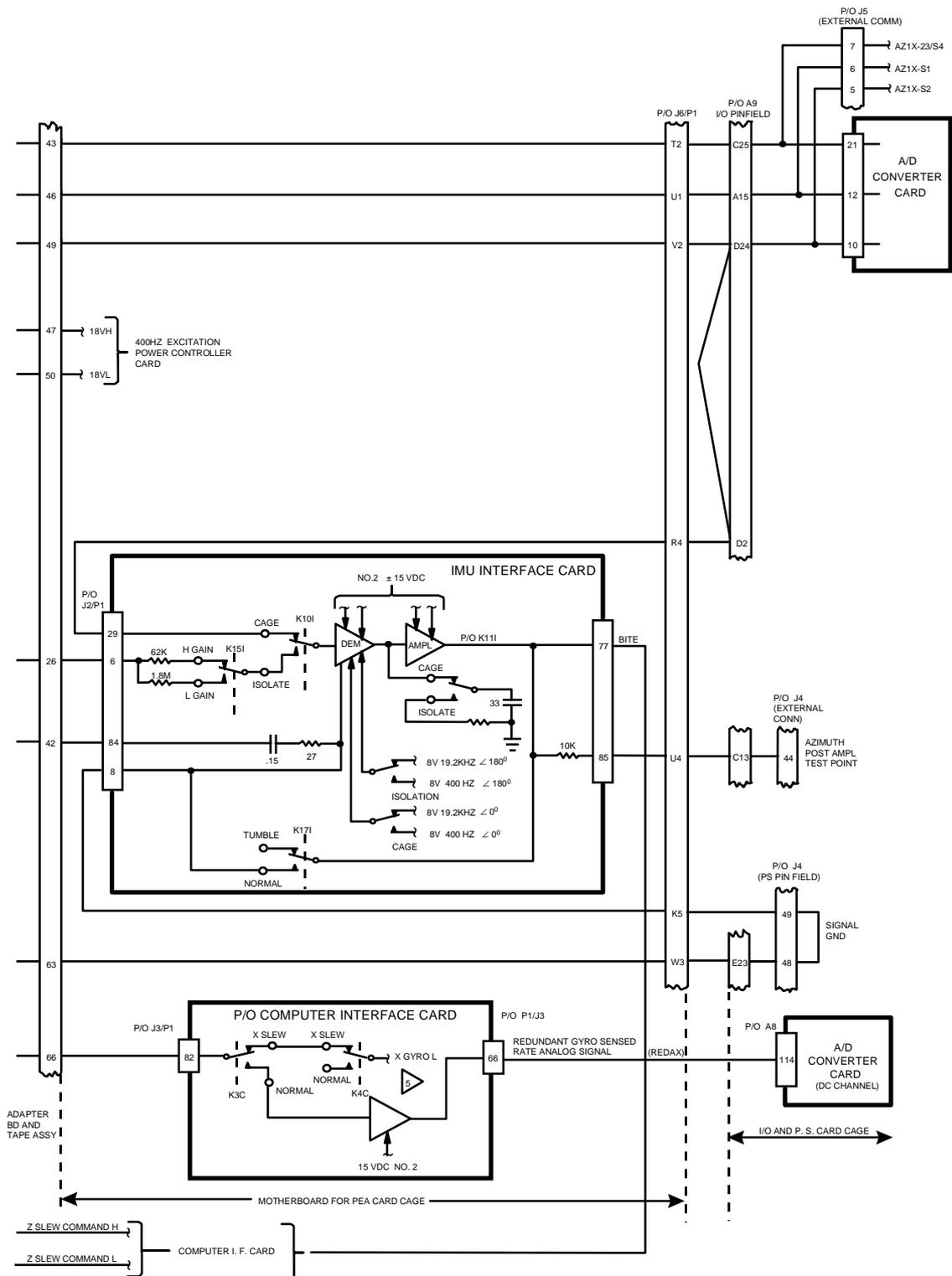


Fig. 6-6 (Sheet 2 of 2) - Schematic diagram, azimuth cage and isolation loops.

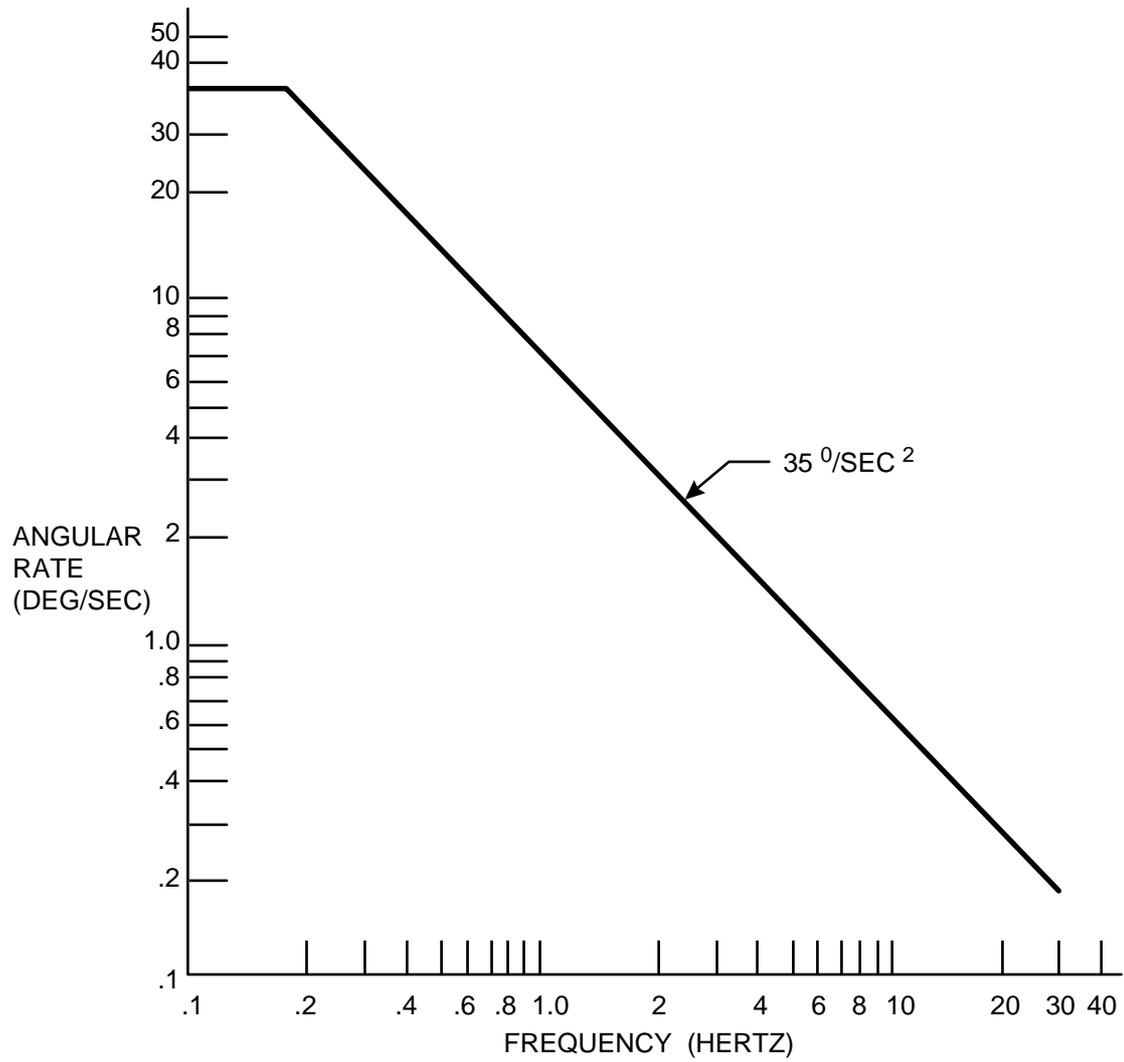


Figure 6 - 7 - Angular rate capability.

6.2.1.1.1 Gimbal Angle Outputs

| | |
|---------------|---|
| Resolver Null | With both the azimuth axis vertical and the HAINS housing level, the single-speed outputs of each gimbal resolver are 0 degrees +/- 30 arc minutes, max. This tolerance includes the null set and its stability and the stability of the DC component of each resolver error curve. |
| Accuracy | The error relative to the shaft position of the eight speed electrical zero for the one-speed outputs of the azimuth, pitch, and outer roll resolvers are +/- 20 arc minutes, maximum, over 360 degrees. The errors of each of the eight-speed Azimuth, Pitch, and Outer Roll resolvers are calibrated over 360 mechanical degrees for the azimuth and outer roll axes, and +/- 80 degrees for the pitch axis, and the calibration data is stored in memory for automatic compensation of the errors. Similarly the inner roll linear 19.2 kHz pickoff errors are calibrated and the data is used for automatic compensation. |

6.2.1.2 Resolver Description

Precision gimbal angle readouts are provided by dual speed resolvers having 1-speed (1X) and 8-speed (8X) outputs on each of 3 gimbals, Azimuth, Pitch, and Outer Roll. The resolver outputs are sent to the A/D converter where they are converted into a digital form directly representing the angles. The 4th gimbal, Inner Roll, has a linear pickoff excited by 19.2 kHz source which is conditioned by the PE No. 1 card for a coarse A/D input and by the PE No. 2 card for a fine A/D input. The A/D converter converts these signals into a digital form representing the Inner Roll gimbal angles.

The dual-speed resolvers produce outputs proportional to $\sin q$ and $\cos q$ by rotor and stator windings similar to the diagram in Figure 6-8.

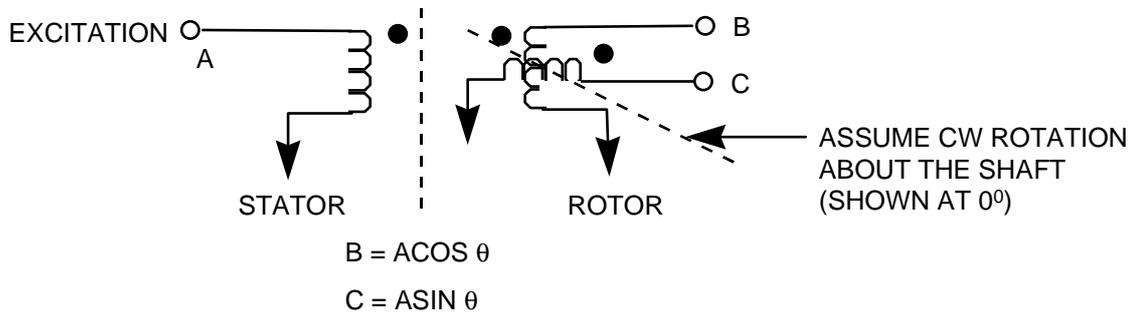


Figure 6 - 8 - Stator windings.

The diagram is simplified when compared to the actual resolvers, but the design theory is the same. As the angular relationships change between the windings on the stator and rotor, the induction in the magnetic field changes, producing an output at points B and C in the diagram. The output varies as the sine and cosine of the angle between the windings.

The A/D converter converts the $\sin q$ and $\cos q$ outputs (both 1X and 8X) into a 13-bit tangent q or cotangent q data word, where q represents the total gimbal angle. The A/D operation, shown in block diagram form in Figure 6-8, is initiated by the multiplexer sequencer.

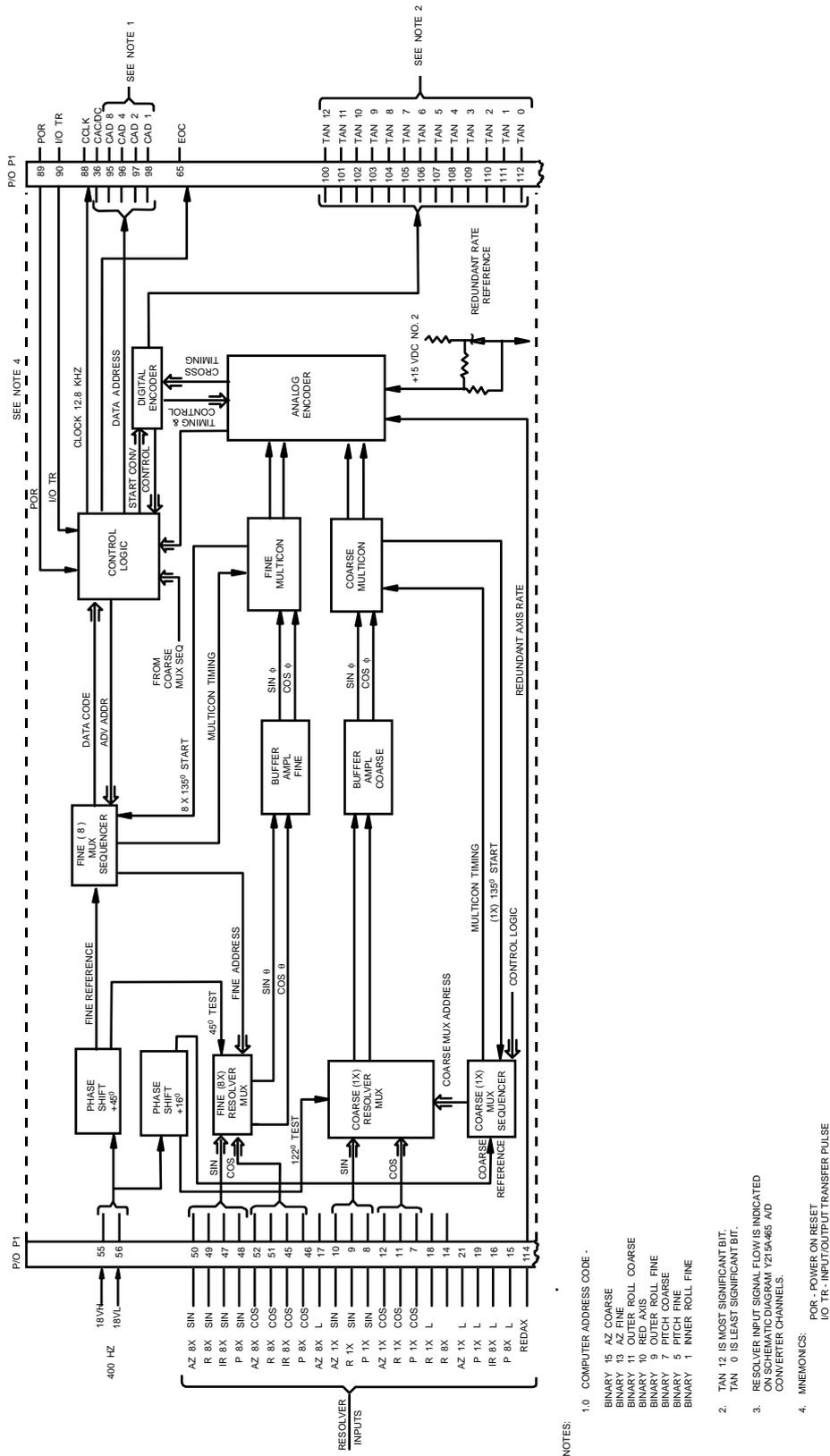


Figure 6 - 9, Functional diagram, A/D converter.

- NOTES:
- 1.0 COMPUTER ADDRESS CODE -
 BINARY 15 AZ COARSE
 BINARY 14 AZ FINE
 BINARY 13 OUTER ROLL COARSE
 BINARY 12 OUTER ROLL FINE
 BINARY 11 RED AXIS
 BINARY 10 RED AXIS
 BINARY 9 OUTER ROLL FINE
 BINARY 8 OUTER ROLL COARSE
 BINARY 7 PITCH COARSE
 BINARY 6 PITCH FINE
 BINARY 5 PITCH FINE
 BINARY 4 INNER ROLL FINE
 BINARY 3 INNER ROLL COARSE
 BINARY 2 INNER ROLL COARSE
 BINARY 1 INNER ROLL FINE
 2. TAN 12 IS MOST SIGNIFICANT BIT.
 TAN 0 IS LEAST SIGNIFICANT BIT.
 3. RESOLVER INPUT SIGNAL FLOW IS INDICATED ON SCHEMATIC DIAGRAM Y215A-465 A/D CONVERTER CHANNELS.
 4. MNEMONICS:
 POR - POWER ON RESET
 I/O TR - INPUT/OUTPUT TRANSFER PULSE

The sequencer awaits the zero crossing of the ac carrier of the first resolver channel to be processed. The zero crossing closes the multiplexer switches of the first resolver channel, and the signals are buffered and passed to the signal conditioning circuitry (MULTICON). The signal conditioning is accomplished by applying the sine and cosine inputs to (RC) networks for a fixed timing interval (135° of the 400 Hz carrier). The appropriate choice of the time of switch closure, effective capacitance, and effective resistance minimize quadrature, harmonics, and noise effects. The resolver outputs, now dc voltages from the signal conditioners, are sent to the analog encoder. The encoder passes the dc signals through an integrator for a fixed time, T, where T is the time required to fill a counter initially set to zero to a counter full condition. The counter is then reset to zero and a reference voltage of the opposite polarity of the input signal (resolver outputs) is applied to the integrator. The counter measures the time required to discharge the integrator circuit to zero as detected by the comparator. By checking the signs of the sine and cosine inputs, the encoder determines whether to make a tangent or cotangent conversion. Assuming $K_1 \sin q$ and $K_2 \cos q$ as inputs, where K_1 and K_2 represent scale factor differences in the inputs, a $\tan q$ conversion takes the form

$$K_1 (\sin \theta) \frac{T}{RC} - K_2 (\cos \theta) \frac{t}{RC} = 0$$

$$t = \frac{K_1}{K_2} (\tan \theta) T$$

For a cotangent conversion:

$$K_2 (\cos \theta) \frac{T}{RC} - K_1 (\sin \theta) \frac{t}{RC} = 0$$

$$t = \frac{K_2}{K_1} (\cot \theta) T$$

The IMU A/D converter then converts the gimbal angle output into a 13-bit tangent q or cotangent q data word, where q represents the gimbal angle. The three most significant bits (MSB's) locate the octant the angle lies in and the converter uses the octant

code to represent the magnitude tan q or cot q (restricted to a value from 0 to 1). As an example, assume the converted 13-bit word is expressed as:

| | | | | | | | | | | | | | |
|-----------------------|---|---|---|------------------------|---|---|---|---|---|----|----|----|----|
| Bit | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 |
| Value | 0 | 1 | 1 | 0 | 0 | 0 | 0 | 0 | 1 | 1 | 0 | 0 | 1 |
| Most significant bits | | | | Least significant bits | | | | | | | | | |

From the first three bits, the converter knows that the angle is in the fourth octant (value of the three MSB's is 3 and 3 X 45° = 135°; adding the value of the 10 least significant bits (LSB's) yields the total angle). Table 6-II indicates when the tangent function is used and when the cotangent function is used.

| Octant | Binary code | Function represented by 10 LSB's |
|--------|-------------|----------------------------------|
| 1 | 000 | Tan |
| 2 | 001 | Cot |
| 3 | 010 | Cot |
| 4 | 011 | Tan |
| 5 | 100 | Tan |
| 6 | 101 | Cot |
| 7 | 110 | Cot |
| 8 | 111 | Tan |

TABLE 6-II, A/D FUNCTION DECISION

In the example, the three MSB's indicated the angle was in the fourth octant. Referring to the table, the 10 LSB's represent the tangent function. Therefore,

$$\text{Tan } q = \frac{25}{2^{10}} = \frac{25}{1024} = .0244$$

The angle q = 1.3985°, and the total angle is 135 + 1.3985° or 136.3985°.

The 1X resolver outputs are sampled every 12 msec using one basic circuit in a multiplex scheme to generate dc voltages proportional to sin q and cos q, when again q is the particular gimbal angle being sampled. The 8X resolvers are similarly sampled but at a 5 msec rate using a different sampling circuit

than that used for the 1X outputs. One converter and multiplexer are used to convert all of the 1X and 8X sampled data from dc voltages to a 13-bit binary digital word proportional to $\tan q$ or $\cot q$. The data then enter a look up data read only memory (ROM) to obtain the whole angle equivalent to $\tan q$ or $\cot q$. The whole angle data is represented by a 13-bit binary word with the MSB representing an angle of 180 electrical degrees.

The interpolator card receives both 1X and 8X resolver data; however, only the 8X data is interpolated. The 1X data enters the card only for storage and is buffered via a serial multiplexer until the Orbiter computer commands the data to be transmitted. The main purpose of the interpolation is to provide a data snapshot of all gimbal angles in the IMU that represent a good estimate of the set of angles existing at an instant of time.

The interpolation scheme for the gimbal angle data is shown in Figure 6-10. The last two 8X data samples taken from the A/D converter are shown in the figure. The interpolator retains only the last two samples for each gimbal and performs a linear interpolation between the samples at an update rate of 64 times between samples. Since the A/D converter samples data every 5 msec (every two cycles of the 400 Hz resolver excitation), the interpolation update is done every 78 μ sec. When the Orbiter computer command arrives at the IMU (shown at 12 msec in figure 6-10), the interpolator freezes the data corresponding to a time exactly 5 msec prior to the present time (shown at 7 msec). This scheme minimizes the relative data staleness between all 8X data channels by essentially forcing each channel to have the same amount of absolute staleness.

The IMU sends separate 1X and 8X data words to the GPC's where the words are combined to form one precision angle word. The 1X output is used primarily to determine the octant of the total angle; and, consequently, the 1X resolver accuracy is not critical. The 8X resolver, however, precisely locates the actual angle within the octant specified by the 1X output, and its accuracy is therefore very critical. The specifications for the resolvers are shown in Table 6-III.

| | 1X | 8X |
|----------------------|---------------------|---------------------|
| Accuracy | 10 arcmin, 1s | 21 arcsec, 1s |
| Excitation frequency | 400 Hz \pm .015 % | 400 Hz \pm .015 % |
| Excitation voltage | 18 V RMS \pm 5 % | 18 V RMS \pm 5 % |

TABLE 6-III, RESOLVER SPECIFICATIONS

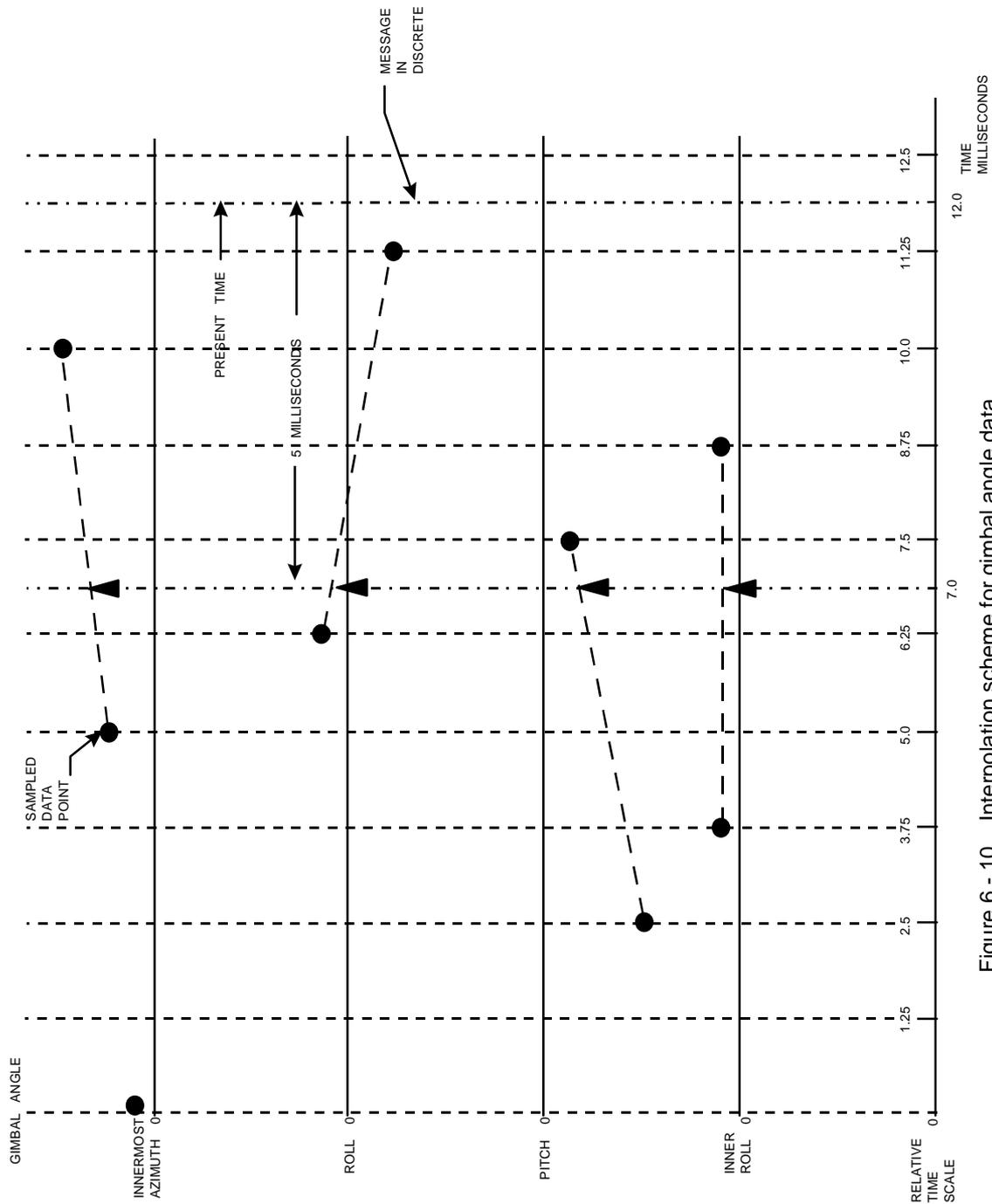


Figure 6 - 10, Interpolation scheme for gimbal angle data.

6.2.1.3 Gyro Description

The IMU contains two two-degree-of-freedom gyros used for platform stabilization. One gyro is oriented such that its input axes are aligned with the X- and Y-platform axes and the second gyro is oriented such that one input axis lies along the platform Z-axis. The other input axis of the Z-gyro lies in the XY-plane and is used as a platform rate detector. Its function will be

discussed in the built-in test equipment (BITE) portion of this report.

The two gyros are identical in construction. The gyro wheel is suspended by a dry flexure joint (Figure 6-11) which not only supports but also pivots the wheel.

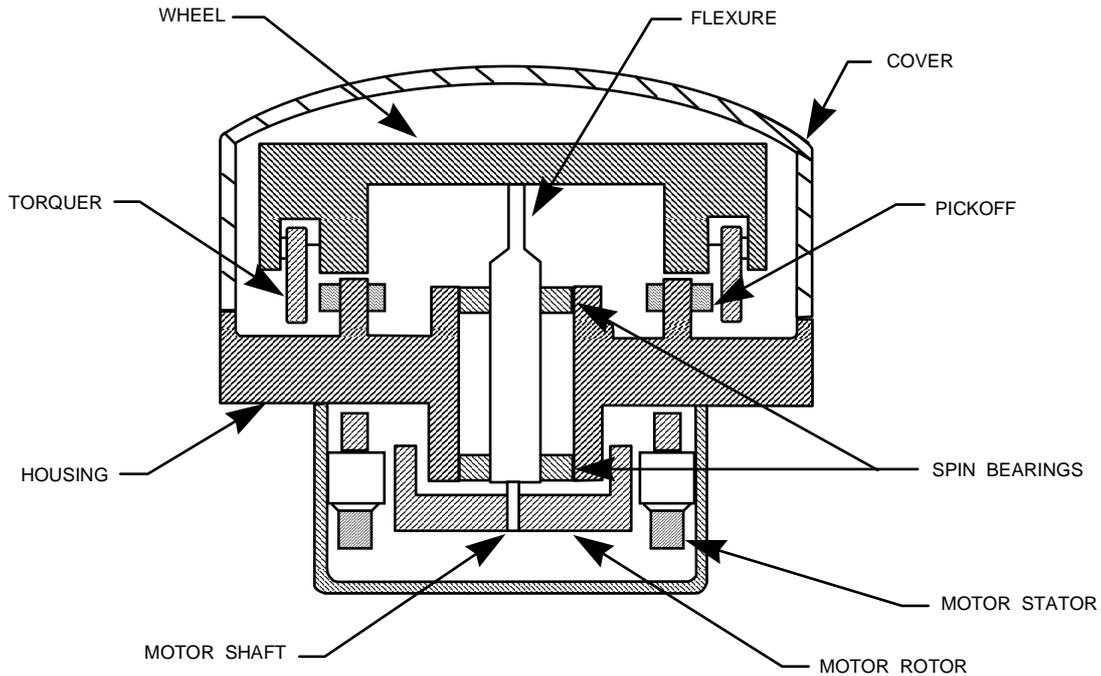


Figure 6 - 11, Gyroflex gyro mechanical schematic.

The design of this joint is such that all spherical friction restraints are eliminated. The joint is the part of the shaft driven by a synchronous-hysteresis motor. The deflection of the gyro wheel due to an input torque is measured by a two-axis pickoff. A two-axis torquer provides compensation torquing for gyro drift. The size and weight of the gyro are shown in Table 6-IV.

| | |
|----------|----------|
| Diameter | 2.13 in. |
| Length | 1.70 in. |
| Weight | 0.67 lb |

TABLE 6-IV, GYRO SIZE AND WEIGHT

The motor which drives the gyro wheel is a two-phase synchronous-hysteresis motor excited by a 19-V peak-to-peak two-phase, 480-Hz square wave. The motor shaft is supported by a pair of precision ball bearings which are preloaded to raise the natural frequency of the wheel well above the synchronous speed.

As previously stated, the suspension system connects one end of the spin motor shaft to the wheel. The system consists of a two-piece flexure which provides support along and perpendicular to the spin axis. In addition to supporting and driving the wheel, the suspension system also provides angular freedom about two perpendicular axes. The combination of axial and radial flexures is designed to have equal spring rates, thereby achieving near-perfect isoelasticity. High radial and axial stiffness attained with these flexures also raises the natural frequency of the wheel well above the synchronous speed. The wheel is then free to pivot about two orthogonal axes without frictional torque. The pickoff, which senses wheel deflections, operates on the differential transformer principle and consists of four U-shaped, soft iron cores around which are wound four pairs of coils. Thus, a pair of coils is located in each quadrant of the pickoff stator. This stator is an integral part of the gyro housing, with the face of the stator perpendicular to the spin axis of the gyro.

The required gyro performance is summarized in Table 6-V.

| Parameters | Range | 1 σ values |
|---|--|---------------------------------|
| X, Y, Z GYRO BIAS (RESTRAINT DRIFT DFX, DFY, DFZ) | ± 3 deg/hr | 0.006 deg/hr ALL ATTITUDE |
| X, Y, Z GYRO SCALE FACTOR KTX, KTY, KTZ | 0.5 +0.185 ARC-SEC/PULSE 0.5 -0.050 ARC-SEC/PULSE | 200 PPM 200 PPM |
| G-SENSITIVE DRIFT (X, Y GYRO & Z GYRO): MASS UNBALANCE DRIFT DIX,DIY,DIZ QUADRANT DRIFT DSX,DSY,DSZ,DOZ | ± 2.0 DEG/HR/G ± 2.0 DEG/HR/G | 0.15 DEG/HR/G (ALL ATTITUDE) |
| G-SQUARED DRIFT (X,Y GYRO & Z GYRO) | - | 0.005 DEG/HR/G |
| GYRO RANDOMNESS | - | 0.0021 DEG/HR |

TABLE 6-V, GYRO PERFORMANCE CHARACTERISTICS

The gyro randomness requirement of 0.0021 deg/hr, one sigma, is specified to assure that the 60 arc-second, one-sigma, azimuth alignment requirement is achieved. This is a gyrocompassing requirement. The other gyro performance requirements comply directly with Orbiter IMU Procurement Specification MC409-0126.

6.2.1.3.1 Slewing and Pulse Torquing Loops

The IMU uses the gyros not only for platform stabilization but also for platform repositioning. Each gyro has torquing coils on each axis such that inputs to these coils cause a deflection of the gyro in that direction. Since the stabilization loops are always trying to null the gyro deflection, purposely causing a deflection will cause gimbal movement and hence platform movement. Under control of the GPC's, the platform can be slewed at a rate of 72 deg/min. The slewing loops of the IMU are used to quickly position the gimbals of the IMU to desired orientations. After arriving at approximately the desired gimbal orientation, the gyro pulse torquing loops, which have better resolution but slower rotational speed, are used to finely position the gimbals to the desired orientations.

6.2.1.3.1.1 *Slew Capability*

In this submode, the computer can independently command the X, Y, and Z gyro axes to receive slew commands in the required sense for a desired time period. The commands are sent via Input Data Word No. 2 to the HAINS. The HAINS I/O decodes this word and sends appropriate torquing commands in the form of fixed direct

current (DC) levels to the analog torquers of the pertinent gyros.

The X, Y, and Z axes are capable of being slewed at a minimum rate of 56 deg/min under control of commands from the Orbiter GPC. Figure 6-12 is a functional diagram of the slewing loops. The GPC's issue slew commands to the IMU which excite relays in the slew logic. This in turn causes a differential voltage of approximately 30 V dc to be applied across the main torquer coil (slew torquer) of the pertinent gyro axis to which a slew command has been issued. The slew direction is controlled by the polarity of the differential voltage. The slew commands cause the gyro to precess, which causes the pickoff to sense an off null condition. The isolation loop electronics then drive the proper gimbal torque motor in a direction to attempt to null the pickoff output. Since the slew command causes the torquer coil to stay energized, the gimbals will continue to try to null the gyro output until the slew command is removed. The GPC's check each computer cycle to determine if the desired gimbal angles have been reached. When the desired angles are achieved, the slew command is removed and the stabilization loops null the gyro output to maintain the stable member orientation. The slew logic is capable of slewing the platform at a rate of 72 deg/min about any one axis or any combination of axes to within 1° of the desired angles.

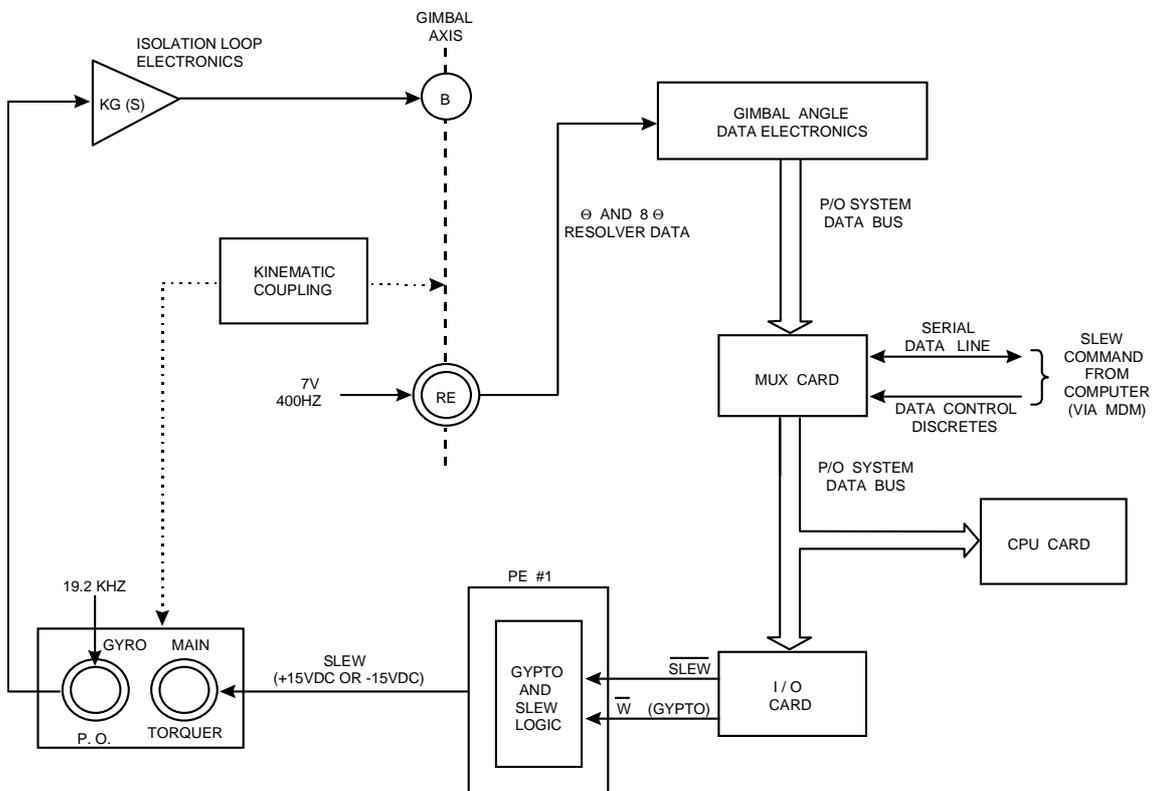


Figure 6 - 12. Typical Gimbal Slewing Loop.

6.2.1.3.1.2 Pulse Torquing Capability

In this submode, the computer can independently command that fine gyro torquing signals in the form of pulses be sent to the X, Y, and Z gyro fine torquer axes. The commands are sent via Input Data Word No. 1 to the HAINS. Five bits in this word are used to designate the fine torquing to be accomplished for each gyro axis. One of these bits indicates the desired torquing sense, the other four indicate the number of pulses to be issued to the pertinent gyro torquer.

During flight, the IMU platform is inertially controlled. The Gyro Pulse Torquing (GYPTO) commands can be issued to the gyros in any of the flight modes (launch, orbit, entry). The GYPTO loops are used when it is desired to accurately orient the stable member to a desired position or when it is desired to maintain a given orientation by compensating for effects such as, earth's rate and/or gyro bias drift rates. A typical GYPTO torquing loop functional diagram is shown in Figure 6-13. This one-channel representation is the same for all three GYPTO channels.

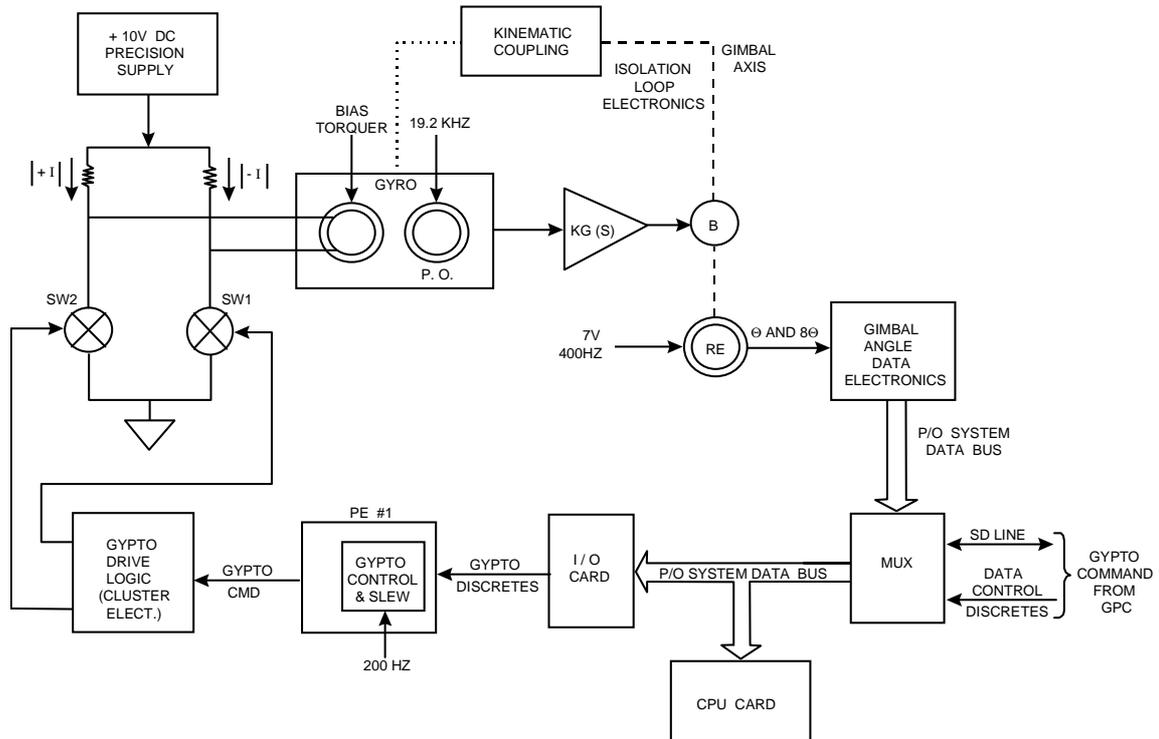


Figure 6 - 13. Typical GYPTO Torquing Loop.

The individual GYPTO commands are determined by the Orbiter GPC and are sent to the HAINS in Input Word No. 1. The commands are processed by the CPU which converts the torquing command into a torquing rate. In addition, the internal software will, if

needed, compensate the torquing rates for gyro misalignments and restraint drift. For the Input Data Word No. 1 (see Table 6-XXIII in section 6.4.1.2 for more detail), the most significant command bit of the 5 bits/channel contains the sign (sense) of the input command and the remaining 4 bits indicate the number of precision DC current torquing pulses that are to be applied to the pertinent gyro. Each torquing pulse lasts for a period of 1/200 second = 5 milliseconds, and causes a nominal angle of 0.5 arc-second to be torqued out. The isolation loop responds to gyro torquing commands in the same manner as previously described for slewing.

The GYPTO electronics (Figure 6-14) are contained in a hybrid module located on the platform cluster. Synchronized GYPTO torquing pulses from the platform electronics (W_x , W_y , and W_z) are used to switch a precision current source to the GYPTO torquing coils. The W_x , W_y , and W_z control is determined by the operational flight program (OFP) and issued through output discretes to the PE No. 1 card. The synchronized W_x , W_y , and W_z outputs are then fed to the GYPTO hybrid module in the platform. Since the issuing of the torquing commands would normally vary with the computer instruction times and direct memory access (DMA) times, a precise current waveform is maintained by synchronizing the GYPTO to the 200 Hz clock. The OFP determines, and spreads out, the required number of torquing pulses during a 40 msec period.

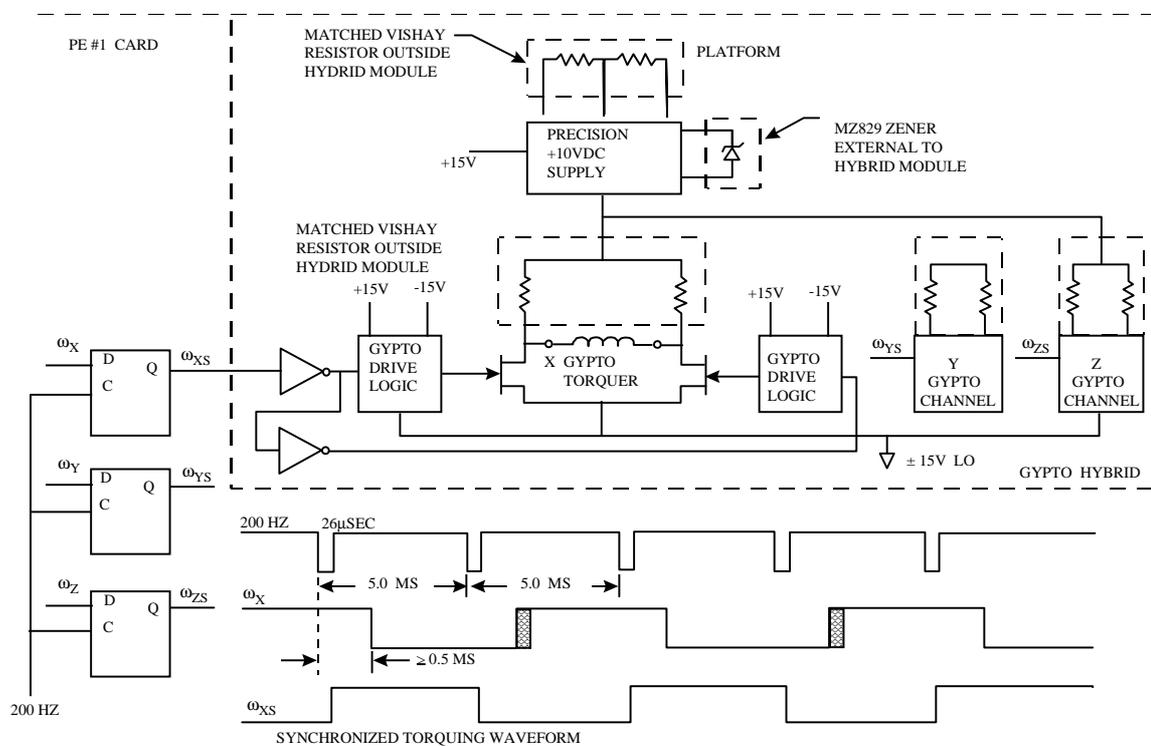


Figure 6 - 14. - GYPTO Functional Schematic

To explain how the GYPTO loops function, assume a command for two positive pulses has been received by one of the loops. An UP/DOWN counter will be counted down at a 200 Hz rate until the counter indicates zero pulses. In this case, a switch would close to allow a positive precision current to enter the gyro bias torquer (much smaller than the main torquer) for two 5 msec pulse periods (10 msec total). After the 10 msec, the circuitry automatically causes the precision current to toggle from plus to minus at 5 msec intervals until the next GYPTO command is received. The countdown of the next GYPTO command will not begin until a full toggle cycle is complete.

Since the maximum input word update rate from the GPC is 25 times per second (every 40 milliseconds) and the GYPTO clock has a frequency of 200 Hz +/-0.015% (5 milliseconds per GYPTO pulse), then a maximum of 8 GYPTO pulses could be issued to the gyros between GPC iterations (updates). However, the GYPTO and GPC clocks are independently derived and, in general, their frequencies are asynchronous and not exact multiples of one another. Therefore, it is possible that it could take a greater period of time for the HAINS to issue a maximum number of GYPTO pulses (8 pulses) than the time between GPC updates. Consequently, the GYPTO scale factor was chosen so that its maximum torquing rate is never less than 90 deg/hour. The GYPTO scale factor falls in the range from 0.45 to 0.685 arc-second per 5 millisecond pulse, this corresponds to a maximum GYPTO rate of 100 deg/hr +37% / -10%. After calibration, the uncertainty of the GYPTO scale factor is required to be 200 parts per million, one sigma. The maximum possible GYPTO torquing rate is 137 deg/hr. For higher rates, the slew circuitry and slew torquers are used.

6.2.1.4 Redundant Gyro Axis Rate Channel Description

The HAINS has two dual axis GYROFLEX gyros, thereby providing a total of four gyro axes. Sensed rates from three of these axes are used to stabilize the stable member about its three mutually orthogonal axes (X, Y, Z). The output from the fourth gyro axis (R-axis) is not required for basic HAINS functioning and is used by the GPC for BITE and redundancy management.

Figure 6-15 shows the orientation of the redundant axis with respect to the X- and Y-gyro axes. The redundant gyro sense axis lies in the plane of the X and Y gyro sense axes and is displaced 26.5 +/- 1.5 degrees from the X gyro sense axis. Because of the location of the redundant axis, the sensed rate is:

$$w_R = w_X \cos(26.5^\circ) + w_Y \sin(26.5^\circ)$$

where w_R is the sensed rate, w_X is the rate about the X-axis, and w_Y is the rate about the Y-axis.

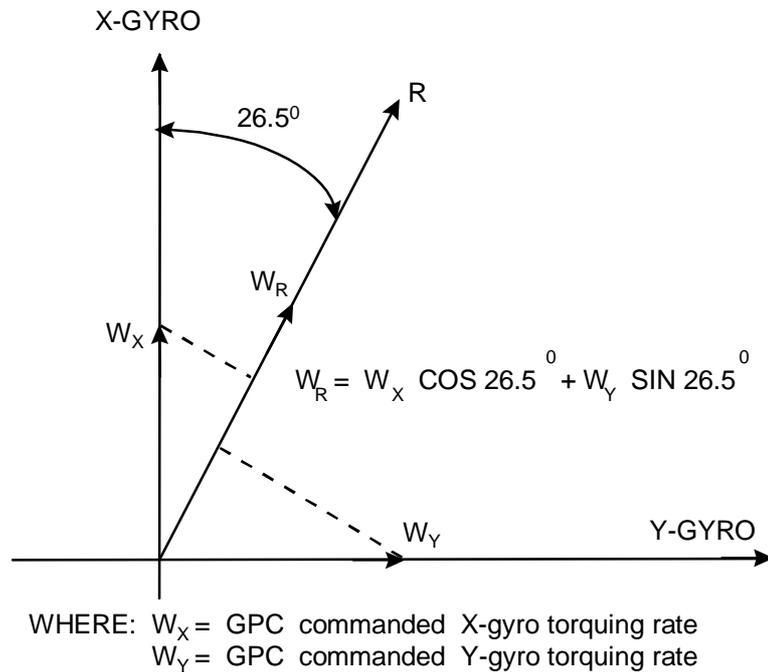


Figure 6 - 15, Redundant gyro axis orientation.

A current (dc) proportional to w_R is obtained by putting the redundant gyro axis in a rate capture loop also shown in Figure 6-16. The current is passed through a resistor in a buffer amplifier to generate a voltage proportional to w_R . This dc voltage is then converted into a digital data word every 10 msec by the A/D conversion circuit and discretely sampled by the GPC's.

A 19.2 kHz AC voltage, proportional to the redundant rate, is measured at the gyro pickoff output. This voltage is then demodulated and amplified on the GWS card. The DC output is digitized by the A/D card processor and filtered by the OFP and sent out, via MUX, as Output Word No. 2. The format for Output Word No. 2 is described in Table 6-XXVII (shown in section 6.4.1.4). The DC output is also used to close the redundant gyro capture loop.

The value of the redundant rate is compared with the computed (expected) rate of the stable member about the redundant axis and the difference between them represents an error drift rate. The computed rate is obtained using the X and Y GYPTO command rate information which is available in the GPC. The error drift rate, therefore, represents a deviation from the expected drift rate and is caused mainly by excessive gyro drift rate uncertainty. If the error drift rate should exceed a predetermined level, it would indicate that a problem exists in the HAINS. Consequently, the redundant gyro rate signal is a computer-aided BITE signal.

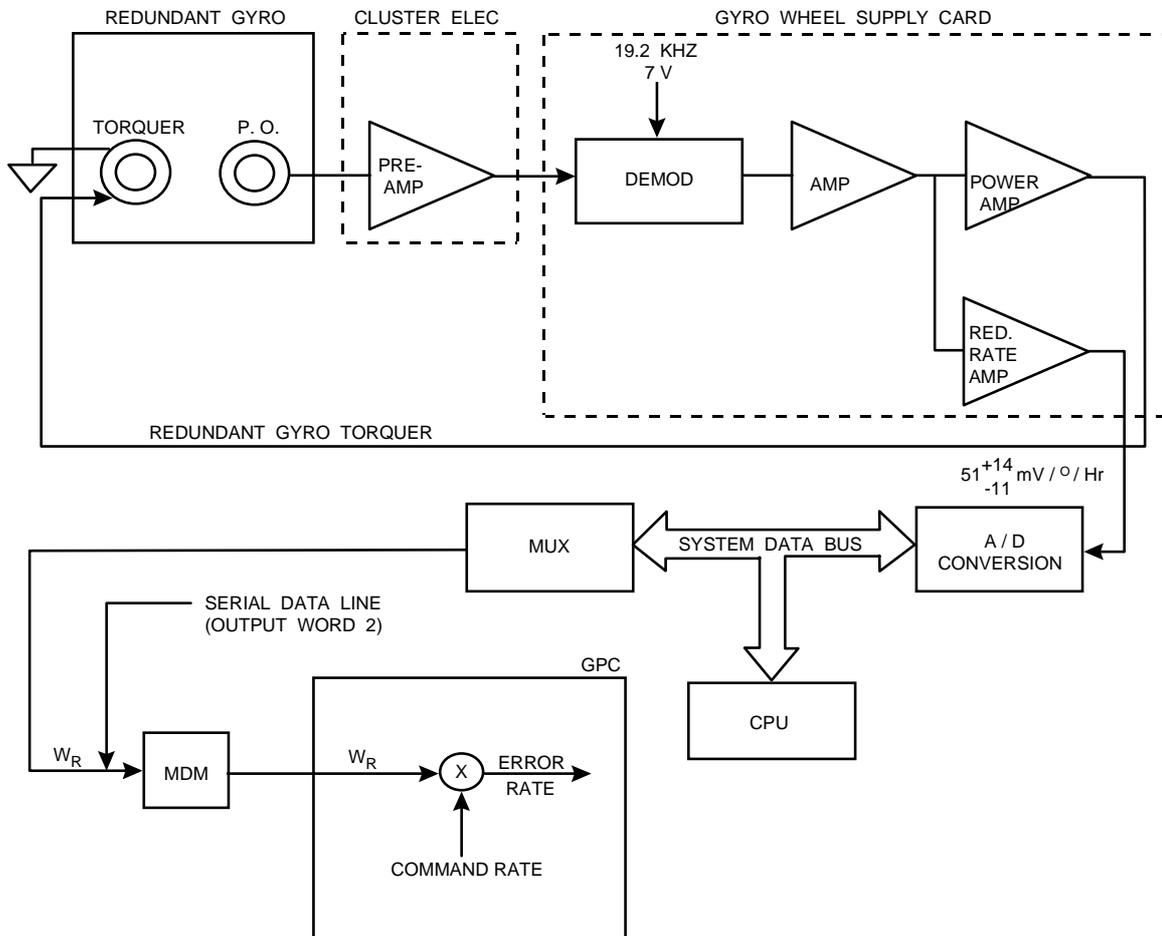


Figure 6 - 16. - Redundant Gyro Sensed Rate Channel

6.2.1.4.1 Performance/Design

The Redundant Gyro loop, along with its transfer function, is shown in Figure 6-17.

There are four parts to the system: capture loop, redundant gyro monitor electronics, HAINS internal software, and GPC software in the Orbiter.

The capture loop is driven by the relative displacement between the case and the rotor. The gyro pickoff converts the displacement into an electrical signal which is amplified, filtered, and used to drive the torquers which maintain the

redundant axis near its null position. Both the GYPTO and slew torquers are used for feedback torquing.

The redundant gyro monitor (RGM) electronics amplify the signal sent to the torquers, sample the result, and convert it to digital.

The software is in two parts, the HAINS software and the GPC software, both are displayed in Figure 6-17. In the HAINS internal software, the redundant gyro signal is compensated for redundant gyro channel error parameters calibrated at the factory. This includes mass unbalance, bias, delta bias, and scale factor. The corrected signal is put through a first order filter:

$$X_{n+1} = (.995) X_n + .005 U_n \quad X_0 = 0$$

where U_n is the signal, and X_n and X_{n+1} are successive values of the filtered signal. The filter time constant is approximately 2 seconds. The result is extrapolated to be consistent with the data request from the GPC. The RGM output word is quantized with an LSB of 0.008625 deg/hr. These instructions are implemented at a 100 Hz (10 msec) rate.

When the data reaches the GPC, the three least significant bits are dropped, because the GPC software was designed to work with three fewer bits. The utilized LSB is nominally 0.138 deg/hr, which is about one fifth of the 0.7 deg/hr. BITE threshold. The data is compensated using the latest RGM calibration data. In the Orbiter's GPC, the corrected signal is put through a first order filter with the same constants as the HAINS filter. This filter, and most of the GPC redundant gyro computations, are executed every 160 ms in flight. The mass unbalance computation is an exception which is run at 960 ms in flight. The filter has a time constant of about thirty seconds.

Most of the channel elements are linear but there are several exceptions. Three of them are quantizations and two of them are delta bias computations. Of these, the 0.138 deg/hr quantization is the most important.

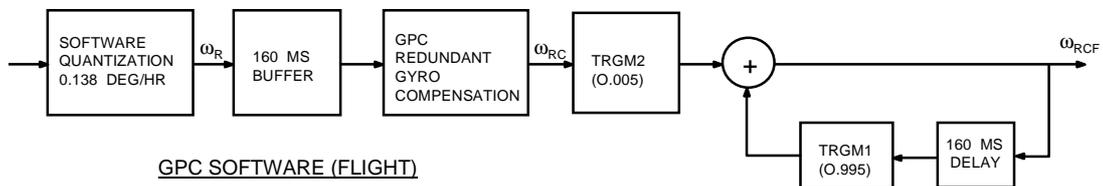
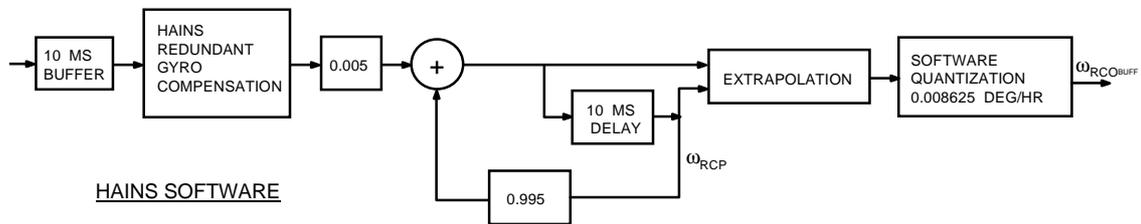
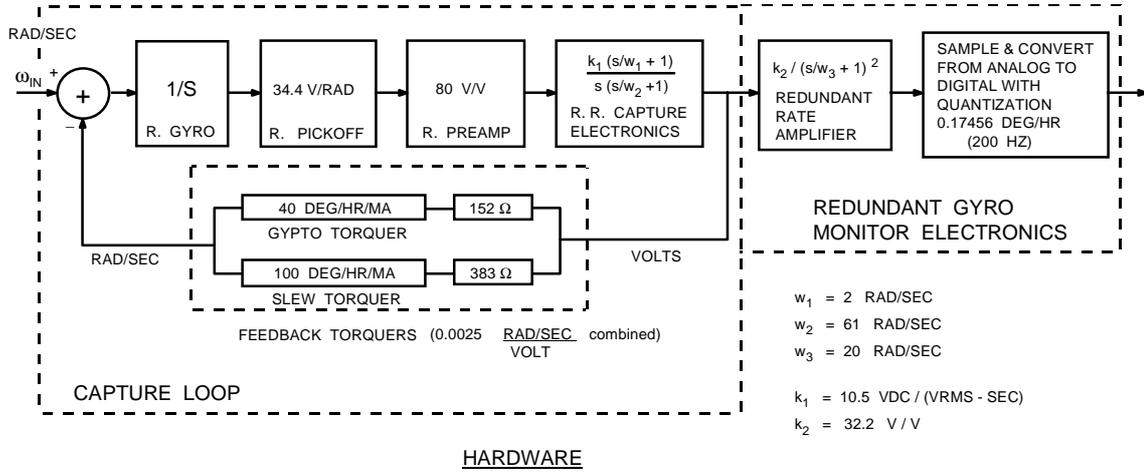


Figure 6 - 17. - Redundant Gyro Loop

Table 6-VI below lists the redundant gyro performance characteristics.

| Parameters | Range | 1 σ values |
|----------------------|----------------------------------|-------------------|
| SCALE FACTORS | | |
| X (KXR) | 0.505 to 0.821 DEG/HR/DEG/HR | 4,000 PPM |
| Y (KTR) | 0.252 to 0.409 DEG/HR/DEG/HR | 4,000 PPM |
| Z (KZR) | -0.175 to 0.175 DEG/HR/DEG/HR | 4,000 PPM |
| MASS UNBALANCE DRIFT | | |
| DIR | +/- 1.64 DEG/HR/G | 0.05 DEG/HR/G |
| DOR | +/- 0.82 DEG/HR/G | 0.05 DEG/HR/G |
| DSR | +/- 2.00 DEG/HR/G | 0.05 DEG/HR/G |
| BIASES | | |
| DFR | +/- 5.0 DEG/HR | 0.05 DEG/HR |
| DDFR | +/- 0.233 DEG/HR | 0.05 DEG/HR |

TABLE 6-VI, REDUNDANT GYRO PERFORMANCE CHARACTERISTICS

6.2.1.5 Attitude Software Interface

In the inertial mode of operation (further defined later in this systems brief), the IMU subsystem operating procedure (SOP) is responsible for the GPC commands to the IMU and it also interprets the data received from the IMU.

The SOP is divided into several different modules but this section of the report will only address those modules concerned with the attitude channel of the IMU.

6.2.1.5.1 Resolver processing (module name: GMD_RES_PROC)

Before being processed by this module, the IMU resolver data is processed by the HAINS OFP. The 1X and 8X resolver hardware values are sent to the A/D card where the combined resolver value is calculated. Like the GPC software described below, the A/D card firmware performs a validity test of the combined hardware. If the absolute difference between the 1X hardware value and the combined resolver value exceeds 22.5°, the software must correct for the singularity by adding or subtracting 45°. Two reasonableness checks are performed next.

1. If the absolute difference between the 1X hardware value and the combined resolver value is greater than 0.7 degrees, then the following occurs:

- set appropriate bit (2-12) in the A/D BITE Summary Word,

- set the OFP 'ADJUST' term value to 5 (initially = 0).
2. If both (or neither) the sine and cosine of each resolver angle is ≥ 0.707 , then the following is performed:
- set appropriate bit (2-7) in the A/D BITE Summary Word,
 - set the OFP 'adjust' term value to 5 (initially = 0).

The AD_READ module in the OFP then reads the combined hardware value and sends it to the OFP's RES_PROC module. In this module, the resolver data is manipulated to get ready to send to the GPCs. The combined hardware value is set equal to the new 1X resolver value and the 8X resolver value is derived from the combined hardware. The 'ADJUST' term from the A/D card firmware is now added to the 1X resolver value. It should be noted that the value for the 'ADJUST' term was arbitrarily set to five to make sure the GPC Resolver Limit BITE threshold of 1 deg is tripped. The following equations represent how the resolver data is manipulated in the HAINS OFP software:

$$\begin{aligned}
 H/W_{\text{combined}} &= 1X_{\text{LO}} + 8X \\
 \text{if } 1X - H/W_{\text{combined}} &\geq 22.5 \text{ deg (add or subtract 45 deg)} \\
 \text{if } 1X - H/W_{\text{combined}} &> 0.7 \text{ deg (ADJUST = 5)} \\
 \text{if both sin and cos } H/W_{\text{combined}} &>\text{or}< 0.707 \text{ (ADJUST =5)} \\
 1X_{\text{new}} &= H/W_{\text{combined}} \\
 8X_{\text{GPC}} &= \text{extrapolated from } H/W_{\text{combined}} \\
 1X_{\text{GPC}} &= 1X_{\text{new}} + \text{ADJUST}
 \end{aligned}$$

The $8X_{\text{GPC}}$ and $1X_{\text{GPC}}$ are then transferred to the GPC RES_PROC module. The first step of this module is to convert the resolver readings to compensated mechanical angles. The inner roll gimbal angle is first determined. If the inner roll angle is less than 180° , as determined from the inner roll $8X_{\text{GPC}}$ data word, then:

$$\text{ANGRL} = \frac{\theta_{\text{RL}}}{8} - \text{IRO}$$

where θ_{RL} is the angular value of the inner roll $8X_{\text{GPC}}$ word and IRO is the inner roll resolver offset (bias). The division by 8 is necessary because the $8X_{\text{GPC}}$ word is constructed the same as the 1X word (explained later in the input/output (I/O) section of this systems brief).

The software next computes the combined resolver angles (H/W) for roll, pitch, and azimuth. The logic takes the three most significant bits (MSB's) from the $1X_{\text{GPC}}$ words ($1X_{\text{LO}}$) (for roll, pitch, and azimuth) to locate the octant, combines them with the corresponding 8X words ($8X_{\text{GPC}}$) to form the total angle, and then subtracts the I-Loaded resolver bias (OFFSET).

$$\text{GPC computed} = H/W = 1X_{\text{LO}} + 8X_{\text{GPC}} - \text{OFFSET}$$

The GPC computed combined angle is then subtracted from the corresponding $1X_{\text{GPC}}$ angle reading. If any absolute difference exceeds 22.5° , the software must correct for the singularity that

can occur at multiples of 45°. Small errors in the 1X and 8X resolvers can cause the 8X to overflow (past 45°) before the 45° bit of the 1X word is set. This will cause the combined angle to be in error by as much as 45°. For this reason, the validity test is performed and 45° is added or subtracted to make the combined word agree more closely with the 1X word.

Validity test:

1X - H/W = ERROR
if |ERROR| > 22.5 deg then add or subtract 45 deg

The resolver outputs are required to be modulo 360°; therefore, if the combined angle is negative (more bias than measured angle), 360° is added to the combined angle. If the combined angle is greater than 360°, the software subtracts 360°.

Next the resolver outputs are limit checked and if the absolute difference between the 1X_{GPC} and the GPC computed H/W value is greater than 1 degree, then the Resolver Limit Fail BITE is set (discussed later in the BITE section). To correct a resolver bias, a G-MEM can be performed to uplink a new value to the I-Loaded OFFSET term. This would remove the Resolver Limit Fail BITE, but the PLATFORM Fail BITE (determined by the HAINS OFP) would still be annunciated.

The resolver processing module also checks for transmission failures (COMM faults (CF)). If a CF is detected, the software propagates the gimbal angles under the assumption that the rate of change of the gimbal angles is the same as detected during the two cycles preceding the CF. The module is executed at a 1.04 Hz rate during OPS 1 and 3, and at a 6.25 Hz rate in OPS 2 and 8. When the star tracker software is active in OPS 2 and 3, this module is executed on demand from that software.

6.2.1.5.2 Gyro compensation (GMK_GYO_COMP)

The purpose of the gyro compensation module is to compensate the GYPTO rates for gyro errors and misalignments. Specifically, the module computes desired torquing rates in cluster coordinates while compensating for gyro drift error and gyro mass unbalance error. The torquing rates are also adjusted for gyro misalignments and gyro torquer scale factors.

During platform repositioning operations (calibrations, alignments), the torquing rates are equal to the required compensation plus the computed torque rate required to complete the repositioning sequence. During most inertial operations, the torquing rates are equal to the compensation required to offset measured gyro errors. During noninertial operations, this module is executed at a 3200 msec rate. During inertial operations, it is executed every 960 msec.

6.2.1.5.3 Gyro torque processing (GMF_GYO_TORQ)

The GYO_TORQ module converts the computer-derived torquing rates into the required number of pulses of each gyro axis of each IMU. The torquing rates are computed in the above GYO_COMP module and passed to this module.

The gyros are pulse torqued to maintain their orientation or to reposition the platform. This module actually computes the number of pulses to be sent to each gyro axis to compensate for gyro errors and/or reposition the platform for alignment or calibration operations. Any pulses that were sent to the IMU but not executed, as determined by a transmission failure, are added to the next computed number of pulses. During OPS transitions, this module saves the pulses that were not sent to the IMU during the transition period and adds those pulses to the first cycle after the transition.

The GYO_TORQ module is executed at a 25 Hz rate during ground and preflight operations, and at a 160 msec rate during all inertial operations.

6.2.1.5.4 Large angle torquing (GMM_LAT_FUNC)

The LAT_FUNC module is used to rotate the cluster from its present orientation to any desired orientation with respect to a given reference. The module computes the present cluster to desired cluster transformation matrix from which it then determines the direction and duration of the required slew commands. When the platform reaches its desired position, the software commands a bi-directional slew to remove slew direction sensitivity effects. After the bi-directional slew, the software commands digital torquing rates to reach the final desired orientation of the platform. Once the orientation is reached, the commands to torque the platform are removed and a torquing complete flag is set.

6.2.1.5.5 Redundant gyro monitor comp (GMY_RGM_COMP)

The redundant gyro monitor (RGM) measures the inertial drift rate of the IMU about the redundant axis. The RGM_COMP module compensates the redundant gyro rate for gyro errors in support of the BITE function of the RGM.

This module compensates for CF's by assuming that the rate is changing by a constant amount equal to the change sensed on the last two reads prior to the CF (linear extrapolation). Similar to the attitude gyros software, software also compensates the raw RGM rate for mass unbalance drift and redundant gyro-axis drift. The compensated RGM rate is then filtered and smoothed by a low pass filter.

The RGM_COMP module is executed at a 6.25 Hz rate. The module is not executed during alignments or while catch up torquing (after an OPS transition) is in progress.

6.2.2 Velocity Channels

Acceleration data is provided by three single-axis accelerometers (X, Y, and Z) located on the innermost gimbal which together form a triad of three mutually orthogonal accelerometer axes. The accelerometers, a high accuracy version of the Kearfott Corporation baseline MOD VII, are single-axis, gas damped torquer restrained pendulous devices capable of sensing a wide range of accelerations. The sensitive element is a pendulum containing torquer coils and capacitance plates.

An alternating voltage (8V, 19.2 kHz) excites the capacitance bridge which produces a pickoff voltage due to the relative displacement of the pendulum to case resulting from sensed accelerations. The pickoff signal is amplified by an integrally mounted amplifier. The preamplifier output voltage becomes the low output impedance servo error signal which is fed to the Digital Accelerometer Loop (DAL) electronics on the platform electronics card. The resulting feedback signal is in the form of precise current pulses, where the restoring coil current pulse rate is a measure of acceleration and each pulse is equivalent to an increment of velocity.

6.2.2.1 Accelerometer Description

The X, Y, and Z accelerometers sense Orbiter acceleration components along their respective axes. Each accelerometer has its own Digital Accelerometer Loop (DAL). The DAL is a terminal pulse rebalance loop which produces accurate current pulses to restore the pendulum to null and provides an incremental velocity pulse train to the HAINS computer. Nominal steady state acceleration capability is 20 g's with a pulse resolution of 0.0075 ft/sec.

Under conditions of little or no vehicle acceleration, the loop will limit cycle due to pulse quantization. Alternative positive and negative pulses will displace the pendulum around its null position.

The two pulse line outputs of each (X, Y, or Z) DAL channel are connected to Gate Array Pulse Accumulators located on the I/O circuit card. As the positive delta V or negative delta V pulses occur, they are accumulated in the Pulse Accumulators. The contents of the pulse accumulators are snapshot (read) by the HAINS computer/OFP internal software via memory mapped instructions.

After the CPU/OFP internal software reads the pulse count from the accumulator, it provides scale factor temperature compensation of each velocity channel and scales the pulse count to either the high or low gain pulse rate as requested. When the Message-In command (set high) is received from the GPC, the HAINS causes the processed velocity output word count to be frozen. Then, the X, Y, and Z velocity words are sequentially transmitted via output words 10, 11, and 12 to the GPC.

The accelerometer loop block diagram is shown in Figure 6-18.

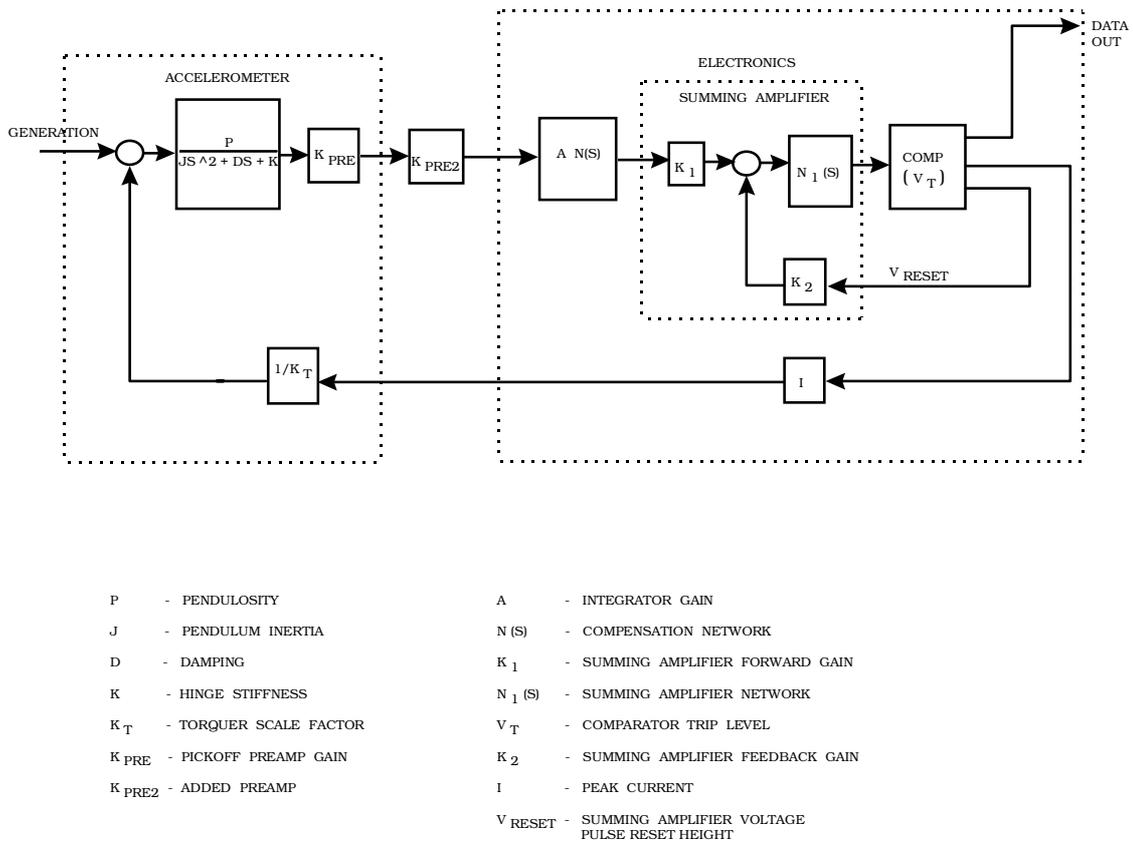


Figure 6 - 18. - ACCELEROMETER LOOP BLOCK DIAGRAM

6.2.2.1.1 Performance Design

The acceleration parameter performance given in Table 6-VII summarizes the all-attitude performance capability of the velocity channels. It can be seen that for the Ascent/Abort Once Around (AOA) Mission, the acceleration bias scale factor and asymmetry accuracies are better than those for the Extended Mission. This reflects the more critical requirements of the Ascent/AOA Mission. To achieve this performance, the acceleration scale factor and bias are calibrated immediately

prior to a flight. The Ascent/AOA All-Attitude performance can be achieved for continuous operation covering a maximum of 15 hours of preflight time plus 2 hours of flight time. The Extended Mission performance is valid for a period of 170 flight hours and includes 5 HAINS turn-ons.

| Parameters (over +/- 6 range) | Range | 1σ values |
|--|--|--|
| X, Y, Z GYRO BIAS (RESTRAINT) KOX, KOY, KOZ | ±1000 MICRO-G | 30 MICRO-G, BOOST/AOA MISSION (17HR); 40 MICRO-G EXTENDED MISSION |
| X, Y, Z ACCELERATION SCALE FACTOR (HIGH GAIN) KIX, KIY, KIZ | ±107,000 PARTS PER MILLION (PPM) | 25 PPM, BOOST/AOA MISSION (17HR); 60 PPM, EXTENDED MISSION |
| ACCELERATION ASYMMETRY KSY, KSZ | ±2100 PPM | 25 PPM, BOOST/AOA MISSION (17HR); 40 PPM, EXTENDED MISSION |
| X,Y,Z ACCELERATION NON-LINEARITY QUADRATIC COEFFICIENT, K2 CUBIC COEFFICIENT, K3 | - - | 5 MICRO-G/G ² 1 MICRO-G/G ³ |
| VIBRATION INDUCED ACCELERATION ERRORS (DURING VIBRATION) | - | 680 MICRO -G (MAX) |

TABLE 6-VII, ACCELEROMETER PERFORMANCE CHARACTERISTICS

To achieve the required performance, the platform unit containing the accelerometers is maintained at a constant temperature via the TCA (Temperature Control Amplifier) Loops. In order to compensate for variations due to temperature changes of the DAL electronics, located on the PE No. 1 circuit card, DAL scale factor temperature compensation is provided via the OFP internal software. The DAL temperature sensor, mounted on the PE No. 1 circuit card, is connected to an A/D card DC channel input. The CPU/OFP reads the PE No. 1 cards temperature as recorded by the A/D card. The CPU/OFP then modifies the accelerometer scale factors for each channel based upon the temperature and the PAL calibration parameters stored in EEPROM memory.

The DAL electronics is powered by +5, +15, and -15 Vdc dedicated supplies. These voltages are generated on the PE No. 1 circuit card and derived from the DC/DC card's +/-19 Vdc supplies. Each of the DAL voltages is connected to an A/D card DC channel input. The CPU/OFP internal software monitors the DAL voltages, as recorded by the A/D card and compares the measured voltage levels to BITE threshold levels.

6.2.2.1.2 Acceleration Measurement Performance

The acceleration channel has only one scale factor setting, nominally 4300 pulses/sec/g.

However, in order to meet transparency requirements, the scale factor is changed by the internal Operational Flight Program (OFP). The OFP multiplies the internal scale factor by:

0.2171001 (934 pulses/sec/g) for low gain operation

and

2.171001 (9,335 pulses/sec/g) for high gain operation

The low gain scale factor setting is used in all flight modes. The high gain scale factor setting is primarily used during HAINS calibrations and alignments on the ground.

6.2.2.1.3 Definition of Acceleration Terms

The following is a basic acceleration error model (Table 6-VII).

$$A_m = KO_i + [(1 + KS_i) + (KI_i)] a_i + (K_2)a_i^2 + (K_3)a_i^3$$

where,

A_m = Acceleration measured by the HAINS, g's
 KO_i = Bias, g's
 KI_i = Average scale factor (ideally equal to 1 g/g)
 K_2 = Quadratic nonlinearity coefficient, g/g²
 K_3 = Cubic nonlinearity coefficient, g/g³
 KS_i = Asymmetry, g/g (the sign in the above equations the same as the sign of the a_i sense)

Where $i = X, Y$ or Z ,

a_i = Input acceleration, g's

i = X, Y or Z

and,

$$KI_i = [SF(+) + SF(-)]/2$$

$SF(+)$ = Acceleration scale factor for positive orbiter accelerations, defined for a linear fit between 0g and +1g inputs.

$SF(-)$ = Acceleration scale factor for negative orbiter accelerations, defined for a linear fit between 0g and -1g inputs.

$$KS_i = [SF(+) - KI_i] / KI_i$$

6.2.2.2 Velocity Software Interfaces

In the inertial mode of operation (further defined later in this systems brief), the IMU subsystem operating procedure (SOP) is

responsible for the GPC commands to the IMU and it also interprets the data received from the IMU.

The SOP is divided into several different modules but this section of the report will only address those modules concerned with the velocity channel of the IMU.

6.2.2.2.1 Accelerometer processing (GMC_ACP_ACUM)

This module reads the contents of three velocity registers in the IMU and computes the change in velocity since the last read cycle. To monitor the hardware registers, this software has a register of its own to which it adds or subtracts velocity counts. If communication with the IMU is lost, the software assumes the DV computed on the last pass is constant and increments or decrements the software register appropriately. Since the hardware registers are really continuous up/down counters, the software must detect and compensate for register overflow. The software accomplishes this by assuming that a DV (computed) that has a pulse count greater than the value of the MSB in the register must indicate that the register has overflowed. If the DV is greater than 32,768 pulses (2^{15} , 16-bit data word), the software subtracts 65,536 to arrive at the proper DV. If the DV is less than -32,768, the software adds 65,536 to the DV. With this mechanization, the software can accommodate one register overflow in the presence of a CF (i.e., when communication is reestablished, if the hardware register has overflowed once during the CF, the software can track it). If, however, the CF is lengthy or the acceleration on the vehicle is great (ascent), the software is ignorant of how many overflows the hardware has experienced. For this reason, we deselect COMM-faulted IMU's, since the initial DV error between the actual hardware and the software registers could be quite large.

This module also performs an overlimit test to detect velocity channel failures. If the DV measured exceeds 4.8g, the software sets a flag to be used by the BITE summary module. The module is executed at 6.25 Hz for major modes 103 (pre-MECO), 601, 602, 603, 304, and 305. It is executed at 1.04 Hz for all other major modes.

6.2.2.2.2 Accelerometer Pulse Compensation (GMH_ACP_COMP)

The pulse compensation module filters and smooths velocity inputs from the accelerometer processing module for calibration applications, but more importantly for flight applications, this module compensates for measured accelerometer errors such as bias, scale factor, and misalignment. This module is executed at the same rate as the accelerometer processing module.

6.2.2.2.3 Accelerometer pulse transformation (GML_ACP_TRSF)

During calibrations, this module transforms the sensed velocity from IMU coordinates to north, west, up coordinates and subtracts out the local gravity from the vertical component. During flight modes, this module transforms the sensed velocity from IMU coordinates to Mean of 1950 (M50) coordinates. The module also performs a velocity underlimit test to detect the three-axis fail to null type of failure.

In the flight mode, the software uses the reference to stable member matrices (REFSMMATS) to accomplish the transformation to the M50 coordinate frame.

$$\Delta V^{M50} = [T_{M50}^{CL_i}]^T \Delta V^{CL_i}$$

where i = IMU 1, 2, or 3

The underlimit test is performed by first forming a normalized acceleration for each IMU.

$$A_N = \sqrt{\left(\frac{\Delta V}{\Delta t}\right)^2 + \left(\frac{\Delta V}{\Delta t}\right)^2 + \left(\frac{\Delta V}{\Delta t}\right)^2}$$

If $A_N > 1.93 \text{ ft/sec}^2$ for one or more IMU's and $A_N < 1.61 \text{ ft/sec}^2$ for one and only one IMU, then the software sets a velocity underlimit fail flag for that IMU that has $A_N < 1.61 \text{ ft/sec}^2$. The flag is used by the BITE_SUM module which is discussed later in this report.

6.2.3 Power and Grounding

The sole source of primary input power to the HAINS is 28 Vdc power which is provided at the HAINS J1 connector.

This section summarizes the input power and grounding requirements and indicates the DC and AC voltage excitations that are generated from the primary input power and the scheme used to isolate the three types of grounds that are used.

6.2.3.1 Input power.

The only power to the HAINS is 28 Vdc power from the Orbiter. Each IMU is redundantly powered from two main dc buses via an associated remote power controller (RPC) in accordance with the requirement in Table 6-VIII

| | |
|-----------------|--|
| Nominal voltage | +28 Vdc |
| Maximum voltage | +32 Vdc |
| Minimum voltage | +24 Vdc |
| Peak current | 10.0 amps at 28 Vdc with start-up transients that do not exceed 12.5 amperes for 100 |

TABLE 6-VIII, POWER REQUIREMENTS

6.2.3.2 Generated dc Voltage

The HAINS has its own self-contained power supplies that generate, using the Orbiter's 28 Vdc voltage input, the DC and AC voltages required to operate the HAINS.

When +28 Vdc is supplied to the HAINS with the Operate Mode discrete low (off), it is filtered and used to excite the platform and electronics temperature control circuits and associated BITE discrettes. The HAINS is then in its Warm-up/Standby Mode. During this mode, the Isolated +5 Vdc is generated in the IPC card, and is used to energize the HAINS discrettes.

The Operate Mode discrete is sent to the HAINS (with +28Vdc still applied). This discrete turns on the IPC +28Vdc switched power supply which provides input power to the DC/DC circuit card. Upon receipt of the +28 Vdc switched power from the IPC card, the DC/DC card issues a logic reset signal. This signal is held low for a nominal time of two seconds. This period of time is to allow the DC/DC card output voltage to stabilize. Removal of the logic reset signal (goes high) synchronizes all HAINS logic functions. Table 6-IX lists the generated voltage outputs of the IPC and DC/DC Cards.

| CIRCUIT CARD | VOLTAGE | TOLERANCE |
|--------------|---|--|
| IPC | + 28 Vdc SWITCHED + 5 Vdc ISOLATED 7 V, 400 Hz | N/A +/- 5% VOLTAGE +/- 5% FREQUENCY +/- 0.06% |
| DC/DC | + 5 Vdc +/- 15 Vdc - 15 Vdc + 19 Vdc - 19 Vdc | +/- 2% +/- 1% +/- 1% +/- 5% +/- 5% |
| GWS | 8 V, 19.2 KHz | VOLTAGE +/- 5% FREQUENCY +/- 0.02% |

TABLE 6-IX, HAINS GENERATED OUTPUT VOLTAGES

6.2.3.3 Power Consumption

Warm-up/Standby Mode

During the initial warm-up period, when the HAINS is in the Standby mode and the input voltage is 28 Vdc with the ambient temperature at plus 35 degrees F, peak power consumption does not exceed 260 watts. Once temperature stabilization has been achieved the average power consumption does not exceed 75 watts.

Operate Mode

In the operate mode (after stabilizing), the average power consumption does not exceed 200 watts and peak power transients do not exceed 280 watts. Table 6-X is a summary of typical input power requirements in the various modes of operation.

| HAINS MODE | +28 VDC ORBITER POWER, WATTS | |
|--------------|------------------------------|---------|
| | TYPICAL | MAXIMUM |
| STANDBY | 35 | 260 |
| OPERATE MODE | 150 | 280 |

TABLE 6-X, POWER CONSUMPTION

Power Interruption

Loss of input power does not cause damage to the HAINS. The HAINS powers down if the input voltage falls below the low voltage normal power transients.

6.2.3.4 Grounding Requirements

The HAINS uses the following four types of grounds:

- o Input power ground (+28 Vdc low)
- o 5V Isolated return (unipoint ground/signal ground)
- o 28V filtered low
- o Chassis ground

These grounds are isolated from one another in the HAINS. Separate Input Power Ground and chassis ground connections to the HAINS are made via the external J1 Power Connector. External 5V Isolated Return connections are made at operational connector J2 (digital discrete) and test connectors. Input power ground, 5V Isolated Return, and Chassis ground are tied together external to the HAINS in the Orbiter.

Input Power Ground - The Input power ground (+28 Vdc low) is used as the return to the Orbiter of the +28 Vdc power. It is used to convey Input power to the Input Power Conditioner (IPC) circuit card via an EMI filter.

5 Vdc Isolated Return (Unipoint Ground/Signal Ground) - This ground is the return for the +5 Vdc Isolated Power supply located

on the IPC circuit card. It is a HAINS requirement that all hardwired discretes issued by the HAINS must be referenced to the 5V Isolated Low (Unipoint Ground/Signal Ground). In order to accomplish this, Light Emitting Diodes (LED's) have been mounted on the PE No. 2 and IPC circuit cards. The internal (HAINS unit) side of each LED is referenced to 28 Vdc Filtered Low while the external (GPC) side of each LED is referenced to the +5 Vdc Isolated return. The output circuit utilizes the +5 Vdc Isolated power and the LED's optical coupling provides the required isolation.

28 Vdc Filtered Low - The 28 Vdc Filtered Low is the general system ground for most internal HAINS power supplies and signals. The 28 Vdc Filtered Low is connected to the Input Power Ground through a filter on the IPC card and an EMI filter. The 28 Vdc Filtered Low does not come out through any of the HAINS operational connectors, and it can only be accessed via the external test connectors.

The central point of convergence (distribution) of the 28 Vdc Filtered Low is located in the wire-wrap pinfield of the DC/DC circuit card. The distribution of the various signals/voltages, plus the circuit card design of the individual local circuits (e.g., circuit cards) minimizes undesirable voltages due to common ground coupling.

Chassis Ground - The chassis ground is basically used to:

- o Ground the various frames of the HAINS equipment (with one exception),
- o Ground the spare pins on all (6) external HAINS connectors.

The one exception referred to above is in the Inertial Platform. In this unit, inward from the isolated side of the platform isolators, the frame (including all platform gimbals) is connected to +28 V Filtered Low. The +5 Vdc Isolated power supply used to generate discrete outputs in the Warm-up/Standby and Operate Modes is referenced to Orbiter signal ground.

6.2.4 System Cooling and Temperature Control

6.2.4.1 Cooling Air

The HAINS is designed to meet its performance requirements over the specified environments with cooling air meeting the requirements of Table 6-XI.

| | |
|----------------------------------|------------------------------|
| Pressure drop range (in. of H2O) | 1.70 to 2.20 |
| Nominal air flow rate (lb/min.) | 0.8 to 14.7 PSIA |
| Minimum inlet air temperature | +35 F |
| Maximum inlet air temperature | +90 F |
| Maximum exhaust air temperature | +150 F |
| Surrounding air pressure | 8 to 18 PSIA |
| Air cleanliness level | Visibly clean per MA0110-301 |
| Emergency conditions | |
| Surroundings air pressure | 8 PSIA |
| Minimum air flow rate (lb/min.) | 0.48 at 80 deg F maximum |

TABLE 6-XI, COOLING AIR REQUIREMENTS

This cooling air is required for both ground and flight operation of the HAINS. The cooling air inlet and outlet fittings for the HAINS are standard 1.5 inch fittings with an external bead for the air hoses.

The HAINS is cooled by forced air as indicated in Figure 6-19. Air is drawn through the inlet port located on the upper right hand side of the aft end of the IMU. Protection from EMI interference is provided by a honeycomb EMI filter inside this port. Cooling air is drawn through this port and then through the card stack. Each circuit card has an air plenum (Figure 6-20) through which air flows to cool the card either by an aluminum fin heat exchanger or a heat transfer plane. The exit air from the card stack plenum is drawn out of the HAINS IMU through the outlet port located on the inertial platform is controlled by a bi-metallic spring valve (air modulator), which regulates the amount of inlet air drawn through the platform. The higher the temperature, the greater the amount of cooling air provided to the platform. The exit air from the platform is drawn out through a flexible air boot which connects the platform to the exit air plenum.

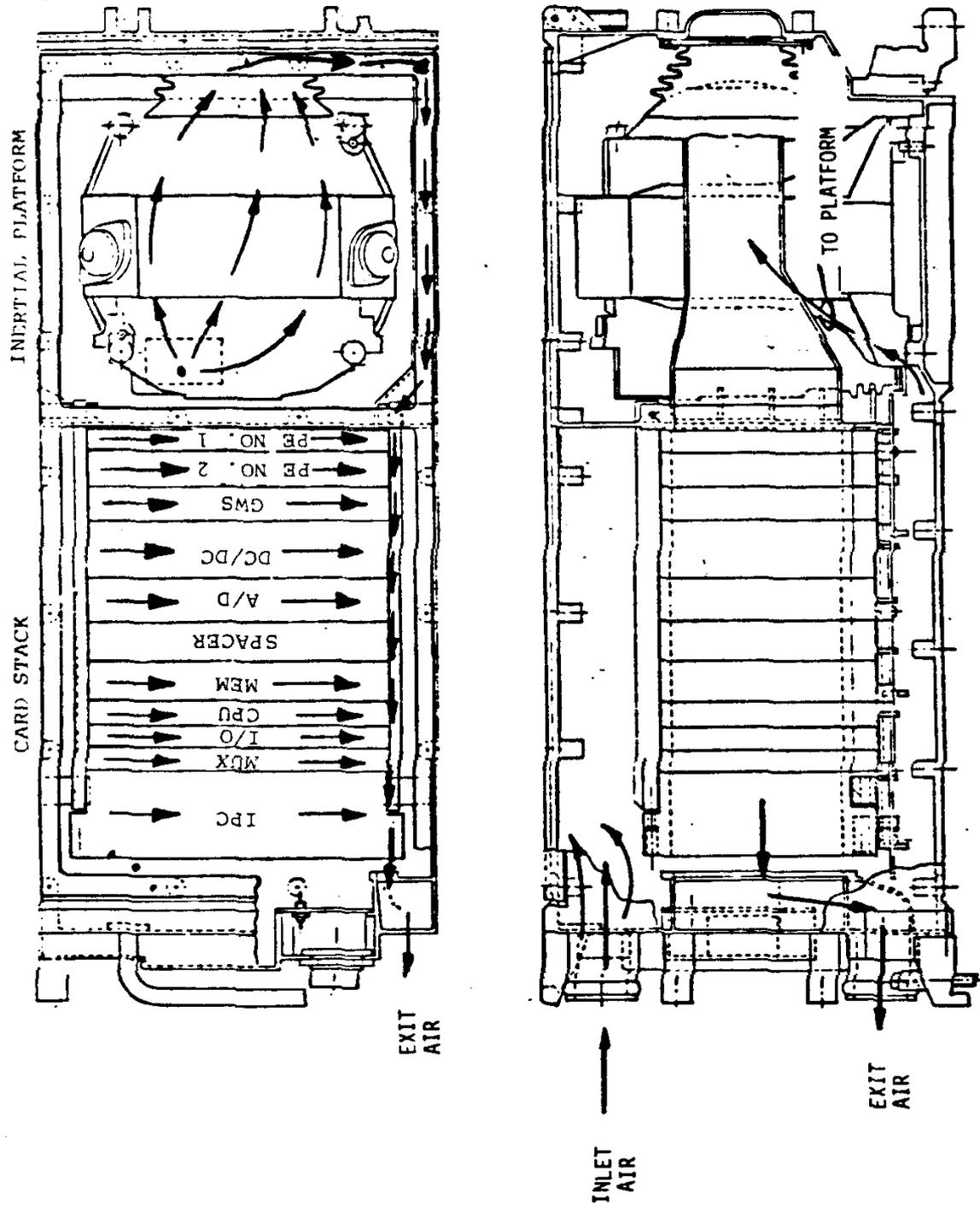


Figure 6-19, Airflow Diagram

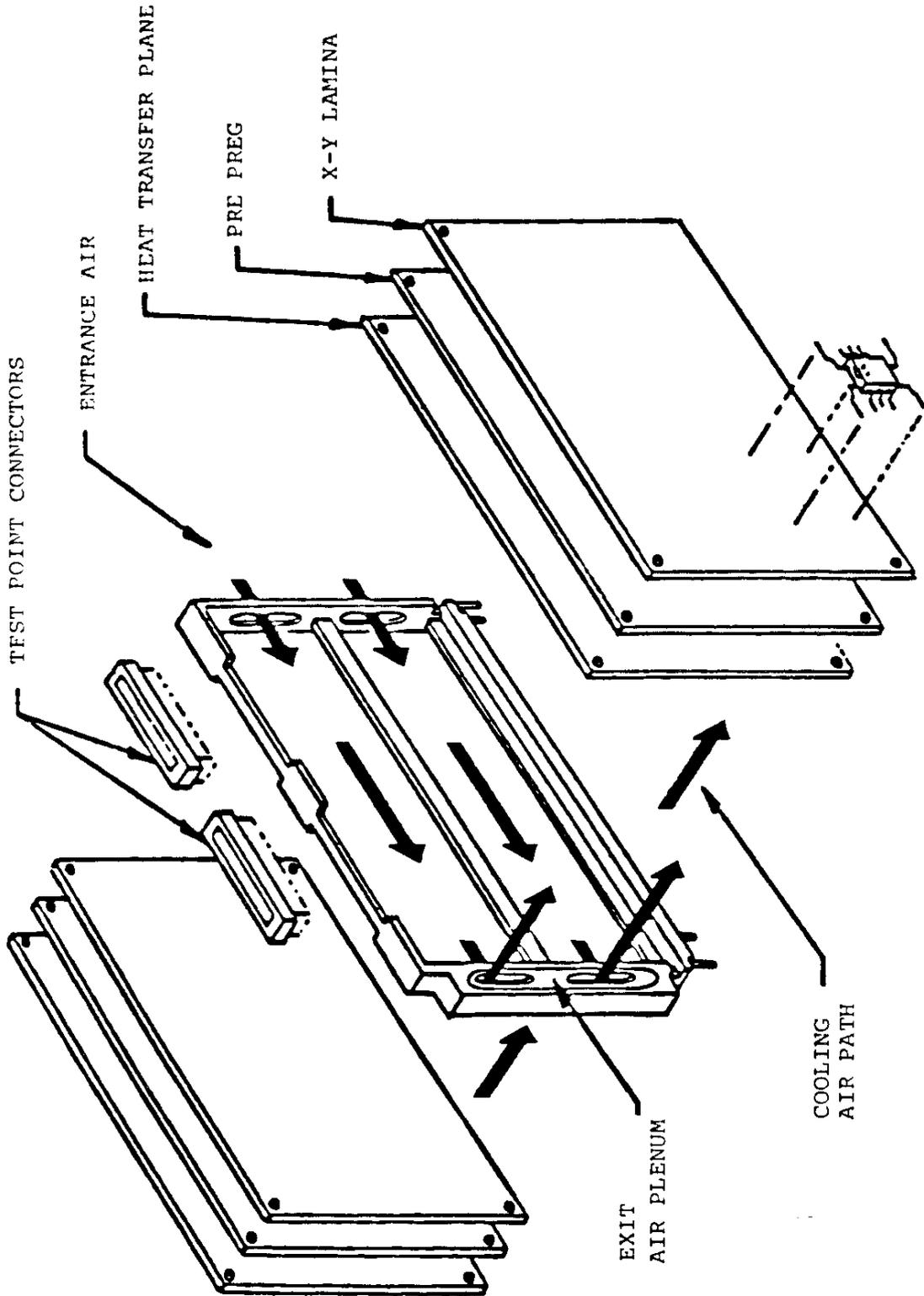


Figure 6-20, Card Cooling Air Distribution

Cooling air is required by the HAINS to ensure that the thermal control systems for the inertial components of the stable element can properly regulate the temperatures of these components. At ambient temperatures between +90 degrees F and +125 degrees F and with the cooling air at its high temperature extreme (+90 degrees F), some degradation in performance may occur, but there is no danger to the HAINS.

The HAINS warm-up time is primarily a function of the cooling temperature and flow rate.

If cooling air is lost, power to the HAINS must be removed within 10 minutes to prevent damage to it. The KT_70 IMU was able to tolerate up to 45 minutes without cooling before risking hardware damage. It is a concern that during critical mission phases (e.g. ascent) the crew may not reconfigure a failed fan before risking damage to the HAINS. This topic was discussed at the On-Orbit Flight Techniques Panel meeting #122 in 1991. The following table (Table 6-XII) from this technique meeting illustrated how the hottest cards are towards the front of the IMU.

| Resistor # | PE1 | PE2 | GWS | DC/DC | A/D | MEM | CPU | I/O | MUX | IPC |
|-----------------------|-----|-----|-----|-------|-----|-----|-----|-----|-----|-----|
| 193-195 (400-401)* | 125 | 128 | 131 | 134 | 134 | 135 | 135 | 135 | 134 | 134 |
| 196-197 (402) | 127 | 126 | 127 | 128 | 128 | 131 | 131 | 132 | 132 | 132 |
| 198-203 (403-407) | 116 | 119 | 121 | 123 | 124 | 128 | 128 | 130 | 132 | 134 |
| 204-209 (408-411) | 118 | 120 | 122 | 123 | 126 | 132 | 132 | 134 | 137 | 139 |
| 210-211 (412) | 135 | 125 | 127 | 129 | 130 | 136 | 136 | 138 | 139 | 139 |
| 212-214 (413-414) | 127 | 129 | 130 | 131 | 131 | 137 | 137 | 137 | 137 | 137 |
| 215-217 (415-416) | 127 | 129 | 129 | 133 | 132 | 134 | 134 | 136 | 136 | 136 |
| 218-220 (417-419) | 126 | 128 | 128 | 133 | 133 | 134 | 134 | 135 | 135 | 135 |

*for A/D and MEM cards

Table 6-XII Temperature Along Card Frame
(Boundary Conditions - from System Analysis)

Further testing showed that the maximum junction temperature of 125 degrees C was exceeded at component U10 between 9 and 10 minutes following the loss of cooling (90 degrees F, steady state with 0.014 lb/min). U10 is a critical micro circuit which control the timing associated with the MUX transmit/receive functions. In other words, U10 control the transmission and reception of commands to and from the GPC. The U10 failure will first result in a commfault which indicates that the flight computer operating system (FCOS) could not successfully update the HAINS. A few seconds after the commfault, one or more of the following BITES may be annunciated:

MUX Fail
Transmission Word 1 Fail
Transmission Word 2 Fail

Ultimately, the HAINS will be deselected by redundancy management (RM). And the IMU will be unrecoverable.

6.2.4.2 Environmental Conditions

The HAINS is capable of meeting the operational performance requirements specified herein during and after exposure to conditions in Table 6-XIII.

| | | | |
|-------------|---------|---|--|
| Temperature | Minimum | +15 deg F | Compliance with operational performance requirements between +35 deg F to +90 deg F |
| | Maximum | +125 deg F | |
| Pressure | Minimum | 12.36 psia 10.2 psia 8.0 psia | Continuous (30% O ₂) for 165 hours with +90 deg F maximum cooling air flowing at a minimum rate of 0.54 lbs/min. For 3 hours (rate of change of pressure will not exceed 9.0 psia/min) |
| | Maximum | 18 psia | |
| Humidity | Minimum | 8% Relative | |
| | Maximum | 100%Relative | |

TABLE 6-XIII, OPERATIONAL ENVIRONMENTAL CAPABILITY

6.2.4.3 Temperature Control Circuitry

The purpose of the temperature control circuitry is to maintain a constant temperature within the platform gyroscopes and accelerometers. There are three separate Temperature Control Amplifiers (TCA's) in the external platform electronics (PE2) which maintain the constant operational ambient temperature. The platform is separated into three independent thermal zones containing its complement of heaters and sensors. An overall view of the control circuitry is provided in Figure 6-21.

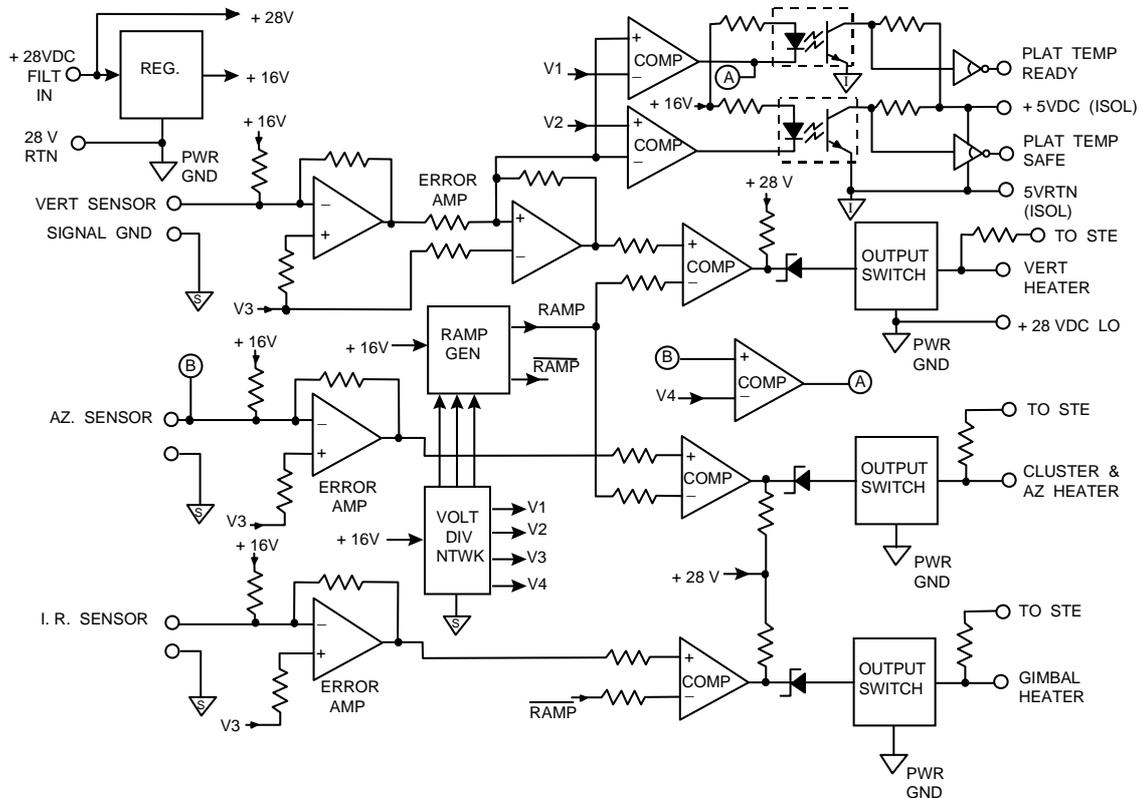


Figure 6 - 21. - Three-Channel TCA.

Gimbal Temperature Control Amplifier

The gimbal TCA is in the outermost zone and consists of control circuitry on the PE No. 2 card, a temperature sensor (thermistor) located on the inner roll gimbal, and 14 heaters distributed throughout the platform. It is the temperature sensor on the Inner Roll gimbal which provides the error signal for the control electronics. There are 5 heaters on the Inner Roll gimbal, 3 on the Pitch gimbal, 4 on the Outer Roll gimbal, and 2 on the sealed housing. There are also two thermostatic switches which protect the platform in case of over-temperature.

Vertical Temperature Control Amplifier

The vertical TCA is in the first of the two inner zones and consists of control circuitry on the PE No. 2 card and a temperature sensor on the Vertical gyro. In addition, the Vertical TCA contains the heaters for the upper cluster, the Vertical gyro, and also the X and Y accelerometers.

Azimuth Temperature Control Amplifier

The Azimuth TCA is in the second inner zone and consists of control circuitry on the PE No. 2 card, 2 heaters on the Azimuth gyro, a heater on the Z accelerometer, 2 heaters on the cluster, and a thermistor on the Azimuth gyro, which provides an error signal for the control circuitry.

Operation of the TCA's

The TCA pulse width modulates the current to the heaters. The lower the temperature, the higher the ratio of current ON/OFF time. At temperatures below the threshold, the control circuitry will send continuous current to the heaters until the operating temperature is reached. Two discrettes, Platform Temperature Ready and Platform Temperature Safe, are generated by the TCA circuitry. These discrettes are described in detail later.

All three thermal zones are protected from TCA malfunctions by safety thermostats. These thermostats are bimetal, snap-action thermal switches that open upon temperature rise. There are five such safety thermostats. See Table 6-XIV for thermostat platform locations.

| Thermostat | Location on Platform |
|-----------------------|--|
| S5, 150 ±5 deg F | on the platform housing in series with its heaters |
| S4, 163 +12, -8 deg F | on the pitch gimbal in series with the outer roll, pitch & inner roll heaters |
| S3, 174 ±8 deg F | on the lower platform support in series with the azimuth gyro, z accelerometer & lower support heaters |
| S1, 174 ±8 deg F | on the upper support in series with a vertical gyro, X & Y accelerometer upper support heater circuit |
| S2, 174 ±8 deg F | on the upper support in series with a vertical gyro, X & Y accelerometer upper support heater circuit |

TABLE 6-XIV, SAFETY THERMOSTATS PLATFORM LOCATION

6.2.5 Electronics and Clocks

6.2.5.1 Electrical Interface

The external operational interface between the HAINS and other aircraft avionics equipment is through three connectors (11, 12, 13) located on the front Panel of the HAINS housing. Three test connectors (14, 15, 16) are provided on the front panel of the HAINS, but are not used operationally.

6.2.5.2 Platform Electronics

The platform electronics, which is mounted on the azimuth gimbal along with the inertial components, contains four gyro preamplifiers, three gyro quadrature spring rate compensations circuits, and a GYPTO electronics hybrid. A block diagram of the

platform electronics, along with its associated components, is shown in Figure 6-22.

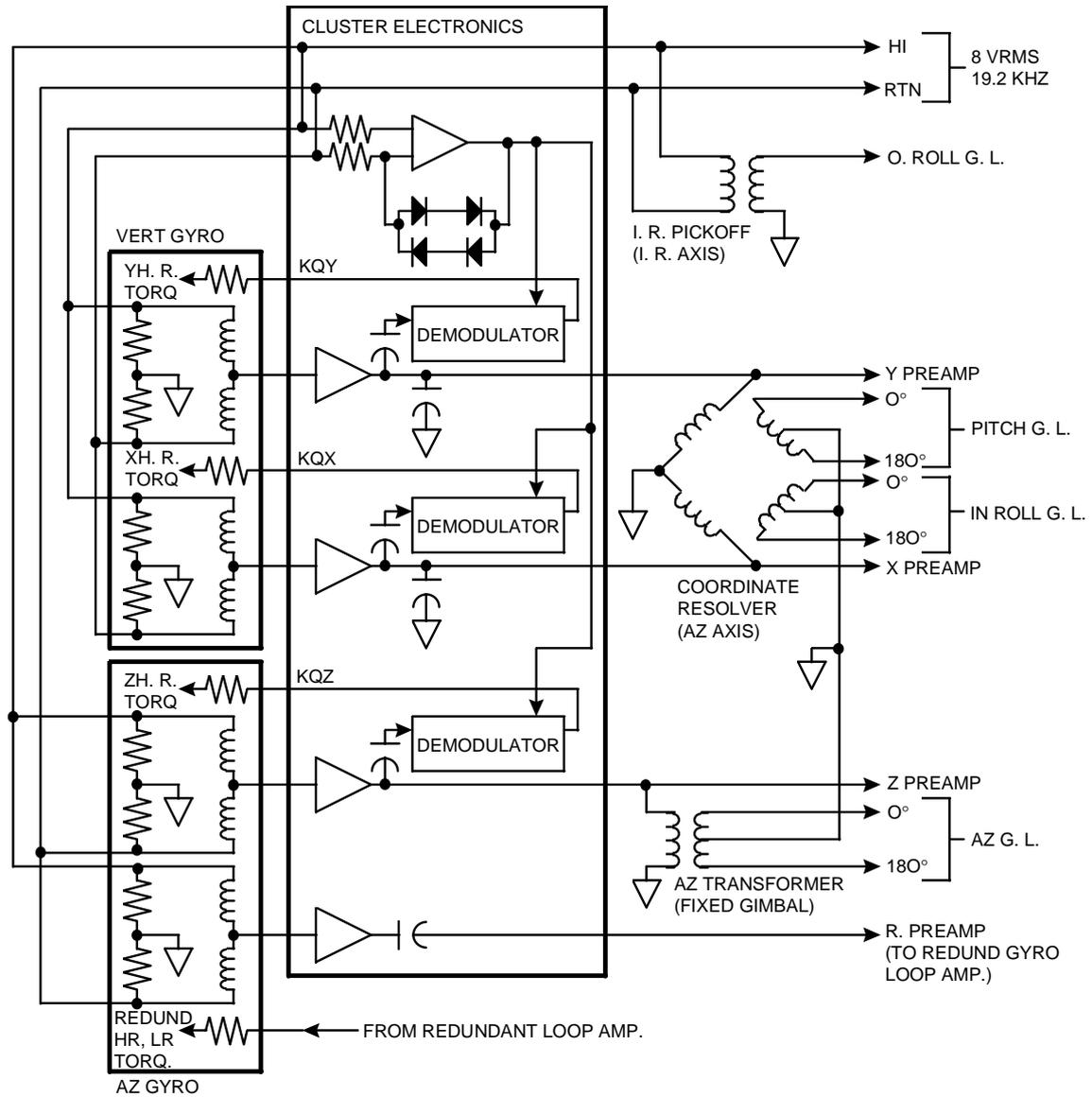


Figure 6-22, Cluster Electronics

6.2.5.2.1 Preamplifiers

The preamplifiers provide voltage amplification for the gyro pickoff signals and buffer networks by presenting high impedance loads. In the case of the X and Y channels, the impedance level of 2000 ohms is achieved by resonating the coordinate resolver inductance with capacitors.

In addition to providing the error signal for the gimbal loops, the Y and Z preamplifier outputs are monitored by the BITE detector circuits on the Platform Electronics #1 Card.

6.2.5.2.2 Spring Rate Compensation

A characteristic of a two-degree of freedom, tuned rotor gyro is a drift which is generated due to the mechanical spring rate torque whenever a displacement angle occurs between the shaft and flywheel. This is held to an initially low value in the construction of the gyro. It is reduced to a negligible value by means of the spring rate amplifiers which feed a portion of the pickoff signal back to the slew torquers to minimize the flywheel displacement angle.

6.2.5.2.3 GYPTO Torquing Electronics

The GYPTO electronics are contained in a hybrid module located on the platform cluster. Synchronized GYPTO torquing pulses from the platform electronics (Wx, Wy and Wz) are used to switch a precision current source to the GYPTO torquing coils. The Wx, Wy and Wz control is determined by the operational flight program and issued through output discrettes to the PE1 card. The PE1 card synchronizes these discrettes to the binary 200 Hz clock. The synchronized Wx, Wy and Wz outputs are then fed to the GYPTO hybrid module in the platform. A functional schematic is shown in Figure 6-23. Since the issuing of the torquing commands would normally vary with the computer instruction times and direct memory access (DMA) times, a precise current waveform is maintained by synchronizing the GYPTO to the 200 Hz clock.

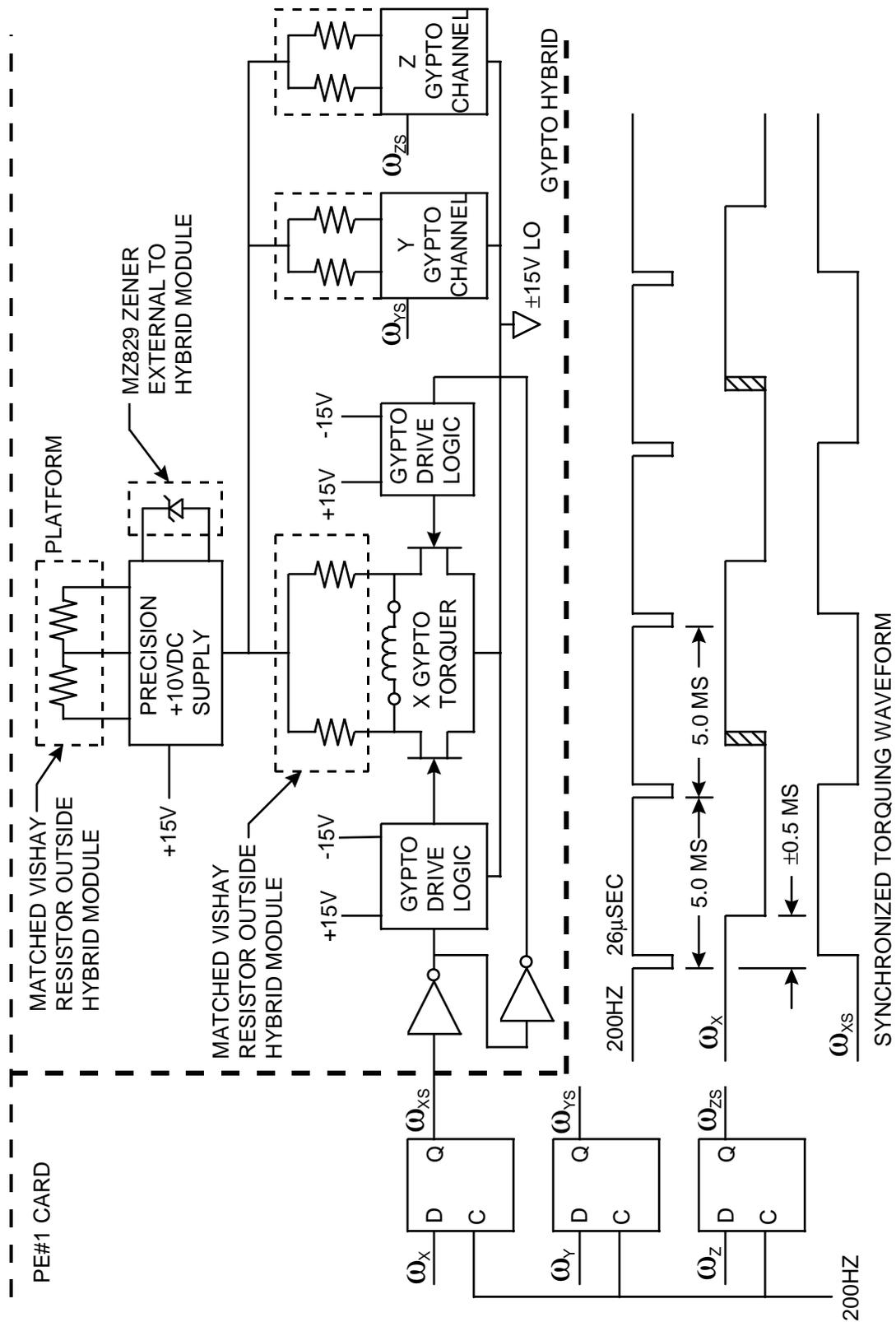


Figure 6-23, Gyro functional schematic

6.2.5.3 Clocks

6.2.5.3.1 System Clocks

Several system clocks are utilized throughout the HAINS to provide a synchronous time base for the various logic and control functions. This section summarizes the various clocks and their functions.

DC/DC Clocks

A 9.83 MHz crystal oscillator and countdown circuitry on the DC/DC card produces 4.9 MHz, 4.9MHz, and 1.2288 MHz pulse train outputs. The BITE circuitry makes a gross frequency check of the 1.2288 MHz signal.

GWS Clocks

The GWS card receives 4.9 MHz and 4.9MHz- inputs from the DC/DC card and through countdown circuitry produces 19.2 kHz, 1.2288 MHz, and 200 Hz. The 200 Hz signal is sent to both the GYPTO logic circuit of the PE No. 1 card and the Interrupt Controller on the I/O card. The 1.2288 MHz signal is sent to the DAL circuitry on the PE No. 1 card.

The 19.2 kHz is sent to the PE No. 1 and IPC circuit cards. The PE No. 1 card, through a power amplifier stage, converts the 19.2 kHz square wave into an 8 Vrms sinusoidal signal used in the platform. The IPC card divides the 19.2 kHz clock to obtain a 400 Hz clock and subsequently produces the 7 Vrms resolver excitation signal utilized by the platform and A/D circuit card.

The 4.9 MHz signal is also passed through a programmable divider which provides 1.875 Hz and 4.8 kHz for the sequencer circuit on the PE No. 1 card. The 1.875 Hz signal is also sent to the CPU watchdog timer on the CPU card. The countdown circuitry also ,provides the PE No. 1 card with 120 Hz.

CPU Clocks

A 20 MHz crystal on the CPU card provides the input for the CPU's timing, clock, and control circuitry. The CPU also produces the 5 MHz signal used on the I/O card.

MUX Clocks

The 16 MHz clock provides timing for the MUX circuitry. In addition, through countdown circuitry, it provides an 8 MHz signal for the A/D card's timing and control logic circuitry and 5a 2 MHz signal for the I/O card's control logic.

6.2.6 H/W Physical Alignment

The mechanical design of HAINS enables it to be interchangeably mounted with KT-70 IMU's on the Navigation Base of an Orbiter.

6.2.6.1 Mounting

The mechanical alignment of the housing to the Navigation Base is obtained by a precision fit between 6 mounting surfaces on its housing and 6 mounting surfaces on the Navigation Base.

Two specially designed mounting feet are used at the fore end of the HAINS to prevent possible bowing of its structure that could occur if a temperature differential occurs between the HAINS mounting feet and the mating Navigation Base (which have different thermal coefficients of expansion). The special mounting feet allow translation expansion or contraction of the HAINS relative to the Navigation Base.

6.2.6.2 Mounting Alignment Repeatability

The mechanical mounting design is able to maintain the alignment of the HAINS housing to the Navigation Base to within 20 arc-sec (1 sigma), when subjected to 125 installation cycles, 75 with the Navigation Base horizontal and 50 with the Navigation Base vertical.

6.2.6.3 Hardware Alignments

HAINS Housing to Orbiter Alignment:

The HAINS fore axis is aligned so that it is pitched down 10.6 degrees below the roll axis of the Orbiter.

IMU Housing to Navigation Base Alignment:

The HAINS housing reference axes are aligned to a set of Navigation Base reference axes, as defined by the Navigation Base mating surfaces, to within 20 arc-seconds (1 sigma) by means of the IMU mounting pad design. This alignment is exclusive of any misalignment in the Navigation Base.

Alignment Accuracy:

Table 6-XV summarizes the accuracy for parameters whose misalignments are calibrated. The calibrated values are inserted into the HAINS Non-Volatile EEPROM memory.

Gimbal Angle Resolver Alignment:

Gimbal angles are defined equal to zero when the gimbals are aligned with their rotational axes mutually orthogonal and individually parallel to a right-handed orthogonal set of HAINS base reference axes, defined by the mounting surfaces of the HAINS case (See Fig. 6-24). All gimbal resolvers are electrically zeroed at the above orientation.

| PARAMETER | RANGE | ONE-SIGMA (ARCSEC) |
|---|-----------------------------|--------------------|
| MISALIGNMENT OF GYRO AXES RELATIVE TO STABLE MEMBER AXES | +/- 3600 ARCSEC | 60 |
| MISALIGNMENT OF ACCELEROMETER AXES RELATIVE TO STABLE MEMBER AXES | 90 DEG +/- 10 MILLI-RADIANS | 10 |
| GIMBAL ANGLE RESOLVER OFFSETS | | |
| OUTER ROLL, RRO | 0 deg +/- 700 ARCSEC | 34 |
| PITCH, PO | | 31 |
| INNER ROLL, IRO | 0 deg +/- 600 ARCSEC | 35 |
| AZIMUTH, AZO | 0 deg +/- 500 ARCSEC | 30 |
| | 0 deg +/- 600 ARCSEC | |
| MISALIGNMENT OF OUTER ROLL GIMBAL AXIS TO IMU BASE | | |
| ABOUT Y BASE AXIS, RGPO | 0 deg +/- 300 ARCSEC | 36 |
| ABOUT Z BASE AXIS, RGAP | 0 deg +/- 400 ARCSEC | 35 |
| NONORTHOGONALITY BETWEEN PITCH AND OUTER ROLL GIMBAL AXES, DP | 90 deg +/- 200 ARCSEC | 15 |

TABLE 6-XV, HARDWARE ALIGNMENT ACCURACY

Accelerometer Alignment:

The accelerometer triad is aligned so that X, Y, Z sense axes are nominally aligned to the respective X, Y, Z stable member axes. The following is a definition of the nominal stable member axes and their relationship to the platform gimbal axes which are also defined (refer to Fig. 6-24):

The X gimbal axis and the X stable member axis are parallel to the navigation base X (roll) axis at the zero degree gimbal reference (vehicle horizontal), with positive X forward and positive gimbal angles defined in the sense of a right-handed rotation of the HAINS case, relative to the stable member, about the positive X axis. The Y gimbal axis and the Y stable member are parallel to the navigation base Y (pitch) axis at the zero degree gimbal reference, with positive Y to the right when facing forward and positive gimbal angles defined in the sense of a right-handed rotation of the HAINS case, relative to the stable member, about the positive Y axis. The Z gimbal axis and the Z stable member axis are parallel to the navigation base Z (yaw)

axis at the zero degree gimbal reference with positive Z downward and positive gimbal angles defined in the sense of a right-handed rotation of the HAINS case, with respect to the stable member, about the positive Z axis. The redundant gimbal axis, at the zero degree gimbal reference, is in the same direction as the outer X gimbal axis, with positive gimbal angles in the same sense as those of the outer axis.

The Z stable member axis lies along the azimuth gimbal axis of rotation. The X stable member axis is perpendicular to the Z axis and lies in the plane defined by the Z axis and the X accelerometer input axis. The Y stable member axis completes an orthogonal right-handed set. With resolvers zeroed, the stable member axes are nominally parallel to the IMU base reference axes.

6.3 OPS MODES

The IMU has two basic modes of operation, the standby mode and the operate mode. In the standby mode, the IMU is powered, but inertial functions are not active. In the operate mode, the IMU is used in a variety of applications. While in the operate mode, the IMU is inertial and is used for calibrations, preflight alignment, flight, and on-orbit alignment.

Table 6-XVI defines the logic settings for each of the HAINS operating modes. Each of the indicated modes of HAINS operation, including mission related details are described in the subsequent sections.

| MODE | VEHICLE POWER +28VDC VEHICLE POWER INPUT | OPERATE MODE DISCRETE INPUT | *ACCELEROMETER GAIN (VELOCITY) SCALE FACTOR SETTING |
|----------------|---|--------------------------------|--|
| OFF | 0 | 0 | - |
| WARMUP/STANDBY | 1 | 0 | - |
| OPERATE | 1 | 1 | 0 or 1 |
| CALIBRATE | 1 | 1 | 0 or 1 |

*Provided by bit 4 of Serial Data Input Word No. 2

+28 Vdc & OPERATE Mode Command Discrete Logic:
1 = presence of input
0 = absence of input

Accelerometer Gain (Velocity) Logic:
1 = HI scale factor
0 = LO scale factor (flight mode)

TABLE 6-XVI, MODE OF OPERATION LOGIC

6.3.1 Off Mode

In this mode no vehicle power is supplied to the HAINS. Within environmental constraints, the unit can be in this mode both on the ground and during any of the mission phases without harm.

6.3.2 WARM-UP/STANDBY Mode

This mode is entered when +28 Vdc vehicle bus power is externally switched into the HAINS with no OPERATE mode command discrete present. Under this condition, the vehicle bus power is used to directly energize individual Temperature Control Amplifier's (TCA) that automatically control power to each of three platform

heaters; the azimuth and vertical gyro heaters, and gimbal heater.

Temperature sensors (thermistors) are mounted at control points within the platform to generate signals that command each TCA to supply heat as needed.

The HAINS is capable of remaining in the STANDBY mode and maintaining thermal stability for a minimum of 170 hours per flight. The maximum warm-up time of the HAINS is as given in Table 6-XVII.

| WARM-UP TIME | AMBIENT CABIN TEMPERATURE | INLET AIR LINE TEMPERATURE | VOLTAGE |
|----------------------|---------------------------|----------------------------|---------|
| 50 minutes (max) | 75 deg F | 75 deg F | 28 Vdc |
| 120 minutes (max) | 35 deg F | 35 deg F | 28 Vdc |

TABLE 6-XVII, WARM-UP TIMES

The vertical gyro and azimuth gyro temperatures are monitored by the TCA circuitry. When the temperature of the vertical gyro exceeds 146.5 degrees F plus or minus 3 degrees F and when the temperature of the Azimuth gyro exceeds 150 degrees F plus or minus 3 degrees F, the Platform Temperature Ready discrete is set to its true state.

This discrete is sent to the GPC to announce that the cluster is near its operating temperature. The OPERATE mode command can now be issued to the HAINS in order to put it in its IN OPERATE mode.

The vertical gyro temperature is monitored for a safe temperature condition. The Platform Temperature Safe discrete is issued in its true state when the vertical gyro temperature is below 156.5 degrees F plus or minus 3 degrees F.

These discrettes provide a BITE function by providing a positive indication that the HAINS thermal control systems are properly functioning. This is particularly important if the HAINS is to be kept in its WARM-UP/STANDBY mode for several or more hours such as might occur on the launch pad or while in Orbit. If these discrettes were not provided and the thermal control circuitry failed to deliver power to the platform heaters, there would be no indication of trouble in the WARM-UP/STANDBY mode.

After the Platform Temperature Ready discrete has been issued, the HAINS is in its STANDBY mode. The HAINS is capable of remaining in the STANDBY mode and maintaining thermal stability for a minimum of 170 hours of flight.

NOTE: The CAPRI Temperature Ready and CAPRI Temperature Safe discrettes used on the KT-70 IMU are not used in the HAINS IMU.

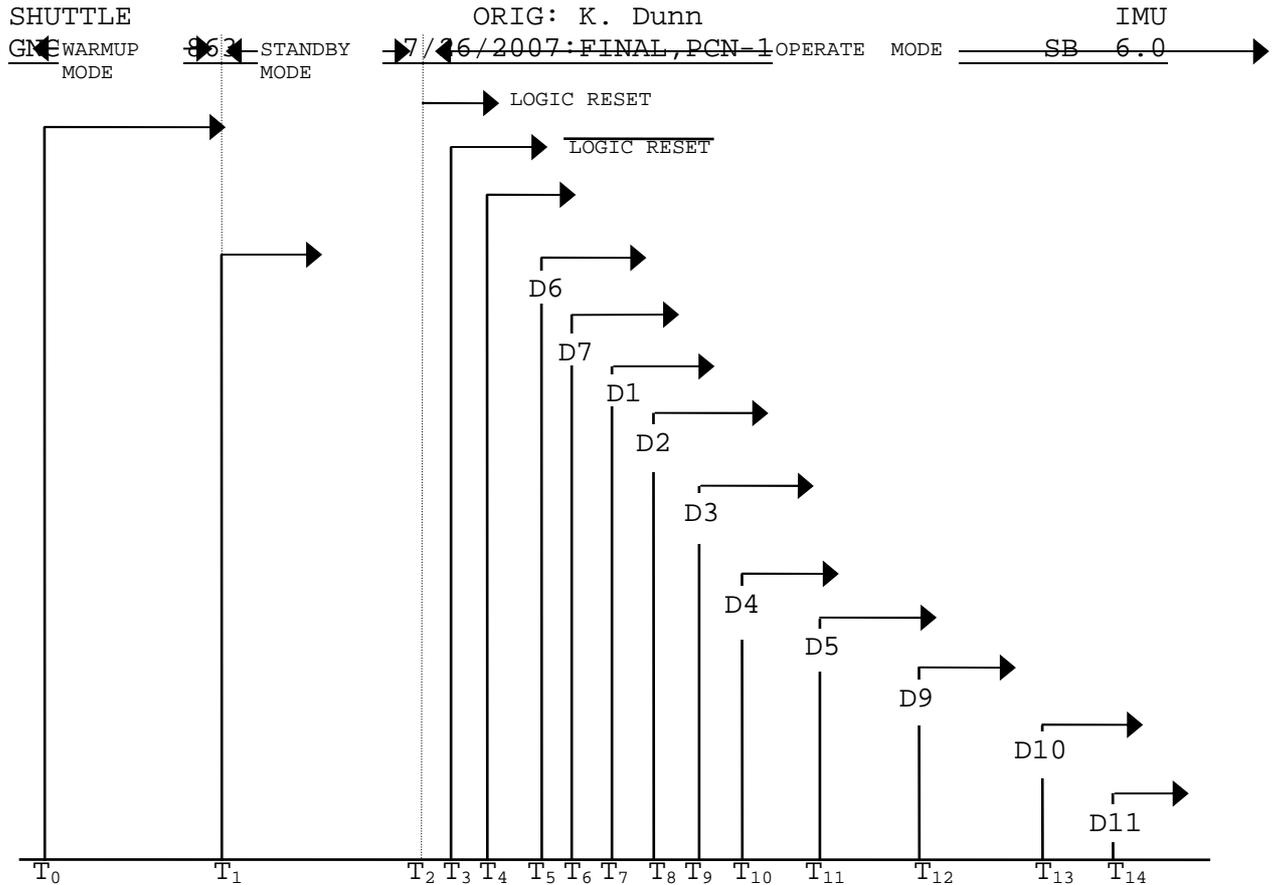
These discretes are set to their true state upon application of 28 Vdc power. The GPC then recognizes these discretes as being in an operational state upon application of 28 Vdc power.

6.3.3 Start-up Sequence

Figure 6-25 indicates the automatic start-up sequence that takes place within the HAINS. A sequencer on the PE No. 1 Card of the HAINS provides the required start-up sequence discretes. These discretes provide the required start-up switching action such as commanding the cage submode, individually starting up each two-axis GYROFLEX gyro, and appropriately setting the gains in each gimbal isolation loop. Selected discretes are transmitted to the GPC via Output Word No. 1.

In the cage submode, each of the four gimbals is caged to the null of the sine winding of its respective readout resolver.

The HAINS start-up sequence is completed approximately 85.4 seconds after issuance of the Logic Reset (LR) discrete. The pitch and outer roll gimbals are caged at 13.0 sec following LR while azimuth and inner roll are caged at 14.9 sec following LR. Signals from the Mode Control Logic Circuitry are used to put the network into the cage loop configuration. The caging command signal is derived from the pertinent SINE or COSINE winding of the 1X resolver and is fed into the caging demodulator that is referenced to 7V, 400 Hz. The signal travels through the network whose output is used to drive the gimbal torquers. The gimbals are then positioned at their readout resolver nulls and the gimbal loops are inertially controlled by the gyros. Zero torquing rates are applied to the gyros at this time. Figure 6-26 shows the start-up sequence timing for special EEPROM Write program entry and validity of digital data output.



* OUTPUT WORD NO. 1 BITS 15-19
TIME TO OPERATE MODE ~ 85.4 sec

NOTES:

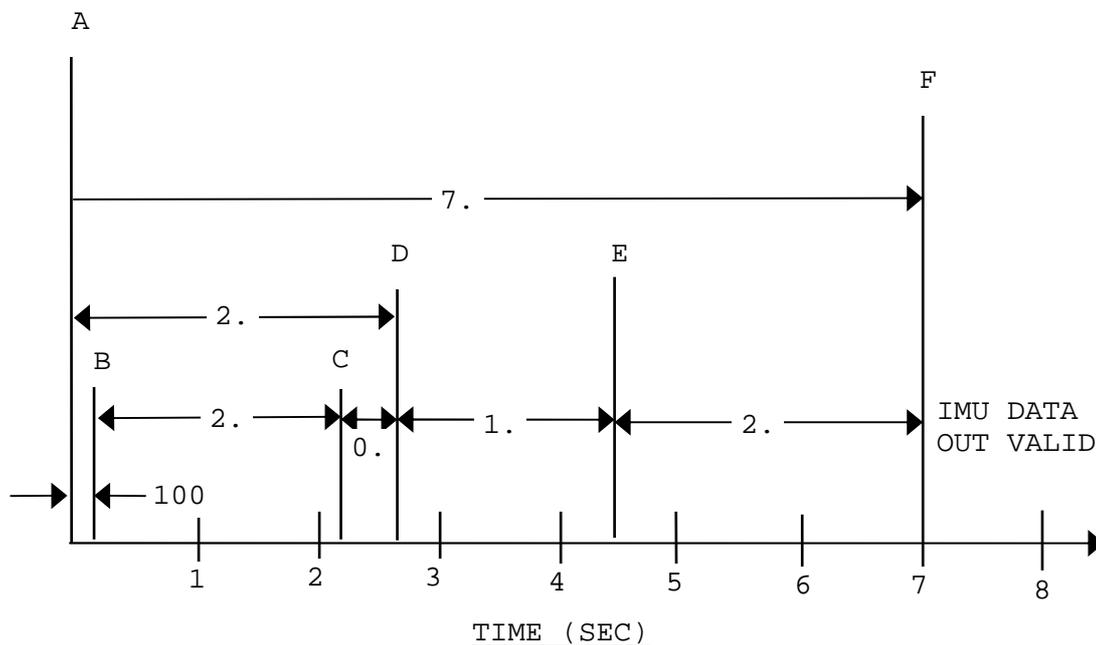
- 1 87.2 SEC DELAY OF OPERATE MODE. TURN ON IF 28 Vdc POWER OR OPERATE MODE DISCRETE IS INTERRUPTED PER NOTE 2.
- 2 THE ABSENCE OF THE OPER MODE DSC FOR A PERIOD IN > 1 SEC MIN. TO 15 SEC MAX WHILE +28 Vdc IS APPLIED, WILL CAUSE THE IMU TO GO INTO THE WARMUP/STANDBY MODE.
- 3 COOLING AIR TEMP: +35 DEG F < T < +90 DEG F
AMBIENT AIR TEMP: 0 DEG F < T < 120 DEG F
- 4 AFTER PLAT TEMP RDY DSC IS ISSUED THE IMU IS IN STANDBY MODE.
- 5 THERE IS A 100 MSEC MAX TIME DIFFERENCE BETWEEN OPERATE MODE DISCRETE AND THE START OF LOGIC RESET.
- 6 WARMUP TIME AND TIME OF ISSUANCE OF PLAT AND TEMP RDY DSC'S IS TEMPERATURE DEPENDENT AND WILL BE ACHIEVED WITHIN THE MAXIMUM TIMES INDICATED BELOW AT SPECIFIC TEMP AND AT 28 Vdc INPUT POWER:

| MAX TIME (MINUTES) | AMBIENT TEMP (DEG F) | INLET AIR TEMP (DEG F) |
|-----------------------|-------------------------|---------------------------|
| 50 | 75 | 75 |
| 120 | 35 | 35 |

FIGURE 6-25, IMU START-UP SEQUENCE TIMING

| <u>TIME RANG</u> | <u>DURATIO</u> <u>N</u> | <u>SEQUENCE</u> | <u>COMMENTS</u> |
|------------------------------------|----------------------------|-----------------|--|
| T ₀ to T ₁ | SEE NOTE 6 | WARMUP | |
| T ₁ to T ₂ | SET BY USER | | PLATFORM TEMP SAFE DISCRETE PLATFORM TEMP READY DISCRETE |
| T ₂ to T ₃ | 2.4 SEC (MAX) | LOGIC RESET | RECEIVE OPERATE MODE COMMAND FROM GPC |
| T ₃ to T ₄ | 6.5 SEC | LOGIC RESET | APPLY 60 Hz BRAKING TO VERT GYRO |
| T ₄ to T ₅ | 6.5 SEC | | REMOVE VERT GYRO BRAKING. APPLY 60 Hz BRAKING TO AZ |
| T ₅ to T ₆ | 1.9 SEC | D6 | REMOVE AZ GYRO BRAKING. CAGE PITCH AND OUTER ROLL GIMBALS. |
| T ₆ to T ₇ | 1.7 SEC | D7 | CAGE INNER ROLL AND AZIMUTH GIMBALS. |
| T ₇ to T ₈ | 4.7 SEC | D1 | START VERTICAL GYRO WHEEL AT 240 Hz. |
| T ₈ to T ₉ | 4.3 SEC | D2 | CLOSE PITCH AND INNER ROLL GIMBALS IN LOW GAIN ISOLATION MODE. CLOSE OUTER ROLL GIMBAL LOOP. ACTIVATE EPA BITE. * |
| T ₉ to T ₁₀ | 2.1 SEC | D3 | SWITCH PITCH AND INNER ROLL GIMBALS TO HIGH GAIN ISOLATION MODE. SWITCH VERT GYRO WHEEL EXCITATION TO APPROX. 530 Hz. START AZIMUTH GYRO WHEEL AT 240 Hz. |
| T ₁₀ to T ₁₁ | 4.3 SEC | D4 | CLOSE AZIMUTH GIMBAL LOOP IN LOW GAIN ISOLATION MODE. SWITCH AZIMUTH GYRO WHEEL EXCITATION TO APPROX. 530 Hz. |
| T ₁₁ to T ₁₂ | 17.1 SEC | D5 | SWITCH AZ LOOP TO HIGN GAIN ISOLATION MODE. CLOSE REDUNDANT GYRO AXIS LOOP. SWITCH X SLEW TORQUER DRIVE FROM THE YGYPA NUTATION DAMPER TO THE SLEW SWITCH HYBRID. ENABLE THE SLEW SWITCH HYBRID TO RESPOND TO S/W SLEW AND GYPTO CMDS. ACTIVATE GIMBAL LOOP BITE. * |
| T ₁₂ to T ₁₃ | 32.0 SEC | D9 | START GYRO WHEEL SPEED LOCK-IN SEQUENCE. SWITCH Z SLEW TORQUER DRIVE FROM THE RGYPA NUTATION DAMPER TO THE SLEW SWITCH HYBRID. |
| T ₁₃ to T ₁₄ | 4.3 SEC | D10 | ENABLE PITCH SWITCH CIRCUIT |
| T ₁₄ to OPER | | D11 | ACTIVATE GWS BITE AND ISSUE THE IN OPERATE MODE DSC |

FIGURE 6-25, IMU START-UP SEQUENCE TIMING (CONTINUED)



- A = RECEIVE OPERATE MODE DISCRETE.
- A TO B = 100MS IMU POWER STABILIZATION.
- B TO C = 2.0 SEC LOGIC RESET, DATA OUT UNAVAILABLE.
- C TO D = DATA OUT UNCERTAIN.
- D TO E = "PRESET" DATA OUT.
CPU DETERMINES IF SPECIAL EEPROM WRITE PROGRAM ENTRY DESIRED. IF DESIRED, ENTERS SPECIAL PROGRAM.
- E = CPU LOCKOUTS ENTRY TO SPECIAL EEPROM WRITE PROGRAM.
- F = SOFTWARE INITIALIZATION COMPLETE. DATA OUT VALID.

PRESET DATA OUT

| *OUTPUT WORD#1: BITE/STATUS WORD | | |
|---|--------------------|--------------------|
| <u>BIT #</u> | <u>DESCRIPTION</u> | <u>LOGIC STATE</u> |
| 4 | HAINS GOOD | HI |
| 5, 6, 7, 8, 9, 10 | | |
| 11, 12, 13, 14 | FAIL BITS | LO |
| 15, 16, 17, 18, 19 | SEQUENCER BITS | HI |
| * OUTPUT WORDS #2 TO #14: ALL BITS ARE SET TO LO. | | |

FIGURE 6-26, HAINS OUTPUT DATA INITIALIZATION SEQUENCE

6.3.4 Operate/Calibrate Mode

There is no functional difference in the operation of the HAINS in the OPERATE and CALIBRATE modes. It is the test/vehicle computer programs that differentiate between these modes. Therefore, insofar as the HAINS operation is concerned, the OPERATE and CALIBRATE modes are considered one mode called the OPERATE/CALIBRATE mode. A description of this mode follows: Entrance - To enter, the HAINS OPERATE/CALIBRATE mode from the WARM-UP/STANDBY mode, the HAINS receives an OPERATE mode command discrete from the Computer (via the MDM) This enables +28 Vdc bus power to reach the inputs of the power supplies.

Within 2.1 seconds maximum after receipt of the OPERATE mode command discrete, all of the HAINS DC and AC voltages have been generated and a Logic Reset (LR) discrete is then internally issued within the HAINS to initialize the HAINS circuitry

After achieving the operate mode, the IMU issues an in operate mode discrete to the GPC. The IMU is now inertial and can be used for calibration, alignment, or other inertial uses on the ground or in flight. There is no difference in the status of the IMU hardware for any inertial application. The GPC's control the IMU to accomplish different tasks.

6.3.4.1 HAINS Calibration and Acceptance Test

Through system level testing of the HAINS, calibration parameters can be computed to compensate for system errors. The HAINS testing, both at the factory and while in any customer facility, results in the determination of the calibration parameters. These calibration parameters along with calibration parameters obtained from tests performed at platform level and card level form the basis for the values stored in EEPROM. A list of all the EEPROM parameters (including calibration parameters) can be found in the HAINS Technical Description.

Following the storage of calibration parameters to EEPROM, any subsequent testing will result in delta compensation; the difference between the whole error and the internally compensated value. These delta compensation terms would then have to be compensated by the external test computer or GPC.

Prior to shipment, each HAINS IMU undergoes a series of tests to determine the values of its calibration parameters and to determine if its operational capability satisfies all system requirements as defined in the Procurement Specification.

At the completion of testing, the final set of calibration parameters is written to a region of EEPROM for internal compensation in the HAINS. This enables the establishment (with the exception of 3 redundant gyro scale factor terms) of a

generic set of I-LOAD parameters for use as input for Hangar or Launch Pad calibrations.

Calibration and evaluation is performed at component level, during integration tests, and during the formal Acceptance Test Procedure (ATP).

6.3.4.1.1 Component Level Testing

Thirty parameters are calibrated at component and card level, and are written to EEPROM for compensation before the start of an ATP. These parameters are 6 accelerometer linearity terms, 6 DAL temperature breakpoints, 12 DAL temperature slopes, and 6 DAL temperature scale factors.

6.3.4.1.2 Integration Tests

Prior to the start of an ATP a series of integration tests are run. All output signals on all connectors are measured and recorded. Numerous tests are performed to check the overall goodness of the HAINS; for example, gyro nutation and gimbal flip. A full functional test is run and a baseline set of calibration parameters is obtained. This is accomplished by running Multicals (The terms Multical and Hangar Cal refer to the same multiposition calibration.), Inner Roll Tracking Cal, and Attitude Calibration and Accuracy Tests (ACAT). During the ACAT test resolver tracking parameters are obtained and written to EEPROM for compensation. These consist of 192 Roll axis, 192 Azimuth axis, and 97 Pitch axis parameters.

6.3.4.1.3 Acceptance Test Procedure (ATP) Tests

The Acceptance Test Procedure that must be run and passed on each unit before shipment consists of the following tests.

6.3.4.1.3.1 *Operational and Up Mode Test*

This test monitors changes in state of various discretes such as Input power, Operate Mode command, Serial I/O, Platform Temperature, and In-Operate Mode and prints occurrences of each. The timing of each of the run-up discretes is checked as well as the gimbal angles during the run-up sequence. Values are compared to limits and pass/fail indications are printed.

6.3.4.1.3.2 *Multical Test*

This test calibrates 39 platform parameters by positioning the cluster to 13 different orientations with respect to a North, West, Up reference frame with the HAINS case in one fixed position. Gyro drift and accelerometer measurements are taken at

the various positions and a data reduction routine combines the data in a set of solution equations to determine the parameters required. The parameters are printed along with their respective limits and pass/fail indications are determined and printed. This program is run 6 times during the course of the ATP, 3 times each at 2 case orientations. Just prior to the start of the Launchpad Verification Sequence, the final values are written to EEPROM for compensation.

6.3.4.1.3.3 *Inner Roll Tracking Calibration Test*

This test calibrates the Inner Roll axis over a fine (± 2 degree) and course (± 10 degree) range. After the gimbals are initially positioned, the Outer Roll loop is opened via the test console and the Inner Roll axis is moved in quarter of a degree increments through the fine range and one degree increments through the course range. Data is taken and stored at each point. At the end of the test, a data reduction routine is invoked and 17 fine and 21 course calibration parameters are printed along with their respective limits and a corresponding pass/fail indication. These values are then written to EEPROM for compensation during remaining ATP tests.

6.3.4.1.3.4 *Attitude Calibration & Accuracy Test (ACAT)*

In addition to the 481 tracking parameters calibrated during Integration testing, the ACAT program calibrates 9 additional gimbal alignment parameters.

The test is run in 3 parts, the Roll test, the Azimuth test and the Pitch test. Each test involves orienting the axis in question to a vertical position and rotating the rotary axis of the test table in 10 degree increments for a 360 degree rotation. Data is recorded at each position which compares resolver readout to table position. This information is used to compute mean and standard deviation values of resolver accuracy which are subjected to pass/fail criteria. Data is also recorded at each point on the level resolvers and the accelerometers. At the end of the test, a data reduction algorithm is invoked which computes and prints the parameters along with their limits and respective pass/fail indications.

This test is run 4 times during the course of the ATP. At the end of the fourth run, the 4 values obtained for each parameter are averaged. Just prior to the start of the Launchpad Verification Sequence the average values are written to EEPROM for compensation.

6.3.4.1.3.5 *Short Term Stability Test*

The Short Term Stability Test is a performance rather than a calibration test. The test table is oriented in the KSC calibration position. The test then consists of positioning the

cluster to the first 3 calibration positions. This puts each accelerometer up and each gyro north. In each position, 15 consecutive ten minute estimates of drift and accelerations are recorded. A data reduction program computes means, standard deviations, and initial point deviations for the north gyro drift, the up acceleration, the point to point differences, and the redundant gyro drift. Values are compared to respective limits and all data, results, limits, and pass/fail indications are printed.

6.3.4.1.3.6 *Gimbal Flip Test*

The Gimbal Flip Test is also a performance test. The ACAT program is run and the Gimbal Flip option is selected. The table and clusters are positioned so that the case is level and pointed north and the Pitch axis is aligned along the rotary axis of the table. The table is then moved to a rotary angle of 86 degrees and tilts are estimated and recorded. The table is then moved to a rotary angle of 94 degrees to induce a gimbal flip and the tilts are estimated and stored again. This procedure is repeated until 10 sets of data are collected. The mean and standard deviation of the 3 tilt measurements are computed and the standard deviation is compared to a limit. All data, results, limits, and pass/fail indications are printed.

6.3.4.1.3.7 *Environmental Tests*

During the ATP, 2 types of environmental tests are run, thermal cycling and vibration.

Thermal Test

With the unit in the thermal chamber, a baseline set of gyro drift data is obtained at 70 degrees F. The temperature is then raised to 90 degrees F and then lowered to 35 degrees F and twelve 5 minute drift estimates are obtained at each temperature extreme. A data reduction program compares each of these 24 sets of drift values (X, Y, and Z) to the baseline data, prints the data, the baseline, the differences, a limit, and the pass/fail indication for each point.

In addition to this, the ATP thermal testing includes two hours of operation with cooling air at 100 degrees F.

Vibration Test

This test is run in three parts. In each part a different axis (X, Y, or Z) is aligned along the vibration axis of the test table. Once an axis is aligned, tilts are estimated (30 second estimates) until they are stable. (Three consecutive estimates within 2 arc-seconds on all three channels.) The vibration is then turned on and another tilt estimate is obtained. Finally, the vibration is turned off and three more tilt estimates are obtained.

The last estimate before vibration is averaged with the first point after vibration and the average is compared to the vibration point and a pass/fail is recorded. The differences between the pre- and post-vibration data on the level accelerometer axes and the vertical resolver axis are also compared to their respective limits, and pass/fail indications are recorded for these as well.

6.3.4.1.3.8 *Heading Sensitivity Test*

This test calibrates three gyro heading sensitivity drift terms, one at each of three launch orientation skewed cluster positions. These parameters are written to EEPROM at the completion of the test for compensation during the ATP Launchpad Verification Sequence. For each of three skew orientations, the test consists of a series of eight, two position gyrocompass measurements followed by an absolute accuracy test which separate heading error from the data. Means and standard deviations for north, west and up tilt error are computed and compared to limits at the end of the gyrocompass portion of the test. Data, results, and pass/fail indications are printed. At the completion of the Absolute Accuracy portion of the test, the mean and standard deviation of the heading error for the eight runs are computed and compared to limits. Data, results, and pass/fail indications are printed. Finally, the heading sensitivity drift is computed and printed.

6.3.4.1.3.9 *Launchpad Verification Sequence*

The Launchpad Verification Sequence is a test that is run to show the HAINS meets the hold time criteria that is required prior to launch. The sequence consists of a Gyrocompass test, a Velocity and Tilt Initialization Test, a tuned Inertial Mode Test and an Absolute Accuracy Test. The test is also used to calibrate the up axis drift parameter.

The test is run in three cluster skew orientations. The Gyrocompass and Velocity and Tilt tests mathematically correct for any tilt or heading errors. The cycle structure is then changed to mimic inertial mode operation on the vehicle and a two hour Inertial Mode Test is run. The tilts measured during Velocity and Tilt and at the end of the two hour Inertial Mode are compared to limits and printed along with their respective pass/fail indications. At the completion of the Inertial Mode Test, the cycle structure is restored to the normal ATP mode of operation and the Absolute Accuracy Test is run to determine the up axis drift value. The value is compared to a limit and printed along with a pass/fail indication.

6.3.4.1.4 Attitude Determination

The attitude determination submode sets the accelerometer loops to high gain and then computes the transformation from navigation base to the present cluster in the north/west/up (NWU) reference coordinate system. This transformation is a prerequisite for running all subsequent options involving ground alignment and calibration. Therefore, it should be noted that this option must be executed for each IMU every time that the vehicle/ navigation base is moved or whenever the transformation data may have been destroyed in GPC memory. The attitude determination function utilizes resolver and velocity data to determine the orientation of the navigation base with respect to an NWU coordinate frame. A gyrocompassing technique is used to determine the position of north, west, and up relative to cluster position. This, in conjunction with the gimbal angles, is used to define the required navigation base-to-NWU transformation. It should be noted that a specific cluster orientation is not required, but generally the cluster will be near the initial caged position as attitude determination will normally be performed shortly after run-up. The duration of this submode is approximately 4 minutes.

6.3.4.1.5 Ground Calibrations

The accuracy of the IMU is extremely dependent on ground calibrations of 61 IMU parameters. The values for these parameters are stored on the Orbiter GPC mass memories and are used in the software as compensation for hardware inaccuracies. Three levels of calibration (CAL) exist for the IMU:

- a. Factory CAL
- b. Hangar CAL
- c. Preflight CAL

Table 6-XVIII indicates which parameters are computed in each type of CAL.

| Parameter | CAL level | | |
|---|-----------|--------|-----------|
| | Factory | Hangar | Preflight |
| KTX(Y,Z) - Gyro torquer scale factor | X | X | |
| DFX(Y,Z) - Gyro drift | X | X | X |
| DIX(Y,Z) - Gyro mass unbalance drift | X | X | |
| DSX(Y,Z), DOZ - Quadrature drift | X | X | X |
| BETAXY(XZ,YX,YZ,ZX,ZY) - Gyro to accelerometer misalignment | X | X | |
| KOX(Y,Z,XH,YH,ZH) - Low and high gain accel. bias | X | X | X |
| KIX(Y,Z,XH,YH,ZH) - Low and high gain accel. scale factor | X | X | X |
| DELTAZX(YX,YZ,ZX,ZY) - Accel to accel misalign. | X | X | |
| KSX(Y,Z) - Accelerometer asymmetry | X | X | |
| RRO - Outer roll offset | X | | |
| IRO - Inner roll offset | X | | |
| PO - Pitch offset | X | | |
| AZO - Azimuth offset | X | | |
| DP - Non-orthogonally between pitch and outer roll gimbal | X | | |
| RGAO - Misalignment of roll to X stable member | X | | |
| RGPO - Misalign of roll to Y stable member | X | | |
| HS11 (21,31,41,51,61) Heading | X | | |
| HS12 (22,32,42,52) sensitivity | | | |
| DFR - Redundant gyro drift | | X | X |
| DIR - Redundant gyro input axis mass unbalance | | X | X |
| DOR - Redundant gyro output axis mass unbalance | | X | X |
| DSR - Redundant gyro spin axis mass unbalance | | X | X |
| KXR (YR,ZR) - Redundant gyro scale factor | | X | X |
| M12B (13B,23B) - Difference vector between relative attitude measured by resolvers and accelerometers for IMU pairs | X | | |

TABLE 6-XVIII, IMU CALIBRATION PARAMETERS

6.3.4.1.5.1 *Factory CAL*

The Factory CAL involves 127 hr of testing. The test is both operator and computer controlled. For the most part, though, the test is automatic in that the cluster is automatically slewed through a predetermined sequence of positions. At each position, accelerometer and gyro data are collected.

At the end of the entire sequence of positions, the data are used to compute the CAL parameters. The equations used to compute these parameters are based on an assumed error model of the IMU. This concept will be discussed further under preflight CAL's.

Two sets of parameters that bear special mention are the heading sensitivity and the $M_{_}B$ terms, since these are computed only at the factory. The heading sensitivity terms are correction terms (fine tuning) to compensate for differential gyro drifts at various attitudes. Drift data are taken at each possible launch attitude for the IMU, so the IMU can be used as either IMU 1, 2, or 3. The test itself is accomplished by positioning the platform at each of the 12 possible launch attitudes. While in that position, the IMU is torqued at Earth rate (to compensate for effects of Earth rotation) and a least squares fit of accelerometer data is computed. The data are used to estimate the north and west drifts of the platform and the resulting heading sensitivity correction. This test is run only after the other calibration parameters have been computed so that any platform drift is due to heading sensitivities.

The $M_{_}B$ terms are necessary for the gyrocompass goodness test that is accomplished at the end of the gyrocompassing portion of the preflight alignment. These parameters are a measurement of the resolver errors in an IMU and are determined by measuring the attitude of the IMU with accelerometers and resolvers simultaneously. Measuring attitude with the accelerometers provides only two axes (level axes) of information, so only these two axes are checked in the gyrocompass goodness test.

6.3.4.1.5.2 *Hangar CAL*

The purpose of the hangar CAL is to update a subset of the parameters that were originally computed at the factory. A hangar CAL is required every 4 months.

Under GPC control, the hangar CAL routine will calibrate the three IMU's simultaneously by torquing the platforms to 13 predefined orientations shown in Figure 6-27. At each position, data are collected to obtain drift and vertical acceleration estimates. As shown in Figure 6-28, data are collected in three of the cardinal positions to compute gyro misalignments and torquer scale factors. In two other cardinal positions, estimates of west gyro drift are obtained, and a least squares fit of the data is used to estimate up axis drift.

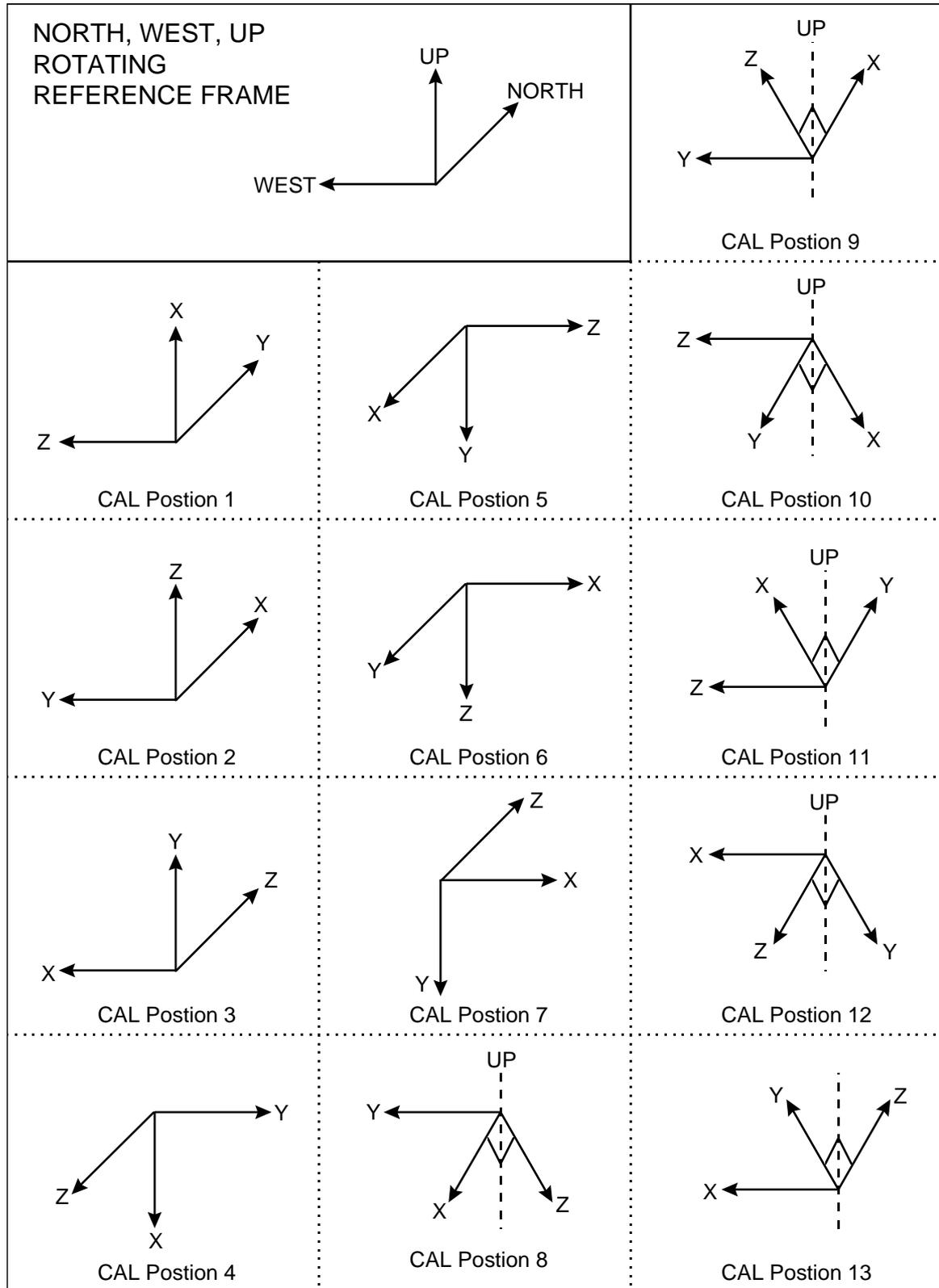


Figure 6-27, IMU Cluster CAL Positions

| Parameter | Time per position (min) | | | | | | | | | | | | |
|-----------|-------------------------|----|----|----|----|----|----|---|---|---|---|---|---|
| | 33 | 38 | 38 | 64 | 13 | 64 | 10 | 7 | 7 | 7 | 7 | 7 | 7 |

| | Position | | | | | | | | | | | | |
|-----------|----------|---|---|---|---|---|---|---|---|----|----|----|----|
| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 |
| KOX, KOXH | X | | | X | | | | X | X | X | X | X | X |
| KOY, KOYH | | | X | | X | | | X | X | X | X | X | X |
| KOZ, KOZH | | X | | | | X | | X | X | X | X | X | X |
| KIX, KIXH | X | | | X | | | | | | | | | |
| KIY, KIYH | | | X | | X | | | | | | | | |
| KIZ, KIZH | | X | | | | X | | | | | | | |
| KSX | X | | | X | | | | X | X | X | X | X | X |
| KSY | | | X | | X | | | X | X | X | X | X | X |
| KSZ | | X | | | | X | | X | X | X | X | X | X |
| DELTYX | X | | X | X | X | | | | | X | X | | |
| DELTZX | X | X | | X | | X | | X | X | | | | |
| DELTZY | | X | X | | X | X | | | | | | X | X |
| DFX | X | X | X | X | X | X | | X | X | X | X | X | X |
| DFY | X | X | X | | X | X | | X | X | X | X | X | X |
| DFZ | X | X | X | X | | X | X | X | X | X | X | X | X |
| DIX | X | X | X | X | X | X | | X | X | X | X | X | X |
| DIZ | X | X | X | X | X | X | X | X | X | X | X | X | X |
| DSX | X | X | X | X | X | X | | X | X | X | X | X | X |
| DSZ | X | X | X | | X | X | X | | | | | X | X |
| KTX | | X | X | | X | | | | | | | | |
| KTY | X | X | | | | X | | | | | | | |
| KTZ | X | | X | X | | | | | | | | | |
| SXY | X | X | X | X | X | | | | | X | X | | |
| BXZ | X | X | X | X | | X | | X | X | | | | |
| SYX | | X | | | | | | | | | | | |
| BYZ | X | X | X | | X | X | | | | | | X | X |
| SZX | | | X | | | | | | | | | | |
| SZY | X | | | | | | | | | | | | |
| DOZ | X | X | X | X | | X | X | X | X | X | X | X | X |

Figure 6-28, HANG_CALA parameters vs. required positions and time.

To delete the unused high gain calibration positions 8 through 13, OI-22 CR 89786C added two new I-LOADs that specify desired Hangar CAL positions. See table 6-XIX.

| I-LOAD | MSID | CONTROL WORD |
|------------|-----------|------------------|
| HCA_HI_POS | V99U9976C | 1111111000000000 |
| HCA_LO_POS | V99U9975C | 1111110111111000 |

Table 6-XIX, Hanger Cal I-Loads

Each of the first 13 bits represent a cluster position. If the bit is set to one, then that position will be used during the Hangar calibration. Not performing calibrations for the unused high gain positions 8 through 13 saves one hour of run time.

The HANGAR CAL collects the gyro calibration data and the high-gain accelerometer calibration data with the accelerometer gain set to high for each of the orientations specified by the high-gain cluster positions control word. Low-gain accelerometer calibration data are then collected with the accelerometer gain set to low gain for each orientation specified by the low-gain cluster positions control word, except position 7. Low-gain calibration data is not collected for position 7.

For the HAINS IMU, high gain & low gain accelerometer data are identical. The high gain mode was retained to maintain compatibility with the KT-70 IMUs. The HAINS is used in the OPERATE/CALIBRATE mode with the accelerometer scale factor set at low gain.

After the final position, the error terms (deltas) associated with each calibration parameter are computed. All parameters are automatically updated at the end of calibration. The HANGAR CAL requires 5 hours to complete.

6.3.4.1.5.3 Preflight CAL

The preflight CAL, normally accomplished on launch day at approximately T-5 hr, updates a subset of the parameters previously computed in the hangar CAL. The CAL requires 2 hours to complete, so if a parameter is out of spec on the first CAL, there is sufficient time to repeat the CAL without a launch impact. More details are provided here for preflight CAL, since this is the only CAL routine flight controllers have the opportunity to see.

Similar to the hangar CAL, under GPC control the IMU is positioned to seven predefined orientations (the first seven shown in Figure 6-27). Also similar to the hangar CAL, the first pass through the seven positions is to collect gyro calibration data and high-gain accelerometer data, while the second pass through the positions (no. 7 is again skipped on the second pass) is to collect low-gain accelerometer data.

To get an appreciation for what the software actually does during the CAL routine, the following step-by-step explanation is provided for one of the positions. Assume that this is the first pass through and the accelerometer gain setting is high.

- a. Form the transformation from north, west, up (NWU) to the CAL position. The rotations required to achieve the desired position are predefined. The rotations assume the cluster coordinate system is coaligned with NWU.
- b. Compute the transformation from the navigation base (NB) to the desired cluster (DC) position.

$$T_{NB}^{DC} = [T_{NWU}^{DC}] [T_{NB}^{NWU}]$$

T_{NWU}^{DC} was computed in a. and T_{NB}^{NWU} was computed in the IMU attitude determination routine (discussed earlier).

- c. Using the matrix T_{NB}^{DC} , reposition the IMU's.
- d. Since this repositioning will be inaccurate due to possible resolver errors, accelerometer data are collected to determine the error in the orientation. For example, in position 1, the X-axis accelerometer should sense $-1g$, while the other two axes should sense zero g. This method assures the correct alignment of the level axes only. Rotational errors about up will be transparent.
- e. The measured misalignment is torqued out to achieve the desired orientation precisely.
- f. Collect acceleration data to measure north and west drift (remember, up axis drift can not be measured with the accelerometers) and to measure the gravity vector. The data are least squares filtered to improve the accuracy. Data are collected for 120 sec for position 1 and 360 sec for positions 2 through 7.
- g. Store the filtered estimates as follows:

$$ETDN(K) = ETD_K(2) + \left[\frac{\theta_K(1)}{G} + \left(\frac{ETD_K(1)}{2(RAD)} \right) TGM \right] W_u$$

Drift Est = Filtered Drift + (Correction for Earth Rotation)

$$AH(K, NPFA) = \theta_K(3)$$

Measured mag of G vector = Up component of filtered accel vector

where ETD_K = Filter estimated drift vector (NWU coords)

θ_K = Filter estimated accel vector (NWU)

TGM = Data collection period
 RAD = Conversion from radians to arcseconds
 G = Local gravity constant
 W_u = Up component of Earth rate
 AH = Vertical component of accel vector-high gain
 $NPFA$ = Present CAL position number
 K = IMU number (1, 2, or 3)

These two equations, simply explained, are: the drift equals the measured (filtered) drift plus a correction term for Earth rotation and the measured magnitude of the gravity vector equals the up component of the filtered acceleration vector.

- h. When all seven positions have been completed, switch the accelerometer gain setting to low and repeat the process for positions 1 to 6. The only data collected in low gain are:

$$AL(K, NPFA) = \theta_K(3)$$

where AL = Vertical component of ACCEL vector-low gain

- i. When all low gain positions have been processed, plug the data into the CAL equations to compute the new CAL parameters.

Again, the CAL process is based on an assumed error model of the IMU. As an example, consider the CAL of the X-axis accelerometer. When the X-accelerometer's input axis is pointed down (position 4), it should measure one g (local value of gravity). Similarly, when the input axis is pointed up (position 1), it should measure -1g. Errors in the accelerometer due to scale factor and bias errors will result in the accelerometer's reading less than or more than the ideal one g. The assumed error model is portrayed in Figure 6-29.

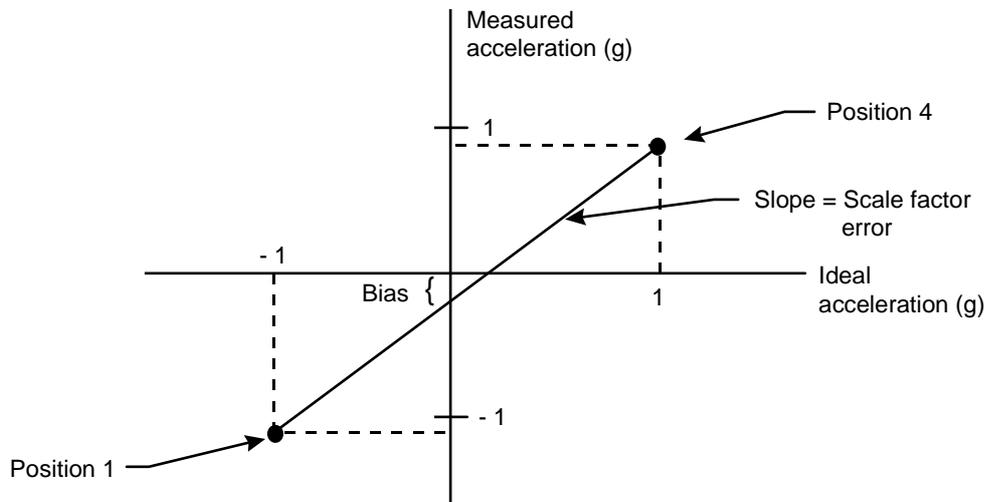


Figure 6 - 29. - Accelerometer Cal theory.

The equation used to compute the X-axis low-gain scale factor is

$$KIX = - \left(\frac{AL(K,1) + AL(K,4)}{2(G)(10^{-6})} \right)$$

which actually computes the slope of the line shown in Figure 6-39. The X-axis accelerometer bias term (low gain) is computed as:

$$KOX = - \left(\frac{AL(K,1) + AL(K,4)}{2(G_0)(10^{-6})} \right)$$

where G_0 = standard acceleration due to gravity, 32.2 ft/sec². This can be thought of as the zero offset, the difference between the ideal acceleration and the measured acceleration.

The X-axis gyro bias term equation is a little more complex, but really boils down to a fairly simple and logical equation after some cancellation.

$$DFX = ETDX(K,2) - \left[\frac{KOXH(K) (W_U)}{(10^6)} \right] \frac{G_0}{G}$$

In words, this equation is:

$$\text{Gyro bias}_x = \text{ESTIMATED WEST DRIFT} - \left[\frac{(\text{ACCEL BIAS}_x) (\text{EARTH RATE}_{up})}{(10^6)} \right] \text{GRAVITY CORRECTION TERM}$$

Substituting the accelerometer bias equation into the DFX equation yields:

$$DFX = ETDX(K,2) - \left[\frac{AH(K,1) - AH(K,4)}{2(G_0) (10^{-6})} \right] \frac{W_U}{10^6} \frac{G_0}{G}$$

After cancellation:

$$DFX = ETDX(K,2) - \left[\frac{AH(K,1) - AH(K,4)}{2G} \right] W_U$$

which reveals that computing the gyro bias term is simply a matter of estimating the drift about the proper axis (in this case, west) and correcting for the Earth's rotation (W_U).

A complete listing of all of the CAL equations can be found in the IMU Subsystem Operating Program Functional Subsystem Software Requirements document.

6.3.4.1.6 Alignments

6.3.4.1.6.1 *Preflight Alignment (GC_ALIGN & GC_ALIGN1)*

The preflight alignment is used to geographically align the IMU immediately before flight. The IMU is first aligned to the local vertical, then azimuth is aligned (gyrocompassed) to the accuracy shown in Table 6-XX. This performance can be achieved for Orbiter sways up to an 8 inch peak-peak amplitude and twists up to 4 arc minutes, both at a frequency from 0 to 0.5 Hz. The preflight alignment is performed under GPC control.

| Parameter | Error, 1s |
|------------------|-----------|
| North axis error | 20 arcsec |
| West axis error | 20 arcsec |
| Up axis error | 60 arcsec |

TABLE 6-XX, PREFLIGHT ALIGNMENT ERROR SPECIFICATIONS

The preflight alignment must be preceded by an attitude determination. The attitude determination is normally scheduled 5 minutes before the preflight alignment, but may be accomplished anytime after the preflight CAL.

During the preflight alignment, different alignment orientations may be selected by the GPC. These are defined as skew orientations. In flight, the inertial sensors are uniquely oriented at one of three skew orientations (no two IMU's have the same skew orientation). The primary reason for doing this is to enable failure isolation between two IMU's when only those IMU's are operating. A secondary reason for the skewed orientations is to assure that only one platform can go through a gimbal flip at any time. This minimizes the possibility of a platform tumble condition occurring.

The preflight alignment consists of a gyrocompass alignment (GC_ALIGN)(azimuth align) and a continuous alignment (GC_ALIGN1). The gyrocompass alignment requires 38 min to complete. The continuous alignment mode will execute from the completion of the gyrocompass alignment to T-4 minutes.

The first step of the preflight alignment is to verify the accelerometer loops are in low gain. (With OI-22, the accelerometer gain is no longer set to high gain when entering the module.)

Next, the matrix for the desired initial gyrocompassing orientation relative to NWU is computed.

$$T_{NWU}^{GC} = T_{NB}^{GCD} [T_{NB}^{NWU}]^T \begin{bmatrix} 0 & -1 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos(THT) & \sin(THT) \\ 0 & -\sin(THT) & \cos(THT) \end{bmatrix}$$

where: T_{NB}^{GCD} = Transformation matrix from the NAV base to the desired gyrocompass attitude

T_{NB}^{NWU} = Transformation matrix from the NAV base to the north, west, up coordinate frame

The third and fourth matrices in the equation represent a 90° rotation about the up axis and a rotation matrix which will allow the platform to drift (with Earth rate) an equal amount on either side of the desired orientation.

Using the above matrix, T_{NWU}^{GC} , the platform is positioned (slewed) to the first gyrocompass orientation.

The platform is monitored for 3 minutes to allow the instruments to thermally stabilize in the new positions. During this time, all inertial functions of the IMU are active but no compensation for Earth rotation is applied.

After thermal stabilization, accelerometer data are collected for 90 sec to estimate platform misalignment. These misalignments are used to compute torquing rates required to remove the misalignments. This process is actually a fine tuning of the platform positioning accomplished above. Since the accelerometers are much more accurate than the resolvers, the data collected by the accelerometers provide a good check on and correction to the platform positioning. The process works by assuming that no accelerations should be measured on the north and west axes. Any accelerations that are measured constitute alignment error.

Once the orientation is precisely established, data are collected for 10 min. The platform is torqued about the north axis to compensate for Earth rotation. The outputs from the least squares filtered accelerometer data are the drift rates about north and west, and the measured acceleration in the north and up axes. The drift about north is computed as:

$$ETDN = ETD(2) + W_u \left(\frac{\theta(1)}{G} + \frac{1}{2} ETD(1) \frac{\text{Data Take Time}}{\text{RAD}} \right) - \frac{W_N \theta(3)}{G}$$

Simply stated, the drift rate about the north axis equals the tilt rate of the west axis, plus a term correcting for the up component of Earth rate.

The second gyrocompass position is merely a 90° rotation about the up-axis from the first gyrocompass position. The matrix defining the second orientation follows.

$$T_{NWU}^{GC} = T_{NB}^{GCD} \left[T_{NB}^{NWU} \right]^T \begin{bmatrix} 0 & -1 & 0 \\ 1 & 0 & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos(\theta) & \sin(\theta) \\ 0 & -\sin(\theta) & \cos(\theta) \end{bmatrix}$$

Note that the third matrix shown in the previous matrix equation is missing. The cluster is rotated to this new position, the position is corrected as before, and data are then collected for 10 min. As in the first position, the north axis is compensated for Earth rotation. In addition to the acceleration measurements ($\theta(1)$ and $\theta(2)$) and the estimated drift measurements (ETD(1) and ETD(2)), three misalignment vectors between IMU pairs are computed along with the misalignment rate vector for each IMU pair. From the misalignment and misalignment rate data (using resolvers as the source for the data), the relative attitude errors for each IMU pair are formed. The accelerometer data are then used to compute the relative attitude error between each IMU pair. The equations used to compute the attitude error from accelerometer data are:

$$MA_{ijN} = \frac{\theta_j(2) - \theta_i(2)}{g} + \left(\frac{\text{data take period}}{2(\text{RAD})} \right) (ETD_j(2) - ETD_i(2))$$

$$MA_{ijW} = \frac{\theta_j(1) - \theta_i(1)}{g} + \left(\frac{\text{data take period}}{2(\text{RAD})} \right) (ETD_j(1) - ETD_i(1))$$

Both of these equations are seen as the average measured acceleration (with perfect accelerometers and perfect alignment, this should be zero).

The above relative misalignments, as measured by the accelerometers, are subtracted from the relative misalignments measured by the resolvers (only the north and west channels are compared). In similar fashion, the relative azimuth misalignment is computed. The difference in the azimuth channel is that a factory calibrated heading sensitivity term is subtracted from the accelerometer measured attitude error.

Now that all three axes' relative misalignments have been computed, the software converts the relative misalignments to

body coordinates and subtracts a factory-calibrated relative resolvers error vector (MijB term discussed in the calibration section) from the misalignments. From these relative misalignments, then, the software performs a reasonableness test. This test, commonly called the gyrocompass goodness test, checks the relative alignment between each IMU pair to assure a good preflight alignment. If any of the relative misalignments exceed the threshold for the test, a gyrocompass fail flag is set and the alignment must be repeated.

After the goodness test, gyrocompass continuous alignment mode (GC_ALIGN1) is entered. GC_ALIGN1 was added to replace VEL_TILT in OI-22. The basic function of this module is to maintain the IMUs in a NWU reference frame, measure the drift after a defined time frame and then torque out the measured errors. The continuous alignment is active in both OPS-9 and OPS-1.

The IMU clusters will be in the second gyrocompass position from the gyrocompass alignment. The only movement of the platforms will be the torquing out of error (no platform repositioning in GC_ALIGN1). The complete cycle of GC_ALIGN1 is 480 seconds (8 minutes). The gyrocompass drift is monitored for 470.4 seconds (7 minutes, 50.4 seconds) and the actual torquing out of measured errors is allotted 9.6 seconds.

There are torque out limits for tilt (north and west) and azimuth (up). If these limits are exceeded when GC_ALIGN1 completes a measurement cycle, a fail flag will be set indicating that the correction was limited. The correction limited flags will be reset at the end of the following GC_ALIGN1 cycle when the new misalignments are calculated. The current values for the limits are shown in table 6-XXI.

| Axis | Limit |
|--------------------|------------|
| Tilt (north, west) | 88 arcsec |
| Azimuth (up) | 200 arcsec |

Table 6-XXI, GC_ALGIN1 Correction Limits

When an OPS transition is commanded (either 9 to 1, or 1 to 0), GC_ALIGN1 will terminate and the module will be initialized and restarted after the OPS transition is completed. The errors measured during the GC_ALIGN1 cycle interrupted by the OPS transition will not be torqued out during that cycle. The first complete cycle of GC_ALIGN1 after the OPS transition will represent the errors accumulated during the interrupted cycle and the current cycle. Errors due to both cycles will then be torqued out. Given this, it is expected to see increased misalignments after an OPS transition. There is no limit on the number of OPS transitions that can be performed between OPS-9 and OPS-1. The only restriction is that GC_ALIGN1 is terminated when an OPS transition is commanded and thus no torquing out of misalignments will be performed. Therefore, it is highly

desirable to allow GC_ALIGN1 to complete a full cycle in either OPS prior to reaching the tilt or azimuth limit. After commanding an OPS transition, a procedural delay of 2 minutes should be allowed for catch-up torquing before issuing another OPS transition command.

6.3.4.1.6.2 *Inertial*

GC_ALIGN1 is terminated when the PLATFORM RELEASE command (event-11) is issued. At T-4 minutes, the PLATFORM RELEASE command is issued and the following events occur:

- IMUs are moded to INERTIAL,
- REFSMMATs are calculated,
- Navigation is initialized,
- PASS REFSMMATs and NAV data are transferred to the BFS,
- BFS Navigation is initialized,
- IMU Redundancy Management is initialized, and
- IMU data is made available to the ground.

Just as in an OPS mode transition, no torquing out of errors will be performed. Therefore the amount of time that the IMUs are drifting prior to launch is 4.5 minutes plus the time of the last interrupted cycle of GC_ALIGN1.

As a cleanup step when exiting GC_ALIGN1, IMU REFSMMATs are determined. The drift and misalignment data are used to compute the final misalignment of the platform about the north and west axes. These final misalignments are then used to compute an update to the transformation matrix, defining the relationship between the NWU coordinate frame and the cluster orientation. This matrix, in conjunction with the Earth-fixed to M50 transformation matrix and the Earth-fixed to NWU, is used to compute the REFSMMAT. The Earth-fixed to M50 transformation is computed in a navigation software module. The module is basically an ephemeris which accepts GMT as an input and returns the matrix T_{EF}^{M50}

The Earth-fixed to NWU transformation is also computed in Nav software and is a function of the launch pad latitude and longitude. The Earth-fixed coordinate frame is defined as follows:

- the X-axis lies in the equatorial plane and intercepts the Greenwich Meridian at the Equator;
- the Z-axis is orthogonal to the X-axis and points out the North Pole from the coordinate system origin at the center of the Earth;
- the cross product of Z x X forms the Y-axis, thereby completing the mutually orthogonal coordinate system.

The two matrices mentioned above, T_{EF}^{M50} and T_{EF}^{NWU} , are multiplied together to form the NWU to M50 transformation matrix.

$$T_{NWU}^{M50} = T_{EF}^{M50} [T_{EF}^{NWU}]^T$$

Given the above matrix and the matrix describing the relationship between the aligned cluster and the NWU frame, T_{NWU}^{AC} , the REFSMMAT's can be computed.

$$T_{M50}^{CL_i} = T_{NWU}^{AC_i} [T_{NWU}^{M50}]^T$$

where $i = 1, 2, 3$ - IMU number

After PLATFORM RELEASE, the IMU's go into the free inertial mode, they are flight ready, and all IMU functions, both hardware and software, remain the same throughout the flight.

It should be noted that after PLATFORM RELEASE (EVENT-11), returning to GC_ALIGN1 is prohibited. In order to realign the IMUs after PLATFORM RELEASE, a TERM/IDLE command followed by an OPS-9 transition and then an ATT_DET and gyrocompass alignment is required.

6.3.4.1.6.3 On-Orbit Alignments

On-orbit alignments are necessary to correct for platform drift caused by gyro errors. The HAINS stable member axes are capable of being fine-aligned under GPC control at any desired orientation with respect to the HAINS base reference system within an accuracy of 60 arc-seconds, one-sigma per axis. This accuracy is premised on the use of calibration constants (tag constants) to compensate for error sources in the HAINS. The stated accuracy does not include the error due to misalignment of the HAINS housing to the Navigation Base.

Three options are available to realign the IMU's. The first option uses star tracker or HUD/COAS data to compute the angular differences between the actual orientation of the platform and the ideal orientation as defined by the REFSMMAT. The platforms are pulse torqued at 25 deg/hr (0.007 deg/sec) to remove the errors (torquing angles). The second option is the IMU-to-IMU align which allows the crew to select an IMU as a reference and align either one or both of the remaining IMU's to the accuracy of the reference IMU. The third option is the Matrix align which uses star tracker or HUD/COAS data as a reference and computes a new REFSMMAT (i.e. the transformation between the actual platform location and M50) for use in determining the true Orbiter attitude. These three options are discussed in more detail below. See FSSR 13 section 4.5.4.7 for even more details.

Each IMU has three different Reference Stable Member Matrices (REFSMMATs) used during on-orbit alignments. The three REFSMMATs

are the Current, Present and Desired. See 6-XXII for more information on the REFSMMATs.

| REFSMMAT Type | Definition | Comments |
|---------------|--|---|
| Current | Where the <u>GPC thinks</u> a platform is w.r.t M50. <u>Used</u> by the GNC GPC software for attitude, velocity and other pointing computations. | |
| Present | Defines the <u>actual</u> present platform orientation w.r.t M50 as determined by an external instrument (e.g. S TRKR, HUD or COAS). | REFSMMAT will contain both Star TRKR and platform errors. |
| Desired | Defines the <u>desired/ideal</u> platform orientation w.r.t M50 for the platform to be after the align is complete. | REFSMMAT is stored in the GNC Major Function Base. |

Table 6-XXII, REFSMMAT definitions

STAR TRACKER OR HUD/COAS ALIGNMENT

The star tracker or HUD/COAS alignment (option 1) is the method normally used to realign the platforms to their ideal orientations, as defined by the Desired REFSMMAT's. To accomplish this type of alignment, line-of-sight vectors on two navigation stars must be measured by either the star trackers or HUD/COAS. The optimal separation between these two stars is 90°, but angles of 90° ± 55° are acceptable to the alignment software. The IMU On-Orbit Alignment (IMU_ORBIT_AL) module firsts uses the ideal (stored) unit line-of-sight (LOS) vectors to determine the transformation matrix from star coordinates to M50, T_{STAR}^{M50} . Then the measured LOS vectors are used to form the transformation matrix from the present cluster to star coordinates, $T_{PLATFORM}^{STAR}$. Multiplying the above two transformation matrices results in the transformation matrix from the present platform to M50 coordinates, $T_{PLATFORM}^{M50}$ (present REFSMMAT). In summary, the IMU On-Orbit Alignment module uses the two stored unit LOS vectors in M50 coordinates and the two measured unit LOS vectors in present cluster coordinates to compute the transformation matrix from the present platform to M50. The stored transformation matrix from the desired platform to M50 (desired REFSMMAT) is then compared to the present REFSMMAT to determine the errors in the IMU's alignment. These errors are displayed to the crew on SPEC 21 and downlinked to the ground as torquing angles. These angles are then torqued out of the platform by the GYPTO loops at a rate of 25 deg/hr (0.007 deg/sec). Upon completion of the IMU alignment, the current REFSMMAT is set equal to the desired REFSMMAT. See Figure 6-30 for a graphical illustration of a star alignment.

IMU TO IMU ALIGNMENT

The second type of alignment, the IMU to IMU alignment, allows one, two or three IMU's to be aligned using an IMU as the reference. Since the reference is provided by another IMU versus star data, an IMU to IMU alignment is only as accurate as the reference IMU itself.

The following six steps describe how the flight software implements an IMU/IMU alignment in OPS 2 or 3.

1. After the reference alignment IMU has been selected on SPEC 21 with an ITEM 14 + (REF IMU) EXEC, the IMU alignment software will compute the transformation matrix from present platform to desired platform in order to obtain the IMU torquing angles. This transformation is computed based on gimbal angles from the IMUs. By reading the gimbal angles of the reference IMU, the transformation from the navbase to the present orientation of the platform, $T_{NAV_BASE}^{PLATFORM_R}$, can be formed. By reading the gimbal angles of the IMU(s) to be aligned, a similar matrix (matrices), $T_{NAV_BASE}^{PLATFORM_I}$, can be formed for the IMU(s) to be aligned. Combining these matrices with the current REFSMMAT's for the reference IMU, $T_{PLATFORM_R}^{M50}$, results in the present REFSMMAT of the IMU(s) to be aligned, $T_{PLATFORM_I}^{M50}$. The torquing angles are then calculated by comparing the present REFSMMAT to the desired REFSMMAT of the IMU(s) to be aligned, $T_{PLATFORM_I}^{M50}$. See Figure 6-31 for a graphical illustration of an IMU/IMU alignment. The torquing angles will be computed for the other non-reference IMUs regardless of whether they have been selected for alignment. The torquing angle computation will be performed every minute until ITEM 16 is executed, at which time the computation will be performed again.

The remaining five steps will be performed for each IMU selected for alignment independently of the other IMUs.

2. The software then computes to see if any axis of the IMU is greater than the amount of slewing obtained for 19.84 sec. In other words, any axis that is misaligned > than 23.81 deg (19.84 sec * 72 deg/min). If this is true, then slew that axis, and any others > 23.81 deg, for 19.84 sec. This process is repeated over and over until all axes misalignment is < 19.84 sec of slewing (23.81 deg). For example, if one axis is misaligned by 180 degrees, then the above process will be performed seven times before proceeding to step 3.
3. The above process is again performed for 5.12 sec (6.144 deg), 0.64 sec (0.768 deg), and 0.32 sec (0.384 deg). The IMU will be slewed until all axes misalignment is < 0.384 deg (0.32 sec * 72 deg/min). Slew time is always one of four fixed values (19.84, 5.12, 0.63, 0.32 sec). Every time the software computes the time required to slew the IMU axis, it will do so

after computing the axis misalignment obtained from the transformation matrix from present platform to desired platform. This means that the slewing time is computed and then performed open loop by setting the slew bit for the appropriate axis and direction for the required time.

4. After slewing to within 0.384 deg is complete, the software will initiate the bi-directional slewing required to reduce the effects of hysteresis caused by the above slewing. First the IMU will be slewed positive for 19.84 seconds (23.81 deg) and then negative for another 19.84 seconds. This rocking motion should bring the IMU back close to where it was before the bi-directional slewing began. It is "close" to the original position since there is some error in the accuracy of the slewing itself (error is small).
5. At the end of the bi-directional slewing, the software computes the amount of misalignment per axis again based on the above transformation matrix computation. This misalignment is used to create a unit vector which will be used to apportion the digital torquing rate of 25 deg/hour. The platform will then be eigen axis rotated at 25 deg/hour (not 25 deg/hr per axis). This eigen axis digital torquing distribution among axes is performed every 0.32 seconds for 160 seconds (requires 500 cycles of 0.32 sec). At the end of 160 seconds the alignment is terminated regardless of whether a successful alignment was performed or not (maybe due to some hardware problem). Remember, during the 160 second period, the maximum axis misalignment which can be torqued out is 1.111 degrees (160 sec * 25 deg/hr). Every 0.32 second period the software can torque out 0.00222 degrees (0.32 sec * 25 deg/hr).
6. During the digital torque period of 160 seconds, the ground can see the torquing angles change ever so slightly around zero (assuming a good alignment). Sometimes the ground will see a + and - toggle in the value of the torquing angle. This is due to one 0.32 sec cycle driving the IMU past perfect alignment and the next cycle driving it back. At the end of the alignment, completion of the 160 second period, the IMU may be misaligned by a maximum amount of 0.00222 degrees (assumes everything is perfect). This happens when the second to last 0.32 sec cycle had the IMU aligned perfectly thereby causing the last 0.32 sec cycle to drive the axis the additional 0.00222 degrees. At this time, the alignment is complete, the torquing angles should be near zero and the IMU gyro deltas on the ground display should also be near zero unless there was some problem with the IMU hardware itself.

If the selected reference IMU is also to be aligned, the other IMU(s) will be aligned to it first, then the lowest numbered IMU already aligned will be used as the reference IMU. Upon IMU alignment completion, the RM limit will be reset to the

appropriate one or two level values (this will be discussed in more detail in a later section).

The error introduced in this process occurs when the current REFSMMAT of the reference IMU, $T_{PLATFORM_R}^{M50}$, and the transformation from the NAV Base to the platform for the reference IMU, $T_{NAV_BASE}^{PLATFORM_R}$, are multiplied together. The assumption made is that the present platform orientation of the reference IMU is equivalent to its ideal orientation. Any misalignment of the platform invalidates this assumption and results in the loss of accuracy of the IMU to IMU alignment. The IMU to IMU alignment is currently used in two situations:

1. To change the REFSMMAT's (platform orientation) of the IMU's being aligned.
2. To correct gross misalignments of the IMU's caused by powerdowns.

In the first case, a new desired REFSMMAT, $T_{PLATFORM_I}^{M50}$, is uplinked prior to the IMU to IMU alignment. When the alignment is complete, the IMU will have a new platform orientation. In the second case, when an IMU is brought back to the operate mode, its platform will be caged and the IMU will be grossly misaligned. A nominal star alignment would take too long to complete, so the IMU to IMU alignment is used.

MATRIX ALIGNMENT

The third and final type of alignment is the matrix alignment. Star data for this type of alignment are collected in the same manner as the star alignment. The difference in this type of alignment is that the IMU is not torqued to the desired REFSMMAT orientation. The transformation from platform to M50 for each IMU is computed as in the star align process (present REFSMMAT). This matrix is then stored in software as the current REFSMMAT for each IMU being aligned. Upon completion of a matrix alignment, the downlinked REFSMMAT is updated and represents the new IMU platform position (current equals present). Since the torquing angles are determined from the desired REFSMMATS, the ground has to uplink the new current REFSMMATS (to the desired slot) in order for the torquing angle to be valid. Again, the platforms are not moved in this type of alignment. This type of alignment is usually used during IMU reference recovery procedures. Figure 6-32 for a graphical illustration of a matrix alignment.

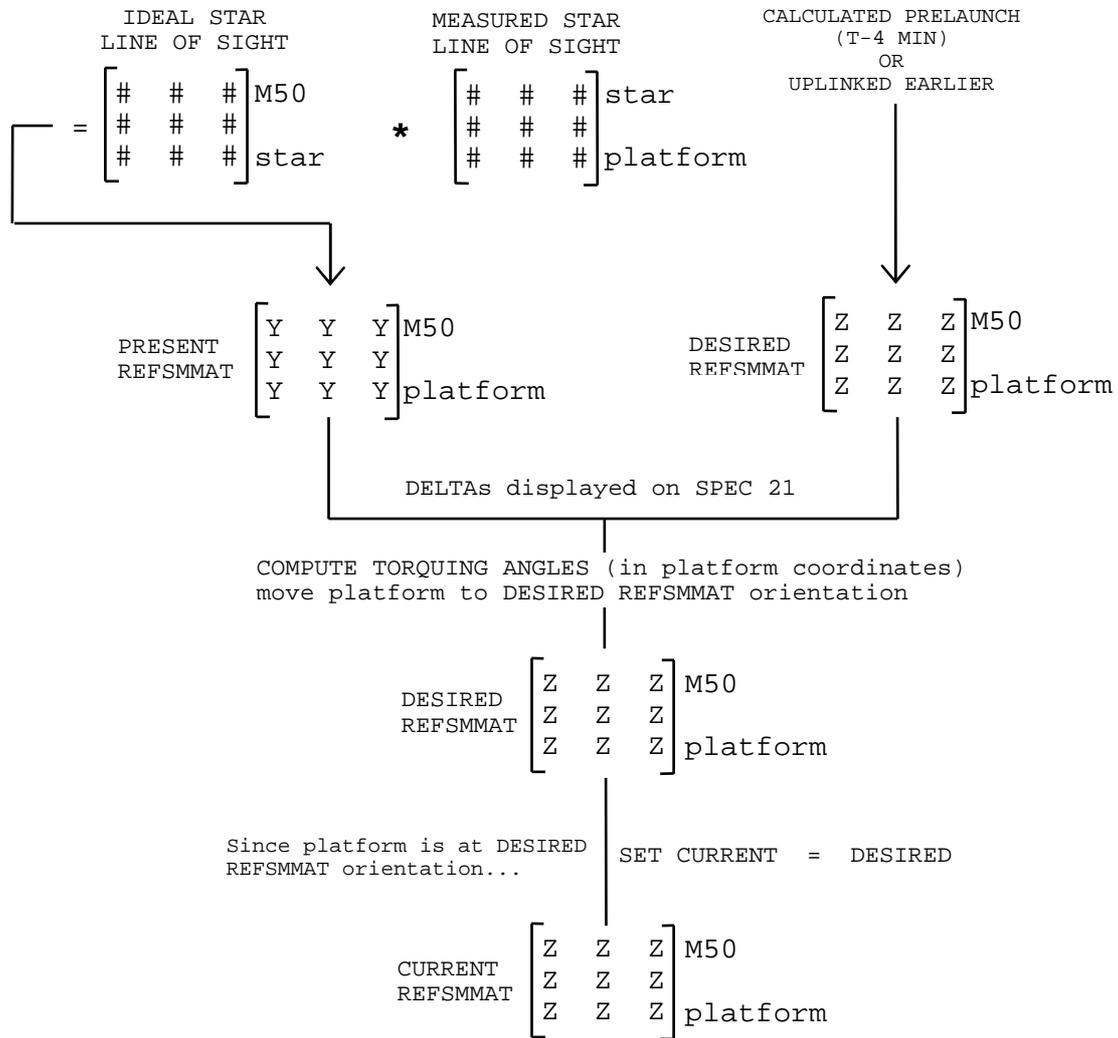


Figure 6-30, Schematic of a Star Align

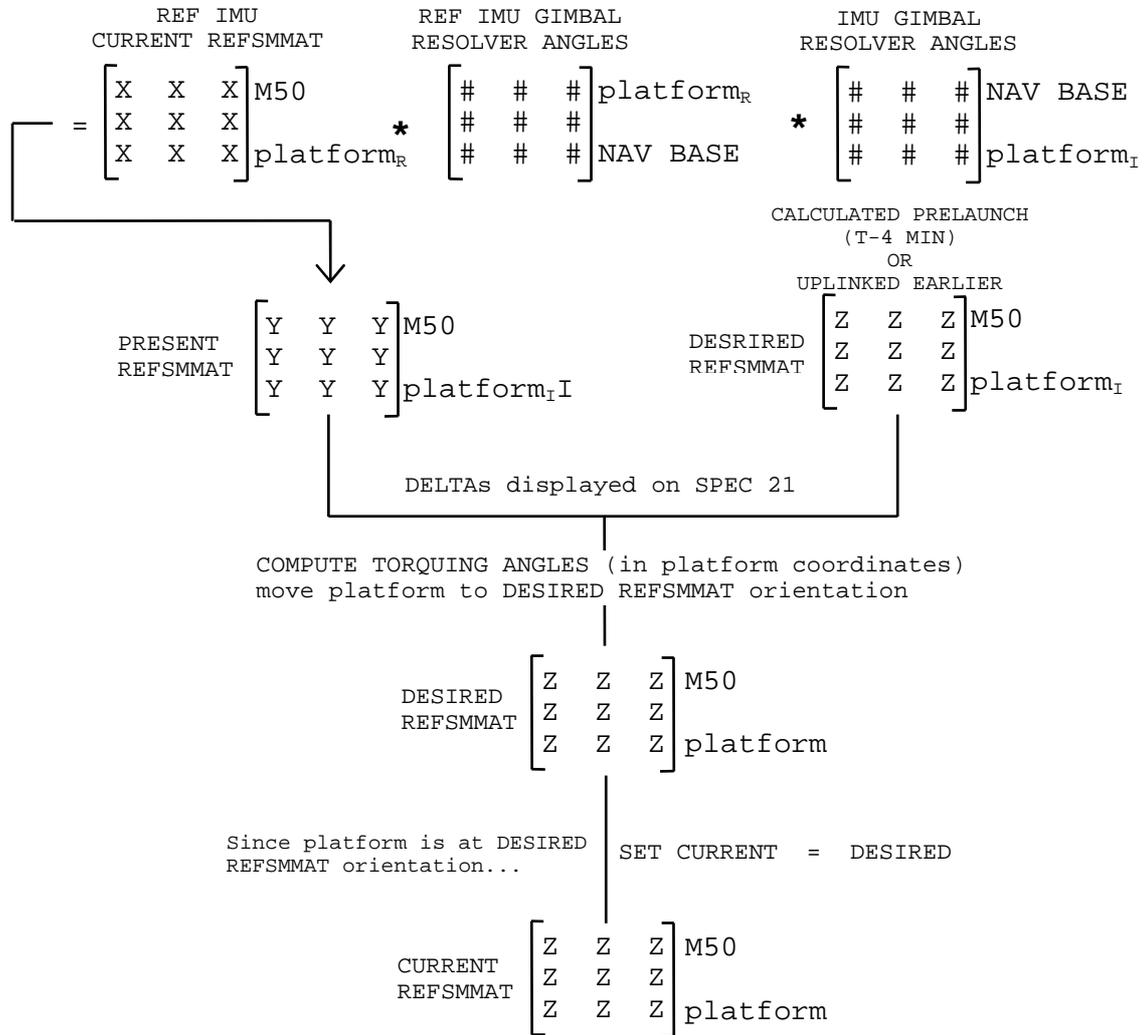


Figure 6-31, Schematic of an IMU-to-IMU Align

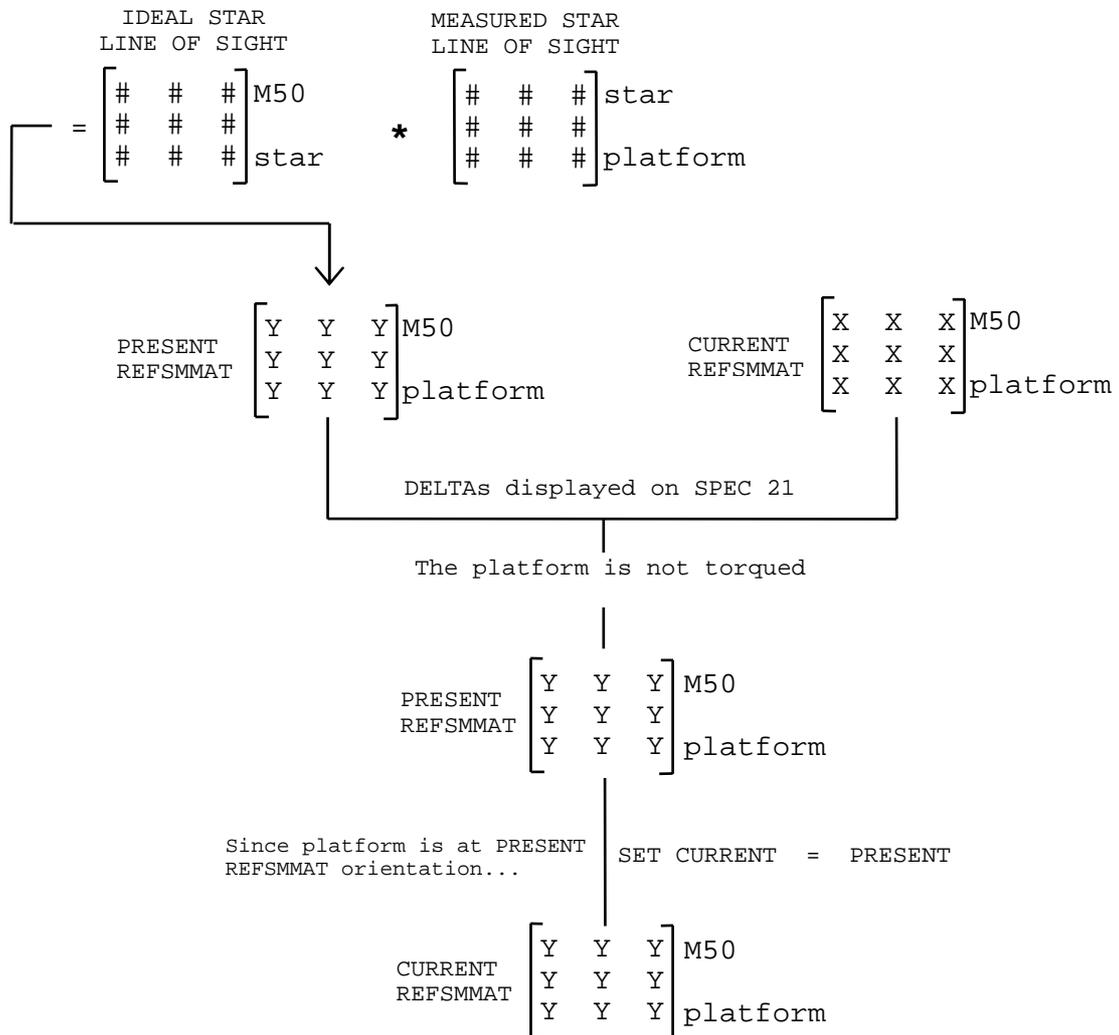


Figure 6-32, Schematic of a Matrix Align

6.4 HAINS INPUT/OUTPUT (I/O) CONTROL

The I/O section of the HAINS provides the basic communication link between the HAINS and the Orbiter General Purpose Computer (GPC) via the MDM Unit.

6.4.1 Description

The I/O controls the flow of input and output serial data to the GPC under the control of discrettes issued by the GPC. One serial data line is used to alternately receive and transmit data at a maximum rate of 25 times per second (every 40 msec). The control and format of the input and output data is separately described below.

6.4.1.1 Input Data

The format of the two input data words that may be sent to the HAINS are indicated in Table 6-XXIII and Table 6-XXIV. The uses of these data words are summarized below:

Input Word 1 This word enables independent Gyro Pulse Torquing (GYPTO) Commands to be sent to the X, Y, and Z gyros. Five bits are allotted for each torquing command using two's complement logic. The most significant bit determines the sense of the command and the remaining four bits the command magnitude.

The least significant magnitude bit has a nominal weight of 0.5 arc-second. After the HAINS receives input Word No. 1, it automatically torques out the commanded angle before the next Input Word Command is received. At the maximum computer iteration rate of 25 Hz, the maximum number of data bits that can be torqued out per iteration is 8. This corresponds to a nominal angle of $8 \times 0.5 = 4$ arc-seconds/40 milliseconds = 100 deg/hr.

Input Word 2 Bit 4 of this word is used to simultaneously set all velocity channels (X, Y, Z) at either Low Gain or High Gain. The Low Gain setting is (used during any mission flight mode (e.g., launch, entry). The High Gain setting is only used when calibrating the HAINS or during HAINS launch pad alignments, both of which are ground operations.

Bits 5 to 10 are used to independently issue slew commands to the HAINS X, Y, and Z gyro axes when it is desired to rapidly rotate the HAINS gimbals to a desired orientation. Once a pertinent command is received, the HAINS will continue to

respond to the last command. Hence, Input Word No. 2 need be issued only when it is desired to change the state of the Velocity gain or Slew Commands.

6.4.1.2 Input Data Control

When the GPC desires to input data to the HAINS, it first sends a Message Out Discrete in its high state to inform the HAINS that it should prepare to receive data from the GPC. Next, the GPC sends a Word Discrete in its high state which the HAINS interprets as a command to receive Input Word No. 1. Next the GPC sends Input Serial Data Word No. 1.

The serial data is sent at a bit rate of 1 MHz controlled by a clock external to the HAINS. After all 20 data bits in Input Data Word No. 1 have been sent to the HAINS, the Word Discrete is set to its low state while the Message Out Discrete is still high. Approximately 8 micro-seconds later, the Word Discrete is again set high to inform the I/O that the GPC is about to send Input Data Word No. 2. The Word and Message Out Discretes are set low in that order and the input transmission cycle to the HAINS is complete.

If only Input Word No. 1 is to be sent to the HAINS, the GPC would issue only one Word Discrete after setting the Message Out Discrete high. When the Message Out Discrete is next set low, the input transmission is complete and only Input Word No. 1 would have been received by the HAINS. Later, when the next Message Out Discrete followed by a Word Discrete is received by the HAINS, the HAINS always interprets the first data word to the IMU as Input Data Word No. 1.

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|---|
| 1-3 | Sync Pulse |
| 4 | Sign of X stable-member commanded rotation. |
| 5-8 | Magnitude - 4-bit code that determines magnitude of the stable-member rotation angle about the X-axis is given in bits 5 through 8. Bit coding shall be: Bit 4: Setting to logic "1" commands positive rotation about the X-axis. Setting at logic "0" commands negative rotation about the X-axis. Bit 5: Most significant magnitude bit. Bit 8: Least significant magnitude bit with nominal weight of 0.5 arc-seconds at logic "1" and 0 at logic "0". |
| 9 | Sign of Y stable-member commanded rotation. |
| 10-13 | Magnitude - 4-bit code that determines magnitude of the stable-member rotation angle about the Y-axis is given in bits 10 through 13. Bit coding shall be: Bit 9: Setting to logic "1" commands positive rotation about the Y-axis. Setting at logic "0" commands negative rotation about the Y-axis. Bit 10: Most significant magnitude bit. Bit 13: Least significant magnitude bit with nominal weight of 0.5 arc-seconds at logic "1" and 0 at "0". |
| 14 | Sign of Z stable-member command rotation. |

TABLE 6-XXIII, INPUT DATA WORD 1 (2's COMPLIMENT)

15-18 Magnitude - 4-bit code that determines magnitude of the stable-member rotation angle about the Z-axis is given in bits 15 through 18.

Bit coding shall be:

Bit 14: Setting to logic "1" commands positive rotation about the Z-axis. Setting to logic "0" commands negative rotation about the Z-axis.

Bit 15: Most significant magnitude bit.

Bit 18: Least significant magnitude bit with nominal weight of 0.5 arc-seconds at logic "1" and 0 at logic "0".

NOTE: Stable-member rotations are with respect to inertial space in accordance with the righthand rule.

19 Spare

20 Parity

TABLE 6-XXIII, INPUT DATA WORD 1 (2's COMPLIMENT) (continued)

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|---|
| 1-3 | Sync Pulse |
| 4 | Scale Factor (SF) - Setting this bit shall cause the HAINS to switch the accelerometer channel to high-gain scale factor (9350 pps/g) for its logic "1" setting and to its low-gain scale factor (935 pps/g) for its logic "0" setting. |
| 5 | Slew X positive (SX+) - Setting this bit to logic "1" shall cause the stable member to rotate about the X-axis in the negative direction at a 56 degree per minute rate. |
| 6 | Slew X negative (SX-) - Setting this bit to logic "1" shall cause the stable member to rotate about the X-axis in the positive direction at a 56 degree per minute rate. |
| 7 | Slew Y positive (SY+) - Setting this bit to logic "1" shall cause the stable member to rotate about the Y-axis in the negative direction at a 56 degree per minute rate. |
| 8 | Slew Y negative (SY-) - Setting this bit to logic "1" shall cause the stable member to rotate about the Y-axis in the positive direction at a 56 degree per minute rate. |
| 9 | Slew Z positive (SZ+) - Setting this bit to logic "1" shall cause the stable member to rotate about the Z-axis in the negative direction at a 56 degree per minute rate. |
| 10 | Slew Z negative (SZ-) - Setting this bit to logic "1" shall cause the stable member to rotate about the Z-axis in the positive direction at a 56 degree per minute rate. |
| 11-19 | Spares |
| 20 | Parity |

NOTE: Stable members rotate about their own axes with respect to inertial space, positive or negative according to the right-hand rule for orthogonal right-handed coordinate systems. With all gimbal angles starting at zero, a climbing right-hand turn by the vehicle shall cause Roll, Pitch, and Aximuth gimbal angles to increase in a positive sense. The same result shall follow simultaneous negative rotations of the stable member when the vehicle is not rotating.

TABLE 6-XXIV, INPUT DATA WORD 2

6.4.1.3 Output Data

The Output Word list is shown in Table 6-XXV. The output words are sequentially issued in the order indicated in this list (i.e., Word 1 is sent first and Word No. 16 last) . Currently the Orbiter's operational software does not look at Output Words 15 and 16.

6.4.1.4 Output Data Control

These output words are transmitted when the GPC issues the Message IN Discrete set high. Otherwise, the basic GPC control operation is identical to that explained for input words except that the Message IN rather than the Message OUT Discrete is used to command Output Words.

The Output Word list may be terminated at any time, by setting the Message IN Discrete low. If this is done, the next time that an output word list is requested, the Discrete may again be set high and the HAINS Output transmissions will again start with Output Word No. 1.

The format of the sixteen Output Data Words that are sent from the HAINS are shown in Tables 6-XXVI through 6-XXXI. The use of these Data Words is summarized below:

Output Word 1 This word is the HAINS MODE/BITE status word. Each data bit provides an indication of the status and proper operation of the HAINS. Bit 4 (HAINS Good) is a logical NOR of the states of bits 5 through 12. If any of the latter bits are set at logic "1" because a fail condition has been detected, then Bit 4 is simultaneously set at logic "0".

Bits 13 or 14 are set to a logic "1" if a failure of parity, gap, or Manchester code error occurs in transmissions of Input Words No. 1 or 2, respectively.

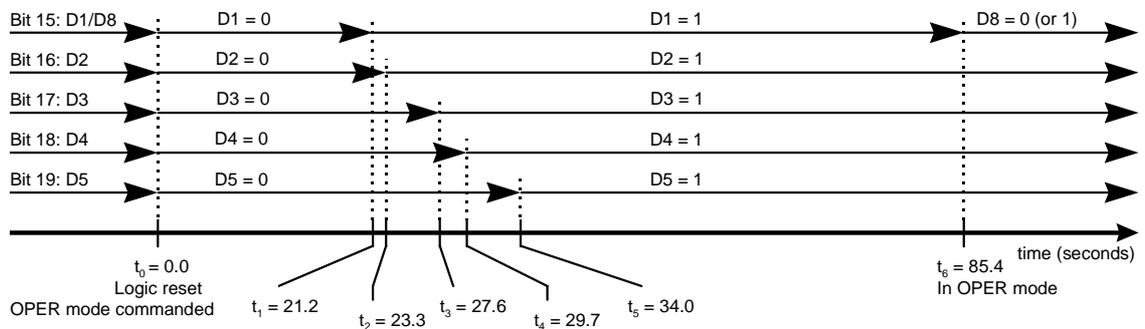
During the initial start-up of the HAINS, Bits 15 to 19 are set at logic "1" and sequence to logic "0" in the order from Bit 15 first to Bit 19 last in accordance with the sequence timing given in Figure 6-33.

Output Word 2 This word contains the value of cluster rotational rate sensed along the redundant gyro axis

- Output Words 3-9 These words give the values of the fine (8X) and coarse (1X) gimbal angle outputs. The angular value for each 8X word is given in electrical degrees and is divided by 8 to transform it to a 1X angle. The 1X and 8X words are combined in the GPC to produce one precision gimbal angle output value for each channel. The basic accuracy is provided by the 8X output and only the 3 most significant bits of the 1X words are used to determine which of the 8 octants that the total angle is in. No 1X output word is provided for Output Word 3 since the Inner Roll gimbal angle is restricted to less than a +/- 11 degree 1X range.
- Output Words 10-12 These words provide the magnitude of velocity along the X, Y, and Z stable member axes. The sense of the velocities is determined in the GPC by continuous subtraction of the value of the previous velocity word transmission from the value of the current. In addition, the GPC keeps track of over-flows that will occur if the velocity magnitude exceed the range of the word (approximately +66,812 cm/sec).
- Output Word 13 This word is identical to the last Input Word No. 1 that was received by the HAINS. The GPC uses this (information to confirm proper HAINS reception of Input Word No. 1 from the GPC (via the MDM).
- Output Word 14 This word is identical to the last Input Word No. 2 that was received by the HAINS. The GPC uses this information to confirm proper reception of Input Word No. 2 from the GPC (via the MDM).
- Output Word 15 This word is one of two BITE summary words maintained by the internal software in order to provide additional fault isolation information. This word is used for test purposes and is not used during normal Orbiter operations. This word is transmitted from the HAINS to the test computer for failure diagnostic use. Each bit of the BITE summary word provides a pass/fail status for a specific BITE test(s). A bit which is set to a logic "1" indicates a fail condition. Output Word No. 15 is a spare word in the KT-70 IMU.

Output Word 16

This word is one of two BITE summary words maintained by the internal software in order to provide additional fault isolation information. This word is used for test purposes and is not used during normal Orbiter operations. This word is transmitted from the HAINS to the test computer for failure diagnostic use. Each bit of the BITE summary word provides a pass/fail status for a specific BITE test(s). A bit which is set to a logic "1" indicates a fail condition. Output Word No. 16 is a spare word in the KT-70 IMU.



Notes:

Bit 15

D1/D8 sequencer discrete - This bit, which initially represents the D1 sequence-up discrete, shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 21.2 seconds after logic reset in order to start the vertical gyro wheel rotating at 240Hz. After the In-Operate-Mode discrete is issued by the HAINS, this bit is designated D8. The D8 bit then represents the velocity channel gain. At logic "0", the channel is set at low gain; at logic "1", it is set at high gain.

Bit 16

D2 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 23.3 seconds after logic reset in order to close the low-gain isolation loops around the vertical gyro axis and to activate the 8-Vrms, 19.2-kHz excitation BITE.

Bit 17

D3 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0", 27.6 seconds after logic reset in order to close the high-gain isolation loops around the vertical gyro axes, switch the vertical gyro wheel excitation to approximately 530Hz, and start the azimuth gyro wheel at 240Hz.

Bit 18

D4 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 29.7 seconds after logic reset in order to close the low-gain isolation loops around the azimuth gyro axis and switch the azimuth gyro wheel excitation to approximately 530Hz.

Bit 19

D5 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 34.0 seconds after logic reset in order to close the high-gain isolation loops around the azimuth gyro, close the redundant gyro capture loop, and activate the gimbal loop BITE.

Figure 6-33, Start Up Output Word 1 Timing Sequence

| <u>Word No.</u> | <u>Information</u> |
|-----------------|--|
| 1 | HAINS mode/BITE status word |
| 2 | W_R redundant gyro axis sensed rate |
| 3 | Θ_{RL} redundant gimbal angle from 8X inner roll |
| 4 | Θ_{XM} X gimbal angle from 1X outer roll resolver |
| 5 | Θ_{XL} X gimbal angle from 8X outer roll resolver |
| 6 | Θ_{YM} Y gimbal angle from 1X pitch resolver |
| 7 | Θ_{YL} Y gimbal angle from 8X pitch resolver |
| 8 | Θ_{ZM} Z gimbal angle from 1X azimuth resolver |
| 9 | Θ_{ZL} Z gimbal angle from 8X azimuth resolver |
| 10 | VX X velocity word |
| 11 | VY Y velocity word |
| 12 | VZ Z velocity word |
| 13 | Echo of command word 1 |
| 14 | Echo of command word 2 |
| 15 | BITE Word 1* |
| 16 | BITE Word 2* |

* Not presently read by GPC software.

TABLE 6-XXV, OUTPUT DATA WORD LIST

HAINS MODE/BITE STATUS

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|--|
| 1-3 | Sync pulses |
| 4 | HAINS Good - This bit shall be set to logic "1" if none of the following conditions occurs: A/D converter failure Platform failure DC/DC failure Discrete power supply failure Digital subsystem failure Circuit card overtemperature MUX failure Resolver failure |
| 5 | <u>MUX fail</u> - This bit shall be set to a logic "1" when either a failure is detected in the end-around transmission test performed on output words, or when the MUX BITE detects a fail condition. |
| 6 | <u>A/D fail</u> - This bit shall be set to logic "1" when the A/D converter BITE detects a fail condition. |
| 7 | <u>Platform fail</u> - This bit shall be set to logic "1" when either the platform BITE or the gyro wheel supply BITE detects a failure. |
| 8 | <u>Digital subsystem fail</u> - This bit shall be set to logic "1" when the BITE detects a failure on any of the following circuit cards: CPU, I/O, or Memory. |
| 9 | <u>DC/DC card fail</u> - This bit shall be set to logic "1" when <input type="checkbox"/> O <input type="checkbox"/> -the DC/DC BITE detects a fail condition. |
| 10 | <u>Discrete power supply fail</u> - This bit shall be set to logic "1" when a BITE fail detection is made of the isolated power supply used to power the IMU discretetes. |
| 11 | <u>Resolver fail</u> - This bit shall be set to logic "1" when either the resolver voltage or current exceeds its allowable tolerance. |
| 12 | <u>Circuit Card Overtemperature</u> - This bit shall be set to logic "1" when a circuit card overtemperature condition is detected. |

TABLE 6-XXVI, OUTPUT DATA WORD 1

- 13 Transmission word 1 fail - This bit shall be set to a logic "1" if a failure of parity, gap, or Manchester coding occurs in input word No. 1
- 14 Transmission word 2 fail - This bit shall be set to a logic "1" if a failure of parity, gap, or Manchester coding occurs in input word No. 2.
- 15 D1/D8 sequencer discrete - This bit, which initially represents the D1 sequence-up discrete, shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 18.6 seconds after logic reset in order to start the vertical gyro wheel rotating at 240Hz. After the In-Operate-Mode discrete is issued by the HAINS, this bit is designated D8. The D8 bit then represents the velocity channel gain. At logic "0", the channel is set at low gain; at logic "1", it is set at high gain.
- 16 D2 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 23.3 seconds after logic reset in order to close the low-gain isolation loops around the vertical gyro axis and to activate the 8-VRMS, 19.2-kHz excitation BITE.
- 17 D3 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 27.6 seconds after logic reset in order to close the high-gain isolation loops around the vertical gyro axes, switch the vertical gyro wheel excitation to approximately 530Hz, and start the Azimuth gyro wheel at 240Hz.
- 18 D4 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 29.7 seconds after logic reset in order to close the low-gain isolation loops around the azimuth gyro axis and switch the azimuth gyro wheel excitation to approximately 530Hz.
- 19 D5 sequencer discrete - This bit shall be set to a logic "1" at logic reset. This bit is set to a logic "0" 34.0 seconds after logic reset in order to close tee high-gain isolation loops around the azimuth gyro, close the redundant gyro capture loop, and activate the gimbal loop BITE.
- 20 Parity

TABLE 6-XXVI, OUTPUT DATA WORD 1 (continued)

W_R REDUNDANT AXIS RATE

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|--|
| 1-3 | Sync pulses |
| 4 | Sign bit. Positive rotation (right-hand rule) of the stable member about its X or Y reference axes causes the sign bit to be set to a logic "1". |
| 5 | Overflow |
| 6 | 70.656 deg/hr |
| 7 | 35.328 deg/hr |
| 8 | 17.664 deg/hr |
| 9 | 8.832 deg/hr |
| 10 | 4.416 deg/hr |
| 11 | 2.208 deg/hr |
| 12 | 1.104 deg/hr |
| 13 | 0.552 deg/hr |
| 14 | 0.276 deg/hr |
| 15 | 0.138 deg/hr |
| 16 | 0.069 deg/hr |
| 17 | 0.0345 deg/hr |
| 18-19 | Spares |
| 20 | Parity |

TABLE 6-XXVII, OUTPUT DATA WORD 2

RESOLVER ANGLES

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|--------------------------------|
| 1-3 | Sync pulses |
| 4 | 180 deg |
| 5 | 90 deg |
| 6 | 45 deg |
| 7 | 22.5 deg |
| 8 | 11.25 deg |
| 9 | 5.625 deg |
| 10 | 2.8125 deg |
| 11 | 1.40625 deg |
| 12 | 0.703125 deg |
| 13 | 0.3515625 deg |
| 14 | 0.17578125 deg |
| 15 | 0.087890625 deg |
| 16 | 0.0439453125 deg |
| 17 | 0.02197265625 deg |
| 18-19 | Spares |
| 20 | Parity |

NOTE: 8X resolver angles are divided by 8 in orbiter GPC.

TABLE 6-XXVIII, OUTPUT DATA WORDS 3-9

VELOCITY WORDS

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|---------------------------------------|
| 1-3 | Sync pulses |
| 4 | 34406.4 cm/sec (low gain) |
| 5 | 17203.2 cm/sec (low gain) |
| 6 | 8601.6 cm/sec (low gain) |
| 7 | 4300.8 cm/sec (low gain) |
| 8 | 2150.4 cm/sec (low gain) |
| 9 | 1075.2 cm/sec (low gain) |
| 10 | 537.6 cm/sec (low gain) |
| 11 | 268.8 cm/sec (low gain) |
| 12 | 134.4 cm/sec (low gain) |
| 13 | 67.2 cm/sec (low gain) |
| 14 | 33.6 cm/sec (low gain) |
| 15 | 16.8 cm/sec (low gain) |
| 16 | 8.4 cm/sec (low gain) |
| 17 | 4.2 cm/sec (low gain) |
| 18 | 2.1 cm/sec (low gain) |
| 19 | 1.05 cm/sec (low gain) = 0.034 ft/sec |
| 20 | Parity |

NOTE: At the high gain velocity scale factor setting, the nominal bit weights are one tenth of the indicated values.

TABLE 6-XXIX, OUTPUT DATA WORDS 10-12

BITE SUMMARY WORD 1

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|------------------------------------|
| 1-3 | Sync pulses |
| 4 | Multiple bit memory error |
| 5 | DAL regulated voltage fail |
| 6 | Arithmetic floating point overflow |
| 7 | Gyro wheel supply |
| 8 | Pulse accumulator fail |
| 9 | DMA fail |
| 10 | Sequencer fail |
| 11 | EEPROM/PROM checksum fail |
| 12 | Circuit card overtemperature |
| 13 | DC/DC fail |
| 14 | Discrete power supply fail |
| 15 | 7 Vac resolver fail |
| 16 | Instruction test fail |
| 17 | RAM self-test fail |
| 18 | Pulse accumulator overflow |
| 19 | Floating point divide-by-zero |
| 20 | Parity |

NOTE: This data word is for IMU test function use only. It is not read by the orbiter GPC.

TABLE 6-XXX, OUTPUT DATA WORD 15

BITE SUMMARY WORD 2

| <u>Bit No.</u> | <u>Bit Assignment/Function</u> |
|----------------|---|
| 1-3 | Sync pulses |
| 4 | Gimbal angle checksum fail |
| 5 | A/D converter interrupt goodness check fail |
| 6 | DOR wraparound fail |
| 7 | A/D test fail |
| 8 | Inner roll reasonableness |
| 9 | MTU fail |
| 10 | MUX tag timer fail |
| 11 | Gimbal loop fail |
| 12 | Missing resolver input/1X, 8X resolver fail |
| 13 | Memory error correction fail |
| 14 | MTU self-test fail |
| 15 | A/D AC channel test fail |
| 16 | A/D DC channel test fail |
| 17 | End-around transmission fail |
| 18 | MUX/DMA parallel fail |
| 19 | Square root of a negative number |
| 20 | Parity |

NOTE: This data word is for IMU test function use only. It is not read by the Orbiter GPC.

TABLE 6-XXXI, OUTPUT DATA WORD 16

6.5 HAINS SOFTWARE

This section describes the Operational Flight Program (OFP) software provided by the processor which consists of the CPU, Memory and I/O circuit cards.

Internal Operational Flight Program (OFP) software implemented in the Memory and Computer Processor Unit (CPU) circuit cards is designed to meet the following objectives:

- Provide transparency between the HAINS and the KT-70 IMU by appropriately scaling and formatting the output data.
- Provide internal compensation for as many I-Load parameter as possible.
- Perform extrapolation and interpolation of data as required in order to meet the data staleness specifications.
- Augment the present compensation I-Load parameters for improved accuracy.
- Provide expanded Built In Test Equipment (BITE) capability in order to detect and annunciate HAINS failures.

6.5.1 Overview

The OFP design uses a modular technique, following the rules for top-down modular design. The majority of the modules have been written in the FORTRAN programming language, the remaining modules, because of time critical application, were written in Assembly language.

The use of the High Order Language and module design allows for easy maintenance and comprehension, and also facilitates future modification.

The OFP code resides in a 24K Word (16 bits) Programmable Read Only Memory (PROM) on the HAINS Memory Circuit Card. Table 6-XXXII is the system memory map. The code is executed by the HAINS Central Processor Unit (CPU) Circuit Card.

The OFP's interaction with the other various HAINS functions is provided by the I/O. The program is contained in less than half of the 48K memory locations (16 bits) available. Only the lower 24K of the available 48K available memory is populated with fusible link PROM's. The execution of the code results in approximately a 70% CPU utilization factor.

The OFP is the means by which acceleration data and attitude data is accumulated, processed, and compensated for use by the Space Shuttle General Purpose Computer (GPC). The OFP also receives the BITE status information from the various circuit cards and provides additional BITE checks of HAINS functions.

The OFP can be classified as six separate processing functions. Each of these functions in turn is made up of one or more modules. The six processing functions and subfunctions are shown in Figure 6-34. The remainder of this section will describe the OFP in terms of these processing functions.

| FUNCTION | MEMORY TYPE | ADDRESS RANGE | |
|-------------------|-------------|---------------|----------------|
| | | HEX | DECIMAL |
| MAIN SCRATCHPAD | RAM | 0000 TO 003F | 0 TO 63 |
| DMA REGION | RAM | 0040 TO 0067 | 64 TO 103 |
| MAIN SCRATCHPAD | RAM | 0068 TO 0FFF | 104 TO 4095 |
| OFP | ROM | 1000 TO 6FFD | 4096 TO 28669 |
| OFP CHECKSUM | ROM | 6FFE TO 6FFF | 28670 TO 28671 |
| GROWTH * | ROM | 7000 TO CFFF | 28672 TO 53247 |
| MUX SCRATCHPAD | RAM | D000 TO D088 | 53248 TO 53384 |
| MAIN SCRATCHPAD | RAM | D089 TO DEFF | 53385 TO 57343 |
| I-LOAD PARAMETERS | EEPROM | E000 TO EFFF | 57344 TO 61439 |
| BITE STORAGE | EEPROM | F000 TO FF00 | 61440 TO 65280 |
| MEMORY MAPPED I/O | - | FFC0 TO FFFF | 65472 TO 65535 |

NOTE: * Upper ROM address field is not populated by fusible link PROM's.

TABLE 6-XXXII, SYSTEM MEMORY MAP

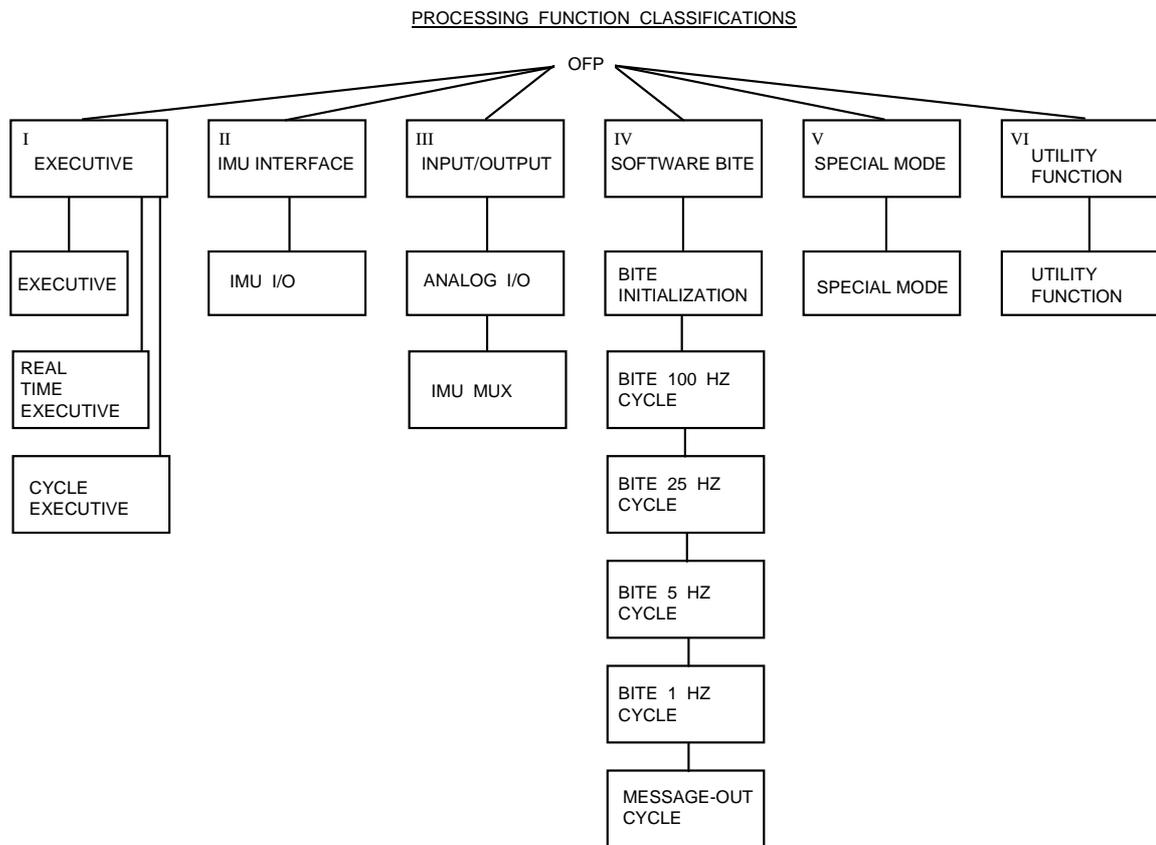


Figure 6 - 34. - OFF Processing Functions - major classifications.

6.5.2 Executive

The Executive Function contains three major functions. These functions are as follows:

- o Executive
- o Real-Time Executive
- o Cycle Executive

6.5.2.1 Executive

The Executive function performs program initialization, sets up calling sequences for different modes (Operational or Special Mode), and buffers data computed in one cycle for use in other cycles.

The program initialization portion consists of setting variables, downloading of EEPROM values and the execution of turn-on BITE. Each of the interrupt cycles has a buffering routine. Data which is needed during the interrupt cycle is unbuffered following initiation of the cycle. Data which is required by other

interrupt cycles is buffered prior to completion of the interrupt cycle.

6.5.2.2 Real-Time Executive

The Real-Time Executive performs the interrupt initialization, scheduling, error detection, and error recovery.

The Real-Time Executive (RTE) performs the initialization of the Interrupt Controller on the I/O Circuit Card Assembly, which handles the Level II interrupts. The RTE also sets up the address for interrupt vectoring (Interrupt vector is HAINS mode dependent) immediately following HAINS turn-on.

During OFP cycling, the RTE determines and executes the appropriate Level II service routine. The OFP Interrupt Priority Structure is shown in Table 6-XXXIII. Also during interrupt handling the RTE insures that the interrupt received is valid and that the test to be executed as a result of the interrupt is not already running. If either occurrence is detected, the RTE initiates the appropriate error recovery routine.

| LEVEL | PRIORITY | DESCRIPTION | ISSUING DEVICE | CYCLE TIME |
|-------|----------|------------------------------------|----------------|---------------------|
| 1 * | | MESSAGE - IN DISCRETE | GPC | UPON REQUEST BY GPC |
| 2 | 1 | A/D COMPLETE | A/D | 5 MSEC |
| 2 | 2 | 200 HZ INTERRUPT | GWS | 5 MSEC |
| 2 | 3 | 100 HZ INTERRUPT | CPU | 10 MSEC |
| 2 | 4 | MESSAGE - OUT DISCRETE | GPC | UPON REQUEST BY GPC |
| 2 | 5 | 25 HZ SOFTWARE INTERRUPT (DOR 1-9) | CPU | 40 MSEC |
| 2 | 6 | 5 HZ SOFTWARE INTERRUPT (DOR 1-10) | CPU | 200 MSEC |
| 2 | 7 | 1 HZ SOFTWARE INTERRUPT (DOR 1-11) | CPU | 1000 MSEC |
| 2 | 8 | SPARE | | |

NOTE: * Level 1 has a higher priority than any level 2.

TABLE 6-XXXIII, INTERRUPT PRIORITY STRUCTURE

6.5.2.3 Cycle Executive

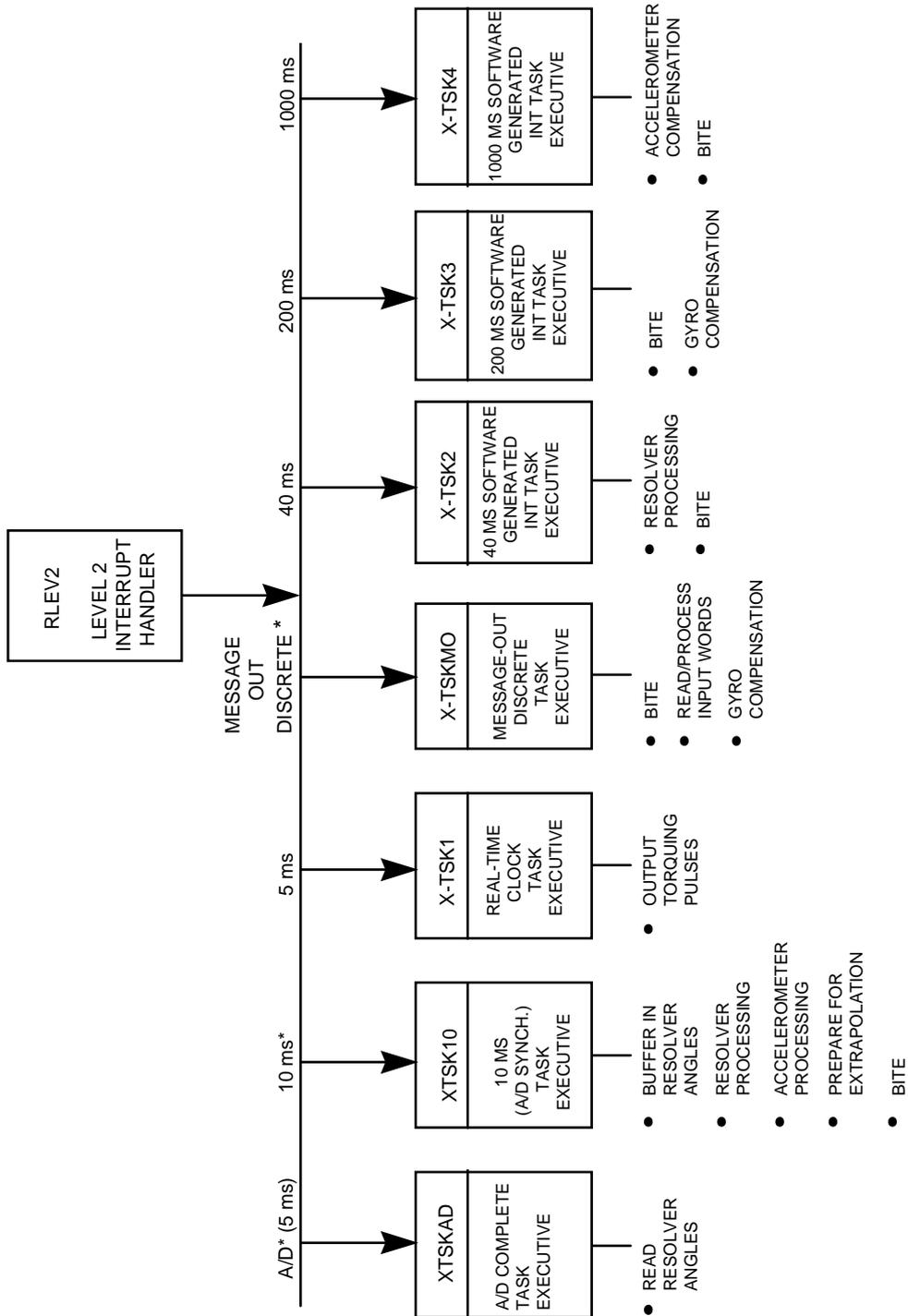
The RTE initiates the Cycle Executive function. The tasks required for an interrupt cycle are performed within this function. Figure 6-35 shows a simplified Hierarchical Chart for the flow from the Level II Interrupt Handler (RLEV2) to the interrupt's task schedule.

Upon completion of the execution of the tasks within an interrupt cycle control is then returned to the RTE.

Following completion of a interrupt cycle the OFP performs background routines (Non-Interrupt initiated). In background the OFP systematically reads data and writes the data back to each RAM location.

This reading and writing to RAM (called RAM scrubbing) will eliminate any single bit errors that may accumulate. A read operation of a RAM location which contains a single bit error will cause the Error Detect and Correct circuitry on the Memory Circuit Card Assembly to correct the error while the data is on the data bus (the RAM location is not altered by EDC) . The subsequent write to the RAM location will result in the storage of the correct data.

Also in background the OFP enables the Level II interrupts periodically. The periodic enable is to ensure that a Single Event Upset (SEU) hasn't caused the Level II Interrupt Controller to be disabled.



* ASYNCHRONOUS TO REAL-TIME CLOCK

Figure 6 - 35. - Interrupt Handler Hierarchical Chart.

6.5.3 HAINS Interface Function

This function provides for the data transfer from the OFP to the Orbiter's General Purpose Computer (GPC) and also provides for the control of the HAINS hardware. The HAINS Interface function consists of the IMU Input/Output.

6.5.3.1 Input/Output

6.5.3.1.1 Level I Interrupt Handler (RLEV1)

The Level I Interrupt Handler (RLEV1) is invoked as a result of the receipt of a GPC issued Message-In discrete (Level 1 Interrupt). It performs the preparation and transmission of the output data required by the GPC.

The RLEV1 performs the following:

- o Output the redundant rate word.
- o Output the resolver angles. The 8X resolver angles (except Inner Roll) must be extrapolated.
- o Output the velocity words. These must be interpolated.
- o Echo back to the GPC command words 1 and 2.

6.5.3.1.2 Gyro Torquing (GYPTO)

The Gyro Pulse Torquing Module (GYPTO), running at 200 Hz, performs both fine digital torquing (+/- 90 arc-sec/sec) for drift compensation and analog slewing (+/- 4185 arc-sec/sec) for logic tilt corrections.

Functional Requirements - The GYPTO module is required to perform the following functions for the X, Y, and Z axes:

- o If an analog slew is required on the X, Y, or Z axes by the GPC command, the gyro is slewed at 4185 +/- 689 arc-sec/sec.
- o Accumulate total torquing pulses as a floating point number.
- o Apply either a positive or negative torquing pulse to the appropriate axis based on the sign of the total torquing pulses.
- o Adjust the pulses left to be torqued to account for the pulse just applied.

6.5.3.1.3 Gyro Compensation (GYRO_COMP)

The purpose of the Gyro Compensation (GYRO_COMP) function is to convert the gyro torquing command from the GPC into a torquing rate, compensate the torquing rate for misalignments and scale factors, and check the slew flags.

Specifically, this function performs the following functions:

- o Convert the gyro torquing command from the GPC into a torquing rate. The rate is broken into a uniform rate over the entire Message-Out cycle at a 5 msec interval.
- o The slew flags are checked to determine if the slewing is required.
- o The torquing rates are compensated to account for gyro misalignments and torquer scale factors.

6.5.3.1.4 5 Hz Gyro Compensation (GY5COMP)

The purpose of the 5 Hz Gyro Compensation (GY5COMP) Function is to compensate the gyro for component errors. These component errors are measured by the system calibrations and are stored in the EEPROM's Tag constants region.

Specifically, this function performs the following:

- o Determines if the gimbal angles are at any of the 3 skews of a gyrocompass test. Also calculates the vector sum of the three acceleration channels and compares it to the 1 g value. Heading sensitivity is applied only if a gyrocompass test is being performed in a 1 g environment.
- o Compensate the torquing rate for the following component parameters:

- Gyro Mass Unbalance
 - Misalignments
 - Torquer Scale Factors
 - Heading Sensitivity (Conditional)
 - Gyro Restraint Drift

6.5.3.1.5 Accelerometer Pulse Accumulation

The READ-DLS module is executed 100 times per second. This module reads in from the Pulse Accumulator circuit on the I/O Circuit Card Assembly the number of accelerometer pulses received over a 10 millisecond period. The reading (snapshot) of the pulse accumulator causes the 16 bit accumulator to reset to zero. The READ-DLS module maintains a total count of all the accelerometer pulses read during the pulse accumulator snapshots. Also the READ-DLS module adds to the total count of the accelerometer pulses which were read at the time of the Message-In discrete (RLEV1 also performs a Pulse Accumulator snapshot).

6.5.3.1.6 Long Term Accelerometer Compensation (AC1_COMP)

The long term accelerometer compensation (AC1_COMP) function performs the compensation of the accelerometer required as a

result of gradual environmental changes and/or component characteristics.

Specifically, the AC1_COMP module compensates for the following at a once per second rate:

- o Compute the temperature dependent scale factors based on the measured temperature in the DAL circuitry area.
- o Compensate the scale factor for overall component characteristics.

6.5.3.1.7 Accelerometer 25 Hz Compensation (A25_COMP)

The accelerometer compensation at a 25 Hz rate (A25_COMP) performs the accelerometer scale factor non-linearity compensation.

The A25_COMP computes the acceleration for each of the axes over a 40 msec period. The non-linearity compensation is then applied if the accelerometer pulse output is within the following ranges:

| | <u>Lower Limit</u> | <u>Upper Limit</u> |
|----|--------------------|--------------------|
| 1) | 2706.1 | 3388.4 |
| 2) | 2987.2 | 3093.0 |

The first pair of limits is used when calibration parameters are not stored in the EEPROM Tag constants region. The second pair of limits is used when calibration parameters are stored in the Tag constants region. The limits are used to determine when the accelerometer is elevated to approximately 45 degrees (0.707g output).

6.5.3.1.8 Accelerometer Pulse Compensation (ACP_COMP)

The accelerometer pulse compensation (ACP_COMP) is executed at a 100 Hz rate. The ACP_COMP compensates the accelerometer pulses for the following:

- o High Gain or Low Gain as commanded by the GPC.
- o Scale Factor, Bias, Misalignments, and Offset errors as computed during system level testing.
- o Scale factor is extrapolated to the next Message-In 8 cycle. The actual multiplication of the accumulated pulses by the compensated scale factor is performed at the time of the receipt of the Message-In.
- o Negative or Positive acceleration.

In addition, the ACP_COMP detects for transition from the 25 Hz Message-In cycle to the 6.25 Hz Message-In cycle. The ACP_COMP accumulated pulses over a 10 msec cycle, 40 msec cycle, and the 200 msec cycle.

6.5.3.1.9 Redundant Gyro Processing (RGM_COMP)

The redundant gyro monitor measures the inertial drift rate of the HAINS platform about the redundant axis of the dual axis Z gyro. The redundant axis nominally lies on the plane formed by the X, Y gyro. It is 26.5 degrees from the X-axis. The RGM_COMP module compensates the redundant gyro rate for redundant gyro errors and commanded rates.

Specifically, the module will compensate for the following at an update rate of 100 Hz:

- o Drifts due to restraint and mass unbalance errors.
- o The torquer scale factor error.

In addition, the RGM_COMP formulates the RGM output word, which includes the sign bit and the overflow bit as well as the actual measurements.

6.5.4 HAINS Input/Output Function

6.5.4.1 Analog I/O

This portion of the HAINS interface function provides the transfer and compensation of the data obtained from the HAINS A/D Circuit Card Assembly.

6.5.4.1.1 Input A/D Data

The READ_AD is executed as a result of the A/D Circuit Card Assembly generated interrupt. This module stores the following A/D supplied data into a buffer:

- o Gimbal Angle Checksum
- o Roll, Pitch, and Azimuth Gimbal Angles
- o Inner Roll Angle
- o Redundant Gyro Rate

In addition, the READ_AD maintains a running count of the received A/D interrupts.

6.5.4.1.2 Resolver Processing at 25Hz

The R25_PROC module performs resolver error compensation at a 25 Hz rate. The A/D converted Azimuth, Pitch, Outer Roll, and Inner Roll readouts are compensated for externally determined (System Calibrations) resolver offsets and tracking errors. Gimbal non-orthogonal errors are also compensated.

In addition, this module activates the Pitch gimbal switch when the Pitch attitude is above 92 degrees and deactivates the switch when the Pitch attitude goes below 90 degrees.

6.5.4.1.3 8X Resolver Processing

The RES_S8X prepares the Azimuth, Pitch, and Outer Roll 8X speed resolver readouts for transmission during the Message-In cycle. Specifically, this module updates the resolver readouts with the compensated resolver error compensation (as computed by other modules). In addition, the RES_S8X determines the rate of change in the 8 speed angle for Azimuth, Pitch, and Outer Roll (required for extrapolation).

The updated resolver readings and the resolver rate of change information are stored for use by the RLEV1 module.

6.5.4.1.4 1X Resolver Processing

The RES_S1X module prepares the output words for the Azimuth, Pitch, and Outer Roll 1X speed resolver readouts and also the Inner Roll readout. These output words are then stored for transmission to the GPC by the MUX.

The RES_S1X module corrects the 1X speed resolver readouts and the Inner Roll for computed errors. This module also extrapolates the 1X speed resolver readouts and the Inner Roll readout for transmission during the Message-In cycle based on the rate of change in the resolver readings.

In addition, the RES_S1X converts the Inner Roll readout from a voltage to an angle.

6.5.4.2 MUX Function

The MUX function performs the non-critical data transfers between the OFP and the MUX Circuit Card Assembly.

6.5.4.2.1 Input GPC Word

Designated bit positions of the command words received from the GPC invoke specific HAINS actions. The MUXIN module evaluates the two command words received from the GPC during the Message-

Out cycle to determine if the GPC is requesting any of the following actions:

- Digital Torquing
- Slewing
- High or Low Gain Accelerometer Setting

The module stores individually into memory the requests for digital torquing, slewing and accelerometer scale factor setting. These individually stored command requests are then available for the other functions to perform the commanded action.

6.5.4.2.2 Input Words for PROVE Configuration

The Program Validation Equipment (PROVE) is used to supply simulated platform data to the OFP for purposes of evaluating the internal software performance. The IN1553 module is executed only when the OFP is being used in a PROVE configuration.

The IN1553 determines if input data is available from the PROVE. If data is available from the PROVE, then the IN1553 module overwrites the locations that normally contain the actual platform data. This simulated platform data is then available for use by the modules which process the platform data.

6.5.4.2.3 Output Words for PROVE Configuration

The OUT1553 module initiates the transfer from the PROVE of simulated platform data. The OUT1553 module would be executed via an externally issued command, only when in the PROVE configuration.

6.5.5 HAINS BITES

The HAINS BITE provides the in-flight functional checks necessary to assure that internal failures, which result in degraded system performance beyond mission critical levels, are detected.

The IMU has built into it an extensive self-monitoring capability (hardware BITES). Additionally, the IMU software in the GPC's provides further checks on the usability of the IMU's (software BITES). This information from both the hardware and the software is used by the redundancy management (RM) software (discussed later) to determine the health of an IMU. The GPC also enunciates the BITE signals to the crew via an alarm and a fault summary message on their cathode ray tube (CRT). This section will describe the general operation of IMU BITES.

6.5.5.1 Hardware BITES

The overall HAINS BITE is made up of both hardware and internal software diagnostics. The internal software BITE also processes both the hardware and the internal software detected fails and sets the appropriate bits in HAINS Data Word 1 for output to the GPC.

6.5.5.1.1 BITE Output

The results from the BITE tests are contained in the two internal BITE words 15 and 16. Each BITE routine has a bit in one of these words which are set in the event of a detected failure.

During the 25 Hz cycle, the SOUTBTE module takes the contents of the two BITE words and logically OR's them with HAINS data Word 15 and Word 16. The two BITE words are then cleared. Appropriate bits from words 15 and 16 are then used to form HAINS data Word 1. HAINS data Words 1, 15, and 16 are zeroed after the Message-In discrete is received.

6.5.5.1.2 Description

The HAINS uses a complement of hardware and internal software tests to provide an effective method of assuring the functional integrity of the LRU. In addition to the LRU fault isolation, the HAINS contains additional BITE features that will aid in fault isolation to the shop replaceable unit (SRU).

The HAINS BITE cannot initiate a system shutdown or circuit deactivation upon detection of a fault. Hardwired discrettes are available via both the operational and test connectors which can be used to initiate a system shutdown via external equipment.

The HAINS BITE will verify the proper operation of the following major functions of the HAINS:

- o Communication Interface
- o System Excitation
- o Velocity Channel
- o Attitude Channel

The following is a summary of these functional checks.

6.5.5.1.2.1 *Communication Interfaces*

The HAINS digital subsystem and MUX are checked for proper operation. In addition, the communication link of the HAINS to the GPC, and internal communication links are either checked by specific Built In Tests (BIT) or by an overall BITE function.

6.5.5.1.2.2 *System Excitations*

The HAINS BITE checks that the power supply voltages, clock signals, resolver current loads, and gyro/accelerometer current loads are within their specified thresholds.

6.5.5.1.2.3 *Velocity Channel*

In addition to the BITE of the velocity channel excitations, the HAINS BITE also checks the channel for a saturated condition.

6.5.5.1.2.4 *Attitude Channel*

In addition to the BITE on the attitude channel excitations, the HAINS BITE on the attitude channel also consists of a reasonableness check between the 1X and 8X resolvers, a check of the availability of an input signal from the resolvers and an A/D conversion accuracy check.

6.5.5.1.3 MUX Fail

The Multiplex Terminal Unit (MUX or MTU) BITE provides checks of the MUX data transfer capability and overall proper operation. The MUX BITE is comprised of the following tests:

- | | | |
|--------------------------------|----------------|------------|
| o MUX Start-up Test | MUX & I/O Card | Power-on |
| o DMA Controller BITE | I/O Card | 25 Hz |
| o DMA Parallel Test | I/O Card | 25 Hz |
| o End-Around Transmission Test | Mux Card | Message in |

When any of the above BITE tests fail, bit five of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Mux Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.3.1 *MUX Start-up Test*

Once at power-on, the MUX performs a self-test of its internal functions as well as a check of the DMA controller (located on the I/O card) and its data lines.

The MTU Circuit Card Assembly performs a self-test of its functions immediately following the receipt of the Logic Reset. A detected failure is transmitted to the OFP via the Discrete Input Registers (DIR) on the I/O Circuit Card Assembly. Specifically, the MUX indicates a detected failure by setting either Multiplex Terminal Unit Fail (MTU) (DIR 1-10) or DMA Grant Fail (DIR 1-11). The internal software (SMTUST) samples both discretes and, if either one is set, it sets bit 14 of BITE Summary Word No. 2.

In addition, the internal software samples the MTU's DMA locations in memory to determine that the MTU Circuit Card Assembly has initiated its operation. This is done by reading location 0062 (HEX) and verifying that its contents are either AAAA (HEX) or 5555 (HEX). If the contents are not as expected, the internal software sets bit 14 of BITE Summary Word No. 2.

6.5.5.1.3.2 *DMA Controller BITE (DMA Request & Grant)*

The MUX performs two periodic checks of its DMA Controller interface. One is a test of the MUX's capability of transmitting a DMA request to the DMA controller and the other being a test of the receipt of DMA Grant from the controller. The MUX tests the DMA request by monitoring its own DMA request line. If a request is not transmitted from the card when it should, the MUX pulses the MTUF (Multiplex Terminal Unit Fail) discrete, which is then latched on the I/O SRU. The MUX also tests the DMA Grant signal by timing the interval between the transmission of a DMA Request and the receipt of a DMA Grant. If this time exceeds 12 microseconds, the MUX pulses the DMGF (DMA Grant Fail) discrete, which also gets latched on the I/O SRU. The internal software samples the MTUF (DIR 1-10) and the DMGF (DIR 1-11) at a 25 Hz rate, if either indicates a failure, the internal software stores a MTU Fail (BITE Summary Word No. 2, bit 9) or DMA Fail (BITE Summary Word No. 1, bit 9). The truth table for these two latched signals is shown in Table 6-XXXIV.

| MTUF | DMGF | DMGF | (Made on I/O Card from MTUF & DMGF) |
|------|------|------|-------------------------------------|
| 1 | 0 | 0 | MUX Hardware Fail |
| 0 | 1 | 0 | DMA Grant Timeout |
| 1 | 1 | 1 | DMA Parallel BITE Test Fail |
| 1 | 1 | 1 | MTUF and DMGF |

All signals are latched in the I/O Card.

TABLE 6-XXXIV, TRUTH TABLE FOR MTU BITE OUTPUT SIGNALS

6.5.5.1.3.3 *DMA Parallel Test*

The MUX tests its DMA parallel data path by performing a write/read/compare operation using complementary data patterns. First the MUX stores a pattern (AAAAH) in RAM, next the pattern is read by the MUX and a comparison made of data read versus data written. If this test passes, it is then repeated using the complement (5555H) of the previous data pattern. If the test fails, the MUX sets the DMPF (DMA Parallel Fail) discrete on the I/O SRU. DMPF is set by first setting MTUF true then clocking it into a latch using DMGF. The internal software samples the DMPF (DIR 1-12) at a 25 Hz rate. If the DMPF is set (indicating a failure), the internal software issues a reset command to the three MTU latches and isolates the fault as follows.

The internal software checks the results of the Multiple Bit Error Test (performed prior to the sampling of DMPF) for a fail indication, if a multiple bit error has not occurred, the internal software stores a MUX DMA Parallel Fail in bit 18 of BITE Summary Word No. 2. This additional check by the internal software helps in isolating a failure. Figure 6-36 illustrates the sequence of DMA tests.

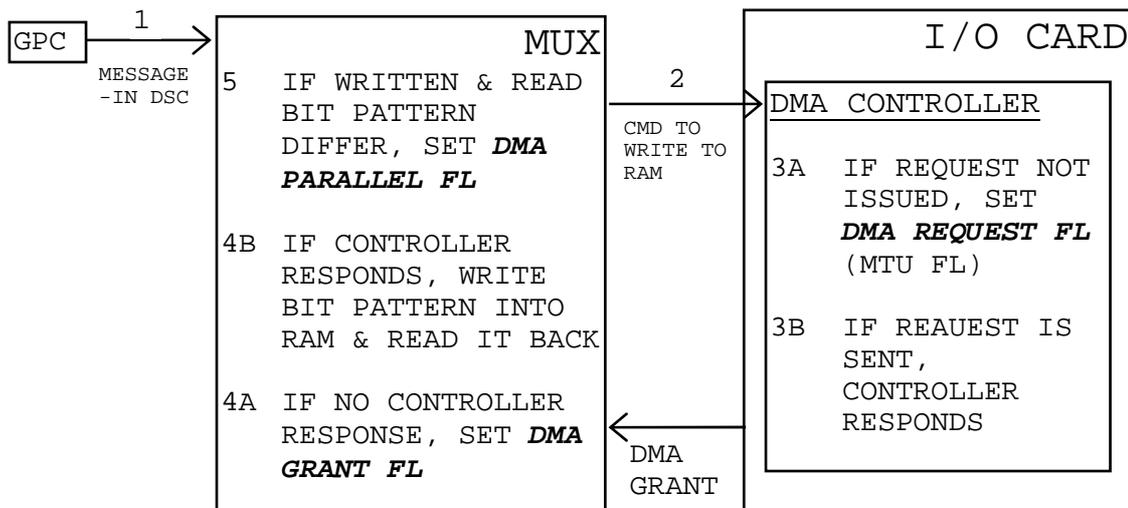


Figure 6-36, DMA Controller and Parallel Test Flow

6.5.5.1.3.4 End-Around Transmission Test

After every transmitted word (by the MUX to the MDM), the MUX performs a check of the Manchester II code, to verify that the word consists of 16 data bits, plus sync, and a parity bit verification. The MUX also checks the contents by taking the serially transmitted data word, bringing it through the word processing section of the MUX and converting it to parallel data. The lower byte of this wrapped around data word is then compared against the lower byte of the original data word. If the two bytes for any of the 16 transmitted words do not compare, the End-Around Transmission Fail bit of the Message Validity Word is set.

The internal software samples the Message Validity Word upon receipt of a Message Out Interrupt, if the End-Around Transmission Fail bit is set, the internal software sets the End-Around Transmission Fail bit 17 of BITE Summary Word No. 2.

6.5.5.1.4 A/D Fail

The A/D Converter BITE monitors the following functions:

- | | | |
|--|----------|----------|
| o AC channel accuracy check | A/D Card | (25 Hz) |
| o DC channel accuracy check | A/D Card | (25 Hz) |
| o Gimbal angle checksum | A/D Card | (100 Hz) |
| o A/D converter interrupt goodness check | A/D Card | (25 Hz) |
| o A/D performed BITE | A/D Card | (25 Hz) |

When any of the above BITE tests fail, bit six of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU A/D Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.4.1 *AC Channel Accuracy Check*

The A/D converter provides an AC channel 45 degree test angle measurement to main memory twenty-five times a second. The internal software compares this angle with the actual value of the angle. If a difference of greater than +/- 0.10 degree exists between the two values, the internal software sets bit 15 of BITE Summary Word No. 2.

6.5.5.1.4.2 *DC Channel Accuracy Check*

The A/D converter provides the +10 Vdc channel value to main memory. Twenty-five times a second, internal software compares this value with the actual 10 Vdc value. If a difference of greater than +/- 2% exists between the two values, the internal software sets bit 16 of BITE Summary Word No. 2.

6.5.5.1.4.3 *Gimbal Angles Checksum*

The A/D converter adds the value of the Azimuth, Pitch, and Roll angles, and DMA's the result into memory. One hundred times a second the internal software adds the value of these same angles and compares the result to the A/D converter value. If the checksum fails, the internal software sets bit 4 of BITE Summary Word No. 2.

6.5.5.1.4.4 *A/D Converter Interrupt Goodness Check*

The internal software insures that the A/D converter is maintaining communications by keeping a count of the number of A/D interrupts received in a 40 msec period. During normal processing, the A/D converter issues an interrupt to the CPU every 5 msec, so an expected count would then be 8 interrupts. The internal software checks the A/D interrupt count once every 40 msec for a count equal to or greater than 6 and less than or equal to 10, since the last sampling of the A/D interrupt count.

If the count is not within these limits, the software initiates an A/D reset routine and sets bit 5 of BITE Summary Word No. 2.

6.5.5.1.4.5 A/D Performed BITE

The A/D microprocessor performs several internal BITE functions to check the proper operation of the A/D card and Platform. Any failure detected by the A/D is indicated in the internal A/D BITE status word which is transmitted to the internal software.

The A/D performs a check of its PROM, RAM, DC conversion channel, and AC conversion channel. In addition, the A/D also performs a check of the inputs from the Platform in the form of a 1X/8X resolver comparison and a check for any missing resolver inputs.

A failure detected by any of the following tests is processed as an A/D test fail which results in the setting of bit 7 of BITE Summary Word No. 2:

- o A/D PROM Test - The A/D periodically performs a checksum its PROM. Any mis-verify that occurs will result in the setting of the A/D PROM fail, bit 0 of the internal A/D BITE status word.
- o A/D RAM Failure - All the A/D converter's RAM locations are checked for unique patterns of both 1 and 0 states. This test is performed during the A/D initialization following power turn-on. If this test fails three times consecutively, the A/D sets bit 1 of the internal A/D BITE status word.
- o DC Offset - The A/D periodically checks the DC offset voltage to assure that the voltage is within specified upper and lower limits. If the voltage exceeds either limit three times consecutively, the A/D sets bit 11 of the internal A/D BITE status word.
- o AC Offset - The A/D periodically checks the AC offset voltage to assure that the voltage is within specified upper and lower limits. If the voltage exceeds either limit three times consecutively, the A/D sets bit 12 of the internal A/D BITE status word.
- o DC Gain - The A/D periodically checks the DC gain voltage to assure that the voltage is within specified upper and lower limits. If the voltage exceeds either limit three times consecutively, the A/D sets bit 14 of the internal A/D BITE status word.
- o AC Gain - The A/D periodically checks the AC gain voltage to assure that the voltage is within specified upper and lower limits. If the voltage exceeds either limit three times consecutively, the A/D sets bit 15 of the internal A/D BITE status word.

6.5.5.1.4.6 *Recurring A/D Fail BITES*

If a HAINS A/D Fail BITE or a Digital Subsystem Fail BITE continues to reoccur multiple times, it is recommended to mode the affected IMU to standby and then back to operate. Cycling back to operate could possibly correct the software problem, clear the BITE and regain operation of the affected HAINS (as a suspect IMU). However, if only one transient IMU BITE is annunciated, the IMU performance should be monitored and considered suspect for the remainder of the mission.

The HAINS OFP software has the capability to detect and correct any single bit error found in the software. However, multiple bit errors can only be detected resulting in the annunciation of the Digital Subsystem Fail BITE. None of the HAINS BITES (except for initialization BITES) are latched; however, if a multiple bit error (MBE) is detected the BITE will continue to reoccur. This is due to a utility function in the HAINS OFP which continues to read and rewrite all of the HAINS memory periodically. When ever the RAM is accessed, the HAINS error detection and correction circuitry is invoked thus reannunciating the MBE BITE each cycle. This reading and writing to the RAM memory performed by the OFP is called RAM Scrubbing. It has been confirmed that moding a HAINS to standby and then back to operate is one way to clear a multiple bit error. If the BITE continues to occur once the moding steps are complete, a real internal HAINS software problem may exist.

It should also be noted that any time multiple BITES are received right after moding from standby to operate, the HAINS should be moded back to standby and then operate. This will clear any BITES detected upon initialization.

Prior to moding a HAINS to STBY or powering it off, consideration should be given to waiting until all BITE information has been permanently stored in the HAINS memory. Whenever a BITE is annunciated on a HAINS, the appropriate BITE word information is immediately and permanently written to EEPROM. In addition to this, every three hours if a BITE has occurred in the previous three hour cycle, additional information regarding the BITE and IMU status is then written to EEPROM. This additional information is used during post flight analysis. The three hour cycle begins immediately when the IMU modes to operate. This additional information data is not required for post flight analysis and is only highly desirable.

6.5.5.1.5 Platform Fail

Hardware BITE is incorporated on the Platform Electronic (PE) card to monitor the status of the Platform functions. Internal

software monitors Platform functions and records any failures. The Platform BITE provides coverage of the following functions:

- o Hardware BITE
 - Gimbal Loop BITE (azimuth & IR) PE1 Card 5Hz
 - 19.2 kHz Excitation Current (same as above)
 - Gyro Wheel Supply GWS Card 1 Hz

- o Internal Software BITE
 - Sequencer Test PE1 Card Power-on
 - Pulse Accumulator Overflow PE1 Card 100 Hz
 - DAL Regulated Voltages PE1 Card 25 Hz
 - Inner Roll Reasonableness PE1 Card 25 Hz

- o A/D Performed BITE
 - 1X/8X Gimbal Angle Comparison PE1 Card 25 Hz
 - Missing Resolver Input PE1 Card 25 Hz

When any of the above BITE tests fail, bit seven of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Platform Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.5.1 *Gimbal Loop*

Hardware monitors the proper operation of the Azimuth and Inner Roll gimbals through error signals from the X and Z gyro preamplifier outputs. When the platform is functioning normally, the gimbal loop nulls gyro errors. Should a malfunction occur that either impedes or degrades the nulling action, an error signal is generated. This error signal is detected by a detector circuit whose output is then compared to a reference voltage in the comparator. If the detector output exceeds the reference voltage, the comparator output enables a digital timer. The timer counts for a 125 ms interval, and should the indicated fault condition hold the comparator input above the fail threshold for a time greater than this period, the timer output switches to logic "1" level. This signal sets the Platform Fail discrete (DIR 2-4). The internal software monitors this discrete at a 5 Hz rate, if this discrete is set the internal software sets bit 11 of BITE Summary Word No. 2 and issues a PE1 BITE Reset discrete (DOR 1-0).

6.5.5.1.5.2 *19.2 kHz Excitation Current*

Hardware monitors the current levels in the accelerometer and inner roll pickoff (19.2 kHz excitation) If the current drops below the threshold level set in the comparator, the output of the comparator enables a digital timer. The timer counts for a 125 msec interval and should the current remain below the threshold for a time greater than the interval, the Platform Electronics No. 1 (PE1) card output is switched to logic "0" level, indicating that a Platform Fail (Gimbal Loop) has

occurred. The Platform Fail signal sets the Platform Fail discrete (DIR 2-4) . The internal software monitors this discrete, if this discrete is set, the internal software sets bit 11 of BITE Summary Word No. 2 (Gimbal Loop Fail) and issues a PE1 BITE Reset discrete (DOR 1-0), resetting the counter.

6.5.5.1.5.3 *Gyro Wheel Supply (GWS)*

The Gyro Wheel Supply (GWS) card contains BITE circuitry to monitor the GWS phase-locked servo loop which controls the speed and the position (synchronization angle) of the induction motor that drives the wheel of the gyros.

The GWS BITE is inhibited until the D11 discrete is issued. The BITE compares the actual wheel speed of each gyro against the selected reference frequencies, if a 90 degree phase difference is detected on either gyro for four consecutive 0.27 second sampling periods, the GWS Fail discrete (DIR 2-3) is set (logic level "1" = fail). Internal software stores the failure in bit 7 of BITE Summary Word No. 1.

6.5.5.1.5.4 *Sequencer Test*

This test is performed once during IMU initialization. The internal software measures the elapsed time between the issuance of the logic reset, which enables the Platform start-up sequence and receipt of the Sequencer discrettes 9 (DIR 1-8) and D11 (DIR 1-5) This test is aborted if the Sequencer Inhibit discrete (DIR 1-9) is issued by the System Test Equipment (external test connector).

The internal software samples the Sequencer Inhibit discrete at a 1 Hz rate and aborts the test if set. The internal software measures the elapsed time for the setting of D9 and D11. If either the elapsed time for the setting of D9 exceeds 60 seconds or D11 exceeds 107 seconds, the internal software stores a sequencer test fail in bit 10 of BITE Summary Word No. 1.

6.5.5.1.5.5 *Pulse Accumulator Overflow Test*

Internal software tests the DAL accumulators for overflow during normal operation. The velocity pulses from the Platform electronic card are counted 100 times/second in the DAL pulse accumulators on the I/O card. Since the maximum expected pulse rate is 62,500 pps, the maximum pulse count received in any 10 msec period is 625 pulses. The internal software checks the upper 7 bits of each pulse accumulator for a non-zero value. If any of the upper 8 bits are non-zero, the internal software stores the fail condition, pulse accumulator overflow fail in bit 18 of BITE Summary Word No. 1.

6.5.5.1.5.6 *DAL Regulated Voltages*

Internal software tests the three regulated excitation voltages for an out-of-specification condition. The three DAL regulated voltages, +5 Vdc, +15 Vdc and -15 Vdc, are generated on the Platform Electronics No. 1 card and scaled down for A/D conversion (max DC input to A/D is 10 Vdc) on the Platform Electronics No. 2 card. The internal software acquires the digital representation of these voltages from the A/D once every 40 msec and checks that the voltages are within specification. If any of the three voltages exceed their specified value by 5% the internal software stores the fail condition, DAL regulated voltage fail, in bit 5 of BITE Summary Word No. 1. The specified voltages are unique to each HAINS and are stored in the EEPROM.

6.5.5.1.5.7 *Inner Roll Reasonableness*

The internal software verifies that the Inner Roll angle readout is not greater than +/- 1 degree. This test is disabled by the internal software, when the Platform is near or in the gimbal flip region. If the angle exceeds 1 degree during performance of this test, the internal software indicates the failure by setting bit 8 of BITE Summary Word No. 2.

6.5.5.1.5.8 *1X/8X Gimbal Angle Comparison*

The A/D microprocessor performs a comparison of the 1X gimbal angle to the 8X gimbal angle for Azimuth, Pitch, and Roll. This comparison is performed once every 40 msec and if the differences for the 1X/8X exceeds 0.7 degree, the A/D indicates the failure by setting bit 8, 9, or 10 of the internal A/D BITE status word for Azimuth, Pitch, or Roll, respectively. Upon receipt of a fail indicator, the internal software adds 5 degrees to the failed 1X angle and sets bit 12 of BITE Summary Word No. 2.

6.5.5.1.5.9 *Missing Resolver Inputs*

The A/D microprocessor checks that the SIN and COS values for each of the resolver inputs are not below 0.707. If this condition occurs, the A/D will set appropriate bits in the internal A/D BITE status word. Upon receipt of a fail indication, the internal software sets bit 12 of BITE Summary Word No. 2.

6.5.5.1.6 Digital Subsystem Fail

The Digital Subsystem BITE provides coverage of the following functions:

| | | | |
|---|---------------------------------|-------------|----------|
| o | <u>Software BITE</u> | | |
| | Pulse Accumulator | I/O Card | Power-on |
| | DOR Wraparound | I/O Card | 25 Hz |
| | MTU Tag Timer | I/O Card | 25Hz |
| | CPU Instruction Test | CPU Card | 1 Hz |
| | Floating Point/Divide by Zero | CPU Card | 25Hz |
| | Arithmetic Floating Pt Overflow | CPU Card | 25 Hz |
| | Square Root of a Negative # | CPU Card | 25 Hz |
| | RAM Pattern Test | Memory Card | Power-on |
| | EEPROM Checksum | Memory Card | Power-on |
| | PROM Checksum | Memory Card | 1 Hz |
| o | <u>Hardware BITE</u> | | |
| | Multiple Bit Error | Memory Card | 25 Hz |
| | Memory Error Correction Circuit | Memory Card | Power-on |

When any of the above BITE tests fail, bit eight of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Digital Subsystem Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.6.1 *Pulse Accumulator Test (SPULTST)*

The six 16-bit pulse accumulators which accept incremental velocity outputs are tested for fixed values of positive and negative counts. At power turn-on, after Logic Reset (LR) goes high, the pulse accumulators are placed in the test mode allowing the OFP to simulate the transmission of velocity pulses to the pulse accumulator. Internal software (SPULTST) issues a memory mapped I/O instruction which causes each of the pulse accumulator registers to increment by one count. Repeated execution of this instruction causes additional counts to be incremented to the maximum allowable count of 65,535 pulses. added in the counters. Internal software then issues a DAL read snapshot (DOR2-1) causing the accumulators to go to an operational mode and to read the DAL accumulator registers. A comparison of the DAL pulse count against the expected sum (>65,535 pulses) is then made. If the test fails (expected sum DAL pulse count) internal software stores the failure condition, sets the pulse accumulator fail bit, bit 8 of BITE Summary Word No. 1. This test is performed once only at HAINS power on.

6.5.5.1.6.2 *DOR Wraparound BITE*

A DOR output discrete is wrapped around to an input discrete (DIR) to be interrogated by internal software. This wraparound test determines the proper decoding of memory mapped addresses and the functioning of the DOR and DIR registers. If a bit failure for this discrete occurs, internal software issues an I/O fail discrete (DOR 1-10) and stores the failure condition by setting bit 6 of BITE Summary Word No. 2.

6.5.5.1.6.3 *MUX Tag Timer Goodness*

Every 40 msec, the internal software reads the value of the MUX Tag Timer from the memory mapped I/O location. The internal software compares two successive readings from the data at that location. If there is no difference between the two tag timer values, internal software sets bit 10 of BITE Summary Word No. 2.

6.5.5.1.6.4 *CPU Instruction Test*

Once per second, the internal software performs a functional instruction test to check the CPU microcode. If the test fails, the internal software stores the failure condition in bit 16 of BITE Summary No. 1, and issues a CPU card fail discrete (DOR 1-2). Upon receipt of this discrete, the I/O card sets the Digital Subsystem Good to its false state.

6.5.5.1.6.5 *CPU Software Traps*

The operating system of the HAINS CPU provides trap locations which detect and handle internal error conditions. The traps detect the occurrence of the following errors:

- o Floating Point Divide-By-Zero
- o Arithmetic Floating Point Overflow
- o Square Root of a Negative Number

The operating system sets the appropriate bits in a trap word whenever any of these error conditions are detected. The internal software samples the trap word once every 40 msec, if an error condition is detected, the internal software sets Arithmetic Floating Point Overflow, Floating Point Divide-By-Zero or Square Root of a Negative Number as appropriate (BITE Summary Word 1 - bit 6, BITE Summary Word 1 - bit 19, BITE Summary Word 2 - bit 19, respectively).

6.5.5.1.6.6 *RAM Pattern Test (RSTART)*

This internal software BITE tests the proper operation of all the RAM and associated circuitry of the Memory Circuit Card Assembly. This routine is the first internal software BITE executed and is performed as part of the OFP start-up code. The internal software tests all memory RAM bits for both "1" and "0" states once at Power turn-on. If this bit pattern test fails or the error detection and correction circuit indicates an error, internal software sets bit 17 of BITE Summary Word No. 1.

6.5.5.1.6.7 *EEPROM Checksum Test (XINIT)*

A comparison of the EEPROM checksum is performed during the variable initialization routine (XINIT). The XINIT routine reads out the Tag contents of EEPROM and stores the information into RAM. Following the downloading of the system Tag constants from nonvolatile (EEPROM) memory to RAM, the internal software adds up all the values and obtains a checksum. The checksum obtained following downloading is then compared to the checksum value generated and stored in EEPROM with the Tag constants. If these values do not match or the EDC has detected an error, the Tag constants will be downloaded and the checksum performed again. The failure of this test results in the setting of bit 11 of BITE Summary Word No. 1.

6.5.5.1.6.8 *PROM Checksum*

The internal software performs a checksum on the program/data stored in PROM every second. If the internal software detects a miscomparison between the stored checksum and the calculated checksum, it sets bit 11 of BITE Summary Word No. 1.

6.5.5.1.6.9 *Multiply Bit Error*

BITE hardware on the memory board detects and corrects single bit memory errors and detects multiple memory bit errors. The memory bit error detection circuitry checks the memory for errors every time the memory is accessed. If the circuitry detects a multiple bit error, both the Single Bit Error (SBE) discrete (DIR 1-14) and the Multiple Bit Error (MBE) discrete (DIR 1-15) is set. If a single bit error is detected, only the SBE discrete (DIR 1-14) is set. The internal software samples both discretetes at a rate of 25 Hz, if both discretetes are set the internal software sets bit 4 of BITE Summary Word No. 1. If the SBE (DIR 1-14) is set, the internal software updates the single bit error counter, which is stored in nonvolatile memory at a three hour interval. Following the detection of a fail indication (Multiple Bit or Single Bit Error), the internal software clears the discretetes by issuing a reset.

6.5.5.1.6.10 *Memory Error Detection and Correction Self-Test*

The internal software tests the Error Detection and Correction (EDC) circuit during system initialization. The internal software induces a series of single bit errors and double bit errors in a RAM location and verifies that single bit errors are detected and corrected and that the double bit errors are detected. If the EDC does not respond properly to any of the internal software induced errors, the internal software disables the EDC function and sets bit 13 of BITE Summary Word No. 2.

6.5.5.1.7 DC/DC Fail

The DC/DC Fail BITE monitors the following function:

- o DC/DC DC/DC Card (25 Hz)

When the above BITE test fails, bit nine of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU DC/DC Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.7.1 *DC/DC*

Hardware monitors the regulation of the DC output voltages -5 Vdc, +/-15 Vdc, +/-19 Vdc, and performs a gross frequency check on the 1.2288 MHz clock. If an out-of-voltage condition exists or the clock stops, the hardware issues a Power Supply Fail discrete to the I/O card (DIR 2-9). The out-of-voltage condition exists when the DC voltages exceed +/- 10% about their regulation. The internal software reads the DC/DC Fail discrete and sets bit 13 of BITE Summary Word No. 1.

6.5.5.1.8 Discrete Power Supply Fail

The Discrete Power Supply Fail BITE monitors the following function:

- o Discrete Power Supply IPC Card (25 Hz)

When the above BITE test fails, bit ten of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Discrete Power Supply Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.8.1 *Discrete Power Supply*

The IPC card generates a separate (isolated) 5 Vdc which is used to power the Output Discretes. The 5 Vdc isolated is monitored by hardware for an out-of-voltage condition. If an out-of-voltage condition exists, the hardware issues a Discrete Power Supply Fail discrete to the I/O card. The out-of-voltage condition exists when the 5 Vdc isolated exceeds +/- 10% of the regulated value. The internal software samples the Discrete Power Supply Fail discrete and if it indicates a failure it sets bit 14 of BITE Summary Word No. 1.

6.5.5.1.9 Resolver Fail

The Resolver Fail BITE monitors the following function:

- o 7 Vac Resolver IPC Card 25 Hz

When the above BITE test fails, bit 11 of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Resolver Fail BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.9.1 *Resolver Excitation Fail*

The IPC card generates the 7 Vac signal which is used as the excitation for the resolvers. The 7 Vac is monitored for either an out-of-voltage or an out-of-current condition. If an out-of-voltage or current condition exists, the hardware issues a Resolver Excitation Fail discrete to the I/O card. The out-of-voltage condition exists when the 7 Vac exceeds +/- 11% about its regulation. The out-of-current condition exists when the 7 Vac current either exceeds 350 mA or falls below 150 mA. The internal software samples the Resolver Excitation Fail discrete and if it indicates a failure, sets bit 15 of BITE Summary Word No. 1.

6.5.5.1.10 Circuit Card Overtemp

The Circuit Card Overtemp BITE monitors the following function:

- o Circuit Card Overtemp PE1 Card 1 Hz

When the above BITE test fails, bit 12 of the IMU Mode Status Word (Output Word 1) is set high. This results in an IMU Circuit Card Overtemp BITE annunciation (see Table 6-XXXVI in section 6.5.5.2.8.2).

6.5.5.1.10.1 *Circuit Card Over-temperature*

A temperature sensing circuit monitors the HAINS circuit cards continuously for an over-temperature condition. The circuit consists of a temperature sensor located on the PE No. 1 card. The temperature sensor voltage output is converted by the A/D converter card into a digital word. The internal software determines if the measured temperature has exceeded either the Upper or Lower temperature limit. The limits are stored as Tag constants in EEPROM. In the event that a temperature problem exists, the internal software stores the failure condition in bit 12 of BITE Summary Word No. 1.

6.5.5.1.11 Watchdog Timer Fail

The Watchdog Timer Fail BITE monitors the following function:

- o Watchdog Timer I/O Card 1 Hz

The signal generated as a result of CPU Watchdog Timer overflow is provided off-card, to the MUX card, to generate a false indication of the HAINS Good (BITE/Status Word bit 4 = 0).

6.5.5.1.11.1 CPU Watchdog Timer

CPU operation is monitored by a CPU watchdog timer located on the I/O card. The internal software prevents overflow by resetting the timer once a second. The watchdog timer check is inhibited when the OFP detects an I/O or Memory failure to prevent an erroneous CPU failure flag. If the timer overflows, a CPU Watchdog Timer fail generates a CPU fail flag, causes the I/O card hardware to issue a Subsystem Fail (SSFAIL) latched condition and inhibits the CPU. This Subsystem Fail signal sets the hardware Digital Subsystem Good to its false state resulting in the setting of the HAINS Good bit to a false state (see Figure 6-37).

NOTE: This is the only fail which only sets the HAINS Good bit.

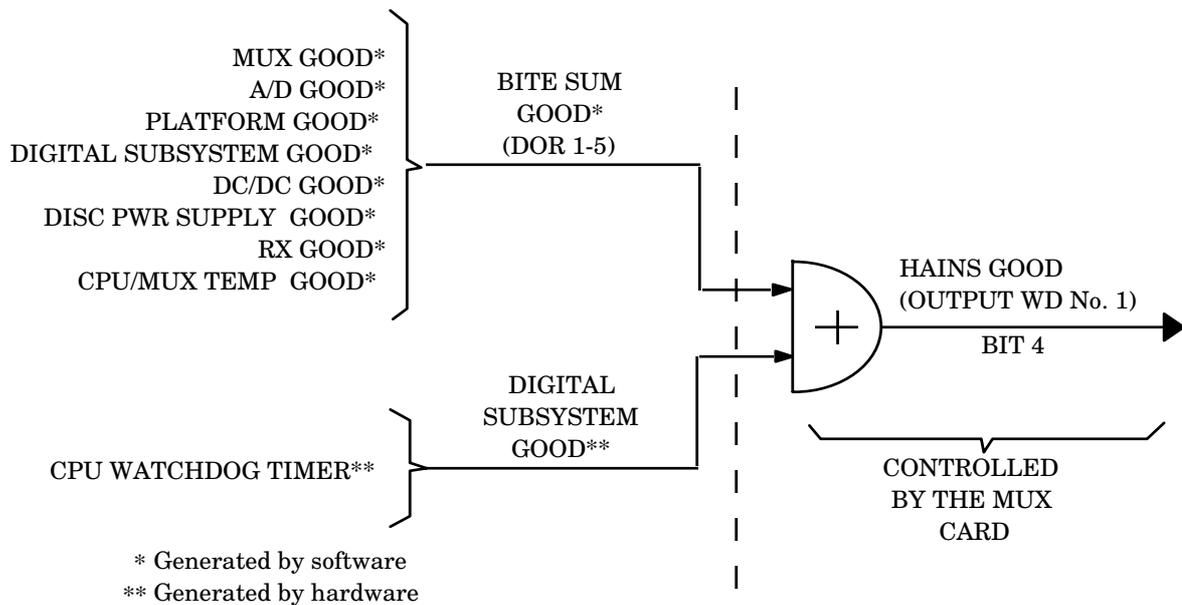


FIGURE 6-37, HAINS GOOD BIT FUNCTIONAL FLOW

6.5.5.1.12 Transmission Word 1 Fail

The transmission word 1 fail is set when the I/O electronics detects a failure of parity, gap, or Manchester coding in the first GPC command word (torquing commands) received by the IMU each transmit cycle. This BITE will cause the PASS to down mode the affected IMU in a dilemma situation.

For a real BITE, the IMU will not process the commanded pulse torquing issued by the GPC. This will result in the IMU drifting at its uncompensated rate. If the BITE is transient, the GPC will continue to reissue the commanded pulse torquing until processed by the IMU, ultimately resulting in no performance degradation.

If the BITE is false, the IMU will process the commanded pulse torquing issued by the GPC. However, the GPC will reissue the pulse torque command causing the IMU to misalign very rapidly, have a high redundant rate, and fail attitude RM.

6.5.5.1.13 Transmission Word 2 Fail

The transmission word 2 fail is set when the I/O electronics detects a failure of parity, gap, or Manchester coding in the second GPC command word (slew commands) as received by the IMU. This BITE will cause the PASS to down mode the affected IMU in a dilemma situation.

For a real BITE, the IMU will not process the commanded slew issued by the GPC. This will result in the IMU effectively being no-oped for IMU to IMU alignments. If the BITE is transient, the IMU will process the commanded slew after the BITE is cleared, ultimately resulting in no performance degradation.

If the BITE is false, the IMU will process the commanded slew issued by the GPC and there will be no impact on IMU operation.

6.5.5.1.14 Communication Good

The communication from the HAINS to the MDM is monitored for proper serial data request/transfer. The false state of this discrete indicates that an error condition has been detected with the MDM issued discrete during the Message-In serial data transfer from the HAINS to the MDM. This discrete is issued by the MUX card. Since this discrete used to indicate Pressure Good on the old KT-70 IMUs, it is not used by Orbiter GPC Redundancy Management (RM) software.

6.5.5.1.15 Platform Temp Ready Fail

The Platform Electronic No. 2 Card contains the three Temperature Control Amplifiers (TCA), that together with associated heater and temperature sensors, maintain the platform at constant temperature. The BITE associated with this SRU consists of temperature sensors that monitor the platform for both an over-temperature and under-temperature condition. The pass/fail indications from this BITE are sent out as output discrettes, Platform Temperature Safe and Platform Temperature Ready.

The Vertical gyro and Azimuth gyro temperature are monitored. When the temperature of the Vertical gyro exceeds 146.5 deg F plus or minus 3 deg F and when the temperature of the Azimuth gyro exceeds 150 deg F plus or minus 3 deg F, the discrete is set to its true state. This discrete is issued by the PE No. 2 card.

6.5.5.1.16 Platform Temp Safe Fail

The Platform Electronic No. 2 Card contains the three Temperature Control Amplifiers (TCA), that together with associated heater and temperature sensors, maintain the platform at constant temperature. The BITE associated with this SRU consists of temperature sensors that monitor the platform for both an over-temperature and under-temperature condition. The pass/fail indications from this BITE are sent out as output discrettes, Platform Temperature Safe and Platform Temperature Ready.

The Vertical gyro temperature is monitored to detect a safe temperature condition. The discrete is issued in its true state when the Vertical gyro temperature is below 156.5 deg F plus or minus 3 deg F. This discrete is issued by the PE No. 2 card.

6.5.5.1.17 Temp Ready Fail

This discrete is set to its true state (+5 Vdc) and not changed. The Temperature Ready and Temperature Safe signals are issued by the MUX card through a pull-up resistor to the +5 isolated supply. These two discrettes are not used operationally and are provided for compatibility to the KT-70 system.

6.5.5.1.18 Temp Safe Fail

This discrete is set to its true state (+5 Vdc) and not changed. The Temperature Ready and Temperature Safe signals are issued by the MUX card through a pull-up resistor to the +5 isolated supply. These two discrettes are not used operationally and are provided for compatibility to the KT-70 system.

6.5.5.2 Software BITEs

The information transmitted to the GPC by the IMU is used not only for attitude and acceleration data, but it is also used to perform certain tests indicating the health of the IMU. The software uses these tests in conjunction with the above hardware BITE to provide assurance that greater than 95 percent of the hardware failures are covered by BITE. Hardware BITE detects only 80 percent of the possible failures.

Software-aided BITE consists of seven separate signals, each of which is discussed below.

6.5.5.2.1 Redundant Rate Fail

If the compensated and filtered redundant axis rate exceeds 0.7 deg/hr, this flag is set.

6.5.5.2.2 Inner Roll Null Fail

The inner roll gimbal is normally kept near 0°. If the angle is outside the range of ±3°, this flag is set.

6.5.5.2.3 Resolver Limit Fail

The software performs a reasonableness test on the outer roll, pitch, and azimuth resolver angles. The 1X resolver angle for each axis is compared with the computed angle for each axis (the resolver offset, which is included in the computed angle, is removed prior to the comparison test). If the difference between the 1X and computed angles exceeds a previously computed limit, the resolver limit fail flag is set. The limit is computed by adding a rate of change factor to a constant limit value (1°). The rate of change factor is computed by dividing the rate of change of a computed gimbal angle by the rate of processing of the gimbal data. Unless fairly rapid maneuvers are occurring, this limit is normally close to 1°.

6.5.5.2.4 Velocity Limit Fail

The velocity limit test actually consists of two tests, an overlimit test and an underlimit test. The underlimit test first forms a normalized acceleration for each of the three IMU's. If on the previous cycle at least two normalized accelerations exceed 1.93 ft/sec², if on the present cycle one or more of the normalized accelerations exceeds 1.93 ft/sec², and one and only one IMU has a normalized acceleration of less than 1.61 ft/sec² in the present cycle, the velocity limit fail BITE signal is set for that IMU. The velocity overlimit test merely compares the measured DV for each IMU to a defined limit. If the limit is exceeded, the velocity limit fail signal is set to one. The limit for the overlimit test in the low-gain mode at a 6.25 Hz processing rate is 720 velocity pulses, which equates to an acceleration of 4.8g. At a 1.04 Hz rate, the limit is 4300 pulses, corresponding to the same g-level.

6.5.5.2.5 Echo 1 Fail

This test is a loop back of input word no. 1. The IMU retransmits input word no. 1 as output word no. 13 (echo word). If the two words are not identical, this signal is set to one.

6.5.5.2.6 Echo 2 Fail

This signal is the same as the fail echo one except input word 2 and output word 14 are compared.

6.5.5.2.7 Accelerometer Scale Factor Fail

The software checks the accelerometer/CAPRI gain bit of output word no. 1. If the bit is not the same as the last transmitted gain setting, this BITE signal is set to one. In other words, if the IMU will not mode to the gain setting requested by the GPC, this BITE is set.

6.5.5.2.8 Failure Indications

The HAINS BITE provides two types of operational failure indications:

- o Hardwired Output Discretes
- o A Serial Digital Output BITE/Status Word (Output Word No. 1)

In addition, several discrete signals are available via the J4 test connector for fault isolation.

6.5.5.2.8.1 *Discrete Inputs*

This section describes both the input and output system-level discretes. These discretes are issued via the J2 connector, and provide indications of HAINS status and BITE. All these discretes (except for Temperature Ready and Temperature Safe) are optically isolated from +28V filter ground and are referenced to +5V isolated ground.

Operate Mode (Command)

The presence of this discrete input to the HAINS (true state) causes the HAINS to be fully powered and operable. The absence of this discrete (false state) for a period in excess of 15.0 seconds, while plus 28 Vdc is applied causes the HAINS to go into the Warm-up/Standby mode, with only the heaters, temperature control, and associated BITE circuitry operable. If this discrete is absent for less than 4 seconds, the HAINS remains in its Operate mode. This discrete is received by the IPC card and optically isolated.

6.5.5.2.8.1 *Discrete Outputs*

The HAINS provides a total of seven output discrettes, which provide continuous system status information. These discrettes are transmitted over the HAINS J2 connector. The output discrettes and their functions are as follows:

In-Operate Mode - The true state of this discrete indicates that the HAINS has received the Operate Mode command discrete from the GPC and is fully powered and operational.

The output circuitry for this discrete is on the Platform Electronic Card No. 2 (PE2) and is generated following the receipt of the Operate Mode command discrete and the Platform Sequence D11 discrete. The false state of this discrete indicates that the HAINS is in the Standby Mode or is powered-down. The normal signal characteristics of this discrete are shown below:

| <u>HAINS MODE</u> | <u>STATE*</u> |
|-------------------|---------------|
| Standby | False |
| Operate | True |

*Positive logic is used; True is +5 Vdc.

This discrete goes true nominally 90 seconds after receipt of the Operate Mode command discrete from the GPC.

The true state of this discrete output from the HAINS indicates that the HAINS has received the Operate Mode command discrete input and is fully powered and operable. The false state of this discrete indicates that the HAINS is in the Warm-up/Standby mode or is powered down. This discrete is issued by the PE No. 2 card.

Platform Temperature Safe Discrete - The true state of this discrete indicates that the Platform temperature is within a safe temperature range, nominally below 156.5 deg F plus or minus 3 deg F. The circuitry for this discrete is located within the Platform and on the PE No. 2 card. This discrete goes to a false state whenever the Vertical gyro temperature exceeds 156.5 deg F plus or minus 3 deg F, regardless of the HAINS mode.

Platform Temperature Ready Discrete - The temperature of the gyros is monitored by circuitry located within the Platform and on the PE No. 2 card. The true state of this discrete indicates that the temperature of the gyros have exceeded a nominal temperature of 146.5 deg F plus or minus 3 deg F for the Vertical gyro and 150 deg F plus or minus 3 deg F for the Azimuth gyro. The output circuitry for this discrete is located on the PE No. 2 card. This discrete remains false, following the application of 28 Vdc, until the temperature of the Vertical and Azimuth gyros reaches the specified range.

Temperature Ready Discrete - This discrete is set to its true state following the application of 28 Vdc to the HAINS and does

not change. The output circuitry for this discrete is located on the Multiplexer Terminal Unit (MTU or MUX) card. This discrete does not provide any monitoring of HAINS functions, but is required for transparency of KT-70/GPC interface.

Temperature Safe Discrete - This discrete is set to its true state following the application of 28 Vdc to the HAINS and does not change. The output circuitry for this discrete is located on the MUX card. The normal signal characteristics for this discrete is the same as the Temperature Ready discrete shown above. This discrete does not provide any monitoring of HAINS functions, but is required for transparency of KT-70/GPC interface.

Communications Good Discrete - The communications from the GPC to the HAINS via the MDM is monitored by circuitry located on the MUX. The true state of this discrete indicates that the timing gap between the Message-In discrete and/or the Word discrettes is within specified limits. The false state of this discrete indicates that the timing gap between the Message-In discrete and/or Word discrettes has exceeded specified value. The MUX also sets this discrete to its false state whenever the MUX is commanded by the test equipment into its Auto Diagnostic Mode or the System Analyzer MUX Mode (SAMM). This discrete goes false during in-flight operations only when an error is detected during a Message-In Transmission Sequence. The output circuitry for this card is located on the MUX card.

HAINS Good Discrete (not used in-flight) - The true state of this discrete indicates that the HAINS BITE has not detected any system abnormalities or failures. This discrete results from the summing of all the BITE indications on the Input/Output (I/O) card. This discrete goes false only in the Operate Mode when the HAINS BITE detects a failure. The output circuitry for this discrete is located on the PE No. 2 card. This discrete is used for ground testing purposes only and is not used in flight. It should not be confused with the HAINS Good bit in the serial output BITE/Status Word (Output Word 1).

The true state (one) of this discrete indicates the BITE circuits have not detected an internal fail condition. This discrete is used for ground testing purposes only and is not used in flight. It should not be confused with the HAINS Good bit in the serial output BITE/Status Word (Output Word 1).

The BITE signals are logically ANDed on the I/O card. The result is sent to the PE No. 2 card where it is optically isolated and issued.

6.5.5.2.8.2 *Serial Output BITE/Status Word*

The HAINS transmits a BITE/Status Word as Serial Output Word No. 1 which provides an overall system status. This word is available at a maximum rate of 25 times a second to the GPC. The Serial Output BITE/Status Word is shown in Table 6-XXVI in section 6.4.1.4.

This word is generated by the internal software and is a combination of all the different BITE functions, a total of 34 checks. The HAINS BITE is comprised of firmware/hardware, performed on individual cards, and internal software BITE, performed in the CPU. All the firmware/hardware detected failures are sent to the CPU (internal software) for processing either as hardwired discrettes, via the I/O card, or as a digital status word, via the Memory card. A listing of all the input BITE signals to the internal software is given in Table 6-XXXV. The internal software then samples the input signals and performs its own functional checks and the results of these are combined to form bits 5 through 12 of the Serial Output Word No. 1.

The HAINS Good bit of the BITE/Status Word is not set directly by the internal software, it is set by the MUX using inputs from the internal software and hardware. The internal software provides a discrete to the MUX which indicates the status of all the BITE routines (BITE Sum Good).

A digital hardware timer which is located on the I/O card monitors the proper execution of the Orbiter Flight Program (OFP) by the CPU. The timer will overflow in the event of a failure which disrupts the OFP execution, resulting in the transmission of a fail indication to the MUX card (Digital Subsystem Good). The MUX uses the BITE Sum Good and Digital Subsystem Good to set the HAINS Good bit (shown in Figure 6-37 in section 6.5.5.1.11.1).

The MUX acquires the first Serial Output Word from a memory location in the memory card following the completion of the transmission of the fourteen or sixteen output words (Message-In cycle) and updates and then stores it in both its output data register and the next consecutive memory location in the memory card. If a Message-Out discrete is received next, then the MUX will update the Transmission Word No. 1 Fail, Transmission Word No. 2 Fail, and HAINS Good bits in the first Serial Output Word as required. The MUX updates the HAINS Good bit by logically ANDing bit 0 of the consecutive memory location and the BITE Sum Fail discrete. Upon receipt of the Message-In discrete the MUX transmits the first Serial Output Word which had been previously (stored in its output data register).

This method of updating the first Serial Output Word results in a staleness of the first Serial Output Word which is equal to either the time between Message-In cycles or the time between the

Message-Out cycle and the Message-In cycle, whichever time is smaller.

General descriptions of HAINS failures are mapped to the BITE/Status Word (Output Word 1) bits in Table 6-XXVI in section 6.4.1.4. Shown in Table 6-XXXVI is the bit mapping of the individual HAINS BITE failures to the BITE/Status Word (Output Word 1) and BITE Summary Words 1 and 2 (Output Words 15 and 16, not used by Orbiter GPC).

| SIGNAL | SOURCE | FUNCTION | LATCH STATUS* |
|----------------------------|---|---|---------------------------------------|
| RESOLVER EXCITATION FAIL | INPUT POWER CONDITIONER (IPC) | 7 VAC CURRENT OR VOLTAGE OUT OF SPEC | NOT LATCHED |
| CIRCUIT CARD OVERTEMP | MUX | CPU OR MUX CARD OVERTEMP | NOT LATCHED |
| MTU FAIL | MUX | TRANSMISSION OF DMA REQUEST | RESET VIA MEMORY MAPPED INSTRUCTION |
| DMA GRANT FAIL | MUX | DMA GRANT NOT RECEIVED | RESET VIA MEMORY MAPPED INSTRUCTION |
| DMA PARALLEL FAIL | MUX VIA THE TOGGLING OF MTU FAIL AND DMA GRANT FAIL | MUX/DMA TRANSMISSION CHECK FAILURE | RESET VIA MEMORY MAPPED INSTRUCTION |
| SINGLE BIT ERROR | MEMORY | SINGLE BIT ERROR DETECTED | RESET VIA DOR |
| MULTIPLE BIT ERRORS | MEMORY | MULTIPLE BIT ERROR DETECTED | RESET VIA DOR |
| GWS FAIL | GYRO WHEEL SUPPLY (GWS) | GYRO WHEEL SPEED ERROR | RESET VIA DOR |
| PLATFORM FAIL | PLATFORM ELEC. NO. 1 (PE 1) | PLATFORM FAILURE DETECTED | RESET VIA DOR |
| DISCRETE POWER SUPPLY FAIL | IPC | +5 VDC ISOLATED VOLTAGE OUT OF SPEC | NOT LATCHED |
| DC/DC FAIL | DC/DC | DC REGULATED VOLTAGES OUT OF SPEC OR CLOCK FAIL | NOT LATCHED |
| A/D BITE STATUS WORD | A/D | FAILURE DETECTED DURING A/D BITE TESTS | NOT LATCHED |
| SUBSYSTEM FAIL | DIGITAL I/O | CPU, WATCHDOG TIMER OVERFLOW | CPU WATCHDOG TIMER RESET |
| END-AROUND TRANSMISSION FL | MUX VIA MUX VALIDITY WORD | BIT WRAP MIS-VERIFY DURING MESSAGE IN DATA TRANSFER | RESET W/ EACH MESSAGE IN TRANSMISSION |
| TRANSMISSION WORD 1 FAIL | MUX VIA MUX VALIDITY WORD | TRANSMISSION ERROR | RESET W/ EACH MESSAGE IN TRANSMISSION |
| TRANSMISSION WORD 2 FAIL | MUX VIA MUX VALIDITY WORD | TRANSMISSION ERROR | RESET W/ EACH MESSAGE IN TRANSMISSION |

NOTES: * REFERS TO LATCHING AND RESETTING ON SOURCE CARD.
DOR - DISCRETE OUTPUT REGISTER
DMA - DIRECT MEMORY ADDRESS
MTU, MUX - MULTIPLEX TERMINAL UNIT

TABLE 6-XXXV, BITE INPUTS TO HAINS INTERNAL SOFTWARE

| BITE TEST/SIGNAL | SOURCE | SRU | RATE* | BITE/STATUS WORD BITE-BIT SET | BITE SUMMARY WORD-BIT SET |
|------------------------------------|----------------|--------|------------------------|-------------------------------|---------------------------|
| MUX START-UP (MTUF LATCH) | MUX & SOFTWARE | MUX | ONCE AT POWER-ON | MUX FAIL-BIT 5 | 2-14 |
| DMA PARALLEL (DMPF LATCH) | MUX | MUX | 25 HZ | MUX FAIL-BIT 5 | 2-18 |
| DMA GRANT (DMGF LATCH) | MUX | MUX | 25 HZ | MUX FAIL-BIT 5 | 1-9 |
| DMA REQUEST (MTUF LATCH) | MUX | MUX | 25 HZ | MUX FAIL-BIT 5 | 2-9 |
| END-AROUND TRANSMISSION | MUX | MUX | PERFORMED @ MESSAGE IN | MUX FAIL-BIT 5 | 2-17 |
| GIMBAL ANGLE CHECKSUM | SOFTWARE | A/D | 100 HZ | A/D FAIL-BIT 6 | 2-4 |
| DC CHANNEL ACCURACY | SOFTWARE | A/D | 25 HZ | A/D FAIL-BIT 6 | 2-16 |
| AC CHANNEL ACCURACY | SOFTWARE | A/D | 25 HZ | A/D FAIL-BIT 6 | 2-15 |
| A/D TEST | A/D | A/D | 25 HZ | A/D FAIL-BIT 6 | 2-7 |
| A/D INTERRUPT | SOFTWARE | A/D | 25 HZ | A/D FAIL-BIT 6 | 2-5 |
| SEQUENCER | SOFTWARE | PE1 | ONCE AT POWER-ON | PLATFORM FAIL-BIT 7 | 1-10 |
| PULSE ACCUMULATOR OVERFLOW | SOFTWARE | PE1 | 100 HZ | PLATFORM FAIL-BIT 7 | 1-18 |
| GYRO WHEEL SUPPLY | GWS | GWS | 1 HZ | PLATFORM FAIL-BIT 7 | 1-7 |
| GIMBAL LOOP | PE1 | PE1 | 5 HZ | PLATFORM FAIL-BIT 7 | 2-11 |
| DAL REGULATED VOLTAGES | SOFTWARE | PE1 | 25 HZ | PLATFORM FAIL-BIT 7 | 1-5 |
| A/D MISSING RESOLVER INPUT | A/D | PE1 | 25 HZ | PLATFORM FAIL-BIT 7 | 2-12 |
| INNER ROLL REASONABLENESS | SOFTWARE | PE1 | 25 HZ | PLATFORM FAIL-BIT 7 | 2-8 |
| 1X/8X ANGLE COMPARISON | SOFTWARE | PE1 | 25 HZ | PLATFORM FAIL-BIT 7 | 2-12 |
| PULSE ACCUMULATOR | SOFTWARE | I/O | ONCE AT POWER-ON | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-8 |
| DOR WRAPAROUND | SOFTWARE | I/O | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 2-6 |
| MTU TAG TIMER | SOFTWARE | I/O | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 2-10 |
| RAM PATTERN | SOFTWARE | MEMORY | ONCE AT POWER-ON | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-17 |
| MULTIPLE BIT ERROR | MEMORY | MEMORY | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-4 |
| MEMORY ERROR CORRECTION CIRCUIT | MEMORY | MEMORY | ONCE AT POWER-ON | DIGITAL SUBSYSTEM FAIL-BIT 8 | 2-13 |
| CPU INSTRUCTION TEST | SOFTWARE | CPU | 1 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-16 |
| FLOATING POINT DIVIDE BY ZERO | SOFTWARE | CPU | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-19 |
| ARITHMETIC FLOATING POINT OVERFLOW | SOFTWARE | CPU | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-6 |
| SQUARE ROOT OF A NEGATIVE NUMBER | SOFTWARE | CPU | 25 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 2-19 |
| EEPROM CHECKSUM | SOFTWARE | MEMORY | ONCE AT POWER-ON | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-11 |
| PROM CHECKSUM | SOFTWARE | MEMORY | 1 HZ | DIGITAL SUBSYSTEM FAIL-BIT 8 | 1-11 |
| DC/DC | DC/DC | DC/DC | 25 HZ | DC/DC FAIL-BIT 9 | 1-13 |
| DISCRETE PWR SPLY | IPC | IPC | 25 HZ | DISCRETE PWR SUPPLY FL-BIT 10 | 1-14 |
| 7 VAC RESOLVER | IPC | IPC | 25 HZ | RX FAIL-BIT 11 | 1-15 |
| CIRCUIT CARD OVERTEMP | PE1 | PE1 | 1 HZ | CIRCUIT CARD OVERTEMP -BIT 12 | 1-12 |

TABLE 6-XXXVI (part 1 of 2) BITE TESTS AND OUTPUT WORDS

NOTES: * EITHER THE RATE AT WHICH THE ERROR SIGNALS ARE SAMPLED OR THE TEST IS PERFORMED BY THE HAINS INTERNAL SOFTWARE.
BITE SUMMARY WORDS 1 AND 2 (OUTPUT WORDS 15 AND 16) ARE NOT USED BY ORBITER GPC.
DMA - DIRECT MEMORY ADDRESS
MTU, MUX - MULTIPLEX TERMINAL UNIT
DOR - DISCRETE OUTPUT REGISTER
PE1 - PLATFORM AND PLATFORM ELECTRONICS NO. 1
IPC - INPUT POWER CONDITIONER

TABLE 6-XXXVI (part 2 of 2) BITE TESTS AND OUTPUT WORDS

6.5.5.2.9 Performance/Design

6.5.5.2.9.1 *BITE Thresholds*

Circuit tolerances over extremes of temperature and component tolerances build up and produce a BITE trip point uncertainty; i.e., a BITE trip band. Where applicable, the trip bands are given in Table 6-XXXVII.

The power conditioner thresholds have been established by adding BITE trip bands to the specification band. As an example, see Figure 6-38.

| | FAILED SIGNAL | NOMINAL VALUE | TRIP VALUE |
|---|---|--|---------------------------------------|
| POWER CONDITIONER (IPC AND DC/DC) | 5 VDC | 5 +/- 0.10 VDC | +/- 3.5 % |
| | 15 VDC | 15 +/- 0.15 VDC | +/- 3.5 % |
| | -15 VDC | -15 +/- 0.15 VDC | +/- 3.5 % |
| | 19 VDC | 19 +/- 0.95 VDC | +/- 3.5 % |
| | -19 VDC | -19 +/- 0.95 VDC | +/- 3.5 % |
| | +5 VDC ISOLATED | 5 +/- 0.25 VDC | +/- 3.5 % |
| | 7 VAC RESOLVER EXCITATION (RMS) | 7 +/- 0.35 VAC (RMS) | +/- 5.5 % |
| | 7 VAC RESOLVER CURRENT | 250 MA | +/- 75 MA |
| | CLOCK CIRCUIT | 4.91 MHZ +/- 50 PPM | LOSS OF SIGNAL |
| PLATFORM AND PLATFORM ELECTRONICS (PLATFORM ELEC NO. 1 AND GWS) | X GYRO PREAMPLIFIER | 6.0 VDC FOR 125 MSEC | |
| | Z GYRO PREAMPLIFIER | 6.0 VDC FOR 125 MSEC | |
| | 19.2 KHZ CURRENT | 25 MA | +/- 10 MA |
| | VERTICAL GYRO SPEED | 90 DEG PHASE ERROR | |
| | AZIMUTH GYRO SPEED | 90 DEG PHASE ERROR | |
| | DAL REGULATED VOLTAGES (+5 VDC, +15 VDC, -15 VDC) | EXACT VALUES DETERMINED DURING INTEGRATION | +/- 5 % |
| A/D CONVERTER | 1X, 8X RESOLVER ANGLE COMPARISON | | DELTA OF 0.7 DEG |
| | MISSING RESOLVER INPUT | | BOTH SIN AND COS INPUTS < 0.707 |
| | DC CHANNEL ACCURACY | +10 VDC | +/- 2 % |
| | AC CHANNEL ACCURACY | 45 DEG | +/- 0.10 DEG |
| | INNER ROLL EXCITATION | +3.6 VDC | +/- 0.5 VDC |

TABLE 6-XXXVII, BITE CRITERIA

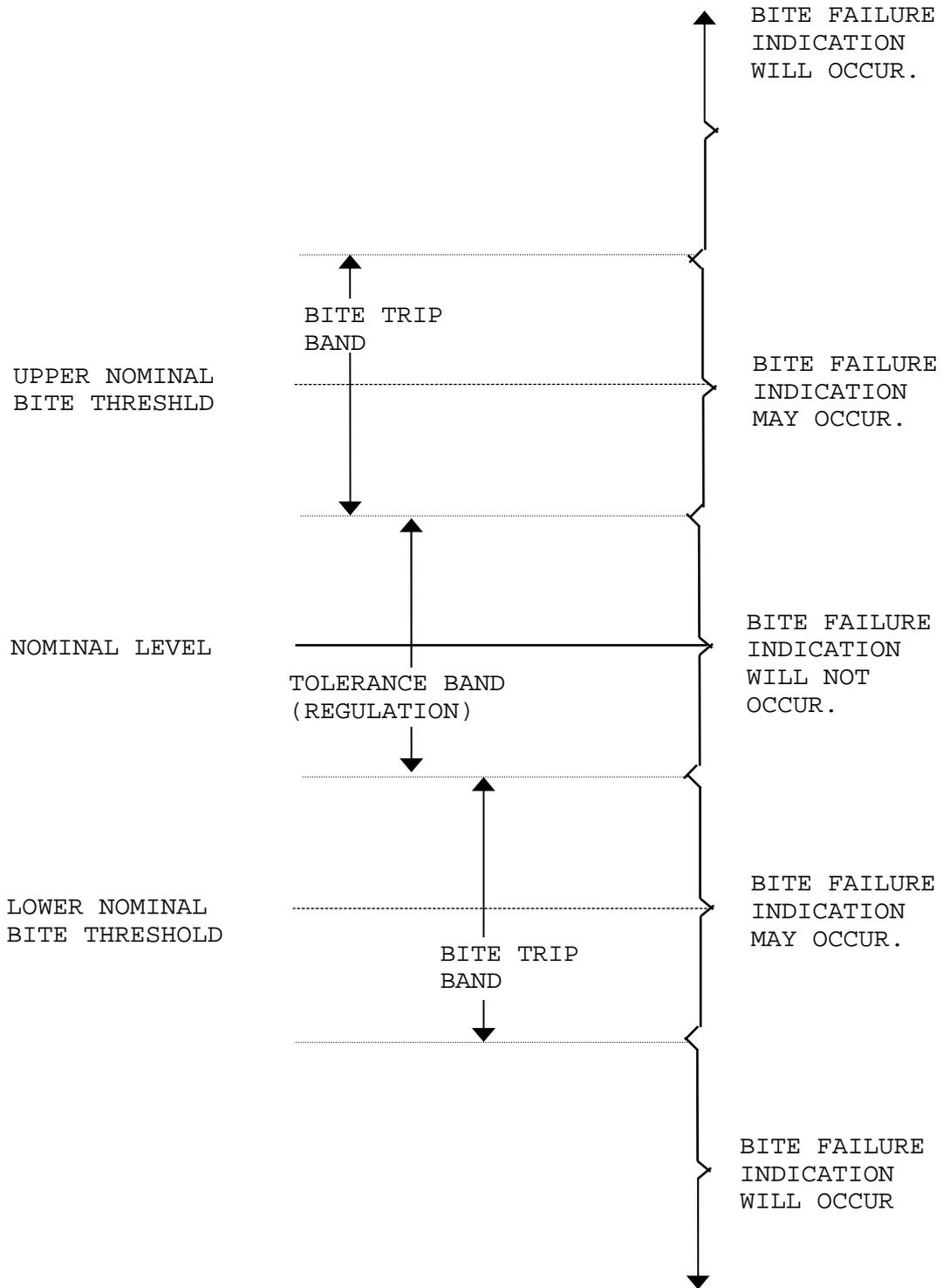


FIGURE 6-38, TRIP BANDS FOR BITE

6.5.5.2.9.2 BITE Summary Words (not used in-flight)

The HAINS has additional output indications which provide additional fault isolation information. The internal software maintains two BITE summary words for internal usage and are accessible as output words 15 and 16 upon request. Output words 15 and 16 are spares on KT-70. The BITE summary words provide a pass/fail status for each of the 33 internal software controlled BITE tests. The two BITE summary words are shown in Table 6-XXX and Table 6-XXXI in section 6.4.1.4, Output Data Control.

6.5.5.2.9.3 *Turn-On Count (STONCNT)*

The STONCNT routine does not perform any diagnostic function. This routine prepares a region of the EEPROM portion of memory to store away the BITE detected fails which may occur during that turn-on cycle.

6.5.5.2.9.4 *Post-Flight Failure Storage*

The HAINS contains 4K Byte of nonvolatile storage on its Memory card which is available for BITE information. This information can only be accessed via the serial communication link using special test internal software. The 4KB block of memory is divided into smaller blocks which contain BITE information for a particular HAINS power-on period. Contained within these blocks is the turn-on count (number of turn-ons since the memory had been erased), a sequence of BITE detected failures during the corresponding turn-on period and various locations which represent the number of times that following events occurred:

- o Spurious Interrupts from the A/D
- o Spurious Interrupts from the MUX
- o Spurious Interrupts from the 200 Hz Clock
- o Single bit Memory Errors Detected and Corrected
- o False Digital Subsystem Failure Indications
- o Spurious Interrupts from the Software Generated Cycles
- o Out-of-Limit Condition for the PE No. 1 DAL Temp Sensor

6.5.6 HAINS Special Mode

A special feature of the OFP is its ability to store BITE failure information in its EEPROM region of memory. Additionally, the OFP can compensate for platform parameter errors which are determined during factory calibration. These compensations are also stored in EEPROM for use by the OFP. This section describes the mode needed to write these calibration parameters to EEPROM, or read BITE or calibration parameters information from EEPROM.

This mode is designed so that any device capable of emulating the multiplexer-demultiplexer (MDM) protocol can enter it. The inputs to the HAINS are the two command words transmitted during a Message-Out cycle. The outputs from the HAINS will be 16 words sent during a Message-In cycle.

A special fail-safe handshake sequence involving the commands issued during the Message-Out cycles which occur within the first two seconds of the OFP initiate is required to force the OFP into special mode. If the specified commands are not received in the correct order and allowed time, then the OFP defaults to its operational mode. The sequence precludes accidental entry into special mode during normal HAINS processing.

Once the HAINS is in special mode, a series of commands, transmitted over the unit's serial communication link to the MDM, causes the OFP to execute several utility subfunctions. These utility routines result in the reading and/or writing of EEPROM.

The OFP will not return to operational mode, once special mode is entered, unless HAINS power is cycled off and then on.

6.5.7 HAINS Utility Function

The utility function is made up of modules which perform the control of the EEPROM, the RAM scrubbing and the data format conversion.

The writing and reading of EEPROM requires that specific signals be set before operation can be performed. As a result of these requirements, the following modules are required:

- EREAD - Provides the reading of EEPROM data.
- EWRITE - Provides the writing of data to EEPROM.
- EERASE - Provides the capability to erase sections of EEPROM so that either BITE information or calibration parameters can be written.
- EDELAY - Allows for a settling of the discrettes used to control the EEPROM circuitry.

The RAM scrubbing function is performed in background (Time between Interrupts) by the WBSCRUB mode. This module reads from designated RAM locations and then writes the data back into the same location. This reading and then writing removes any single bit memory errors which accumulate in the RAM locations.

6.6 IMU RM

Since the Orbiter carries three IMU's independently accomplishing the same functions, a failure detection scheme must be present and a means of selecting the best available data must also be present. Selecting the best available data is known as selection filtering. Checking for and identifying failures is known as fault detection and identification (FDI). Flight software (OPS 1, 2, 3, 6, and 8) executes these functions in IMU RM to insure good data are supplied to all users of IMU data. IMU RM is not active in MM101 before REFS_RDY is set to TRUE (at T-4 min). IMU RM uses a combination of the IMU serial data and the IMU BITE information to accomplish its objectives. The reason for skewed IMU's, as already discussed, is to facilitate the isolation of IMU failures when only two IMU's are functioning. When all three IMU's are operational, RM provides a simple two on one voting scheme to isolate failures. The RM software is divided into two distinct and separate areas, attitude RM and velocity RM. Backup flight system software (BFS) differs from the primary software and will be discussed in section 6.6.3.

The ability of the onboard redundancy management (RM) software to identify an IMU failure depends upon the decision functions, instrument noise, identification logic, and the relative platform geometry of the IMU's. Of these four factors, only the relative geometry, or skewing of the IMU's can be modified.

During the early days of the space shuttle, the ascent and entry phases of flight used a skewing orientation to protect against certain types of critical failures. To meet the initial platform gyro compassing position accuracy requirements, a skewing orientation referred to as the " $\sqrt{6}$ orientation" was used. This relationship allowed the factory determined calibration parameters to be used for platform compensation.

The most optimum orientation for 3 level IMU RM is known as the " $\sqrt{6}$ orientation" and is the current skewing used for all phases

of flight. For many years, the optimum skewing for 2 level IMU RM was the " $\sqrt{2}$ orientation". If an IMU was considered unrecoverable, the $\sqrt{2}$ RESMMATS were to be invoked for the remainder of the flight.

More studies in 1990 and 1991 revealed that the $\sqrt{2}$ skew did indeed provide longer Z-axis failure coverage than the $\sqrt{6}$ skew, but did not provide X and Y axis coverage as well as the $\sqrt{6}$ skew. The decision was made in the Spring of 1992 to use the $\sqrt{6}$ skewing orientation even after suffering the loss of one IMU. The shorter and more optimal failure isolation capabilities of the $\sqrt{6}$ skew will be taken advantage of for entry by resetting RM thresholds at deorbit TIG minus 70 minutes with an IMU-to-IMU align. This provides the best possible RM coverage during the entry time frame and landing is accomplished before the $\sqrt{6}$ skew RM coverage falls below that which the $\sqrt{2}$ skew could provide.

The requirements that led to the development of the $\sqrt{6}$ REFSMMATS are as follows. The space shuttle requires that the IMU's be two-fault tolerant. For the three IMU's, the first failure can be identified by a simple two-on-one voting logic. The $\sqrt{6}$ skew maximizes the separation of the platform axes for the three IMU set. This skewing arrangement is based on three constraints:

1. IMU pairs have identical skewing.
2. Maximize the probability of identifying a randomly oriented X-Y plane failure.
3. Place the line of ambiguity (LOA) in a direction that will enhance the identification of the correlated X-Y plane failure

The skewing analysis resulted in the following $\sqrt{6}$ REFSMMAT:

$$M = \begin{vmatrix} 1/4 & -3/4 & \sqrt{6}/4 \\ -3/4 & 1/4 & \sqrt{6}/4 \\ -\sqrt{6}/4 & -\sqrt{6}/4 & -1/2 \end{vmatrix}$$

The old $\sqrt{2}$ REFSMMAT is shown here:

$$M = \begin{vmatrix} (2+\sqrt{2})/4 & -(2-\sqrt{2})/4 & -1/2 \\ -(2-\sqrt{2})/4 & (2+\sqrt{2})/4 & -1/2 \\ 1/2 & 1/2 & 1/\sqrt{2} \end{vmatrix}$$

The main reason for optimizing the skewing arrangement of the IMU's is to set up the LOA for nominal entry to be sensitive to vertical and downrange errors. If the LOA falls within 30 degrees of the vertical downrange plane, then RM cannot guarantee failure isolation. Therefore, using the proper skewing places

the Z-axis of the IMU platforms in the orbit plane for entry and places the LOA crosstrack to minimize the effects of unidentifiable IMU failures on the critical navigation state components of vertical and downrange position and velocity.

The Z-axis now lies in the critical plane for navigation performance and is made as sensitive as possible for failure identification by projecting Z-axis failures equally into the X-Y plane and along the Z-axis of the other IMU's.

6.6.1 Attitude RM

The first function of attitude RM is to determine which IMU's are candidates for RM processing. Criteria for candidacy depends on whether an IMU alignment is in progress or not. If an on-board align is being performed then the IMU is a candidate if it is not crew deselected, not failed by RM, not slewing and not commfaulted.

If an on-board align is not being performed then the IMU is a candidate if it is not crew deselected, not failed by RM, and not commfaulted. If any of the three criteria are not true for a given IMU, RM will not operate on the data from that IMU.

6.6.1.1 Selection Filter

Once the number of candidate IMU's has been determined, the selection filter (SF) must then decide which IMU should be used as the attitude source. If three IMU's are candidates for RM when an alignment is executed, the previously selected IMU remains chosen unless it is no longer available. If the latter is true, the lowest numbered available IMU becomes the selected IMU. Once the selected IMU ID is determined, no further attitude RM processing is performed until the on-board alignment is complete.

If three IMU's are candidates for RM when an alignment is not being executed, the software determines the errors between the actual relative orientations of the platforms and the ideal relative orientations. This is accomplished by the following quaternion equation.

$$Q_AD_{ij} = Q_SM_{i_SMj_I} * (Q_SM_{i_SMj})^*$$

where: $Q_SM_{SM_I}$ is the ideal transformation between two platforms as determined from the REFSMMATS

Q_SM_{SM} is the actual transformation between platforms as determined by the gimbal angles (the * after this quaternion indicates conjugation)

The vector portion of Q_AD_{ij} is then dotted with itself to form a squared attitude error value. The software next checks the squared attitude errors, looking for the maximum value. The two IMU's which produced the largest squared attitude error are obviously the farthest apart, so the IMU which is not a part of this maximum value is the middle value IMU and is used as the attitude source.

When only two IMU's are candidates, the SF software computes the misalignment between the two IMU's by computing the squared attitude error as described above. This error is then compared to the attitude data good threshold (1° during ascent and orbit operations, 2.5° during entry). If this threshold is violated, the attitude data good flag is set to invalid, the fault identification portion of RM speeds up from a processing rate of 1/16 Hz to a rate of 1.04 Hz to try to isolate which of the two IMU's has the failure. The SF at the two-IMU level does not change the selected IMU attitude source. If the attitude source remains a candidate during the downmode from three to two IMU's, that IMU will continue to be the attitude source. Otherwise, if the selected IMU was no longer a candidate after the downmode, the lowest numbered candidate IMU would become the attitude source.

The attitude SF operations result in two additional downmodes that function only during OPS 2 and 8. If the attitude data good flag is violated as described above, the digital autopilot (DAP) will downmode to free drift indicating no IMU data are available. Also, if only one IMU is a candidate and that IMU has a BITE fail indication, the SF will indicate that the attitude data are bad. Whenever the attitude data good test is violated, the software switches from its normal 1/16 Hz fault isolation rate to a 1.04 Hz rate.

If there are no IMU candidates available, the selected IMU ID is set to the previous value, the attitude data good flag is set to invalid, the ADI attitude flag is set to invalid (OPS 2 only) and the attitude fault identification flag is set to the slower rate.

6.6.1.2 Fault Detection

The second major task of attitude RM is to provide fault detection (FD) capability for the three IMUs. If all three IMUs are available, the three squared attitude errors computed in the SF are compared to the square of the three-level FD threshold. The FD threshold, varies with the different phases of flight, but is generally keyed to the last alignment, whether it is the preflight align or on-orbit alignment. If the squared attitude error for a given IMU pair exceeds the squared FD threshold, a miscompare flag is set which is then sent to the fault isolation software.

If only two IMUs are candidates for RM, the first function of the software is to compute the new FD threshold. As discussed above, this threshold varies with the different flight phases. Once this threshold is computed, it is squared. The next function is to compute the squared attitude error between the remaining two IMUs. The calculation of the squared attitude error differs somewhat from the calculations done in the SF. The equation used to compute the magnitude of the misalignment error is:

$$Q_MAG = Q_SM_iSM_j_O * (Q_SM_iSM_j)^*$$

where: Q_SM_SM is the actual transformation between platforms as determined by the gimbal angles

$Q_SM_SM_O$ is the quaternion describing the relationship between IMU_i and IMU_j at the completion of the latest alignment. Unlike the ideal transformation (Q_SMSM_I) formed from the REFSMMAT's, Q_SMSM_O is formed from gimbal angle reads of both IMU's at the time of completion of the alignment.

The vector portion of the attitude error (Q_MAG) is squared (dotted), then compared with the squared FD threshold. If the threshold is violated, an attitude miscompare flag is sent to the fault identification software.

The following sim case illustrate the difference between how RM calculates IMU attitude errors at the three level and at the two level. During the sim, IMU 1 was commfaulted due to an FF 1 problem at the GPC. During the time that IMU 1 was commfaulted, IMU 3 developed a high drift rate and was manually deselected before it broke the RM threshold. Subsequently, the drift on IMU 3 stopped and it was aligned to IMU 2 via an IMU to IMU align (IMU 1 was still commfaulted during the align). Later a port mode was executed on string 1 (IMU 1 was recovered) while IMU 3 was still deselected. Although all three IMUs showed almost no relative error between each other, IMU 1 and IMU 2 went into an attitude dilemma and IMU 2 was failed by RM because it had a

slightly higher Redundant Rate. The crew then reselected IMU 3 and IMU 2 because there was no evidence of any miscompares between any of the IMUs. At the three level, RM did not fail any of the IMUs.

The reason IMU 1 and IMU 2 went into dilemma is because of how RM calculates the attitude error at the two level. At the completion of the IMU to IMU align, IMU 1 was commfaulted. Therefore, the snapshot of the gimbal angles used to compute Q_SM_SM_O for IMU 1 was inaccurate. At the two level, this inaccuracy was large enough to cause a dilemma. At the three level, Q_SM_SM_O is not used and because there was no error between Q_SM_SM and Q_SM_SM_I for any of the three IMUs, no IMUs were failed by RM.

This next sim case provides another example of how 2 level RM processing can lead to results which differ from what one might expect. During the sim, IMU 1 was recovered by aligning only IMU 1 to IMU 3. At that time, IMU 3 had about a 0.1° misalignment from IMU 2. IMU 1 failed some time later, leaving IMUs 2 and 3. At the two level, IMU 2's drift rate began to increase. As the IMU's Gyro $\Delta 2/3$ approached the RM limit of .75, the flight control team expected IMU 2 to fail out. However the gyro Δs grew to 0.8° and IMU 2 did not fail as expected. Recall that the transformation between IMUs 2 & 3 at the end of the last align was already misaligned by 0.1°. Even though IMU 2 and 3 disagreed by 0.8° via ground computations (which use a method to determine attitude errors similar to the three level method of the onboard software), IMU misalignments reflected in Q_SM_SM_O which was snapshot at the end of the last align introduced uncertainties as to when the value of Q_MAG would in fact reach the RM threshold. This is also an example of why it is a good practice to align the third IMU when recovering only one, unless there are other factors preventing aligning the third IMU.

In both cases, the recovery required performing another alignment of all good IMUs to reset Q_SM_SM_O.

6.6.1.3 Fault Identification

The third and final function of the attitude RM software is the fault identification (FI) operation. If all three IMU's are candidates for RM and an attitude miscompare was detected by the FD software, the FI software attempts to isolate which IMU is responsible for the miscompare. The software checks the number of miscompare flags and if only two flags are set, it records the IMU number common to the two flags (recall there is a miscompare flag for each IMU pair) and starts a counter. No failure is annunciated unless there are three consecutive passes through the FI software which identified the same IMU as the cause of the miscompare. If all three flags show miscompares, the FI software is exited and the failure will never be isolated. In other

words, if two or three IMU's have attitude problems, the software cannot isolate the failures and no failures are declared. If the FI counter equals three (three consecutive votes against an IMU), the IMU failure will be annunciated with a fault message to the crew and a down arrow next to the appropriate IMU number on the crew displays. This module of the software runs at a 1/16 Hz rate, so an attitude failure will require 48 sec to isolate (see Table 6-XXXVIII).

| IMU ATTITUDE FDIR | | Declare Failure | Declare Dilemma | RGM Failure | Processing Rate |
|-------------------|------|-----------------|-----------------|-------------|-----------------|
| Three Level | | 48 sec | ----- | ----- | 1/16 Hz |
| Two Level | Slow | 48 sec | 64 sec | 96 sec | 1/16 Hz |
| | Fast | 3 sec | 4 sec | 6 sec | 1.04 Hz |

Table 6-XXXVIII, ATTITUDE RM RATES

When only two IMU's are candidates and a miscompare flag is set, the two level FI software is active. The first check made by two-level FI is whether a BITE failure is present on either IMU. If one and only one IMU exhibits a BITE failure, then the FI counter is incremented and the IMU with the BITE fail is recorded. No further FI processing will be done this FI cycle. If no BITE fails are present, the software forms four attitude error components. Since the X- and Y-axes have a common gyro and the Z-axis is a different gyro, an attitude error component is formed for the XY-plane and the Z-axis separately. The XY-plane and Z-axis errors are computed in the coordinate system of one of the two remaining IMU's, then transformed to the coordinate system of the other candidate IMU. This is accomplished by first multiplying the following quaternions.

$$AEZ_I = Q_{SM_iSM_j_0} * (Q_{SM_iSM_j})^*$$

Only the components of Q_{SMSG_0} and Q_{SMSG} which, when multiplied together, produce the fourth element (Z) of the resultant quaternion are actually multiplied. This fourth element is then squared to produce the Z-axis squared error in IMU I coordinates. This same methodology is used to produce the Z-axis errors in IMU J coordinates.

$$AEZ_J = (Q_{SM_iSM_j_0})^* * Q_{SM_iSM_j}$$

Again, these errors are squared. As a shortcut, the software then subtracts the Z-axis contributions from the total error squared which was computed in the two-level FD software. This subtraction then leaves the XY-plane error squared. Since the Z-axis components will differ in the two different coordinate systems, the two different results after subtraction already reflect the proper coordinate system. No further transformation is needed. One exception to the source of the total attitude error squared exists. If the FD logic detected an attitude data

good violation, the FI software runs at 1.04 Hz versus the normal 1/16 Hz. Since the attitude error squared is needed at the faster rate for this case, it is computed in the FI software (same methodology as FD). The four error components are then compared to the square of the FI threshold. Similar to the FD threshold, the FI threshold varies with the OPS phase and is a function of time. If three of the four squared attitude errors exceed the FI threshold, the IMU corresponding to the attitude error component that does not exceed the threshold is considered failed after three consecutive violations. The reason for this type of logic is that a single gyro failure will project its error into both the XY-plane and Z-axis of the other IMU, but it will not project onto any other component of itself. If only one, two, or all four squared attitude errors exceed the threshold, the failure is not identifiable. If the FD logic continues to indicate a fault for four consecutive passes but the FI software is unable to isolate the failure, the FI software will indicate that the IMU's are in dilemma, at which time the RGM outputs will be used to resolve the dilemma (at a high or low rate, corresponding to whether the ATT-DG threshold was exceeded).

As previously stated, in OPS 2 and 8 the BITE fail flags are used for fault isolation of the IMU when only one IMU is a candidate for RM processing. If the BITE condition persists for three consecutive passes through FI logic, the last remaining IMU is failed; however, the auto pilot is downmoded to free drift, within one SF cycle.

Note: If both IMU's in dilemma exhibit a BITE, then RGM outputs will be used to resolve the dilemma.

6.6.2 Velocity RM

Velocity RM is also divided into the two major functions of selection filtering and FDI.

6.6.2.1 Selection Filter

The velocity selection filter determines the number of candidate IMUs (or the ID # of candidate IMUs at the two and one level) and stores the information in the IMU velocity status vector. The IMU velocity status vector is then sent to FDI. If an on-board alignment is being performed, the velocity status vector indicates zero IMUs available and is then sent to FDI.

The SF, in addition to determining which IMUs are candidates and which data should be used, also provides a separate candidate status flag (called the navigation selection filter control vector) to the navigation software. An IMU is available to the navigation software (state vector module) if it is not crew deselected, not failed by RM, not commfaulted and has no BITEs

(this last check is used only in OPS 2&8 at the one level). If an on-board alignment is being performed, the above candidacy criteria plus if the IMU is not being slewed are used to determine if an IMU is available to the NAV software.

The IMU velocity SF also provides to NAV a selected velocity vector in M50 coordinates. These total velocities are rectified every 8 sec to limit the errors sent to NAV. Rectification involves calculating the difference (BIAS) between the total velocities from each IMU and the selected total velocity vector. This BIAS for each IMU is then added to the total velocities for each IMU every 8 sec. This results in agreement between the total velocities from each IMU and the selected total velocity every 8 sec. The BIAS (selected minus total velocity) computed at a given rectification cycle is applied to each IMU's total velocity until the new computation on the next rectification cycle (see Figure 6-39).

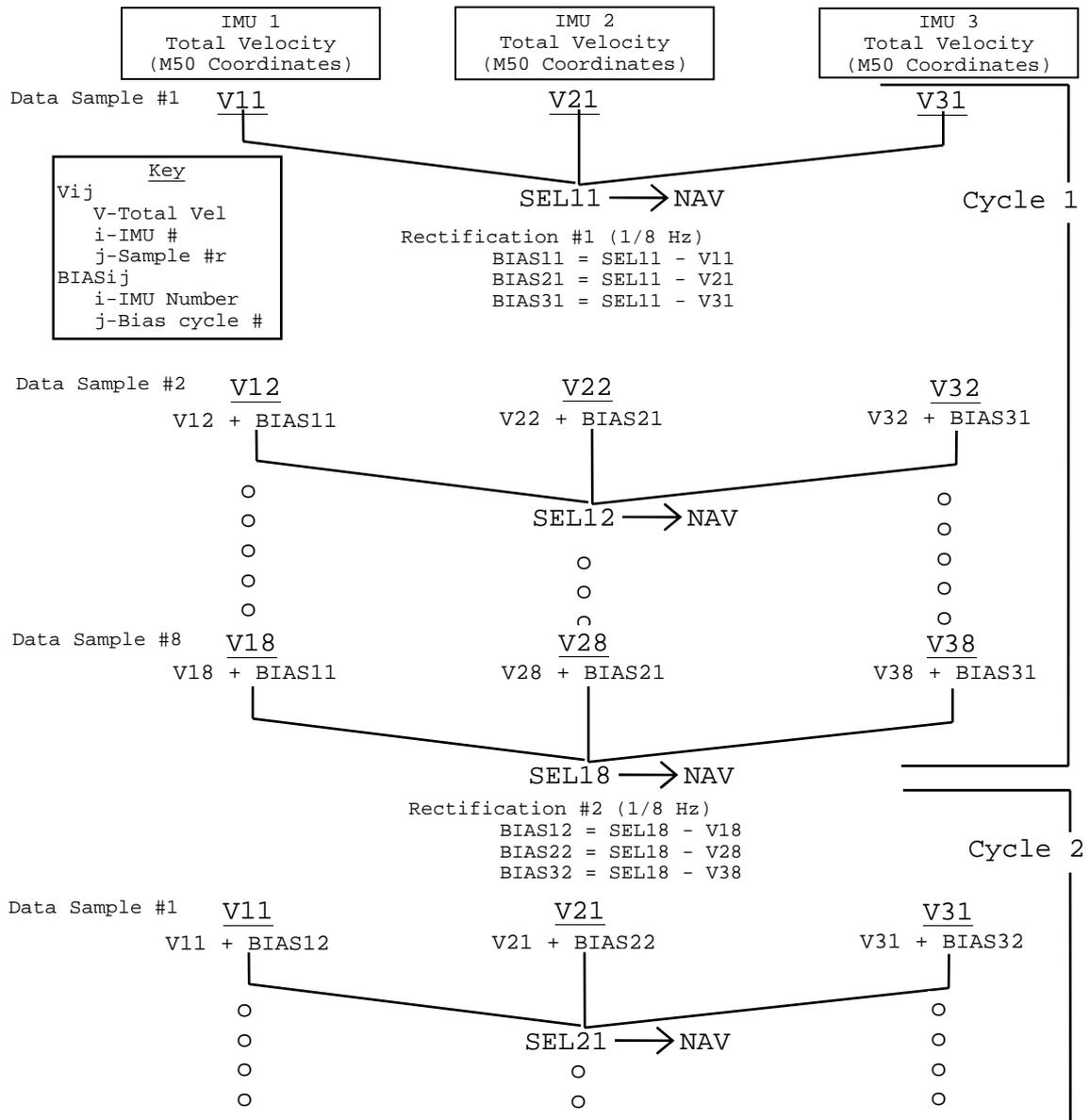


Figure 6-39, Velocity SF Rectification

If three IMUs are candidates, the SF middle value selects the total velocity vector from the three biased M50 velocity vectors. The middle value scheme is done on each component (X, Y, Z) which could result in one component selected from each IMU (in the extreme).

When only two IMUs are candidates, the SF first decides if the data are usable and, if it is good, averages the total velocity inputs from the two IMUs. To check for good data, the software computes a delta velocity between each consecutive RM cycle. The delta velocities for each IMU are then subtracted from each other

to form a delta velocity error vector between the two IMUs. This error vector is then dotted with itself to produce a squared error magnitude. The squared error magnitude is compared to the velocity data good threshold. If the threshold is exceeded, the SF looks for BITE fail flags on each IMU. If only one BITE is indicated, the SF will indicate to NAV that only one IMU is available and NAV will use the velocity data from the IMU that does not have a BITE failure. This downmoding to one IMU is temporary in nature and designed to protect NAV from transients until the FDI logic can permanently isolate the failure. If neither IMU indicates a BITE failure, the data good logic attempts to isolate which IMU caused the data good violation. The delta velocities in M50 coordinates are transformed back to IMU coordinates by multiplying the vectors by the REFSMMATs. The delta velocity of IMU_i is then transformed into IMU_j coordinates then subtracted from the delta velocity of IMU_j. This delta velocity difference between the two IMU's is then separated into two squared components, an XY-plane component and a Z-axis component. The X-component of the delta velocity difference is squared and added to the square of the Y-component to form the XY-plane squared component. The Z-axis component is simply squared to form the squared Z quantity. The same process is performed in IMU_i coordinates culminating in the production of an XY-plane squared error and a Z-axis squared error in IMU_i coordinates. These four components (two in IMU_i coordinates and two in IMU_j) are then compared to a velocity data good identification threshold. Similar to the attitude voting logic, a three out of four voting scheme is used. If three of the error components exceed the threshold, RM temporarily downmodes to the one IMU available case for NAV purposes. The IMU with the error component that does not exceed the threshold is considered bad (same logic as described in the two-level FI section). If one, two, or four of the components exceed the threshold, or if both IMUs indicate BITE failures, or if no data good violation has occurred, the software averages the total velocities from the two good IMUs and passes the averaged vector to NAV as the selected total velocity.

During OPS 2 or 8, if the navigation selection filter control vector indicates no IMUs are available, then the selected total velocity in M50 coordinates is set equal to its previous value and the velocity selection filter is exited. During OPS 1 & 3, if the navigation selection filter control vector indicates no IMUs are available, then the navigation selection filter control vector is set equal to its previous value, and the selection filter process is continued.

Last but not least, during OPS 3 only, the IMU velocity selection filter automatically deselects an IMU when its associated bypass flag is set. An IMU will not be automatically deselected in OPS 3 if it is the last available IMU.

6.6.2.2 Fault Detection/Identification

Two sets of computations are performed in velocity FDI before other processing occurs. As in the SF, a delta velocity vector in M50 coordinates is computed for each IMU over each computation cycle. Also, the FD threshold is computed and squared. The velocity FD threshold varies with the different OPS phases.

If all three IMUs are candidates for RM, the first step of FDI is to transform the delta V vectors from M50 to IMU coordinates. These delta V vectors are then transformed into another IMUs frame and differenced from that IMUs measured delta V as follows.

$$\begin{aligned}EDV_{21} &= DV_2 - (Q_{SM_2SM_1})(DV_1)(Q_{SM_1SM_2}) \\EDV_{31} &= DV_3 - (Q_{SM_3SM_1})(DV_1)(Q_{SM_1SM_3}) \\EDV_{32} &= DV_3 - (Q_{SM_3SM_2})(DV_2)(Q_{SM_2SM_3})\end{aligned}$$

where: EDV is the delta V error
DV is delta velocity
Q_SMSM is as defined in attitude RM

The magnitude squared of the delta velocity error between IMUs is computed by dotting the EDV vectors with themselves. If two of the squared errors exceed the threshold on three consecutive passes through the FDI software, the IMU common to the two errors is failed and deselected.

If only two IMUs are available for RM processing, the delta V's are transformed from M50 to IMU coordinates and the EDV vector is formed for the two IMUs using the same methodology as the three level FDI. The EDV vector is dotted with itself to form the squared velocity error magnitude which is then compared to the velocity FD threshold. If the threshold is violated, FDI will attempt to identify the failed IMU; otherwise, FDI will be exited during this cycle. The first step after a fault has been detected is to check for a BITE failure. If neither IMU has a BITE failure, the software must check further to identify the failure. If, however, only one IMU indicates a BITE failure, that IMU will be failed by RM and will be deselected after three consecutive passes through FDI with the same conditions. If BITE was not able to identify the failure, the software forms four velocity error components as follows:

$$VEXY_J = EDV_JI_X^2 + EDV_JI_Y^2$$

$$VEZ_J = EDV_JI_Z^2$$

$$EDV_IJ = (Q_SM_I SM_J)(EDV_JI)(Q_SM_J SM_I)$$

$$VEXY_I = EDV_IJ_X^2 + EDV_IJ_Y^2$$

$$VEZ_I = EDV_IJ_Z^2$$

These error components are then compared to a velocity FI threshold and the three out of four voting scheme is used. If, on three consecutive passes through FDI, the same IMU is declared bad, that IMU will be failed by RM and deselected. As in previously described voting schemes, if the failure cannot be identified but the error exceeds the fault detection threshold, the IMU dilemma flag is set and remains set until the crew takes action or the fault is identified. All of velocity FDI runs at 1/16 Hz (see Table 6-XXXIX).

| IMU VELOCITY FDIR | Declare Failure | Declare Dilemma | RGM Failure | Processing Rate |
|-------------------|-----------------|-----------------|-------------|-----------------|
| Three Level | 48 sec | ----- | ----- | 1/16 Hz |
| Two Level | 48 sec | 80 sec | ----- | 1/16 Hz |

Table 6-XXXIX, VELOCITY RM RATES

Note: If both IMU's in dilemma exhibit a BITE, then RM processing is exited, with no further FI evaluation performed.

6.6.3 BFS RM

The BFS selection scheme is very similar to the primary flight software when all three IMUs are available. The BFS will midvalue select the attitude source via the same methodology discussed above. The software will also midvalue select the velocity vector on a component by component basis and provide rectification, as previously discussed. The major difference in BFS is that there is no FDI software. Downmoding to less than three IMUs will occur only if there is a CF, the IMU good discrete goes to zero (hardware BITE), or if the crew deselected the IMU in the BFS. Primary flight software RM failures or deselections are not recognized by the BFS.

Thought the BFS will not use an IMU with the IMU good discrete set to bad, there is a way to force the BFS onto IMU 3 (only) if it has a H/W BITE. The following steps can be performed to accomplish this:

1. Deselect IMUs 1 & 2 i the BFS
2. Pro the BFS to OPS 0
 - If MM301 - 303: mode to OPS 301 then OPS 0
 - If MM304 - 305: mode BFS to STBY-HALT to STBY-RUN
 - If Ascent: mode BFS to 106 then to OPS 0 then to OPS 3

It should be noted that when the BFS uses an IMU with a H/W BITE, it will use accelerometer data from that IMU but the ATT_PROC inner loop software will use only RGA data for attitude control. In addition to this, the ATT_PROC outer loop freezes which results in bad guidance needles if the BFS is engaged. The BFS is not considered no-go, but the crew should be warned that even though BFS NAV is good, upon engaging, they should not fly guidance and that a GCA of some type will probably be required.

If downmoding does occur, the lowest number IMU available will be used for both attitude and velocity. If subsequent upmoding occurs (i.e., the CF is cleared, the BITE clears, or the crew reselects), the BFS will not return to a midvalue scheme. The IMU currently in use will continue to be used and the newly reinstated IMU will only be used again if the other two IMUs are not available. In other words, the first IMU to fail will be the last IMU reselected. If a second IMU subsequently is deselected by BFS and then is reinstated, it will be used prior to the IMU that was first deselected. If all three IMUs have been deselected at some time, the BFS still maintains knowledge of the order they were deselected and will select the IMUs based on the first fail - last selected logic. Anytime the "first failed" IMU is being selected when another IMU becomes available, it will remain selected only if the reinstated IMU is a lower numbered IMU. In other words, if IMU 3 is the first failed IMU and the last available IMU when IMU 2 or 1 become available, the BFS will remain on IMU 3 (the first failed). However, if IMU 1 is the first failed IMU and the last available IMU when IMU 2 or 3 become available, the BFS will use IMU 2 or 3. In this latter case, the crew can force the BFS to use the first failed IMU by deselecting the recovered IMU.

Another key point about the BFS is that the BFS will never automatically return to a midvalue scheme if it has ever downmoded. The crew can force the BFS to MVS after it has downmoded by 'PRO'ing the BFS through OPS 0. If an IMU is downmoded in OPS 1 and subsequently recovered (prior to the OPS 3 transition of a nominal mission), the BFS will MVS again in OPS 3 (due to the fact that the BFS was in OPS 0 during orbit). In a TAL scenario, when the BFS is moded directly from OPS 1 to OPS 3, the BFS will not return to an MVS scheme.

6.7 IMU IN-FLIGHT FAILURE HISTORY

As of March 1999, the Shuttle program has experienced 15 in-flight IMU failures and 2 prelaunch failures. (Prelaunch failures are included here because they were seen during GNC

console operations, even though they are not technically an in-flight failure.) No IMU in-flight failures have occurred during the ascent time frame. Ten of the failures were with the old KT-70 IMUs.

6.7.1 KT-70 Flight Failures

Note that the KT-70 IMUs had two accelerometers. An X-Y accelerometer and a Z accelerometer.

6.7.1.1 STS-3

On the first day of the mission, IMU 3 null bias acceleration shifted by $-321\mu\text{g}$ in the Y axis. The bias was updated and remained stable for the remainder of the mission. The probable cause of the shift was particle contamination within the accelerometer that has disturbed the pendulum and caused a small shift from the nominal null position, thus affecting both the X and Y outputs. Another possible cause would be the accelerometer flex tape or transformer intermittence. The IMU was removed for testing.

The test results verified the bias shift and the up axis drift noted during prelaunch was confirmed to be 0.075 deg/hr. A latching noise was noted in the coil-restoring current. After extensive testing at the vendor, the X-Y accelerometer bias shifted back to the former value and the noise was no longer evident. Several attempts were made to induce the latching noise by temperature cycling and vibration tests, but were unsuccessful. Failure of the accelerometer pendulum lower cement joint, the accelerometer pre-amplifiers, or the transformer were all considered possible causes. The IMU was disassembled for component test and analysis, but no firm conclusions were made.

6.7.1.2 STS-6

Post entry blackout, communications were re-established and an 'IMU BITE/T' message was noted to have occurred twice, one second apart. Post flight analysis indicated that the velocity limit fail flag was set for IMU 3. This IMU BITE/T message occurred near transition to major mode 304.

Preflight and post flight analysis of data indicated that the IMU 3 Z accelerometer had a 5000 micro g noise. There is no accelerometer specification limit on noise. This noise level is not considered excessive although it is higher than experienced on other flight accelerometers.

6.7.1.3 STS-7

Shortly after deploying the first satellite, several IMU 1 BITE/T messages were received. Subsequent investigations revealed that the messages were triggered during the resolver limit determination for the outer roll axis. This was caused by the software algorithm that is used for the 1X-resolver-angle minus the whole angle difference determination. The whole angle uses resolver offset compensation whereas the 1X angle is uncompensated. The compensation plus the harmonic error at a roll gimbal angle of about 120 degrees caused the limit to exceed the one degree threshold and produce the BITE/T messages. This harmonic error in the outer roll axis is slightly higher than the error on the other IMUs used, but it is not believed to be a hardware problem. During the remainder of the mission, the IMU 1 BITE/T annunciation was masked using a software read/write procedure and IMU 1 performed nominally for the remainder of the mission.

Post landing, CR 59864 was approved for a short term fix. The CR would change the resolver limit threshold from one degree to 1.5 degrees starting with STS-8.

6.7.1.4 STS-8

During the mission, IMU 2 went to standby. The IMU was reactivated, realigned and reselected for redundant set operations. IMU 2 performed nominally for the remainder of the mission.

It was determined that the most probable cause of the IMU dropping out of "operate" was a transient failure in the command path from MDM FF2 to the IMU. The specific path is question can almost certainly be narrowed to the single "operate" discrete that originates in the MDM (from the GPC software). Although the discrete originates in the MDM, it is also plausible that the transient occurred within the IMU side of the interface. The entire discrete interface was investigated thoroughly during post flight testing at KSC. Extensive closed-loop and open-loop tests were performed to verify the command path, including flexing of interconnecting cables. The open-loop tests were structured to maximize detection of any MDM transient anomalies during a three hour interval as the MDM warmed up from a cold start.

Experience has shown that most transient conditions are extremely difficult to repeat even during extensive environmental testing at the vendors. Given the low probability of repeating the anomaly at even the MDM or IMU, and the inherent redundancy with three IMUs, it was decided that no further action was warranted at that time.

6.7.1.5 STS-9

During the mission, several IMU 1 BITE/T messages occurred. After AOS, the BITE had isolated the problem to a failed DC/DC No. 1 Card in the IMU 1 power supply. The DC/DC No. 1 Card provides plus/minus 15 volts and 5 Volts internally to the IMU. Fifteen minutes after the last BITE announcement, two more BITEs were announced and RM failed IMU 1. The crew power cycled the IMU and did an I/O reset. The IMU was unrecoverable and IMU1 was powered off for the remainder of the mission.

Post mission, failure analysis disclosed a faulty capacitor in the +5 volt power supply. The testing also showed that the capacitor failure was humidity related (degradation of leak resistance) and was associated with capacitors that have a specific lot date code. Since this was a generic problem with the specific lot date, the suspect IMUs on the upcoming mission were removed and replaced with IMUs that did not have the suspect capacitors.

6.7.1.6 STS-32

During the mission, IMU 1 exhibited seven occurrences of erratic Y-axis accelerometer outputs. The accumulated Y-axis velocity indicated a negative acceleration when it should have been indicating a positive accelerometer bias. IMU 1 was eventually failed by RM and then manually reselected. It was noted that though the IMU was failed by RM, no IMU hardware BITEs were announced. For the remainder of the mission, the IMU performed nominally.

Post flight analysis revealed that the inner roll resolver angles decreased at the times that corresponded to the Y-velocity upsets. Due to the time synchronization of the perturbations on both the Y accelerometer and the inner roll, it is most likely that the perturbations have something in common. A common failure mode for the accelerometer and the inner roll resolver was not determined, but the vendor did locate several locations on the circuit board where the two are located close together.

6.7.1.7 STS-40 (Prelaunch)

During the launch attempt on June 1, 1991, IMU 2 (S/N 023) failed the preflight calibration (PFCAL) for X and Y accelerometer shifts of greater than 2 sigma. (Recall that for a KT-70, the X-Y accelerometer is a dual axis unit.) The shifts were 4.15 sigma (KOY), 3.47 sigma (KIYH), -2.24 sigma (KOXH), and 5.54 sigma (KOYH) which violates the prelaunch cal criteria of 2 sigma (and 5 sigma for KOYH). The calibration sigma values are an indication of large shifts in these terms from the previous calibration (done on May 8, 1991.) The OMRSD cal criteria requires a contingency PFCAL to be performed if any terms exceed

the 2 sigma criteria, and are less than 5 sigma. The second PFCAL was performed, realizing that the 5 sigma criteria would have to be waived for the KOYH term, in order to be GO for launch if all terms were less than the required 2 sigma. However, all terms were not less than the desired criteria. Although not required per the OMI, a third PFCAL was performed to obtain more data on the stability of the IMU. Engineering analysis of the data indicated a problem with the stability of the accelerometer, and it was decided to scrub the launch and R&R the IMU. The IMU was removed and sent to the JSC ISL for testing. The problem was duplicated and isolated to a defective computer interface card (CIC) in the IMU. The failed CIC resulted in one axis sensing an input rate at the same time another axis was transitioning through a zero rate value (ref. Flight Problem Report No. STS-40-V-01 in data book). The IMU PFCAL data can be referenced in the STS-40 Post Flight Report, Appendix A.

6.7.1.8 STS-41

During the planning shift preceding the STS-41 entry, IMU 1 (S/N 7) was failed by RM for a violation of the velocity threshold. The Z-axis acceleration output shifted in a series of steps and broke the OPS 2 RM limit of 951 micro-g's. (Recall that that for a KT-70, the Z accelerometer is a single axis unit.) The output continued to grow for eleven minutes before returning to the nominal output of zero acceleration. The IMU 1 output repeated this signature over the next 17 hours with 70 separate violations of the OPS 2 RM limit. The largest magnitude seen was 21,000 micro-g's. The maximum OPS 3 RM threshold is 10,000 micro-g's. The IMU's behavior could not be correlated to any external factor.

Based on the observed performance, IMU 1 was declared failed by the GNC flight control team. The Mission Evaluation Room engineers concurred with this position. Plans to re-skew the remaining IMU's to optimize for entry were made, but not executed due to the short amount of time prior to entry. (Previously the skewing between two IMUs, if the third was failed, would be updated to the $\sqrt{2}$ orientation for entry.) IMU 1 was deselected in the BFS and not reselected in the PASS.

The series of transient outputs persisted over a substantial period of time from the initial RM violation to Entry Interface (over 17 hours). Even though the IMU appeared to perform nominally during the 30 minutes from E1-5 minutes until touchdown, it must be noted that while on orbit there were several occurrences of 30 minutes or more when the IMU appeared to be performing nominally. It is probably a coincidence that the failure did not recur during that 30 minute period.

The IMU 1 failure signature was not similar to any of the previous in-flight failures (STS-3 and STS-32), nor has it ever been seen during ground calibrations or tests. The IMU failure experienced during STS-32 was similar only in that it was a transient accelerometer output which caused the RM violation.

During that flight, the transient was only present for a total of 10 minutes and was accompanied by a transient resolver output. The cause of this failure has been attributed to moisture contamination on an internal electronics card. The STS-3 failure was also an accelerometer problem, but it was a single transient that never returned.

Post-flight testing of IMU 1 by the manufacturer, Kearfott Guidance and Navigation, measured accelerometer outputs up to a maximum magnitude of 0.5 g's (500,000 micro-g's). The cause of the Z axis accelerometer failure has been traced to a short internal to the IMU platform. Inspection of the platform electronics at Kearfott revealed evidence of a short (burn mark) between +28 VDC heater power and Z accelerometer restoring coil low. A gray substance that appeared to be vaporization residue was also noted. Rockwell/Downey and Kearfott engineers agree that the failure is non-generic in nature.

6.7.1.9 STS-44

The mission was terminated early because of an IMU 2 failure. This IMU failed due to a hard-over Z-axis accelerometer output of approximately 15 g's. As a result of the anomaly, two software BITES were annunciated; velocity limit fail and redundant rate fail. The velocity limit fail BITE occurred due to the bad accelerometer data violating a reasonableness constraint. The redundant rate fail BITE resulted from an erroneously sensed redundant rate gyro output. The redundant rate gyro was operating properly; however, the bad accelerometer data fed into the redundant rate gyro calibration equations (see below for details) by 5.7 degrees/hour. This violated the redundant rate BITE limit of 0.7 degrees/hour.

For OPS 1, 2, and 3, the redundant rate mass unbalance drift compensation equation is as follows:

$$DGR = \frac{(DIR)(DVTX) + (DOR)(DVTY) + (DSR)(DVTZ)}{[T(TORQ)]G_o}$$

where: DGR - redundant rate gyro mass unbalance drift compensation (G-sensitive drift)
 DIR, DOR, DSR - I-Loads for redundant rate gyro g-sensitive compensations for input, output & spin axes
 DVTX, DVTY, DVTZ - ΔVx (ΔVy, ΔVz) over 1 torquing cycle (960 msec)
 T[TORQ] - time duration of 1 pulse cycle (960 msec)
 G_o - standard acceleration due to gravity

- assume: 1. DVTX = DVTY = 0
 2. Working with G instead of ΔV (T[TORQ] & G_o go away)

results: $DGR = (DSR)(DVTZ)$
 $DGR = (-0.3740 \text{ deg/hr/G})(14.9 \text{ G})$
 $DGR = \underline{-5.57 \text{ deg/hr}}$

The following equation is used to calculate the redundant rate:

$$wr_c = (RR - DFY - DGR) * SF$$

where: wr_c - calculated redundant rate
RR - measured redundant rate
DFY - bias term determined prelaunch
DGR - g-sensitive term calculated above
SF - redundant rate scale factor
assume: RR = DFY = 0
SF = 1

The calculated redundant rate output, $\underline{wr_c = +5.7 \text{ deg/hr}}$, as observed during the mission.

Though the failure was a hard-over Z-axis accelerometer, null accelerations were noted to be approximately 3300 and -5700 micro-g's in the X and Y axes, respectively. These false indications were due to the way null accelerations are computed. Since the null accels are computed by calculating the changes in the individual IMU's M50 velocities, they are not completely representative of the IMU accelerometers (see below for details).

The following ΔV computation equations are used to determine the IMU null accels:

$$\begin{aligned} DVX &= CMx + DELTXZ (CMz) \\ DVY &= CMy + DELTYX (CMx) + DELTYZ (CMz) \\ DVZ &= CMz + DELTZX (CMx) + DELTZY (CMy) \end{aligned}$$

where: DVX, DVY, DVZ - $\Delta V_x, \Delta V_y, \Delta V_z$
CMx, CMy, CMz - measured velocity counts from x, y, z accelerometers
DEL Tij - I-Load of j accelerometer in the ij plane

assume: 1. CMx = CMy = 0
2. working with μg instead of ΔV counts
3. small angle approximation

results: $DVX = 2.2155983 \times 10^{-4} \text{ rad } (14.9 \times 10^6 \mu g)$
 $DVX = \underline{3301 \mu g}$
 $DVY = -3.7960894 \times 10^{-4} \text{ rad } (14.9 \times 10^6 \mu g)$
 $DVY = \underline{-5656 \mu g}$

Though the X and Y axis null accels were in error, the accumulator velocities showed no change in the X and Y axes at the time of the failure. The following equations are used to

calculate the X and Y accel biases using the hardware velocity accumulator:

$$ACC = \frac{V_f - V_i}{\Delta t} * \frac{10^6 \mu g}{32.2 ft / sec^2}$$

where: V_f & V_i - the final and initial velocity values for the delta t time period

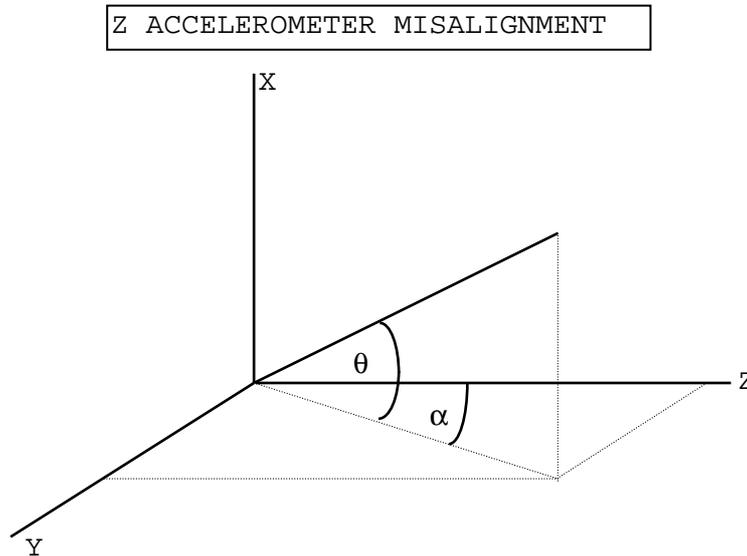
results: $X \text{ ACC} = \frac{1451.77 ft / sec - 1449.71 ft / sec}{60 sec} * \frac{10^6 \mu g}{32.2 ft / sec^2}$

$$X \text{ ACC} = \underline{1066 \mu g} \quad (\text{on board bias} = 1052 \mu g)$$

$$Y \text{ ACC} = \frac{1318.15 ft / sec - 1316.25 ft / sec}{60 sec} * \frac{10^6 \mu g}{32.2 ft / sec^2}$$

$$Y \text{ ACC} = \underline{983 \mu g} \quad (\text{on board bias} = 994 \mu g)$$

The reason the X & Y null accels were in error, though the accumulator velocities showed no change is that the null accel computations do not account for accelerometer misalignment. Since IMU 2's Z accelerometer is misaligned by 45.7 arcsec toward the X-axis (DELTXZ) and -78.3 arcsec toward the Y-axis (DELTXY), residual null accelerations appeared in the X and Y axes (see Figure 6-40).



Preflight hanger calibration terms
 $\theta = \text{DELTXZ}$
 $\alpha = \text{DELTYZ}$

Figure 6-40, IMU Z axis Misalignment

The IMU was unusable for velocity data but alignments were still performed in OPS 2 to keep it as a backup attitude source. However, in OPS 3, the IMU attitude performance degraded severely since the g-sensitive gyro drift calibration equations reacted to the 15 g acceleration and produced a drift compensation to the Z gyro of over 10 degrees/hour. As expected, the X and Y gyros were unaffected. The following equations are used to determine gyro torquing:

OPS 2: the g sensitive terms are set to zero

$$\begin{aligned} T_x &= TRX - DFX \\ T_y &= TRY - DFY \\ T_z &= TRZ - DFZ \end{aligned}$$

Required torque = Align/OPS transition torques - onboard bias

OPS 3: the g sensitive terms are incorporated

$$T_x = TRX - DFX - \frac{(DIX)(DVTX) + (DSX)(DVTY)}{[T(TORQ)]G_o}$$

$$T_y = TRY - DFY - \frac{(DIY)(DVTY) + (DSY)(DVTX)}{[T(TORQ)]G_o}$$

$$T_z = TRZ - DFZ - \frac{(DIZ)(DVTZ) + (DSZ)(DVTY) + (DOZ)(DVTX)}{[T(TORQ)]G_o}$$

where: TRX, TRY, TRZ - sum of align & OPS transition torques
 DFX, DFY, DFZ - on board bias for the x, y, z gyros
 DIX,DIY,DIZ,DOZ,DSX,DSY,DSZ, - I-Loads for X,Y,Z gyro g-sensitive compensations for input, output & spin axes
 DVTX, DVTY, DVTZ - ΔVx (ΔVy, ΔVz) over 1 torquing cycle (960 msec)
 T[TORQ] - time duration of 1 pulse cycle (960 msec)
 G_o - standard acceleration due to gravity

assume: DVTX = DVTY = 0
 Working with micro g's instead of delta V
 (no need for T(TORQ) & Go)
 TRX = TRY = TRZ = 0

results: $T_z = -DFZ - \frac{(DIZ)(DVTZ)}{1}$
 Tz = -0.295deg/hour - (0.71086788 deg/hr)(14.9 g's)
 Tz = -10.9 deg/hr

Post flight testing attributed the failure to the IMU computer interface card (which contains the CAPRI), and not with the actual accelerometer.

6.7.1.10 STS-55

During the second launch attempt, when IMU 2 (S/N 017) was moded to operate, the IMU good discrete was set low resulting in the annunciation of an IMU fail. The DC/DC1, Platform Fail, Transmission Word 1 and I/O Fail BITES were set and the gimbals did not cage to the proper position as required when upmoding to operate. After 32.6 seconds, all BITES except for the Platform Fail BITE cleared. This particular BITE is latched and can only be reset by moding the IMU to standby and then back to operate. This was performed and the Platform Fail BITE cleared. In an attempt to recreate the failure. The IMU was moded to standby and then back to operate an additional five times, but the BITES never repeated.

A review of the data showed that the most likely cause of the failure was the DC/DC1 power supply. A review of prior failure history was also made and it was determined that this particular DC/DC1 card had experienced an unexplained anomaly in 1978. After reviewing all current and prior failure signatures, it was concluded that there was a 10 to 20 percent chance that the IMU would not go to the nominal end of mission and the launch was scrubbed. See the post flight report for more details.

6.7.2 HAINS In-Flight Failures

6.7.2.1 STS-63 (Prelaunch)

During the first launch attempt, IMU 2 (S/N 212) was moded from standby to operate when the HAINS issued the Platform Fail and the IMU Good fail. One second later the GPC issued an IMU fail and a Resolver Limit Fail BITE. About 7 seconds later the GPC started to intermittently issue Inner Resolver Null Fail BITEs. The data indicated that the platform was tumbling so the IMU was moded to standby and then back to operate. The failure indications essentially repeated and the launch attempt was delayed 24 hours to remove and replace the IMU.

During the next day's prelaunch IMU calibration, IMU 2 (S/N 211) was being slewed from position 12 to 13 when the GPC annunciated a Resolver Limit Fail BITE for one second.

The Resolver Limit Fail BITE is set high when the difference between the 1X resolver and the compensated resolver reading minus the resolver offset compensation is greater than a set limit. This test is performed on the pitch, roll and yaw resolvers and the limit is a function of the expected rate of change of the gimbals.

The data showed that at the time of the failure, IMU 2 was being slewed in the X or pitch axis while the platform underwent a gimbal flip. When the pitch axis was slewed past 91 degrees, the roll axis changed 180 degrees as did the yaw axis in less than 0.6 seconds.

The fast movement of the outer roll and the yaw gimbals and the sampling rate of the GPC resulted in the GPC seeing a larger than allowed difference between the 1X and compensated angle which was larger than the BITE limit. When the gimbal flip was complete, the BITE cleared. This scenario had been seen several times in the Inertial Systems Lab (ISL), therefore the cause of the Resolver Limit Fail BITE was understood and the IMU performed nominally during the next launch attempt.

6.7.2.2 STS-65

On MET day 8, IMU 1 (S/N 204) began exhibiting erratic outputs of the redundant rate gyro of varying magnitudes. Because of the extremely short duration of the redundant gyro spikes, there was no measurable change in the IMU alignment. On four occasions, the output exceeded the PASS software BITE limit of 0.7 deg/hr, resulting in the Redundant Rate BITE annunciation (IMU BITE/T 1). Since it would be difficult for the crew to catch the BITE

present to mask, GMEM paper work was turned in to do the BITE mask. After the fourth BITE, the crew managed to performed a BITE mask via Spec 21.

Analysis of the data during the flight indicated that the problem in the IMU was real and not a failure of the BITE circuitry. During instances of the high redundant rate, changes in the resolver outputs seem to indicate that the platform was actually being affected in a very minor way. It was also seen that the gyro deltas between IMUs 1 and 3 would "dither" during this time. However, IMU 1 remained very well aligned with comparisons to IMU 3 and with star data. The IMU was not declared failed for mission termination and landing site downmode until MET day 12 when the magnitude and duration of the spikes had increased such that a measurable perturbation in the IMU platform was seen. Platform drift rates as high as 0.2 degrees per hour were measured. See appendix A of the post flight report for examples of the data.

There were three chits written discussing the IMU 1 entry management. Originally the plan was not to perform any troubleshooting of the IMU. IMU 1 would remain selected for entry with the BITE masked. Should any other IMU fail, the IMU I BITE would be unmasked. After looking at the high frequency oscillations of IMU 1 attitude output when the redundant rate was present, the EG2/Aerojet DAP Principal Function Manager recommended not using it for attitude at the two level. There was a possibility of setting up a yaw jet limit cycle and using excessive propellant. After the first entry day was waved off for weather, a plan was discussed to troubleshoot the IMU since it was getting worse. The plan would take the IMU to standby for 5 minutes and back to operate in hopes that what ever was causing the problem could be reset. Engineering Management did not want to do this for fear that it could make the IMU worse. A compromise plan was to wait until the rate spikes cause a misalignment of 0.25 degrees before performing the troubleshooting. When this occurred, the troubleshooting was performed at MET 14/10:51. The problem did not clear or get worse. About a hour before the deorbit burn an IMU-to-IMU align of 1 to 3 was performed to cleanup misalignments from the spikes. The gyro deltas post landing indicated an IMU 1 misalignment of less than 0.2 degrees.

Post flight testing (IPR GNC-2-18-1012) found unexpectedly high noise levels on the voltage outputs of the DC/DC converter card. This card supplies power to various internal IMU components. Further investigation indicated that the ripples on the DC/DC card are the result of a combination of poor grounding and a generic miswire of one capacitor. The miswired condition is a generic HAINS production problem that by itself does not cause the failure observed during this flight, but does magnify the noise levels when a poor grounding condition is present. The failure on STS-65 was induced by the failure of the ground wire. Since then, several other IMUs have experienced similar DC/DC card voltage ripples and/or redundant rate output when tested in

the lab. This is compelling evidence of a loose ground connection and a generic HAINS anomaly. New DC/DC cards with an added ground strap and the capacitor wiring fixed are being placed in the IMUs the opportunity arrives such as during a vehicle major modification period.

6.7.2.3 STS-75

IMU 3 (S/N 210) experienced X and Y gyro drift rates reaching a maximum of about 16 sigma. It did not annunciate any BITES. The Y gyro was compensated 4 times for a total of 15.5 sigma and the X gyro was compensated twice for a total of 5 sigma.

Engineering reported that the poor gyro performance signature was indicative of a gyro wheel lubrication type problem similar to HAINS S/N 200 and S/N 198. The only difference was that this IMU did not annunciate a Platform Fail BITE. Engineering noted that the data do not indicate a bearing failure whereby the gyro would not run after being placed in standby. Given this, the IMU was powered off and then powered back on 12 hours prior to landing. After being powered back down for the waveoff, the IMU was repowered for entry and exhibited the same behavior observed on orbit. Misalignment at touchdown was 0.14 degrees.

Post landing, the IMU ran for 300 hours before it failed completely.

6.7.2.4 STS-80

At MET 014/13:24:59 IMU 1 (S/N 215) issued a Inner Roll Null Fail BITE. Up until just prior to this time the IMU had been performing nominally. An inflight anomaly was written. (GNC-01) It appeared the outer roll gimbal had failed into a 'open loop' configuration (loss of feedback into the summing junction) and the inner roll then began to compensate for the failure. As the outer loop continued to degrade, the inner roll reached the software limit of 0 ± 3 degree. This was the cause of the BITE message. Due to the failure in the IMU, the relative misalignment was increasing and the IMU was deselected from the SF and RM three minutes after the BITE message to preclude an IMU RM fail message.

This condition persisted for a total of 28 minutes when the outer loop appeared to return to normal 'closed loop' operations. During this time it became apparent that there was a possibility of damaging the hardware until a more thorough analysis of the data could be performed. The decision was made to take IMU 1 to standby. IMU 1 was moded to standby at 014/14:13:33.

After review of the IMU data by engineering and GNC personnel, it was recommended to recover IMU 1 for data gathering purposes.

IMU1 was recovered on the following crew day. For the remainder of the flight (3 days) the IMU was recovered in the crew morning and taken to standby and then OFF during crew sleep. The inner roll gimbal did exhibit noisy behavior for the remainder of the flight.

IMU 1 performed well enough for the remainder of the mission that it was available as a last IMU for orbit or entry. IMU 1 was never reselected to the SF and RM.

IMU 1 was removed from the vehicle post flight for testing in the ISL and then returned to the vendor for repair. Repair consisted of replacing the outer roll bearing assembly, azimuth brush block assembly, and replacement/upgrade of both gyros. The unit has been returned to service.

Additional backup data and anomaly report are in Appendix C of the post flight report.

6.7.2.5 STS-81

IMU 3 (S/N 201) demonstrated the same behavior observed on IMU 3 on STS-75. This IMU exhibited a X/Y drift rate of approximately 9 sigma and about five sigma was removed via uplink. Since the MOIR reported that this IMU had not shown drift trending due to out-gassing as the STS-75 IMU did, the IMU was not powered down, but was moded to standby. Out gassing is when an increase in heat within the housing causes an internal gas to be forced out of the gyro causing the pressure to change. When the pressure changes, drift performance is affected. New units contain gold plating for a better seal to aid in the prevention of out gassing. Each individual gyro acts differently with respect to out gassing.

IMU 3 was removed from the vehicle post flight for testing in the ISL and then returned to the vendor for repair. Repair consisted of replacement/upgrade of both gyros. The unit has been returned to service.

Table 6-XXXX contains data on the IMUs which have experienced the same failure mode as the STS-81 IMU.

| S/N | Status | Hrs of Operation | Cause of Failure |
|-----|-----------|------------------|--|
| 198 | Eng. Unit | 4100 | lubricant contaminated by solvents |
| 199 | Eng. Unit | 2000 | |
| 200 | | 1036 | inadequate lubricant |
| 210 | STS-75 | 1996 | inadequate lubricant |
| 201 | STS-81 | 4300 | 8 th flight, no previous problems |

Table 6-XXXX, Generic Lubrication Failure Data

The vendor identified a problem in the IMU production process in 1991 and implemented 2 corrective actions. One was implemented to ensure adequate lubrication of the gyros and the other ensured that the lubricant was not contaminated by solvent. Twenty IMUs (40 gyros) were produced prior to these corrective actions. The affected IMUs include S/N 200 to 216.

6.7.3 HAINS Gyro Lubrication Deficiencies

On STS-75 (HAINS S/N 210) and again on STS-81 (HAINS S/N 201), an IMU experienced extensive on-orbit drift caused by poor gyro bearing lubrication (see Figures 6-41a and 6-42a). It is estimated that up to 72% of the HAINS gyros were affected by the lubricant problem. The estimate was made by the number of gyros made in the same batch along with the defective gyros. Testing has shown that the problems show up early, and most of the faulty gyros have been screened out.

The signature of the lubrication problems is a divergence in both directions of the gyro that builds quickly (see Figures 6-41b through 6-41d, and 6-42b through 6-42d). During STS-75, beginning on mission day 8, HAINS S/N 210 began to indicate an extreme vertical (X/Y) gyro drift trend of 0.026 deg/hr/day in the X axis, and -0.053 deg/hr/day in Y axis. A compensation between 5 and 6 σ was uplinked to the X axis, and 16 σ to the Y axis. Prior to that time in the mission, the Y axis was trending at -0.0038 deg/hr/day which appeared to indicate a single axis problem in the Y and it was assumed to be manageable via uplinks from a flight support standpoint (see Table 6-XXXXI). Refer to the STS-75 post flight report for more details.

During STS-81, the X and Y gyro drift rates increased to approximately 0.04 deg/hr (7 σ) and -0.05 deg/hr (8 σ), respectively. The worst case measured gyro drift rate trending was 0.028 deg/hr/day (31 σ) and -0.056 deg/hr/day (62 σ) for the X and Y axes, respectively (Systems Anomaly List, GNC page 2). Refer to the STS-81 post flight report for more details.

On both missions, the IMU was taken out of operate until shortly before re-entry. The IMUs operated within mission requirements through re-entry and landing. After both missions, the units were removed from the vehicle and studied. Post landing failure analysis was performed on the STS-75 HAINS IMU (S/N 210). The results showed the gyro bearing had been assembled with insufficient lubricant. HAINS S/N 201, from STS-81, was sent to the vendor where repairs consisted of replacement/upgrade of both gyros. The unit has been returned to service.

No indications were found in pre-mission data that would have predicted the rapid performance degradation during the missions. However, data has shown that if the unit is powered down, the drift remains around its pre-shutdown value. This is an

effective approach for preserving an IMU to support the end of a mission. For the STS-75 failure, a complete power down versus going to STBY was recommended because some gyros, like that in S/N 210, exhibit trending due to out gassing when at the operate temperature. The gyro from STS-81 did not exhibit the out gassing type trending, therefore there was no reason to take the additional risk of powering the heater circuitry down.

There have been four previous failures of HAINS IMUs during ground testing due to gyro bearing problems. Two failure causes have been confirmed by failure analysis. HAINS S/N 200 failure (1990 time frame) was due to insufficient lubricant, and HAINS S/N 198 failure (1992 time frame) was due to contaminated lubricant. Two other gyro failures with similar symptoms have failure analysis pending; HAINS 199, and HAINS 204.

A plan was presented at the February 25th CCB to perform change outs on IMU gyros. The proposed plan was to replace all gyros in HAINS IMUs, including at least two in each vehicle before a space station mission requiring approximately six years to complete. The plan called for the purchase of 24 additional spare gyros, and performing change outs as rapidly as the vendor can process HAINS units and/or supply new spare gyros.

The first IMU was sent to the manufacturer to remove and replace all gyros. The IMU was returned with good gyros, but it also came back with a new problem. Given this, a decision was made by Ron Dittemore at the March 14, 1998 VECB not to perform the HAINS IMU gyro replacement plan. The decision was made to replace the gyros on a failure basis only. This decision was made because it was felt that performing gyro change outs introduced new problems, therefore it was better to change out only gyros which have showed problems.

Several changes were approved for the Intermediate Depot Maintenance Requirements Document (IDMRD) to reflect various technical and editorial updates with respect to IMU maintenance. The major technical changes in MCN OM2669 were approved and included adding a requirement for screening IMUs for performance degradation, conditioning spare gyros annually, and ensuring that the requirements specified in the IDMRD and OMRSD for conditioning IMUs reflect the technical recommendations from the original equipment manufacturer.

| STS-75 | | | |
|---------|--------|--------|-------|
| MET | IMU1 | | |
| | X | Y | Z |
| 3/18:13 | -0.006 | -0.010 | 0.000 |
| 7/10:08 | 0.000 | -0.018 | 0.003 |
| 8/23:15 | 0.020 | -0.050 | 0.000 |
| 9/18:08 | 0.012 | -0.018 | 0.000 |

| STS-81 | | | |
|---------|-------|--------|-------|
| MET | IMU2 | | |
| | X | Y | Z |
| 1/19:37 | 0.000 | -0.002 | 0.000 |
| 4/07:50 | 0.060 | -0.025 | 0.000 |
| 9/23:50 | 0.040 | -0.025 | 0.000 |

Table 6-XXXXI, STS-75 and 81 Gyro Bias Uplinks

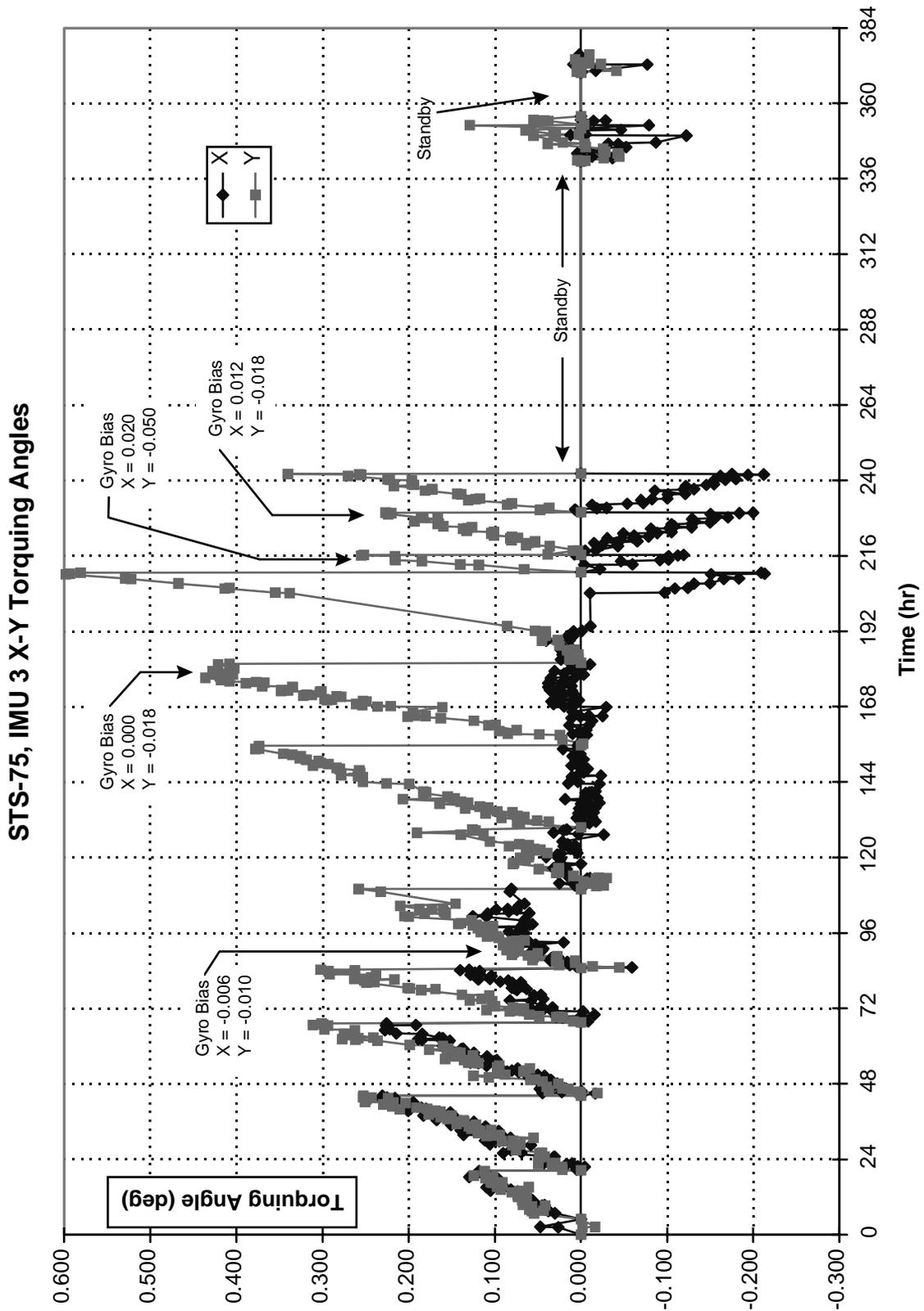


Figure 6-41a, STS-75 IMU 3 X-Y Torquing Angles

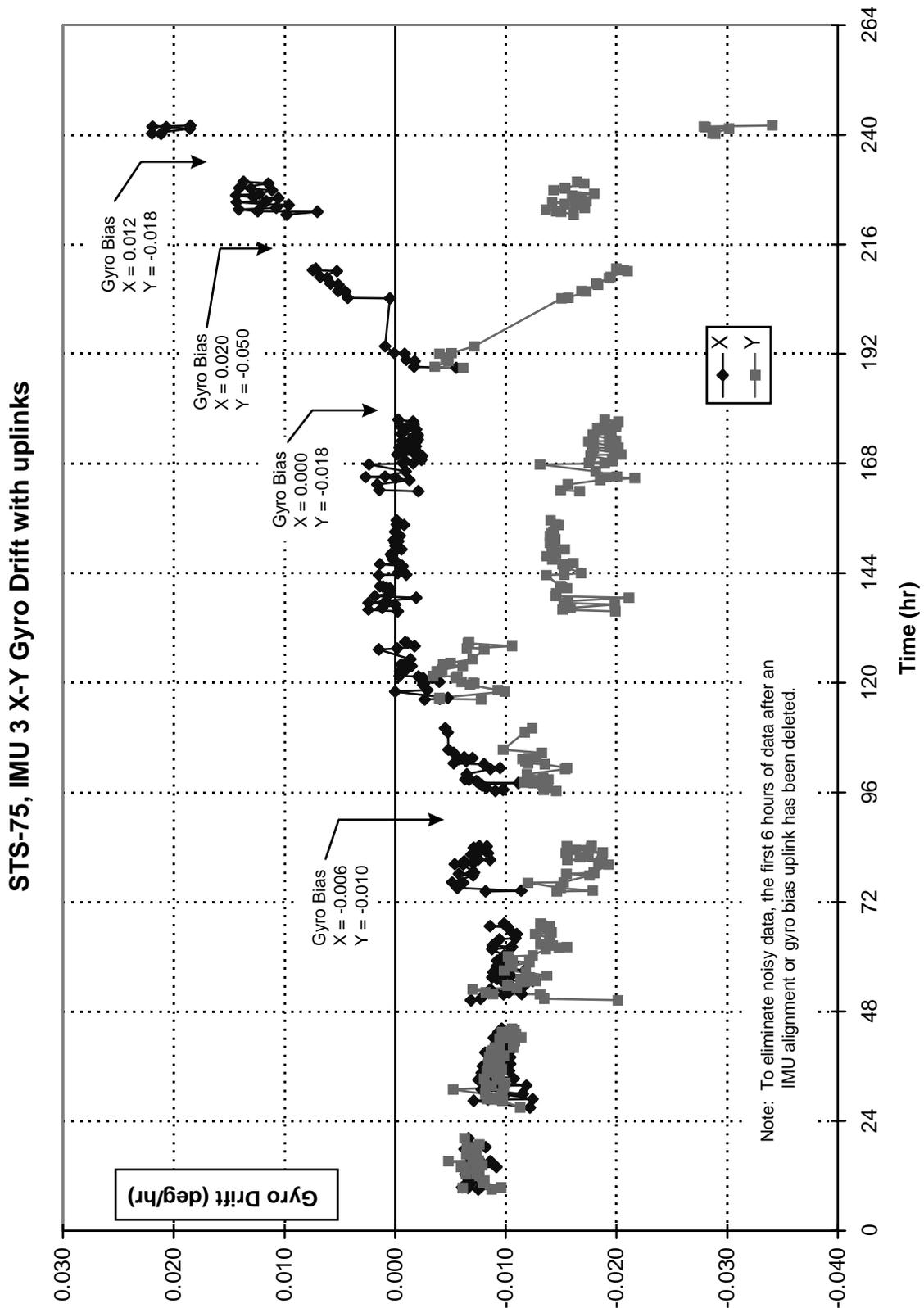


Figure 6-41b, STS-75 IMU 3 X-Y Gyro Drift with uplinks

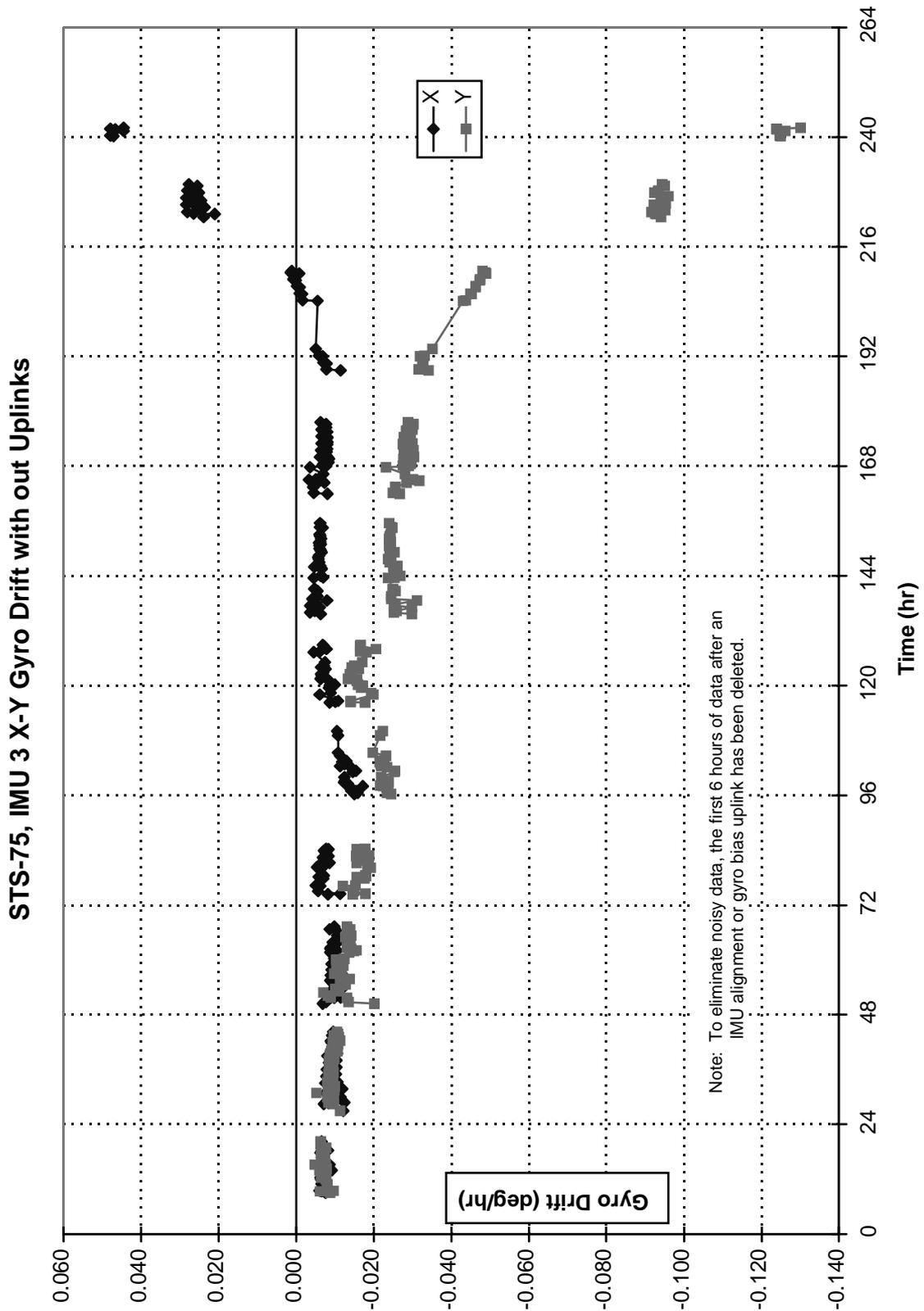


Figure 6-41c, STS-75 IMU 3 X-Y Gyro Drift with out uplinks

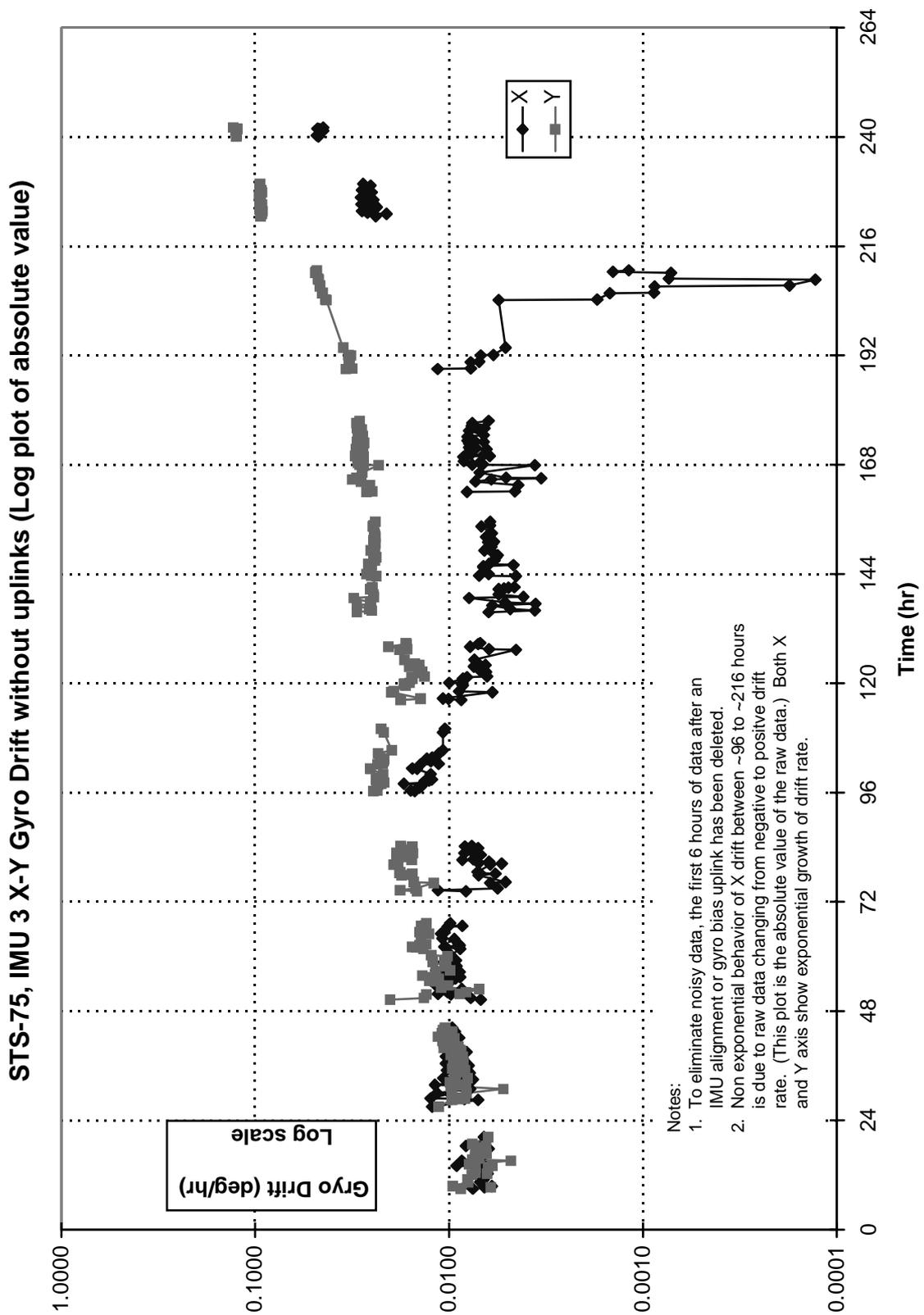


Figure 6-41d, STS-75 IMU 3 X-Y Gyro Drift (log scale)

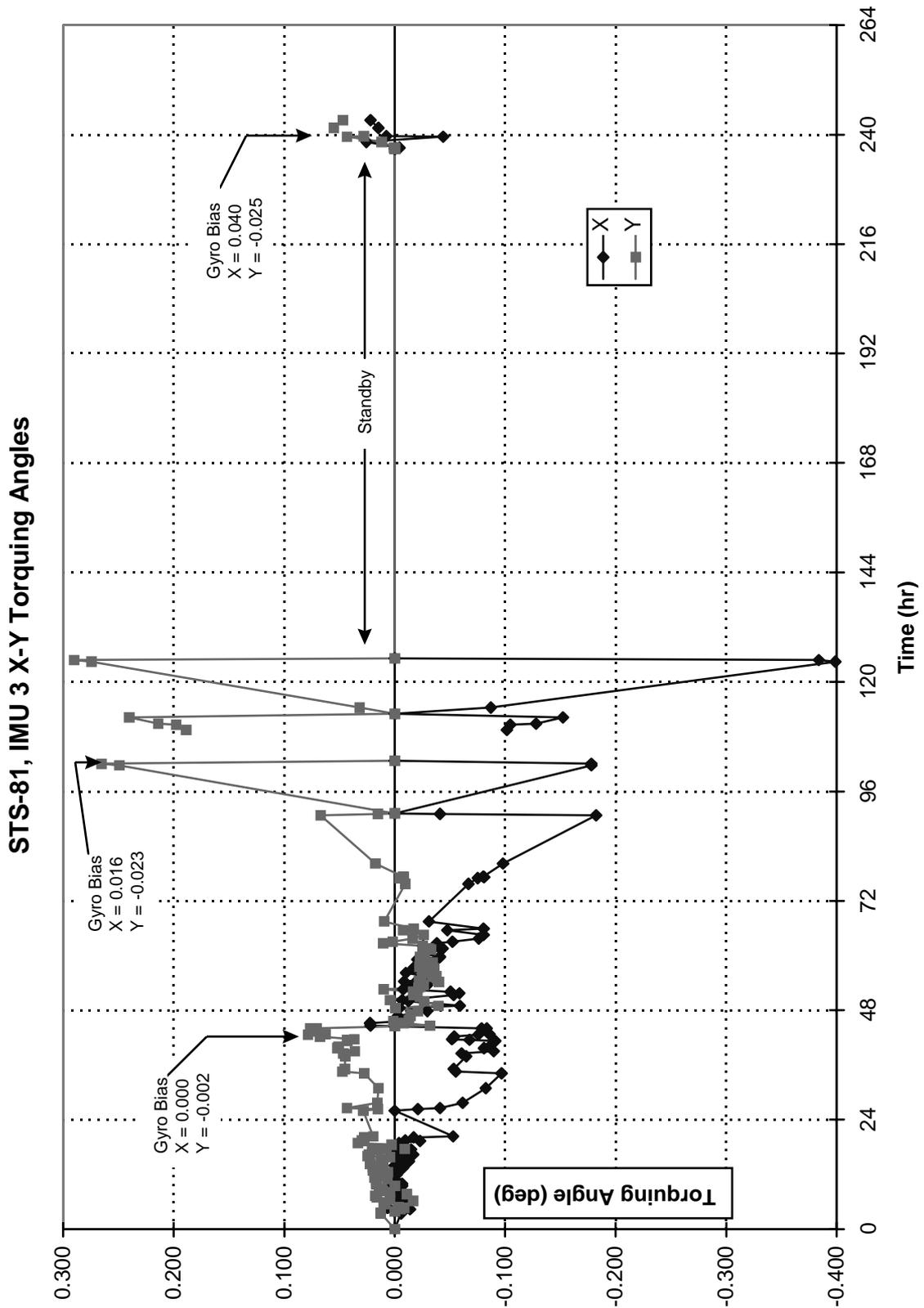


Figure 6-42a, STS-81 IMU 3 X-Y Torquing Angles

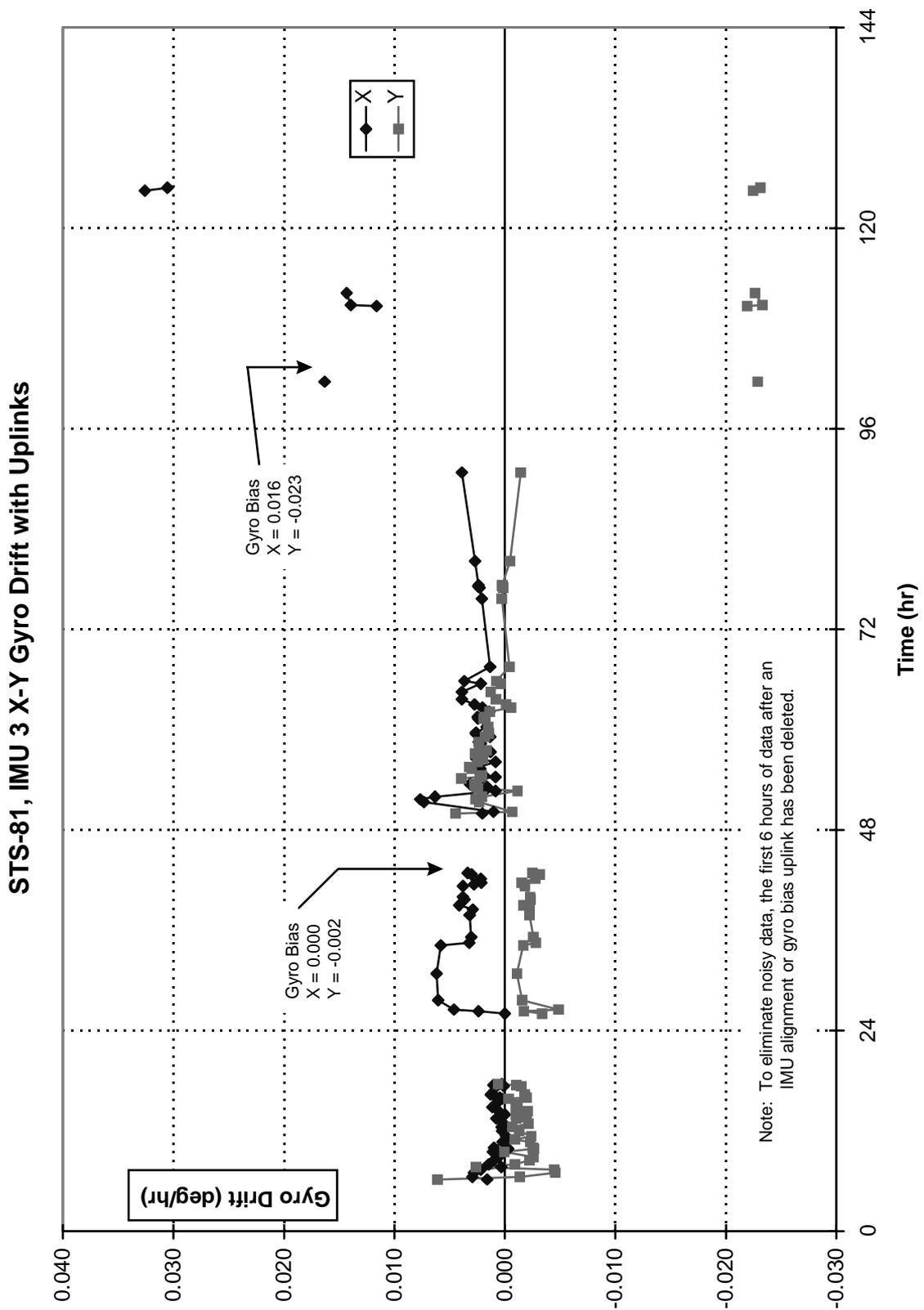


Figure 6-42b, STS-81 IMU 3 X-Y Gyro Drift with uplinks

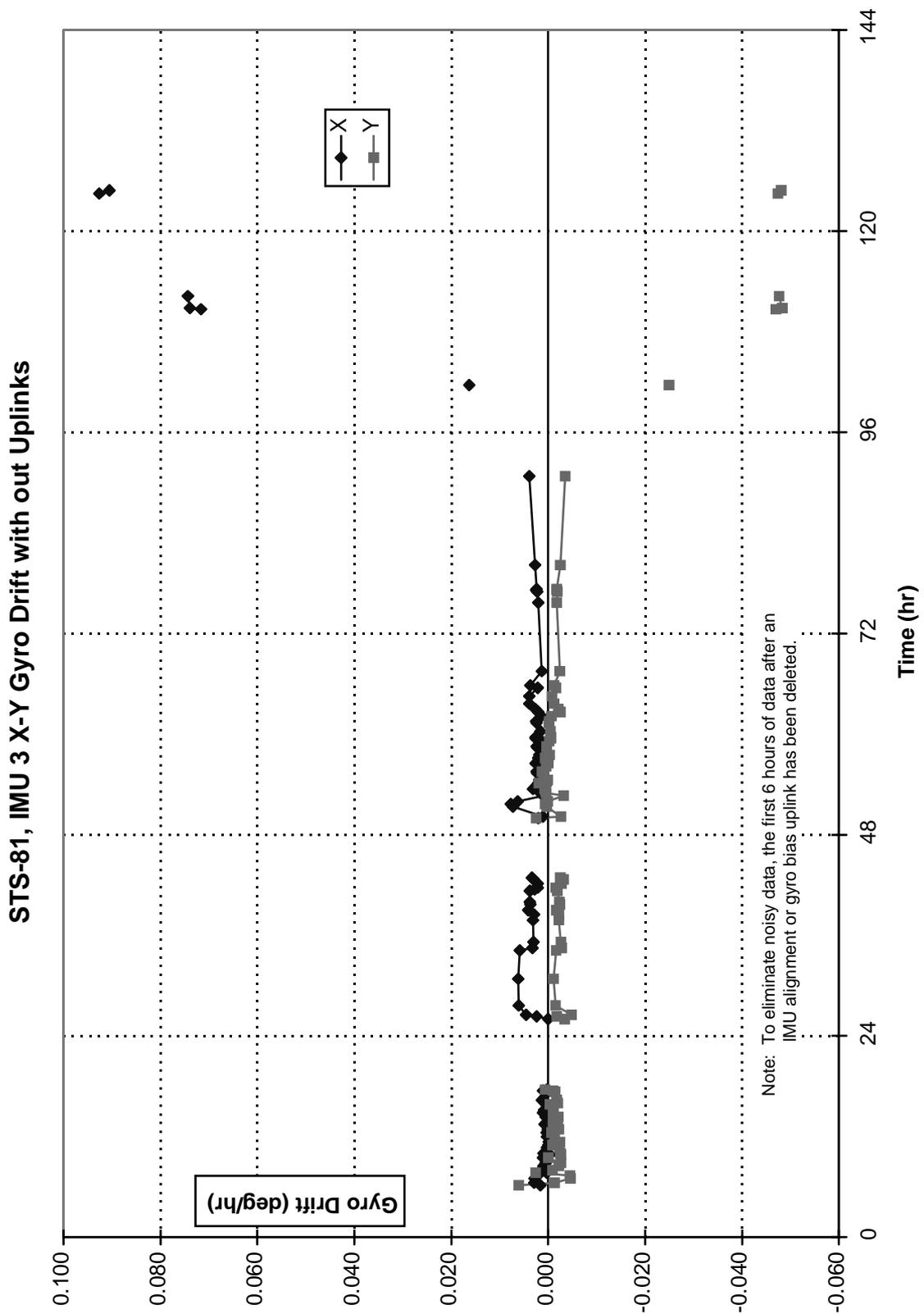


Figure 6-42c, STS-81 IMU 3 X-Y Gyro Drift with out uplinks

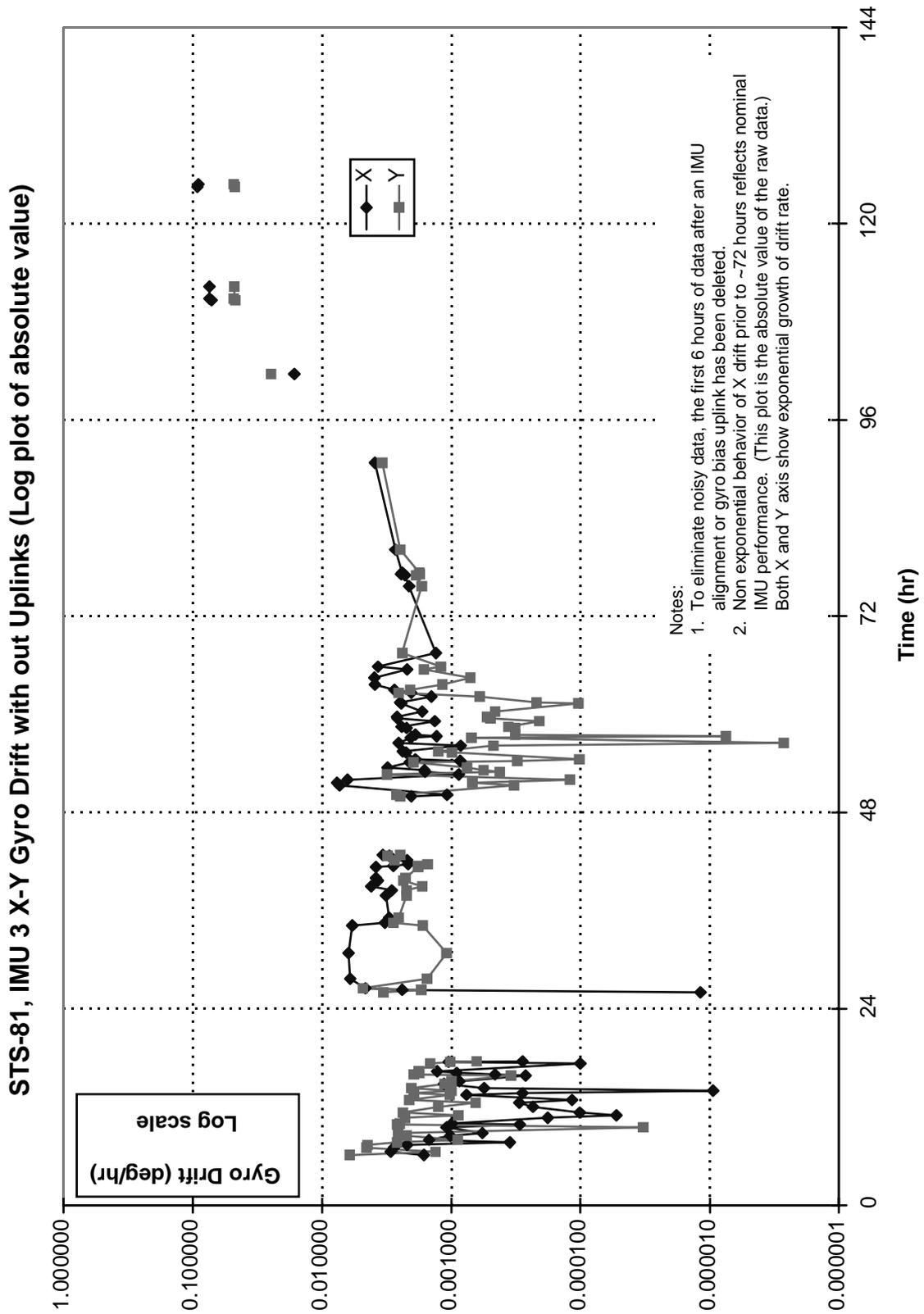


Figure 6-42d, STS-81 IMU 3 X-Y Gyro Drift (log scale)

6.8 Notes of Interest

6.8.1 IMU Inner Roll Resolver Failures

I. Overview -

The IMU azimuth, pitch, and outer roll gimbals utilize 1X and 8X resolvers to measure gimbal angles. The IMU inner roll gimbal, however, uses a linear pickoff to measure gimbal angles. The linear pickoff is excited by a 19.2 kHz source which is conditioned by the Platform Electronics No. 1 card for a coarse (1X) A/D input and by the Platform Electronics No. 2 card for a fine (8X) A/D input. The coarse (1X) signal is used to drive the outer roll gimbal torquer in response to motion sensed by the inner roll linear pickoff, which causes the inner roll gimbal to remain at its null position. The fine (8X) signal is used by the GPCs to determine the inner roll gimbal angle. This 8X input is used by ATT_PROC to calculate the IMU attitude.

II. Failure modes -

Fine (8X) input errors:

Causes: PE card #2 demodulator bad, Power supply problems

Effects: Since the GPCs are only using the fine (8X) inner roll signal for attitude calculations, a problem with only this signal would cause the measured inner roll gimbal angle to be in error. The gimbal itself does not move, but the GPC calculates that it has moved and is therefore not at its null position (which could trigger the INNER ROLL NULL FAIL BITE).

The gyro & resolver deltas for the affected IMU will be erroneous and the IMU may fail RM. The inner roll gimbal angles on IMU SUMM will display off-nominal values. The accel deltas and null accels will remain valid. Since the calculated attitude is in error, Vel Diff calculations (which use resolver data) will show velocity degradation and the IMU may fail for velocity during dynamic flight.

Coarse (1X) input errors:

Causes: PE card #1 demodulator bad, Power supply problems

Effects: Since the coarse (1X) inner roll signal is being used to torque the outer roll gimbal in response to movement sensed by inner roll, a problem with this signal would send an erroneous signal to the outer roll gimbal torquer. The outer roll gimbal would begin to move to null out this bias, which in effect disrupts the orthogonality between the inner roll gimbal and the pitch gimbal. The platform will sense this movement, forcing the

other gimbals to compensate in order to keep the platform inertial. As long as the platform is able to remain inertially stable, the GPC will be able to calculate a valid attitude solution. Accel deltas, Vel Diff, gyro deltas and resolver deltas will all look nominal, but inner roll will display an off-nominal value. Also, since our ground comps assume that inner roll is at null, the calculated reference cluster angles will not match what the hardware is actually doing. If the course (1X) inner roll problem is too large, the gimbals may not be able to compensate enough to maintain the inertial platform and the affected IMUs attitude reference will be lost.

Linear pickoff errors:

Causes: 19 kHz signal error, Power supply problems, Excitation voltage problems

Effects: The effects to the platform itself will be the same as described in Case #2. The erroneous 8X inner roll signal will also cause the attitude calculated by the GPCs to be incorrect. If the platform is able to remain inertially stable, the inner roll signal should settle out close to zero.

The gyro and resolver deltas for the affected IMU will be erroneous and the IMU could fail RM.

III. Comments -

Any failure that prevents the nulling of the pickoff will eventually cause a Gimbal Loop Fail that will drive the Platform Fail BITE annunciation.

In any of the cases described above, the IMU is considered failed and will no longer be available for use. A G-MEM write procedure is an option to compensate for an 8X inner roll problem.

A nominal value for the Inner Roll resolver during sims is zero. A nominal value for the Inner Roll resolver during flight is $\pm 0.12^\circ$.¹

¹ STS-114 flight data

SHUTTLE
GNC/JSC-18863

ORIG: K. Dunn
7/26/2007:FINAL,PCN-1

IMU
SB 6.0

SECTION 7
STAR TRACKER

7.1 SYSTEM OVERVIEW

The Space Transportation System (STS) Orbiter star trackers (ST's) are strapped-down, wide field-of-view (FOV), electro-optical tracking devices. There are two hardware designs in use today, the original image dissector tube (IDT) star tracker and the solid state star tracker (SSST). Their primary function is to obtain precise line of sight (LOS) angular measurements of selected stars and illuminated orbiting objects (targets). They also provide magnitude (brightness) data. Data from the ST's are sent to the general purpose computers (GPC's) in digital formats. The SSST was designed to be interchangeable with the IDT ST. The two ST designs are interchangeable at the mechanical interface and use the same mechanical envelope. However, their weights and center of gravity's are different. The SSST has a heavier Head Assembly, because a solid structure was required to sink the heat from the electronic boards to the Navigation Base so heat would bypass the temperature sensitive Detector Assembly. The SSST was first flown on the first flight of OV105 (STS-49) in May 1992.

7.2 SYSTEM DESCRIPTION

The star tracker/light shade (ST/LS) system is a searching and tracking device which uses a magnetically focused, magnetically deflected ITT F4012RP image dissector tube (IDT) or a 512x512 pixel Charge Coupled Device (CCD) detector to provide the image scanning function. The system consists of an ST for detecting and processing star/target position and magnitude; an LS to allow for star/target tracking at close angles to bright sources; and a quartz window in front of the ST lens to protect it from radiation, dust, and contamination as well as for reflecting the image of simulated star, light emitting diode (LED), into the ST FOV for self-testing the ST. The total FOV is a 10° square that can be completely scanned in 10 sec. The GPC software can dictate a 1° square offset scan to search for a specific star. A cruciform track scan (IDT ST) or 10x10 pixel block read (SSST), centered on the light source, is initiated when the star is detected. The tracker will search out and acquire items of approximately +3 (+4 SSST) to a -7 visual magnitude (including those at relatively close angles to the Sun, sunlit Earth or Moon, or other bright sources). It tracks the acquired star/target with a small tracking pattern which follows the star/target throughout the FOV. When the brightness reaches approximately zero magnitude, the track pattern doubles in size to allow for tracking the brighter and larger targets (up to 8 arc minutes in size). Two-axis position information as well as magnitude information, is provided to the multiplexer/demultiplexer (MDM) in digital format.

The ST/LS is provided with a bright object sensor (BOS) and video signal overcurrent-limiting sensing circuit coupled to a shutter mechanism to protect the IDT from being exposed to excess energy from bright sources which would damage it or degrade its performance. The same BOS is used with the SSST along with a background charge monitoring mode to prevent output of unreliable data. Status of the command to the shutter mechanism is supplied to the MDM in digital format.

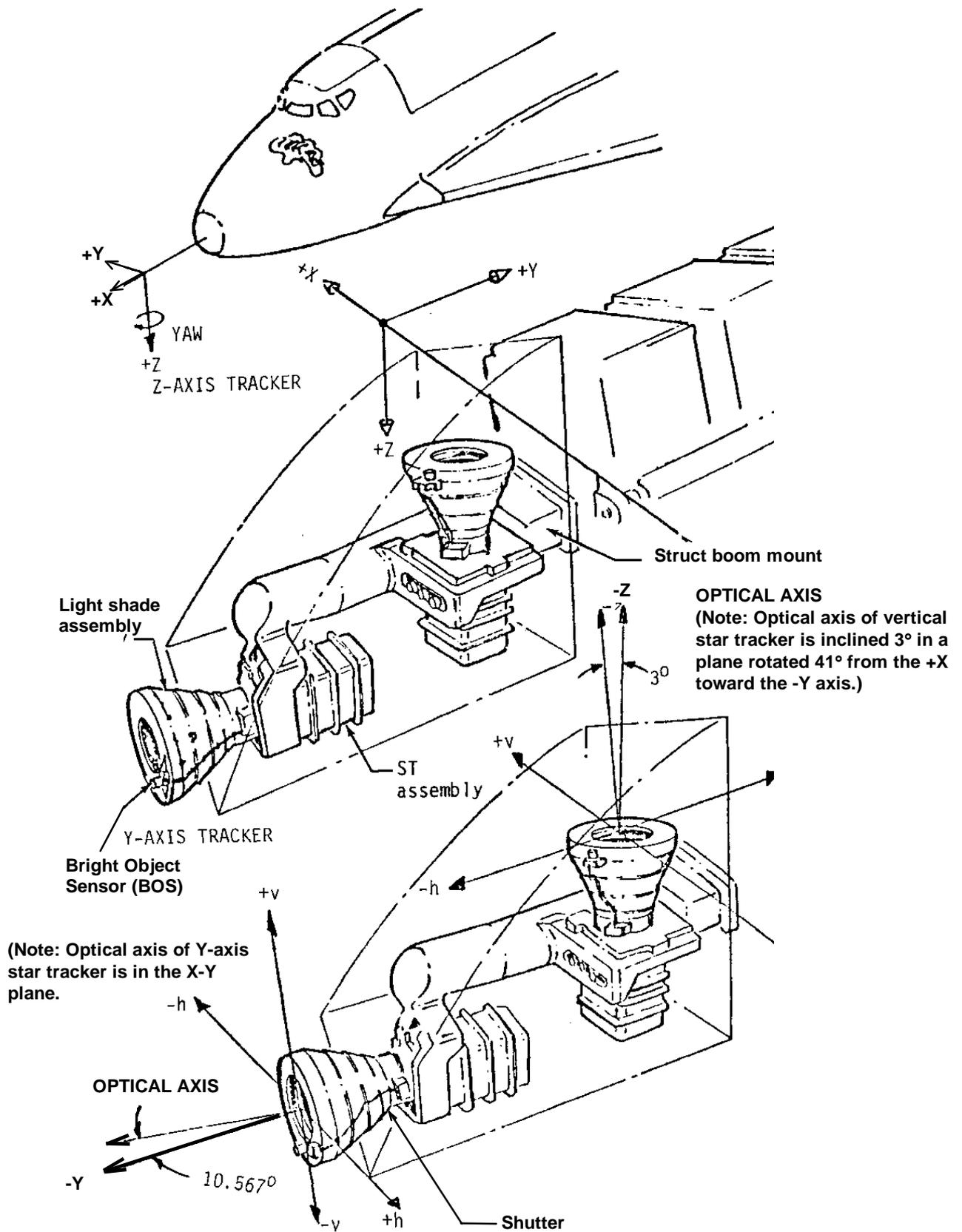


Figure 7-1 - Star Tracker and Light Shade mounting

The Space Shuttle Orbiter contains two ST/LS assemblies as subsystem of the guidance and navigation (G&N) system. Figure 7-1 shows the relationship of the ST/LS with respect to the navigation base.

The ST's are physically located in an unpressurized cavity, forward and to the left of the crew station. They are mounted on the navigation-base boom which extends through a bulkhead from the avionics bay containing the inertial measurement units (IMU's). Since the navigation base provides the mount for the IMU, the ST line of sight is referenced to the navigation base and the Shuttle coordinate system.

The -Y axis ST centerline is approximately 10.5° from the Orbiter -Y axis in the XY plane. The -Z axis ST centerline is approximately 3° from the Orbiter -Z axis in a plane rotated 41° from the Orbiter +X axis toward the Orbiter -Y axis. The location and components of the ST's are shown in figure 7-1. Nominal vehicle coordinates for the ST LOS's are:

| | | | |
|-----|-------------|-----|------------|
| -Z | | -Y | |
| X = | 0.03949845 | X = | 0.1833852 |
| Y = | -0.03433548 | Y = | -0.9830411 |
| Z = | -0.9986295 | Z = | 0 |

Each ST is provided 28 volts DC power through a 3 amp circuit breaker in series with a power switch. The -Y ST is connected to main buss MNB through a circuit breaker is on panel O15 and the -Z is connected to main buss MNA through a circuit breaker on panel O14. Both power switches are on panel O6.

The interface between the ST and the GPC is through a Flight-critical Forward (FF) MDM. The -Y ST is connected to FF3 and the -Z ST to FF1.

7.3 SYSTEM HARDWARE COMPONENTS

The ST subsystem is composed of the following (see fig. 7-2):

- a. Electronics assembly - Contains optics and electronics for converting optical images into digital data.
- b. Adapter plate - Provides common mounts for ST subsystem.
- c. Protective window assembly - Provides a seal that protects and allows electronics assembly to be pressurized as well as a surface for reflecting the image of a simulated star into the ST field-of-view for self-test.
- d. Light shade assembly - Protects lens from glare. Also provides mount for BOS and shutter mechanism.

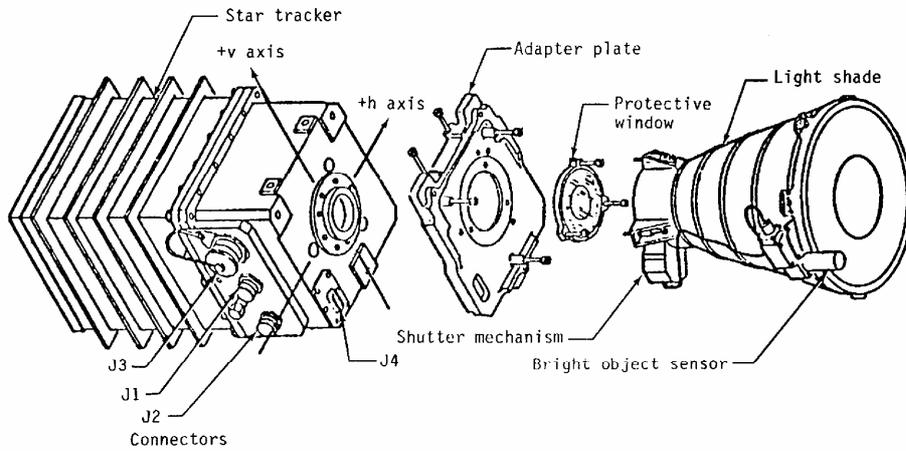


Figure 7-2 Star tracker components

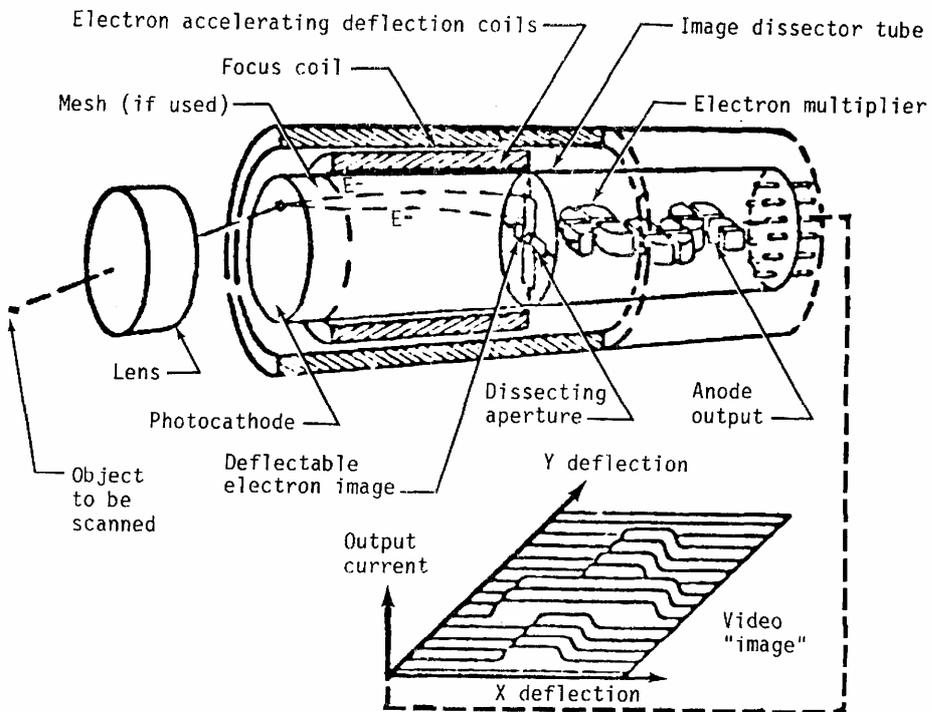


Figure 7-3 Image dissector tube

7.3.1 IDT ST (Deflection and Focus Coils)

An IDT is a type of photomultiplier device in which the sensitive area can be sampled in an elemental fashion using electronic/magnetic scanning techniques. Figure 7-3 shows the tube and its basic components; a photocathode, an aperture plate, and an electron multiplier.

The image dissector tube is the heart of the ST subsystem. Excitation of the photocathode by optical radiation causes electrons to be emitted in proportion to the level and spectral content of the flux applied. The electrons are then accelerated away from the cathode by the mesh potential and pass through the mesh screen holes into a unipotential drift space. While in this volume, a magnetic focus (or electrostatic, depending upon the tube type) field is applied, so that the electrons move through one complete spiral path to arrive at the aperture plate with the same spatial relationship as the optical image on the photocathode. The optical image thereby has been changed into an electron image and transferred in space to the single plane aperture plate within the tube. At the center of the aperture plate, there is a small hole which will admit electrons from the transposed area of the photocathode. The size and shape of this aperture hole is chosen to satisfy the sampling technique and provide the resolution required.

In order to properly generate scan sequences for acquisition and tracking raster operation, it is necessary to apply transverse deflection fields in the drift space. These fields cause the complete electron spatial image to shift orthogonally to the field direction, and as this electron image passes over the aperture, the tube signal output varies directly with the image intensity changes. Thus, the amount of current in the deflection coils required to deflect the beam through the aperture is a measure of the angular offset from boresight of the optical image. Therefore, by scanning the photocathode in a regular pattern (full field), any optical image within the FOV can be located.

As with all other types of photoelectronic imaging tubes, the image dissector employs a photoemissive layer, or photocathode for photon/electron conversions. The surface largely determines the spectral range and detective performance of the sensor.

The electron multiplier and final output anode portions of the IDT are equally important to its operation, and in general, they perform in the same manner as a standard photomultiplier tube.

After passing through the hole in the dissecting aperture, the electron beam enters the electron multiplier where it is amplified to a usable level at the anode output. Acquisition of an image occurs when the output voltage exceeds a predefined detection threshold level. At this time the scan pattern is changed from a full field or offset scan pattern to a cruciform pattern and automatic tracking of the object begins.

The values of deflector current corresponding to a specific location on the photocathode are dependent on tube temperature. A digital correction is applied by the ST electronics. Each tracker has a programmable read only memory (PROM) which contains correct terms for H and V values for its particular image dissector tube. Specific correction values for H and for V are accessed by H and V coordinates and one of four temperature levels. The four temperature levels are indicated by a two-bit binary code consisting of measurements V71X5027B (most significant bit (MSB)) and V71X5028B (least significant bit (LSB)) for the -Z tracker and V71X5527B (MSB)) and V71X5528B (LSB)) for the -Y tracker. The binary values and the corresponding temperatures are

00 = 24° C
01 = 31° C

10 = 38° C
11 = 50° C

These measurements are fixed at 1 and 0 for trackers earlier than serial number 8 and do not reflect tube temperature for these units.

The detection threshold level can be set to any of four levels.

- a. Level 0 - Maximum sensitivity level. Acquires objects to approximately +3 magnitude
- b. Level 1 - Cutoff level approximately +2.4 magnitude
- c. Level 2 - Cutoff level approximately +2 magnitude
- d. Level 3 - Cutoff level +1 magnitude

The detection level is defaulted to level 0 for normal star tracking operations.

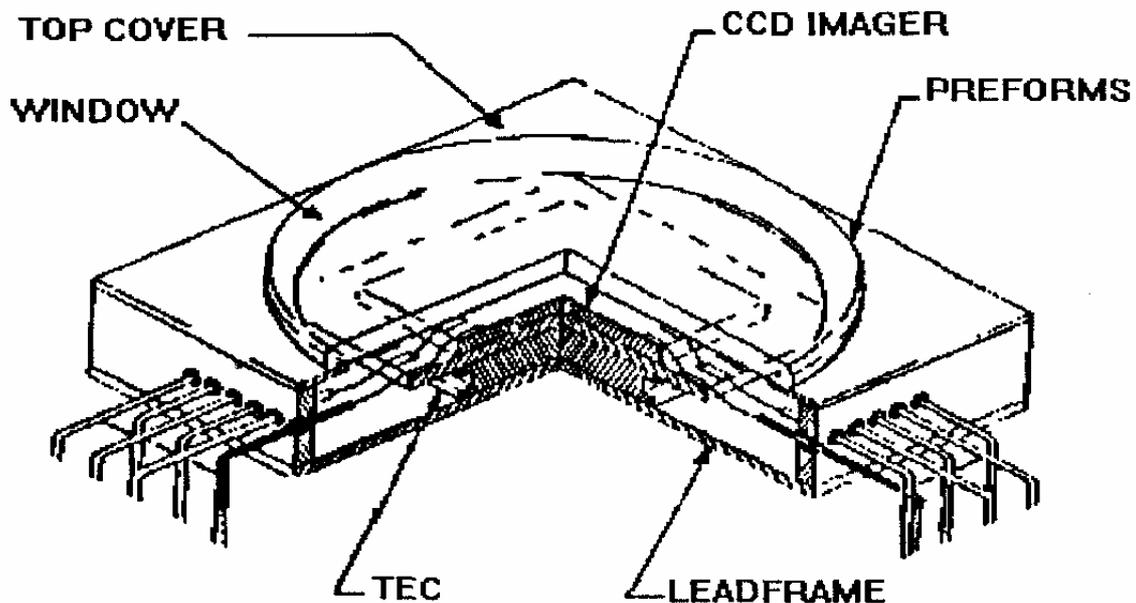


Figure 7-4 SSST Detector Assembly

7.3.2 SSST Electronics Assembly

The optics of the SSST resides in the Head Assembly. Alignment of the optics is mechanically controlled in the Head Assembly. The Tracker Base is an aluminum structure designed for rigidity and the Lens Assembly is heat-fitted into it. The lens contains six doublets and has a focal length of 72 ± 2 mm. The Focal Plane Assembly is mounted to the back of the Tracker Base and shimmed into alignment. The three front pads of the SSST are machined to four fringes and coplanar to two wavelengths of light to ensure proper mounting with the Navigation

Base.

The Detector Assembly within the Focal Plane Assembly contains the “eye” of the SSST (see fig. 7-4). The CCD imager is a 512x512 pixel silicon light sensing device. Photoelectric generated charge accumulates in the pixels under the image. The charge accumulation is proportional to the intensity and the time interval while the light is present. The CCD imager has a resistor diffused into it for temperature measurements. The CCD imager is normally cooled to between 3 and 20 degrees C for nominal performance. A Thermal Electric Cooler (TEC) is mounted under the CCD to provide the cooling. The TEC operates on the Peltier principle. The CCD temperature range is monitored for BITE and telemetry. The binary values and corresponding temperature zone are

| | | |
|----|---|------------|
| 00 | < | 3° C |
| 01 | = | 3 - 11° C |
| 10 | = | 11 - 23° C |
| 11 | > | 23° C |

The SSST contains six electronic boards: Interconnect, Power Supply (PS), Processor, Focal Plane, Communications Input/Output (CIO), Tracker Utility Interface (TUI).

The Interconnect Board serves as the interface between electronic boards. The CIO, TUI, PS and Processor are connected to this board. The Processor Board talks directly to two other boards, the Interconnect and the Focal Plane boards. The processor is the brain of the SSST. It utilizes a Harris RTX 2000 Real Time Express microprocessor for processing. This microprocessor is rated at 10 MHz but running at 6 MHz in this application.

The processor utilizes three types of memory: Programmable Read Only Memory (PROM), Static Random Access Memory (SRAM) and Electrically Erasable PROM (EEPROM). Permanent storage of the processor's program is in 8K of fusible link PROM. Temporary data storage is in 8K of SRAM. Calibration constants unique to each SSST are stored in the EEPROM. There are four sets of calibration constants to reduce the effects of potential radiation damage. A cyclic Redundancy check (CRC) constant is stored at the end of each set of constants. The processor conducts a CRC, calculates a CRC constant and compares it to the constant stored at the end of the EEPROM set of calibration constants to determine if any bit states have changed. If the CRC check is bad, that set is declared failed and the next set of constants are used. During normal operations an EEPROM failure will be annunciated only if all four sets of constants fail CRC. During Self Test the EEPROM failure bit will annunciate a failure for 3 seconds for each set of bad constants and the Star Tracker fail bit will be annunciated for the remainder of that Self Test. The EEPROM can be externally programmed or reprogrammed from the test connector.

Two signals are OR'd into the microprocessor reset, the Power On Reset (POR) and the Watch Dog Timer (WDT). If the SSST power is cycled the microprocessor will be reset. The WDT counter will generate the reset if it counts for 270 milliseconds without getting cleared by the microprocessor.

7.3.3 Shutter Mechanism and Return Spring

The shutter mechanism consists of shutter blades, drive motor, linkages, housing connector, spring, and support. It provides protection to the ST during poweroff and from bright objects during operation. The shutter closes approximately 500 msec after power is removed (fig. 7-5).

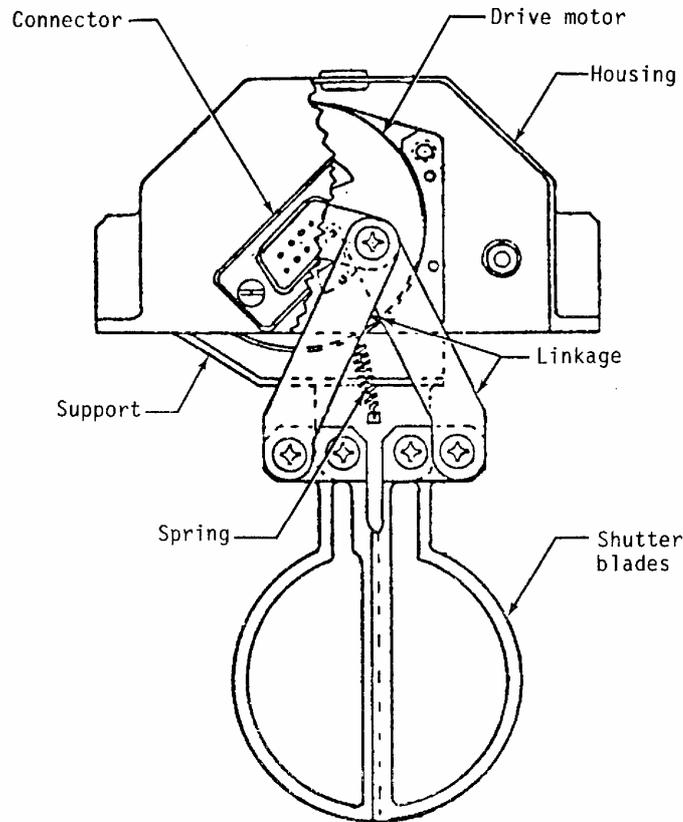


Figure 7-5 Shutter mechanism

7.3.4 Bright Object Sensor

A BOS and shutter mechanism mounted on the LS assembly protect the ST from bright objects approaching the FOV of the ST, endangering it with radiance levels which would damage it or degrade its performance. This hardware is common to both types of star trackers.

Referring to figure 7-1, notice a BOS mounted at the forward end of the light shield so that its FOV extends beyond that of the ST. When the threshold of the BOS is exceeded, the spring-loaded shutter shall be allowed to close through the action of the ST electronics.

The BOS consists of a detector, lens, LED, baffle system, housing, and associated electronics as shown in figure 7-6.

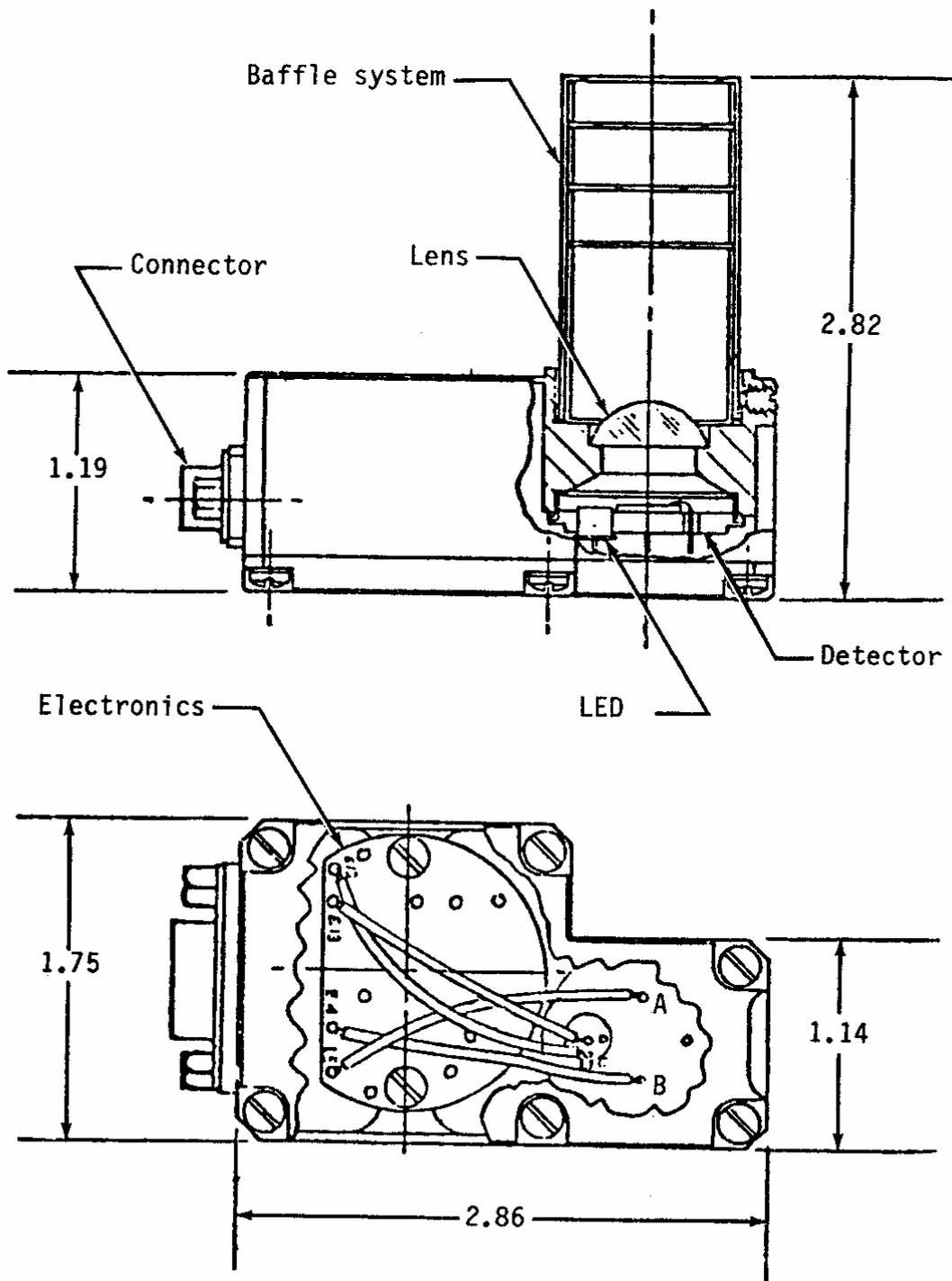


Figure 7-6 Bright object sensor

During normal operation, when a bright object illuminates the detector above its threshold, the BOS issues a bright object alert (BOA) signal to the ST. Upon receipt of the BOA signal, the ST removes power from the shutter motor allowing the spring-loaded shutter to close. Once the bright object has left the FOV sufficiently for the energy level to drop below the threshold, the BOS removes the BOA signal from the ST. The shutter will cycle open when the BOA is

removed or when the crew manually commands open. Note that there is no protection against the crew's opening the shutter when there is a bright object that could cause unit damage to the IDT ST. The CCD in the SSST will not be damaged by sunlight. During self-test, the LED is energized by the self-test mode signal, causing it to illuminate the detector above its threshold. This will cause the BOS to issue a BOA signal as in normal operation for verification that the BOS is operating properly.

The specifications for the relationships between the shutter and specific bright objects are as follows:

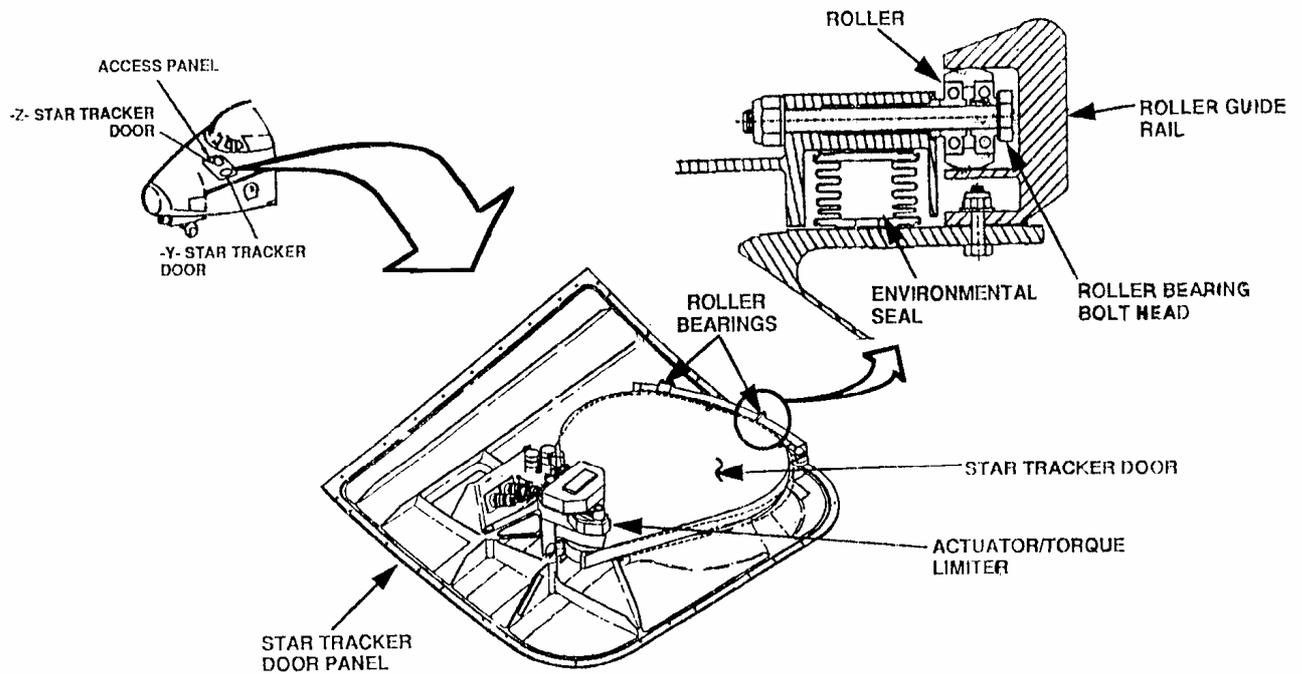
| <u>Bright Source</u> | <u>Shutter Open</u> | <u>Shutter Closed</u> |
|--------------------------------|---------------------|-----------------------|
| Sun limb | $\geq 31^\circ$ | $< 23^\circ$ |
| Earth limb | $> 20^\circ$ | $< 16^\circ$ |
| Target (≥ -12 magnitude) | | $< 8^\circ$ |

If the BOS detects an object brighter than -12 magnitude approaching at rates up to 10 deg/sec, it will close the mechanical shutter that protects the ST system. As a matter of reference, the Moon between the first and third quarters will close the shutter about 9° from the ST line of sight while a crescent Moon will not close it at all. A sunlit horizon will close the shutter at 14° to 17° from the line of sight.

7.4 STAR TRACKER DOORS

The ST units are enclosed within the Shuttle structure and are protected by closed doors during ascent and entry. Each door is opened and closed by two motors operating through a differential gearbox. With both motors operating, the doors are opened or closed in 6 sec. Should a motor, or power to a motor, fail, the operating time is increased to 12 sec. Figure 7-7 illustrates the doors. Rockwell thermal study IL SAS-AA&T-79-615 has shown that entry with one door open produces only local damage to insulation, light shade, and tracker. No thermal study has been done for the two-doors open case, and the possibility exists for severe structural damage to airframe, crew cabin, and navigation base due to plasma flow between the two doors.

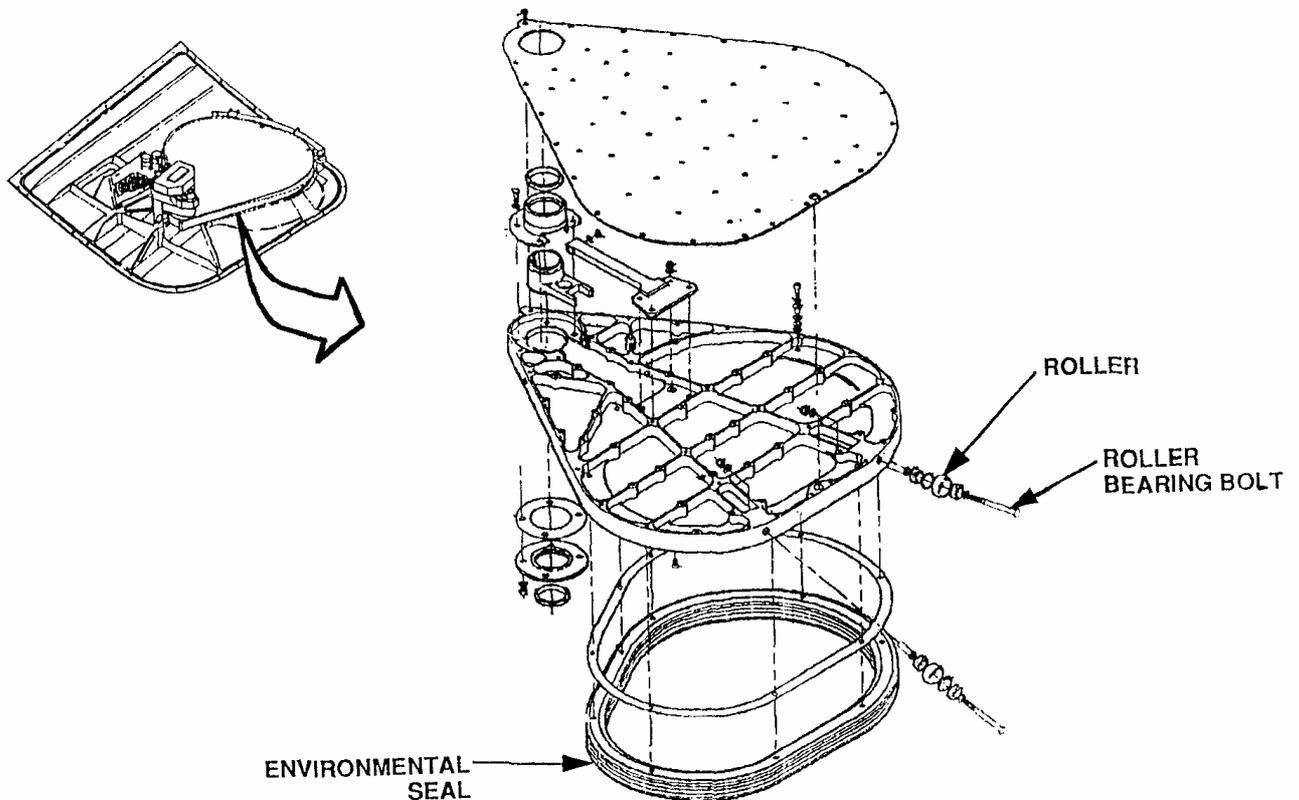
The three-phase ST door motors are controlled by the forward motor control assemblies (FMCA's). Refer to the Shuttle System Handbook Drawing 9.6 sheet 1 STAR TRACKER for the door control logic. Each motor is controlled by 4 logic drivers and relays, two in series for each direction. The ST power must be ON to open a door. Each door has an open and closed limit switch which is used to turn off the motor power when the limit switch contacts close. There is a SYSTEM 1 and a SYSTEM 2 STAR TRACKER DOOR CONTROL switch located on panel O6. Each switch will control one motor for each door. Therefore both switches must be operated to drive both motors on a door. Each relay logic driver is controlled by a logical AND of two inputs. For example, to drive in the open direction, the ST power ON signal is 'AND'ed with the NOT of the open limit switch for one relay and the DOOR CONTROL switch open signal is 'AND'ed with the NOT the open limit switch for the second relay. The two relays in series are closed to provide power to the motor to drive in the open direction. The motor will drive until the open limit switch closes or the crew turns the appropriate STAR TRACKER DOOR CONTROL switch off.



The doors rotate back under the Shuttle skin

The -Z door rotates up and in; the -Y door rotates down and in

(a) Cross-section view of door seal and roller guide



(b) Internal view of star tracker door

Figure 7-7 Star tracker door

A similar series of events using the close drivers, relays, and limit switch occurs when the STAR TRACKER DOOR CONTROL switch is set to CLOSE. The only difference is that the ST power ON signal is not used in the logic and therefore the ST doors can be closed with the ST power switch in the OFF position. When the STAR TRACKER DOOR CONTROL switch is OFF, power is removed from the drivers in the FMCA. This protects the door motors in the event that a limit switch fails to close after a door has reached its limit.

Each of the relays has four poles. The three normally open contacts are used for directing the three phase AC power to the motors. The fourth pole is a normally closed contact used for operational status. The Star Tracker Motor Control Assembly operational status is shown on Shuttle System Handbook Drawing 2.4 sheet 1 MCA DISTRIBUTION. The normally closed contact of several motor control relays are wired in series to a DC power source. The output of this string of relays is on telemetry to give the ground an indication that a motor control relay has been activated. For example, the -Y System 1 motor control relay K13 is connected in series with Air Data Left Probe motor 1 control relay K9, the Vent Doors Right 1/2 motor 1 control relay K17 and three Forward RCS isolation valve control relays to give the Forward MCA 1 Operational Status 1 signal on the downlist.

7.5 ST PERFORMANCE CHARACTERISTICS

The Orbiter ST's exhibit the following performance characteristics and operating features:

- a. Sensitivity - The spectral responses of the IDT ST and the SSST are significantly different. The IDT is approximately visual while the SSST is red-shifted. Therefore, a star will appear different to each type of tracker. For example, a blue star may appear 2.6 magnitude to the IDT ST and 3.1 magnitude to the SSST, and a red star may appear 3.0 magnitude to an IDT ST and 2.0 magnitude to an SSST. The IDT ST will acquire and track the 150 brightest stars based upon the S-20 star intensity scale. One hundred of these are in the on board software star catalog. The SSST is more sensitive and it is predicted that there are 1114 stars that it can possibly track. Beside the 100 stars in the software catalog, there are another 839 candidate stars in a ground uplinkable star catalog that the SSST may track. Approximately ten percent of these uplinkable stars may also be trackable by the IDT ST.
- b. Field of view - The ST total FOV is a $10^\circ \times 10^\circ$ square. The ST can also be commanded to search a square field within the FOV.
- c. Accuracy - The ST total error in measurement of star or target angles does not exceed 1 arc min (1 sigma). Star or target magnitude measurement errors do not exceed an absolute maximum of 0.6 magnitude. (Specification values)
- d. Warmup and acquisition - The ST will meet performance requirements within 15 min after poweron. The ST will search, acquire, and begin tracking a star in the FOV (if available and of sufficient magnitude) within 12 sec of being commanded by the GPC (within 1.2 sec in offset mode). (Specification values)
- e. Bright source protection - The ST's have a bright source detector which causes a protective shutter to close before bright objects can cause damage to the ST's. In addition in the IDT ST, a current limit sensor will trigger the shutter to close if an object being tracked exceeds approximately -8.0 magnitude brightness (target suppress).
- f. Angular rates - The ST's will acquire and maintain accurate tracking for Orbiter angular rates of up to 0.2 deg/sec. The ST's will also maintain lock-on with reduced accuracy at rates up to 0.5 deg/sec. (Specification values)
- g. Break track - The ST's can be commanded to break track of a star or target and begin searching so as not to reacquire the same star or target before the remaining commanded FOV (offset or total) is searched.

7.6 SYSTEM OPERATION

When power is applied, and no other mode is commanded, the ST will automatically start searching the total field of view (TFOV). The search scan pattern for the IDT ST is a left-to-right, right-to-left horizontal (H) direction/axis sweep with a staircase signal applied to the vertical (V) direction/axis to move the horizontal scan progressively from top to bottom (raster pattern). The TFOV search scan pattern for the SSST is performed by reading one frame at a time of CCD pixels progressing from top to bottom, taking 52 frames for the full field of view. Each frame is made up of 5 macro pixel rows of 256 macro pixel columns. Each macro pixel is made of 2 pixel rows by 2 pixel columns. The search scan for both continues until the first star or target that exceeds the commanded threshold level is encountered. When this occurs, the star/target is acquired and track scan begins.

The track scan pattern (cruciform for the IDT ST and a 10 by 10 pixel window for the SSST) centers on the star/target image and provides two-axis position signals at the system output. If

the star/target moves in the FOV due to small vehicle attitude changes, the track scan pattern will follow and remain centered on the star/target image. The star/ target position shall be identified by a positive or negative H coordinate and a positive or negative V coordinate based on the coordinate shown in figure 7-8.

Positive position signals in the H and V directions shall represent positive rotations about the ST/LS H and V axes. The search and track scan patterns for the IDT ST are shown in figure 7-9 and for the SSST are shown in figure 7-10.

In the star track or target track mode, the system will initiate in the full FOV search mode. If the software determines that a star/target should be present in the FOV, a BREAK TRACK command will be issued, the ST will be placed in the 1° offset scan mode, and the ST will begin searching about the point where the star should be located. If a star is found, tracking on the same star continues until it leaves the TFOV or the ST receives a break track command from the flight software.

When one of these events occurs, the search mode resumes, and regardless of the reason for returning to the search mode, it will search the remaining portion of the TFOV.

The ST may be commanded by the flight software to offset scan mode to search a reduced field of view (RFOV) within the TFOV. If a star/target is acquired, it will track that star throughout the TFOV as described above. If the star is lost for the same reasons as stated above, the ST will return to the RFOV position originally commanded and start a search scan of the RFOV. A typical offset scan pattern is also shown in figures 7-9 and 7-10.

The ST may receive a BOA signal from the LS or a target suppress (TS) signal from within the ST indicating a potential danger to the IDT and remove shutter motor power from the LS assembly to allow the shutter to close before damage occurs. The TS operates differently for the two types of ST. For the IDT the TS signal will remain latched on until it is reset by a manual shutter open command. For the SSST all pixels intensities are added to form a background sum. If this background exceeds a magnitude of 2, the tracker goes into a background charge monitor mode to protect against tracking errors exceeding specification values. The H and V positions will be set to zero, the star present signal is inhibited and the TS is set. The shutter remain open to allow the tracker to continue monitoring the background intensity. Should the background charge exceed a magnitude of -0.765, the shutter will be closed to guard against CCD saturation. Again the latched shutter closed signal can be reset by the shutter open command send by the crew or by the flight software. The flight software will send the manual open command for one cycle if there is a target in the FOV, the shutter is closed and there is no BOA.

The ST system may be commanded to perform a self-test by the flight software via a crew command. In self-test, the ST generates a simulated star that is returned into the FOV by a reflective surface on the protective window and is acquired and tracked. The ST shutter is commanded closed during the self-test through activation of the BOS by a separate light source LED that exceeds the illumination threshold. The angular position and magnitude of the simulated star are compared with stored values within the ST to determine if the system is within operational accuracy requirements. At one point in the self-test, erroneous data is used to verify proper operation of the built-in test equipment (BITE).

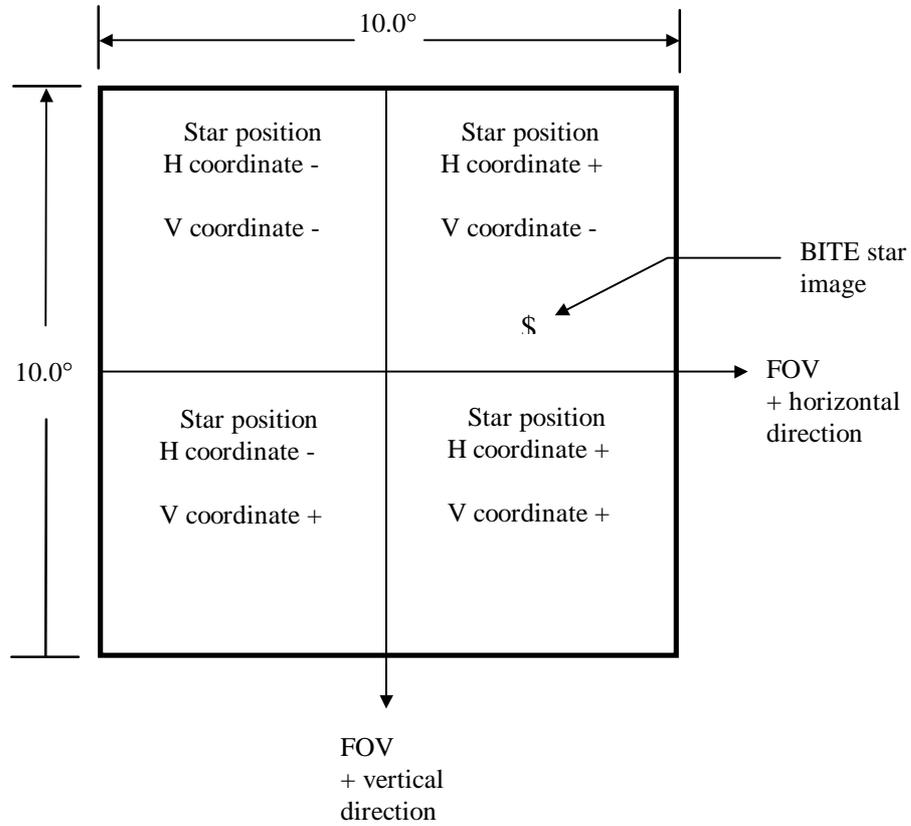
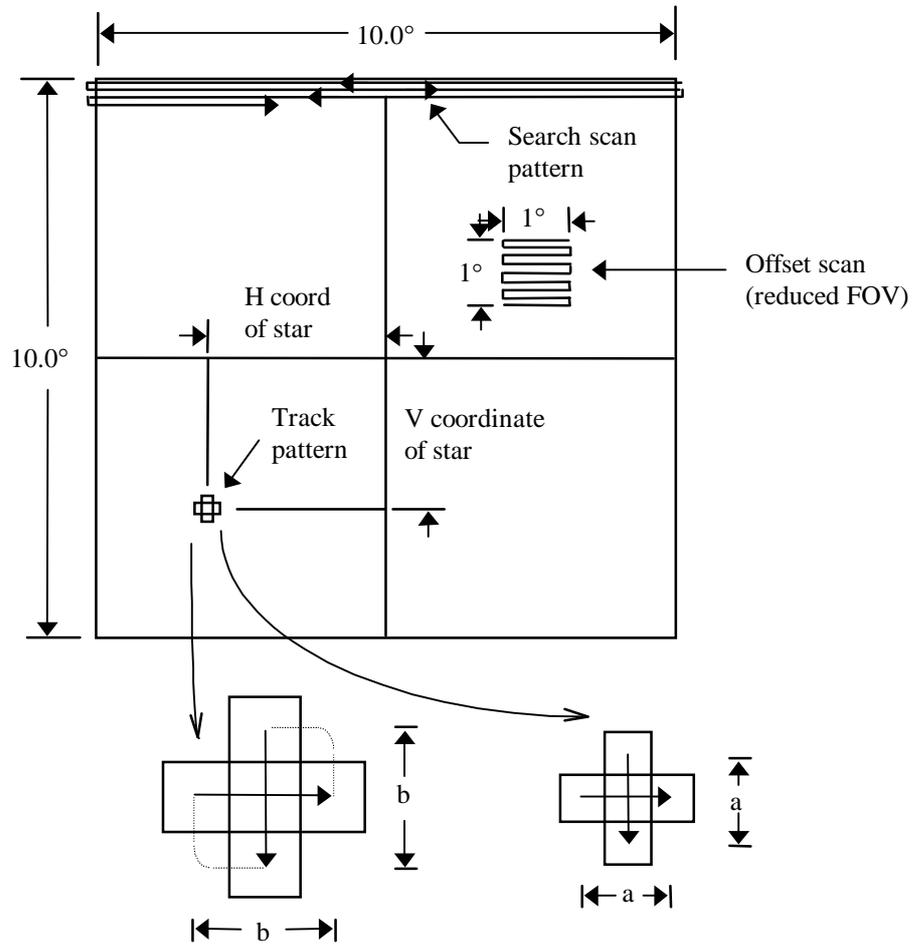
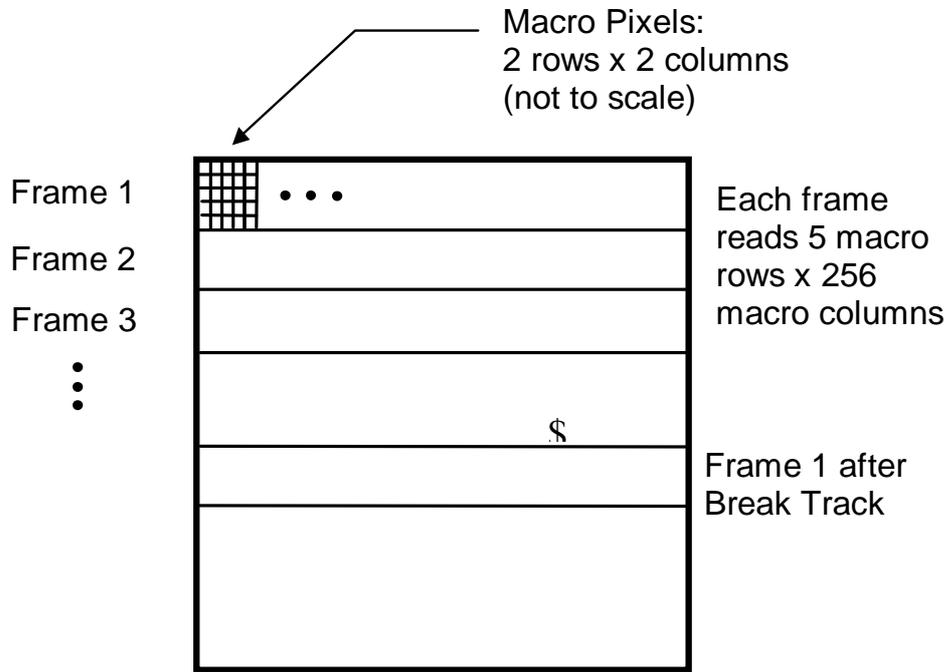


Figure 7-8. - ST TFOV as viewed from inside the tracker looking out at the star field

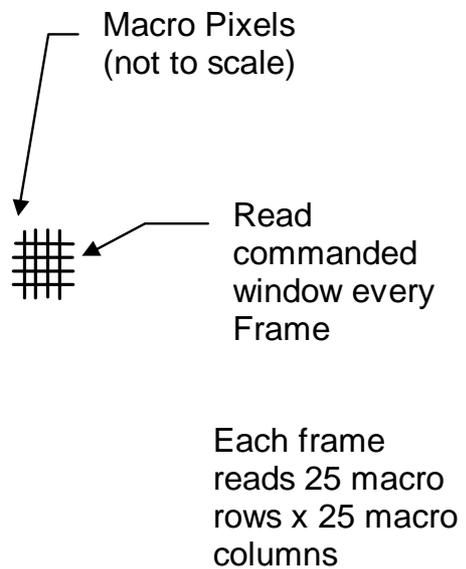


a = Approximately 0.24° for "0" mag or dimmer star/targets
b = Approximately 0.48° for brighter than "0" mag star or targets

Figure 7-9 - ST TFOV as viewed from inside the ST looking out at the star field.



Full Field Search (10 degree x 10 degree)



Off Set (Reduced Field of View) Search
(1 degree x 1 degree)

Figure 7-10 - SSST Search Fields of View

7.6.1 SSST Software Design

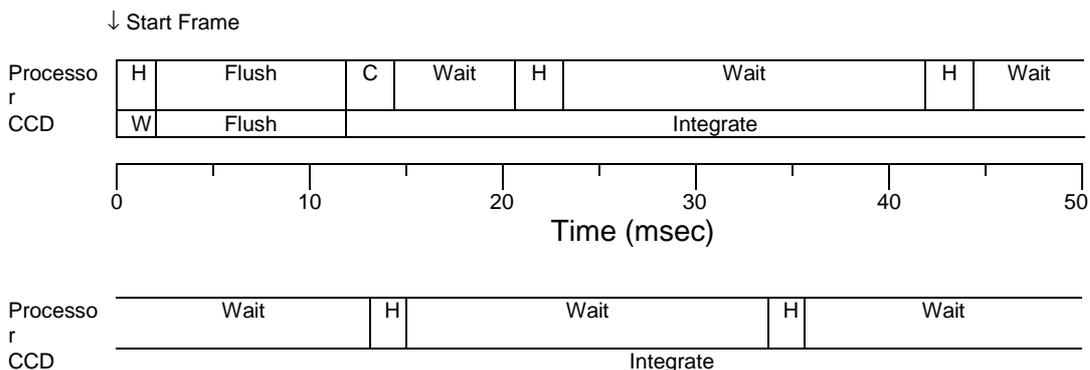
The internal software of the SSST performs the following functions; Initialization, Process command, Poll and control, Search, Track, Background charge monitor, Diagnose, and Output. Initialization is everything that must be done once at the beginning such as setting all flags with initial value and starting timers. The Process command function validates, reads and parses the command word and executes hardware oriented commands, such as “shutter open” as well as setting flags to control the other processes. Poll and control reads hardware status and executes routines based on that status, such as “tracker over temperature”. Search finds candidate targets/stars using modes specified by the command word and initiates acquisition which is the validation of the candidate as a target or star. Track completes acquisition and tracks the position and intensity of a target or star. The Background charge monitor handles the high background charge exceptions discovered in search or track. The Diagnose function is a ground test function for executing commands from the test connector. The Output function constructs the output words and sends them to the MDM.

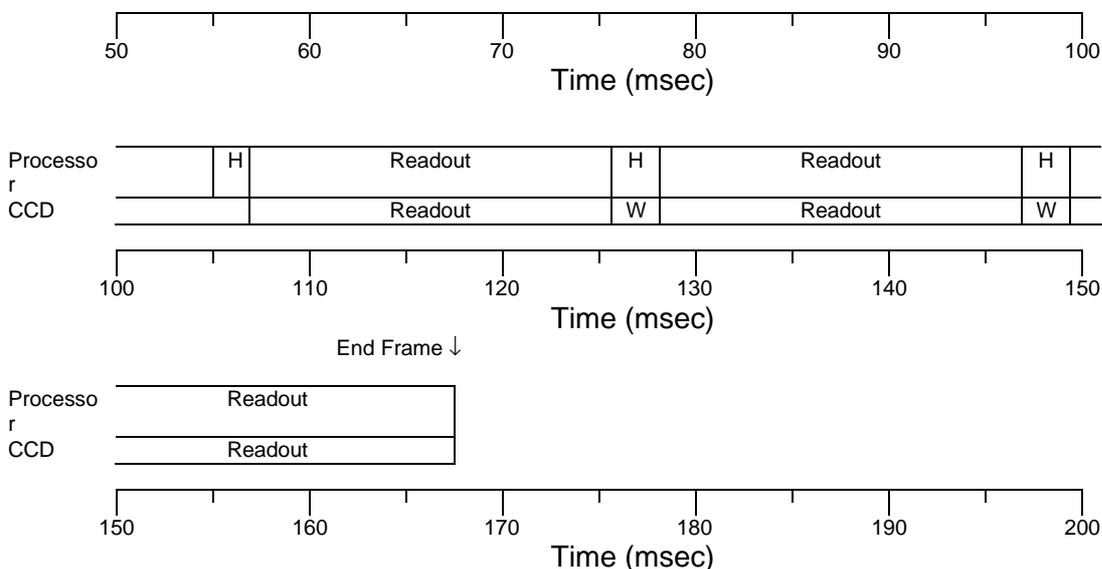
7.6.1.1 Initialization

The Initialization tasks are performed only once when the tracker is powered on or when a reset occurs. They initialize variables and timers used by the succeeding routines. The type of reset, Watchdog timer or power on reset, is determined. The redundantly stored data in EEPROM is tested and if all sets are bad a EEPROM bite is set. This same EEPROM test is performed during self test.

7.6.1.2 Search

The following timing diagram shows the activities that occupy the processor and CCD during each frame when the tracker is in the search mode. The “C” interval refers to “Computing”. It is during this time that the processor does all of its data processing. The “H” interval refers to “Housekeeping”. This interval occurs at regular 21 msec intervals in all of the tracker operating modes. During the “H” interval, the processor takes care of several miscellaneous tasks, such as updating output registers, checking for the receipt of a new command, controlling power to the thermoelectric cooler, and checking and setting various status bits. When the CCD is being flushed (i.e., unwanted charge is being dumped in preparation for a new exposure) or readout, the processor is fully occupied in controlling the process. During CCD integration, the processor is free for other tasks. The frame illustrated will need to be repeated 52 times in order to complete a search of the entire FOV. A RFOV, however, can be searched completely within one of these frames.

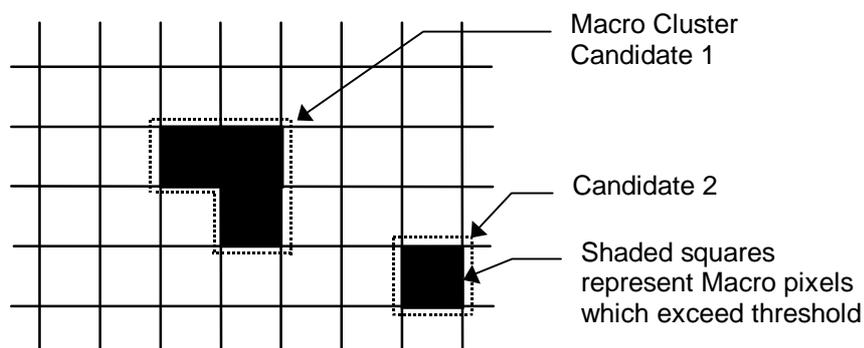




The Search mode combines two rows and two columns of pixels to create each macro pixel it reads. This enables more territory to be covered in the same amount of time. It also maintains awareness of any break track conditions commanded and starts searching after the previously tracked star. Macro pixels that exceeded the threshold are examined for “hot” pixels and adjusted accordingly. They are then re-thresholded and dropped from the list if they didn’t exceed it. Adjacent macro pixels, those sharing sides or corners, are grouped into a macro cluster, the whole cluster being a possible candidate target or star (See illustration below). Search mode also keeps a running sum of all macro pixels read to compute a background measurement.

Upon entering a readout cycle, the locations to be read must be calculated. These are a function of the FOV size and the break track condition. For the full FOV search, it will either jump the previously tracked star or start where it left off the last time (which may also be at the beginning). Once it has calculated the beginning and ending row and column values, it makes sure that all macro pixels lie within the FOV if a reduced FOV search is in process. The readout commences to take place by digitizing and checking the threshold for each macro pixel. If a macro pixel exceeds the macro threshold its position and intensity are stored. Either way, its value is added to the background sum. In the case that all the pixels are so high that they exceed the threshold, the corresponding background will be calculated as high and the tracker will enter background charge monitoring mode (unless the shutter open has been commanded).

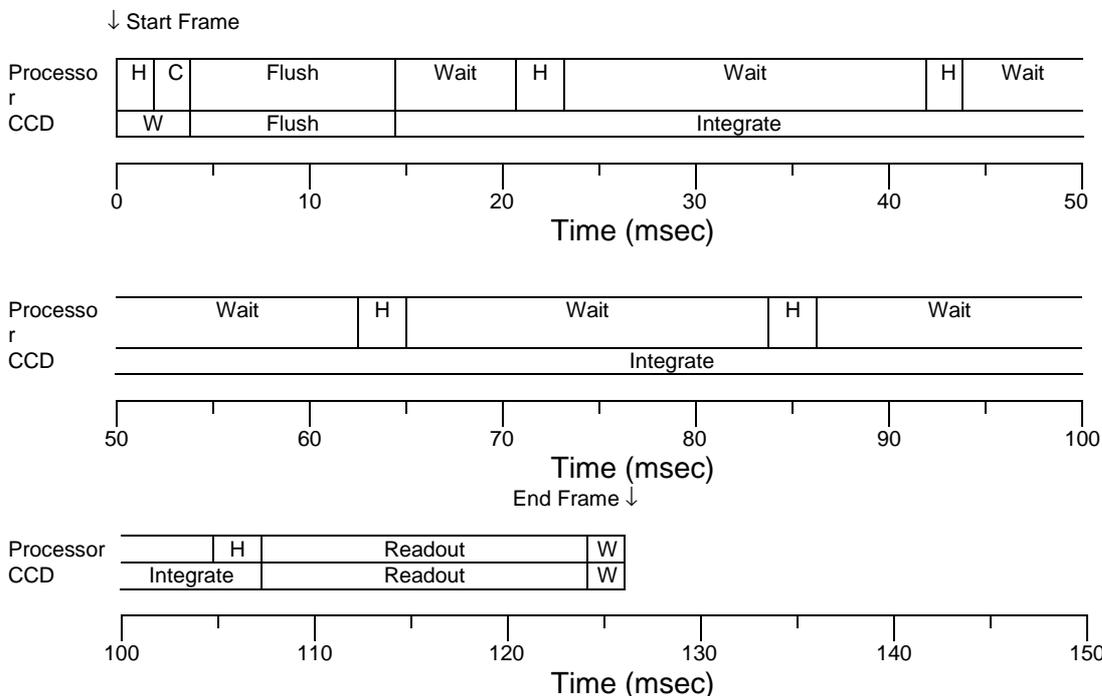
The macro pixels that have exceeded the threshold are clustered, since a target’s energy can be imaged upon more than one pixel (illustrated below). For a given exceedance, the list is checked to see if any of his preceding neighbors are on it and have therefore been associated with a cluster. If so, he is associated with the same cluster. He may even turn out to be the “missing link” between two cluster, and they are combined into a single cluster if that is the case. Information such as peak pixel, power (the sum of the pixels’ values), extent (rows and columns that completely border the cluster) and centroid is kept about each cluster. In the case of macro clusters, the macro centroid is converted to a true pixel location if the cluster is chosen as the target. This gives the location upon which to center the track window in the first frame of acquisition.



7.6.1.3 Acquisition and Track

After exceedances are clustered, the acquisition phase is entered. First, the background is checked to make sure it is not too high. If it is the background charge monitoring mode is entered (if a shutter open command is not in effect). If the background is appropriate, the first cluster encountered is chosen to be the target. After a candidate is chosen, a track threshold is calculated and a frame counter is initialized in order to track the target for three frames before reporting that a star is present.

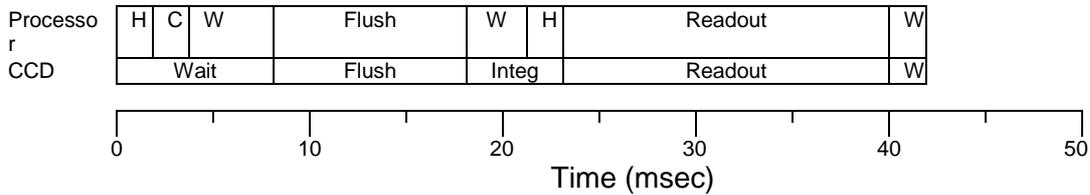
As in search mode, a track frame is made of several 21 msec cycles, each of which begin at a housekeeping interval. The following diagram shows the timing for the track mode.



The CCD integration time for star tracking will be varied according to target brightness. The integration time is shortened or lengthened by removing or adding the 21 msec wait cycles. The integration times available are 9, 30, 51, 72, 93 msec.

↓ Start Frame

End Frame ↓



This diagram shows processor and CCD activities when tracking bright sunlit targets. In this mode, the pre-flush and post-flush processor “Wait” intervals will be varied according to target brightness. The post-flush “Wait” interval can be reduced to virtually zero in order to shorten the integration period to 2 msec for the brightest targets. On the other hand, the post-flush “Wait” can be lengthened to 7 msec (resulting in virtually zero for the pre-flush “Wait” interval) in order to give a minimum integration time of 9 msec. Note that when the integration interval is lengthened to the full 9 msec, this mode is identical to the previously illustrated star tracking mode with its minimum integration time.

A 10x10 pixel track window is read out every frame. This size is sufficient to cover the required target dimensions. The window is centered on the predicted location of the target at the succeeding readout cycle. The predictor takes star image centroids from the five most recent tracker frames. It computes a least squares linear regression for both the H and V coordinates. Predicted star positions for the next frame are based on that linear regression. This action upon the data provides a smoothing effect at the output, as well as giving a measure of how noisy the data really is. Background charge monitoring is initiated if the data is deemed too noisy (unless there is a Shutter Open command in effect).

The few dead or hot pixels that may be present are corrected first, then the data is thresholded and clustered. The target is chosen and a centroiding window is calculated. This window is a square or rectangle that completely encloses the cluster plus a one-pixel-wide border all the way around the cluster. Fine centroiding is done on all the pixels within this window and their values are added to compute the star magnitude. The star present output bit is then set if the tracker has completed the acquisition phase.

Offset values for hot (or cold) pixels were stored in the EEPROM during calibration and redundantly read into RAM during initialization or self test. These values will be adjusted for integration time and added to or subtracted from the pixels. Dead pixels will have their values calculated via interpolation over their neighbors.

The brightest pixel of the target cluster must lie within a certain range to get the best centroiding data. If this pixel gets too bright, it is time to shorten the integration time. First, whole 21 msec wait cycles are subtracted from the integration time. If all of those are used up, then the integration time is shortened in 1 msec increments by changing post-flush waits to pre-flush waits. The opposite process happens if the integration time needs to be lengthened. In this case, any pre-flush waits are converted to post-flush waits 1 msec at a time. When there are no more pre-flush waits, whole 21 msec wait cycles are added until the maximum of four is reached. This makes the longest integration time 93 msec (4 times 21 = 84, plus 7 for the post-flush wait, plus 2 for the housekeeping interval before the readout).

When going from acquisition to track, the predictor’s data base of five points is built one frame at a time. This means that its predictions get better and better as more data is added (for the first three frames, acquisition is still in effect and no star present is reported, so the tracker will not be reporting less accurate values). After five frames, the oldest point is replaced by the newest each frame so the basis of the prediction is always the five previous frames. The best

straight line fit to the data is calculated, and those slope and intercept values are used in all subsequent predictions until the next update. During this process, the deviation of the points to the fit is calculated. If this becomes too large, the data is noisy and background charge monitoring mode will be entered from the main tracking module (if there is no Shutter Open command in effect).

7.6.1.4 Background Charge Monitoring

The background charge monitoring mode is entered when the background is too high coming out of search or when the data is too noisy. The point of this mode is to make quick background measurements and to adjust the integration time if the background is above the design limit. Again, the integration time is shortened in 21 msec intervals until all are used up. In this case, the wait interval directly after the flush becomes a read/compute/wait so the CCD is read immediately after flushing.

Only a few pixels at the top of the array are read in this mode. If they are above the saturation limit (the integration time is 0 and they are still high) then the shutter is latched closed. Otherwise, the tracker stays in this mode with the target suppressed bit set and the shutter open until they are below the design limit; then the tracker goes back to the search mode. While in the background charge mode the target magnitude will be set to the average value of the pixels read, the H and V positions will be 0, and the star present will be inhibited. This mode can be ignored or exited by commanding a Shutter Open and the tracker may be able to track a bright rendezvous target with degraded data.

7.6.2 ST Operating Modes and Software Commands

The ST hardware has several operating modes and receive several commands from the Orbiter flight software.

7.6.2.1 Full Field Scan Mode

The full field scan mode is the normal operating mode and will be entered when power is applied and no other mode is commanded.

Upon application of power to the ST/LS system, the shutter opens, the BOS is activated, and the data input registers are reset (erasing any stored MDM messages). The threshold level is set on the first pass at the most sensitive level, zero, and then at the level specified in the command word from the flight software.

In this mode the ST will search the TFOV, $10^\circ \times 10^\circ$, and acquire and track the first star or target that exceeds the threshold. Once a star or target has been acquired, the ST will continue tracking the star until it leaves the FOV and will then reinitiate a search of the TFOV to acquire another star or target.

7.6.2.2 Offset Mode Command

When the flight software issues this command, it will also command a set of offset coordinates where the candidate star or target should be acquired. The ST will scan the $1^\circ \times 1^\circ$ RFOV, whose center is at those coordinates, and acquire a star/target that is located within the RFOV. If the commanded coordinates are near the edge of the FOV, the 1° field will not extend beyond the TFOV. If a star is acquired in the offset mode, it will continue tracking the star through out

the TFOV until it leaves the FOV. The ST will then return to the commanded offset coordinates and resume offset scan until the star is acquired or the ST is commanded to another mode.

7.6.2.3 Track Scan Mode

When the ST locates an object exceeding the commanded threshold level, the V and H scan switches to a cruciform pattern centered on the object for the IDT. The SSST switches to a centroiding window enclosing the cluster of pixels centered on the object. The intensity of the object is monitored and the position of the cruciform pattern or centroid window is altered to keep it centered on the point of highest intensity. Tracking will continue as the object moves across the FOV at up to 0.2° deg/sec with guaranteed accuracy and up to 0.5 deg/sec at reduced accuracy.

7.6.2.4 Break Track Command

A break track command from the flight software does cause the ST/LS system to stop tracking and revert to the search/acquire/track sequence. It does not return to the coordinates which initiated the scan, but will continue the scan in such a manner that it does not reacquire the previous star/target in the scan of the remainder of the TFOV.

7.6.2.5 Self-Test Command

When self-test is commanded by the flight software, the ST/LS system shall energize an LED in the BOS that illuminates the BOS, exceeding its threshold and causing it to issue a BOA signal. The BOA signal will cause the ST to allow the shutter to close.

In this mode, light from a simulated star is reflected from an aluminized spot on the inner surface of the protective window to a point displaced approximately 4.25° horizontally and vertically. When this mode is selected, the flight software will command the offset mode until the simulated star is located. After acquiring the star, the angular position and magnitude are compared with internally stored standards to verify subsystem accuracy.

The angular position and apparent SMAG are compared with stored values to determine if the observed values are within the following prescribed tolerances:

- a. H angular position ± 146 arc sec (0.04 degrees)
- b. V angular position ± 146 arc sec (0.04 degrees)
- c. Magnitude approximately ± 60 percent (8 or 9 counts)

In the event either of these parameters is out of tolerance, the summary fail bit and a bit denoting the type of failure will be set. Once commanded, the ST/LS hardware remains in self-test mode until another mode is commanded.

7.6.2.6 Shutter Open Command

The shutter open command from the flight software will open the shutter, overriding both the target suppressed or BOA signal that may have closed it.

7.6.2.7 Threshold Set Command

The threshold set commanded from the flight software prevents the ST from acquiring and tracking a star/target of magnitude dimmer than the level commanded in a range of 0 to +3 (0 for all objects and 3 for only the brightest objects). After acquisition of stars/targets meeting the commanded threshold criteria, the threshold set command will be disabled during tracking to prevent loss of track due to fluctuations in star/target intensity. The one exception is the highest commanded threshold (3) for the SSST. The highest threshold command on the SSST, is the only threshold which shortens the CCD integration time. If the target dims after passing the highest threshold, loss of target will occur if it becomes too dim to be tracked by that integration time.

7.7 ST FLIGHT SOFTWARE FUNCTIONAL OVERVIEW

The ST Subsystem Operating Program (SOP) controls and monitors the operation of the two ST's and processes ST data for use by other program elements. The ST software (S/W) interfaces with the ST hardware to perform the moding functions, tracking functions, data enhancement, and self-tests. The S/W specialist function (SPEC 22) provides the user interface for the ST's. The ST responds to inputs from the SPEC and the navigation function and provides responses to the SPEC as required. The ST also provides alignment data to the IMU SOP. The three S/W data words from each ST are read every 160 msec whenever the S/W SOP is active. Block diagrams of the flight software can be found in figures 7-11 through 7-24. A brief description of the software submodes and the software block diagrams follow.

7.7.1 ST Submodes

When in the operate mode, the ST will respond to user interface requests to perform the following submodes; Terminate/Idle, Star Track, Target Track, and Self-Test.

7.7.1.1 Terminate/Idle

This submode terminates software processing associated with other submodes and sets ST output commands and user interface displays to the nominal idle conditions. The terminate/idle or more commonly referred to as TERM/IDLE submode is first entered when the ST is powered up and the SPEC 22 is selected. The terminate/idle submode is reentered whenever other submodes run to completion (such as self-test) or are manually terminated by requesting the terminate/idle submode via the keyboard, or any GPC activity that goes through GNC OPS 0.

7.7.1.2 Star Track Submode

This submode accepts crew-selected stars, performs star catalog searches, generates offset coordinates and issues the required outputs to the ST's for either offset scan mode or full field scan mode tracking. Offset coordinate data is tested for FOV limitations. Star sighting data is compensated for star aberration errors. Appropriate status displays are made available to the crew. The star track submode is normally utilized whenever an IMU is sufficiently aligned to provide a suitable inertial reference. If a crew inputs a required star ID, this submode will return to TERM/IDLE at its completion. If stars of opportunity are to be selected (crew required star ID is zero), this submode will continue until TERM/IDLE or some other submode is requested.

7.7.1.3 Target Track Submode

This submode commands the ST's to search and track a rendezvous target in either the offset scan mode or the full field scan mode. Target tracking utilizes navigation-generated inputs

defining the relative target vector. Status displays are made available to the crew. Target tracking will continue until TERM/IDLE or some other submode is requested.

7.7.1.4 Self-Test

This submode tests the ability of the ST to acquire and track a simulated star. Star magnitude (SMAG) and position data are checked against hardware prestored values for reasonableness. Pass or fail status is made available for display to the crew.

7.7.2 Software Overview

A block diagram of the software overview can be found in figure 7-11. The ST SOP consist of a Supervisor module which will call Self-Test, Star Track, or Target Track software modules based on crew inputs. The Supervisor also has processing for the Crew Optical Alignment Sight (COAS) which will not be discussed here. Both the Star Track and Target Track modules call both the Field of View Test and the Data Filter module. All of these modules will be discussed briefly in the following sections.

7.7.2.1 Supervisor Module

Refer to figures 7-12 through 7-14 for discussion of the Supervisor module. The first thing it does is remove the shutter open command if the unlatch flag is on and the crew manual shutter open command is not on. These steps are for resetting the shutter open command that may have been sent by either the star track or target track modules. The reason they sent the shutter open command will be discussed in later sections. The supervisor does keyboard and display processing for SPEC 22 and calls the appropriate modules for the current crew selected ST submode.

7.7.2.2 Self-Test Module

In the Supervisor, figure 7-13, DO CASE 1 performs the actions for self-test. It checks that the star tracker is not comm-faulted and then takes one of two paths depending on if it is first pass for self-test. For the first pass, the not bright object flag is saved for checking at the beginning of self-test. The self-test module (figure 7-15) is called and if it returns with it's case still 1 (meaning there is a bright object alert), it will mode of the star tracker to term idle canceling self-test. Otherwise the supervisor calls the self-test module every 960 milliseconds until it completes.

The self-test module, figure 7-15, performs ten do cases at the 960 millisecond rate. The module will send an offset mode command to K-load positions, wait for three cycles to check if the bite star was tracked correctly. It then breaks track and commands offset mode to a different position, waits for three cycles again to check that the tracker failed to track the bite star. It then turns off self-test mode and check the last cycle that the tracker did not remain in self-test.

7.7.2.3 Star Track Module

The Star Track Module flow is shown in figures 7-16 through 7-18. If there is no bright object, this module will be performed every 160 milliseconds. This submode will either check for a crew input star within the tracker FOV or search the star catalogs for a star within the tracker FOV. If no star is in the FOV, the 'OUT FOV' status is set and the module waits 5 seconds before moding to term idle for the crew input case or initiating another catalog search.

The star catalog search takes 3.2 seconds to complete looking at all 100 stars (1.6 seconds for the 50 stars in OPS 3). After looking at all stars, the search selects from the stars in the FOV the star closest to the center of the FOV to attempt to acquire. Actually only 99 (49) stars are checked because star 11, Sirius, is not considered a candidate. It is so bright that it may cause secondary images which could cause false tracks.

When a star is determined to be in the FOV, the module first checks the shutter position. If the shutter is closed with no bright object alert present, the module sets the shutter open command and the unlatch flag to 'ON' and then commands offset mode. If after 4.32 seconds in offset there is no star present, the module commands full field mode. If after 40.18 seconds in full field there is no star present, the 'NO TARGET' status is set and as before modes to term idle for the crew input case or initiates another catalog search. Once a star present is acquired the module calls the Data Filter module for 21 counts (3.36 seconds). The Data Filter will return a 'FALSE TRK' status or place the star in the star table.

7.7.2.4 Target Track

The Target Track Module flow is shown in figures 7-19 and 7-20. When in the target track submode the supervisor will call this module every 160 milliseconds. This module works similarly to the Star Track module except it use the target state vector to determine if the target is in the FOV. If it is in the FOV, the module does the same check of the shutter position. If the shutter is closed with no bright object alert present, the module sets the shutter open command and the unlatch flag to 'ON' and then commands offset mode. The module will remain in offset mode for 4 seconds before commanding full field mode if no star present is obtained. The full field mode will continue for 20 seconds before setting the 'NO TARGET' status if no star present is obtained. Once a star present is acquired the module calls the Data Filter module for 21 counts (3.36 seconds). The Data Filter will set the NAV TARGET FLAG and DF DATA GOOD flags 'ON'.

7.7.2.5 Data Filter

The Data Filter Module flow is shown in figures 7-21 through 7-23. This module is call by either the star track or target track modules at a rate of 160 milliseconds. It accumulates and averages the horizontal and vertical positions over 21 data samples. If the orbiter rate is greater than 0.75 degree per seconds, it sets the status to 'HI RATE' and exits. Otherwise, if the submode is Target Track, it sets the NAV TARGET FLAG and DF DATA GOOD flags 'ON' and returns to Target Track module. For Star Track submode it continues to test that the tracked object is the correct star. The module computes the measured line of sight (LOS) and then computes the separation angle between the measured and actual LOS computed from the IMU attitude and the catalog star vector. This separation angle must be within 0.7 degrees in offset mode or 1.4 degrees in full field mode. The Data Filter then computes the separation angle between the first and twenty-first data sample LOS. This must indicate the object has not moved more than 0.0034 degree per second. In other words it is an inertial object.

If there are stars in the star table, the measured and actual catalog separation angles between the LOS of the star in star table slot number 1 are compared. If their difference is greater than 0.2 degrees, one of the stars is considered bad. If there is only the one star in the star table, the star table is cleared and the present star discarded with a status of 'FALSE TRK' because the Data Filter does not know which one is bad. If there are more than one star in the star table, then the present star is considered bad and discarded. It is important to note at this point that the catalog position vector used in this computation for the star in slot number 1 is one that

has been saved in a local variable in the Data Filter module. This will be discussed in more detail later, but it allows the test to be valid for the star in the star table even if its catalog vector to be changed by uplink.

After passing this last test the present tracked star will be placed in star table slot number 1. The Data Filter must now decide which other stars remain in the star table. If there are three stars in the table and the oldest one has been in the table over one hour, it is discarded and the remaining two stars are moved over one slot. If none of the three stars is over one hour old, the Data Filter will select from the four stars (the new star and the three in the table) the two which have a separation angle closest to 90 degrees. These two will be kept and the oldest of the remaining two stars will be discarded.

At this point the Data Filter will save the catalog position vectors of the stars in the star table into local variables. Also the two stars in the star table with a separation angle closest to 90 degrees and also within ± 35 degrees of 90 are selected for use in IMU alignments.

7.7.2.6 Field of View Test

The Field of View Test flow is shown in figure 7-24. This module is called by either the star track or target track modules at a rate of 160 milliseconds. The star or target position vector is transformed into the star tracker coordinate system and horizontal and vertical offset positions are computed. If the absolute value of either the horizontal and vertical offset positions is greater than 4.5 degrees, then the object is not considered in the field of view and the status 'OUT FOV' is returned.

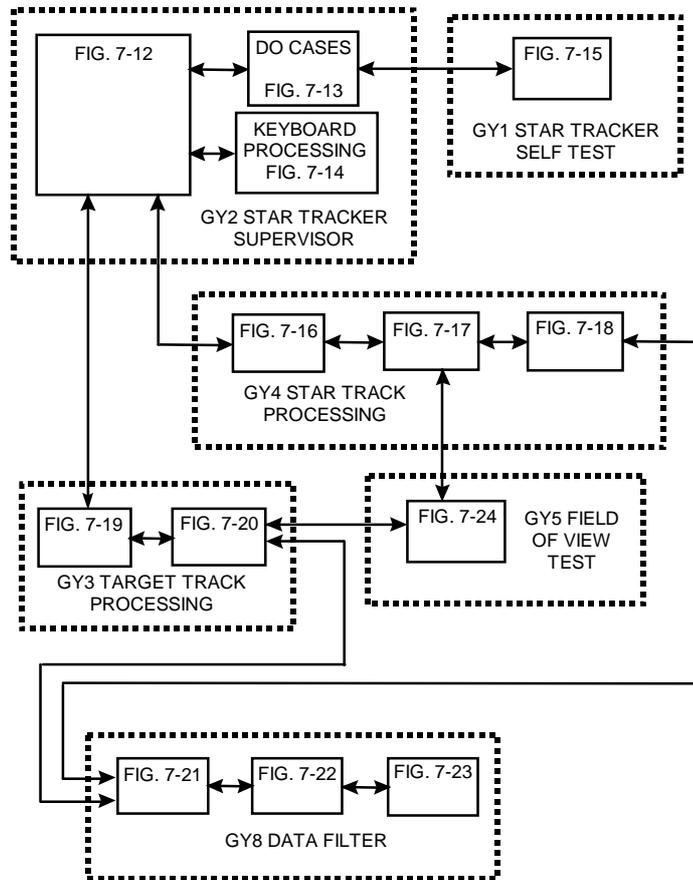


Figure 7-11 Star Tracker Software Overview

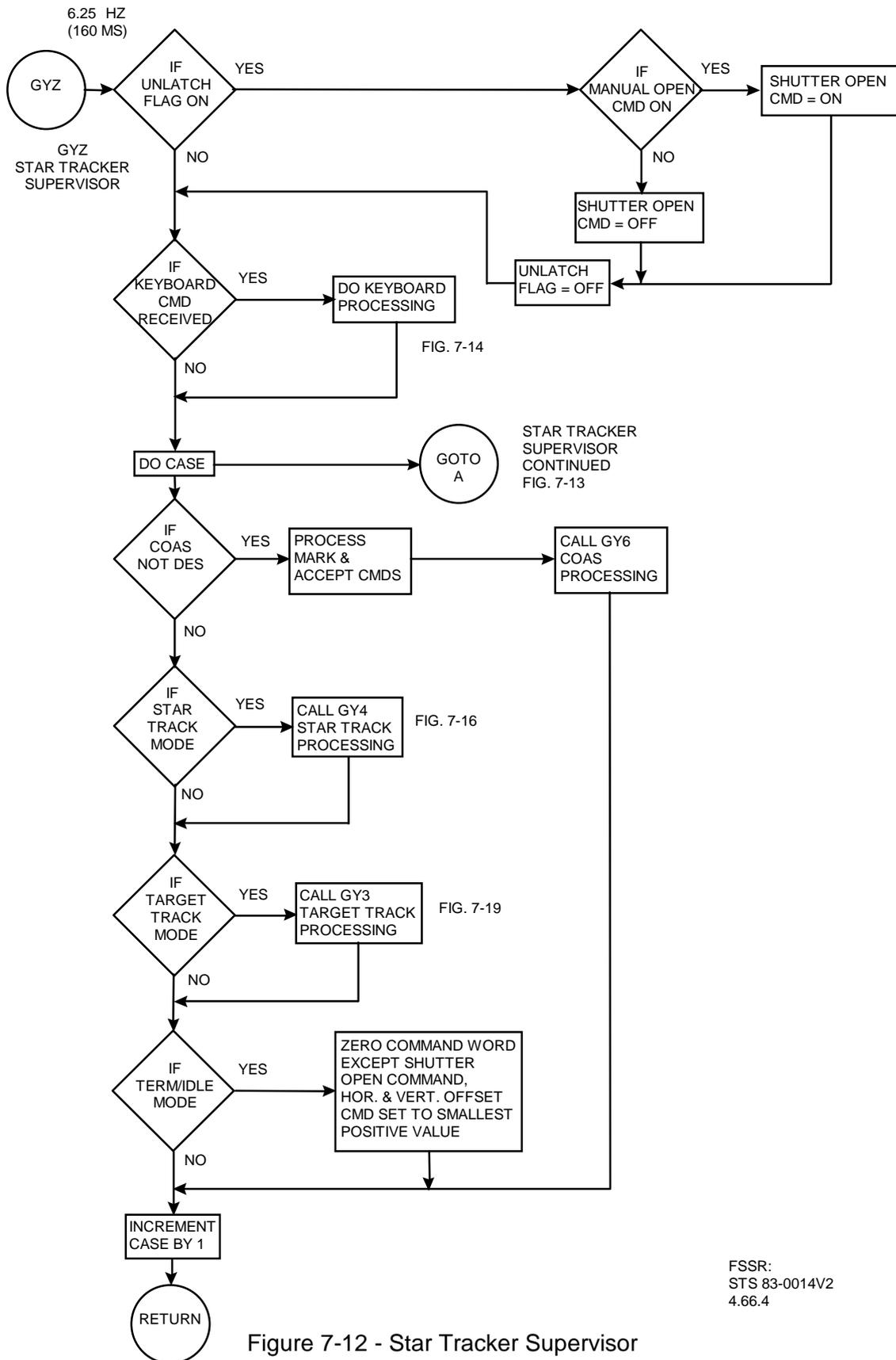


Figure 7-12 - Star Tracker Supervisor

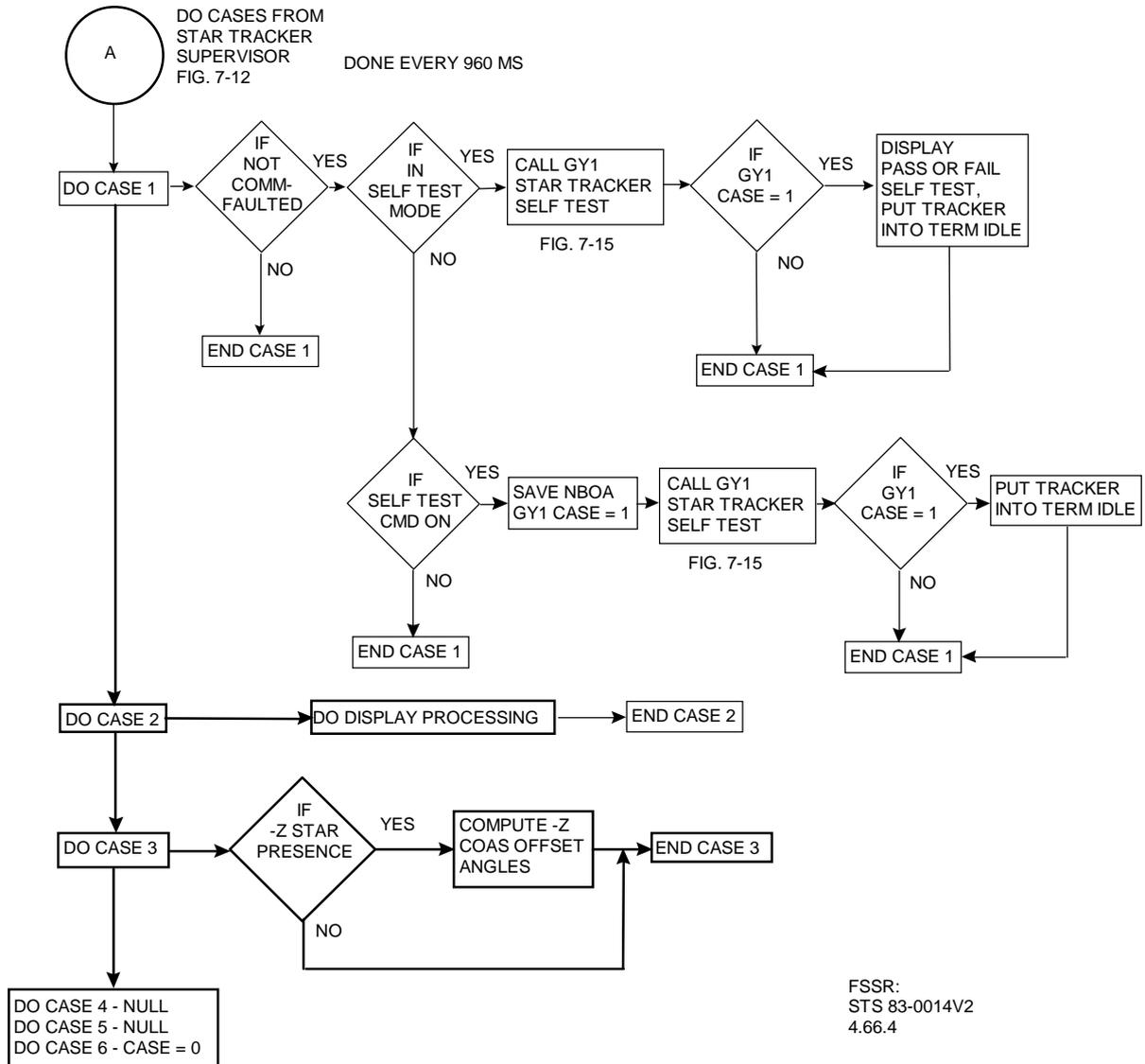


Figure 7-13 Star Tracker Supervisor Continued

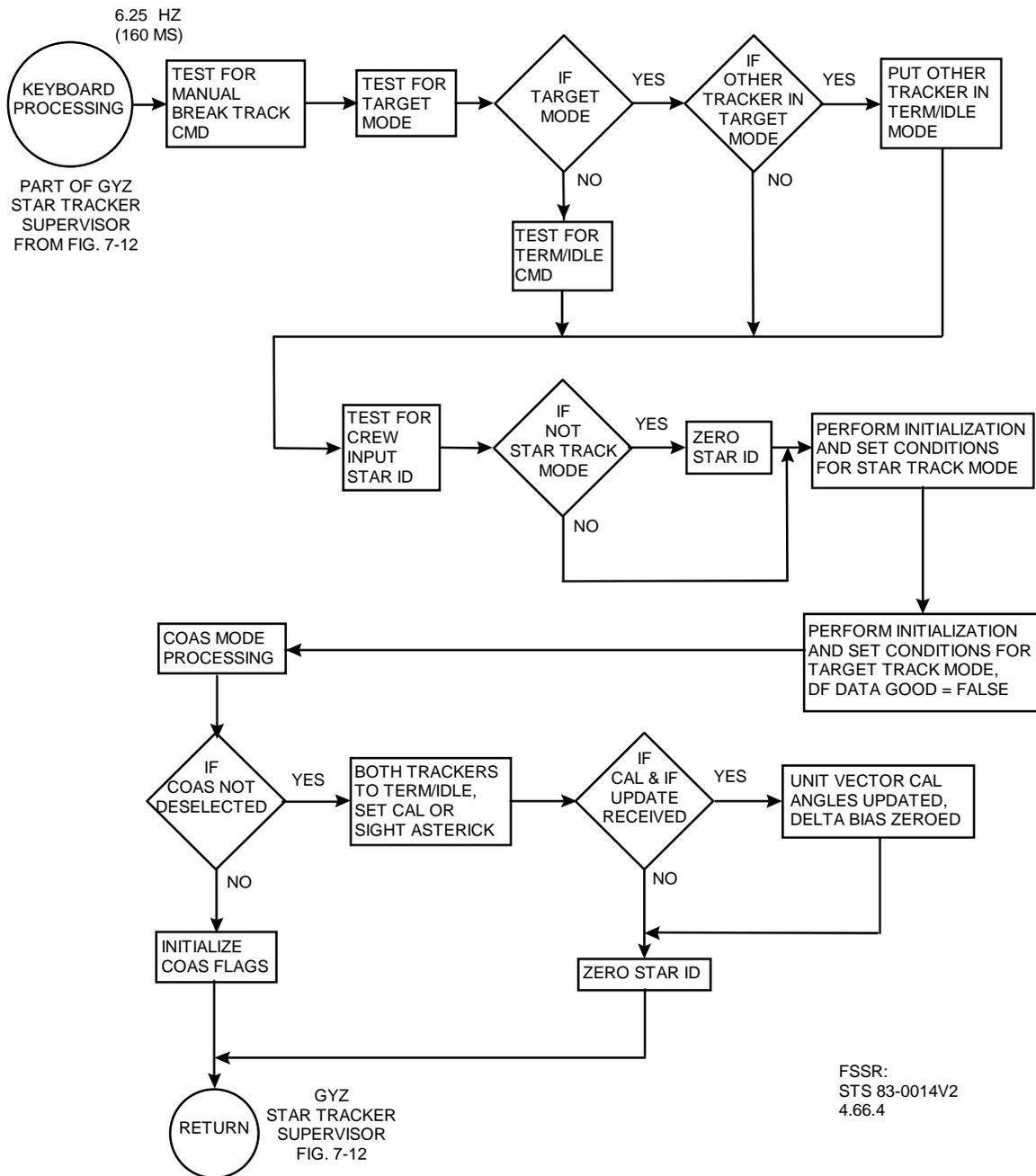


Figure 7-14 Keyboard Processing

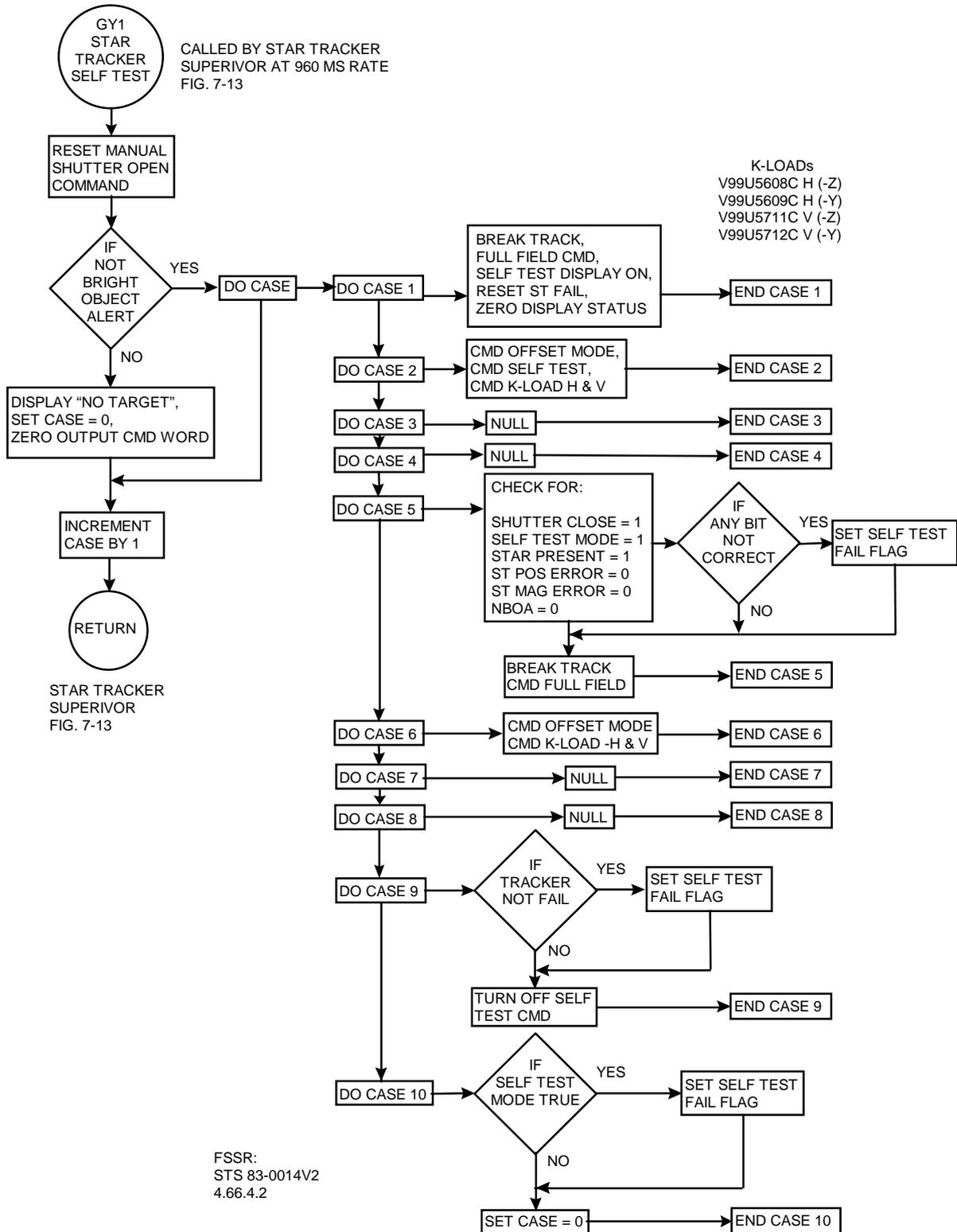


Figure 7-15 Star Tracker Self Test

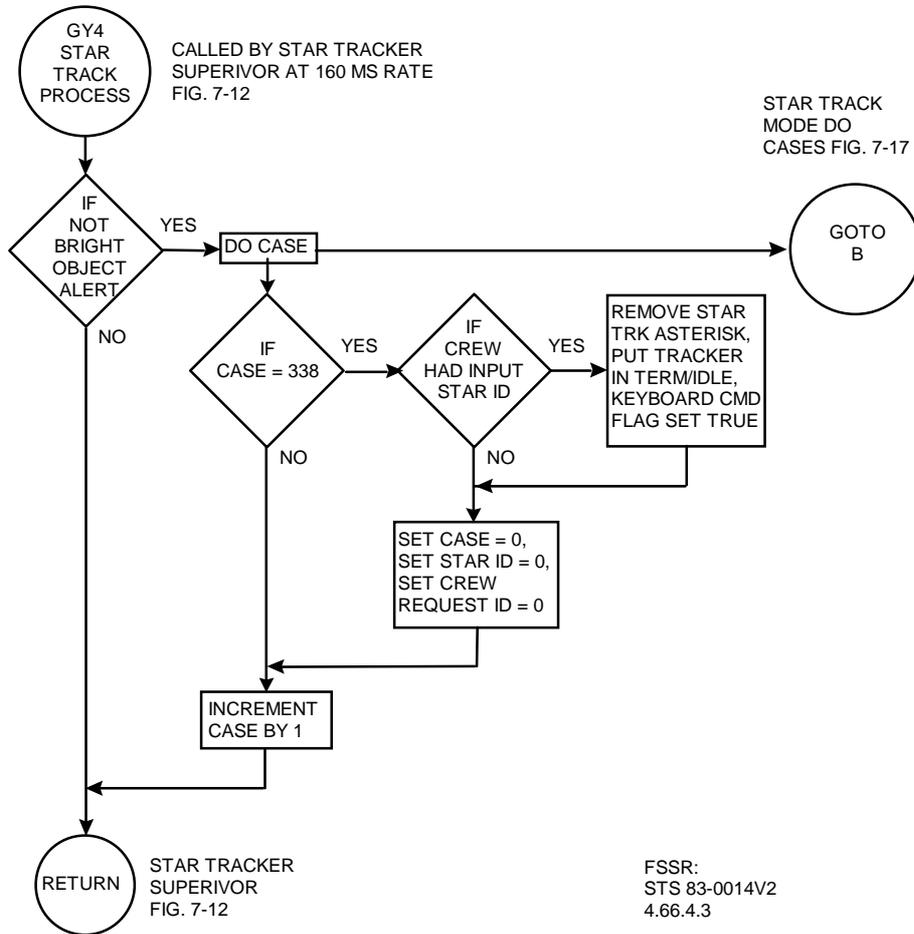


Figure 7-16 Star Track Mode

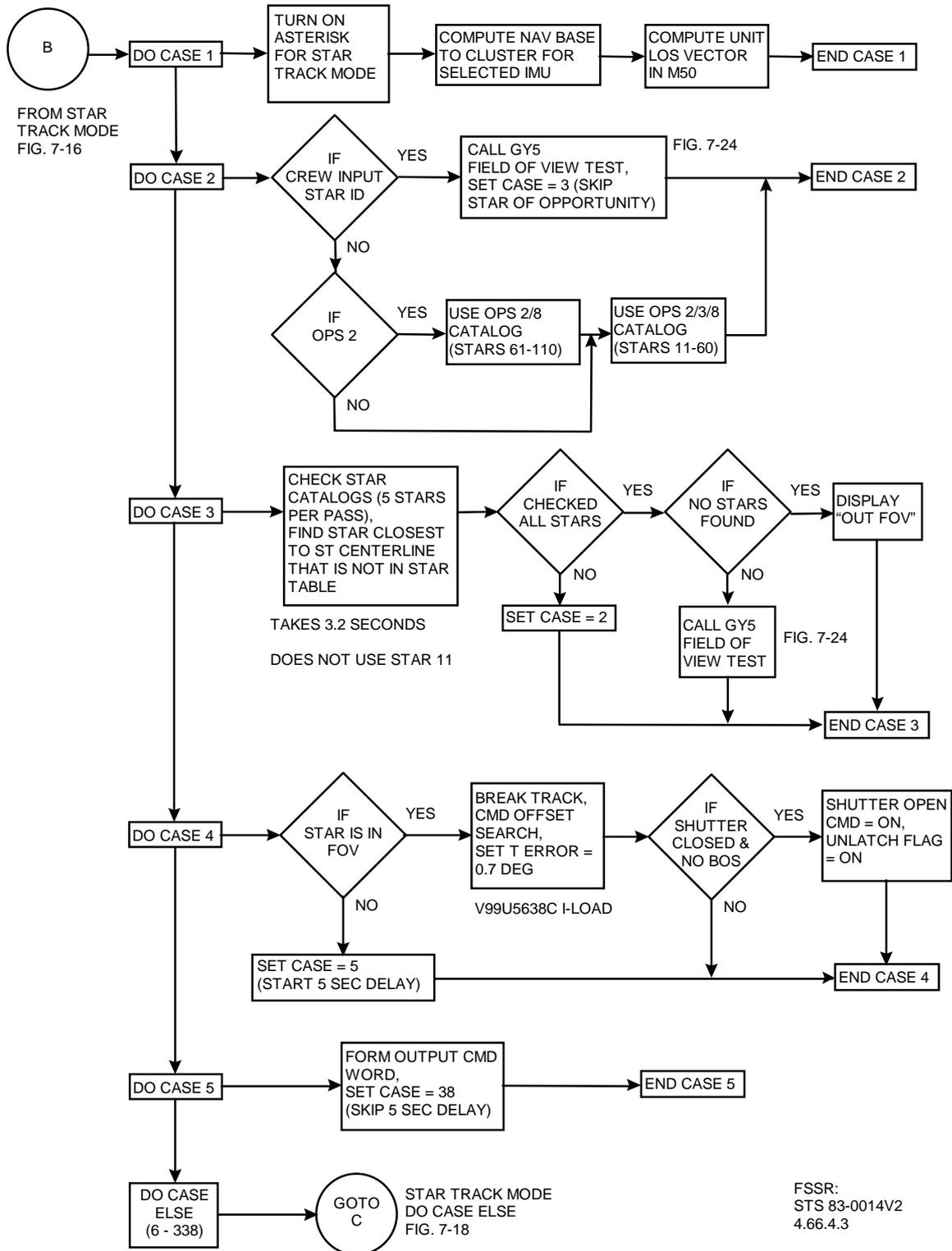


Figure 7-17 Star Tracker Mode Do Cases

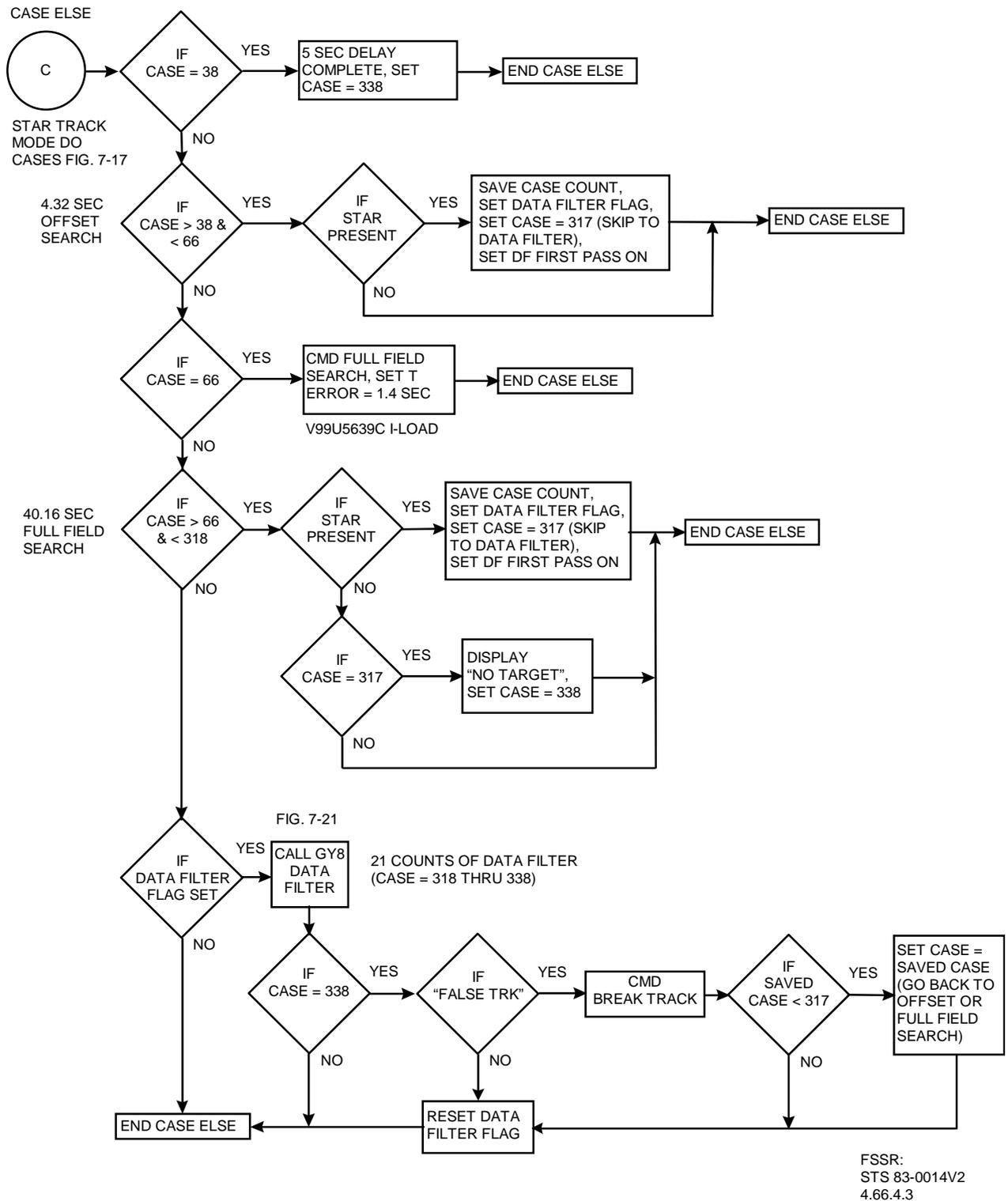


Figure 7-18 Star Track Mode Do Case Else

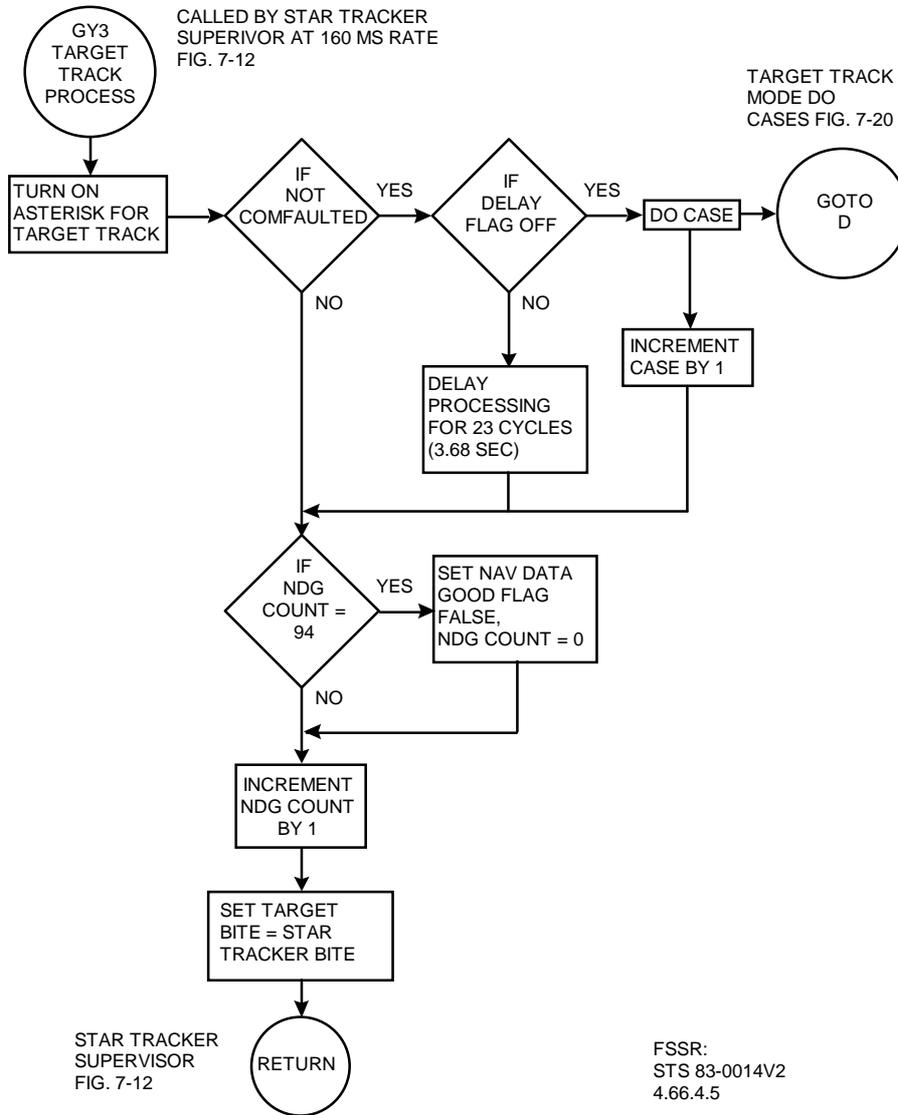


Figure 7-19 Target Track Mode

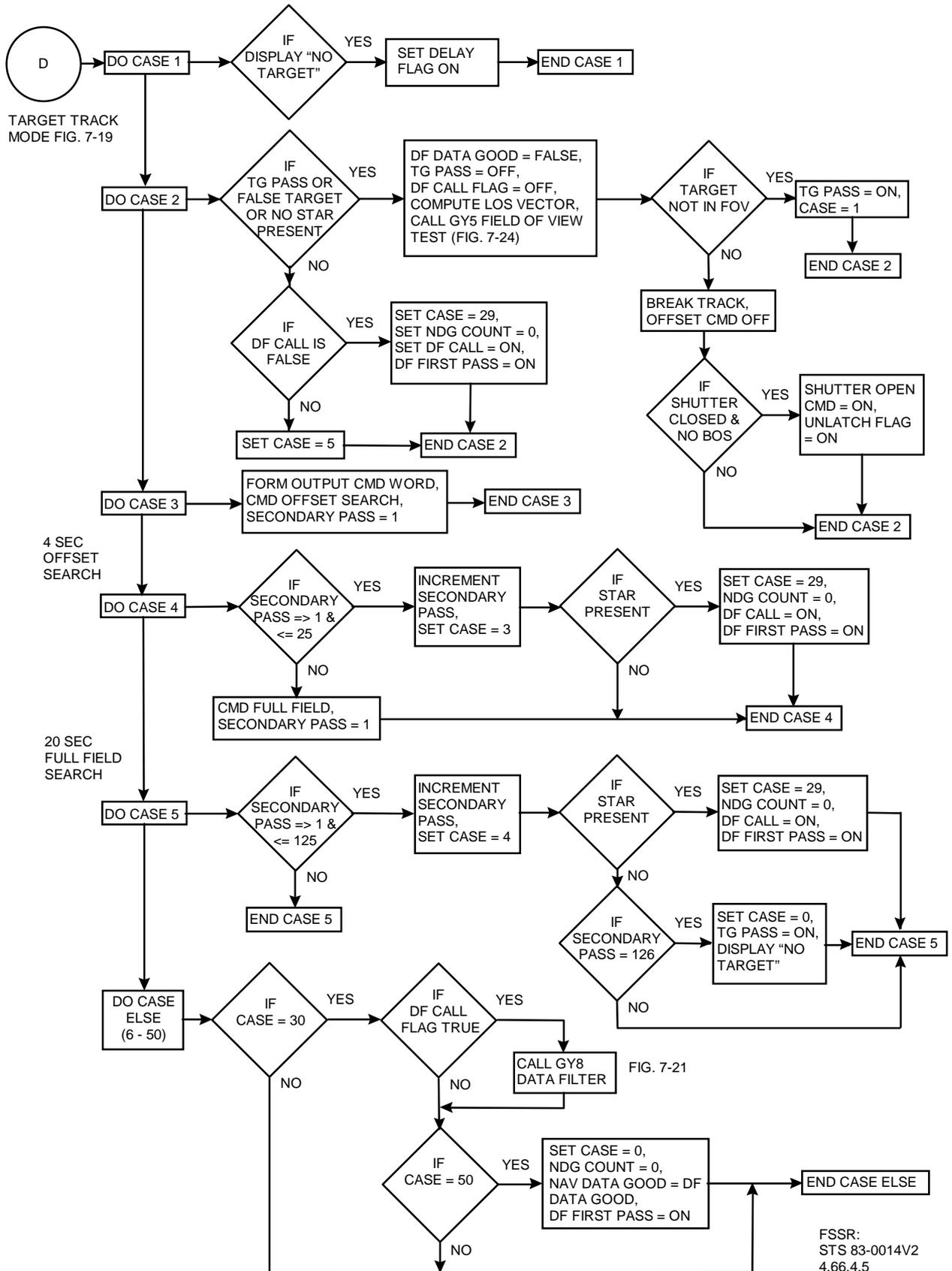


Figure 7-20 Target Track Mode Do Cases

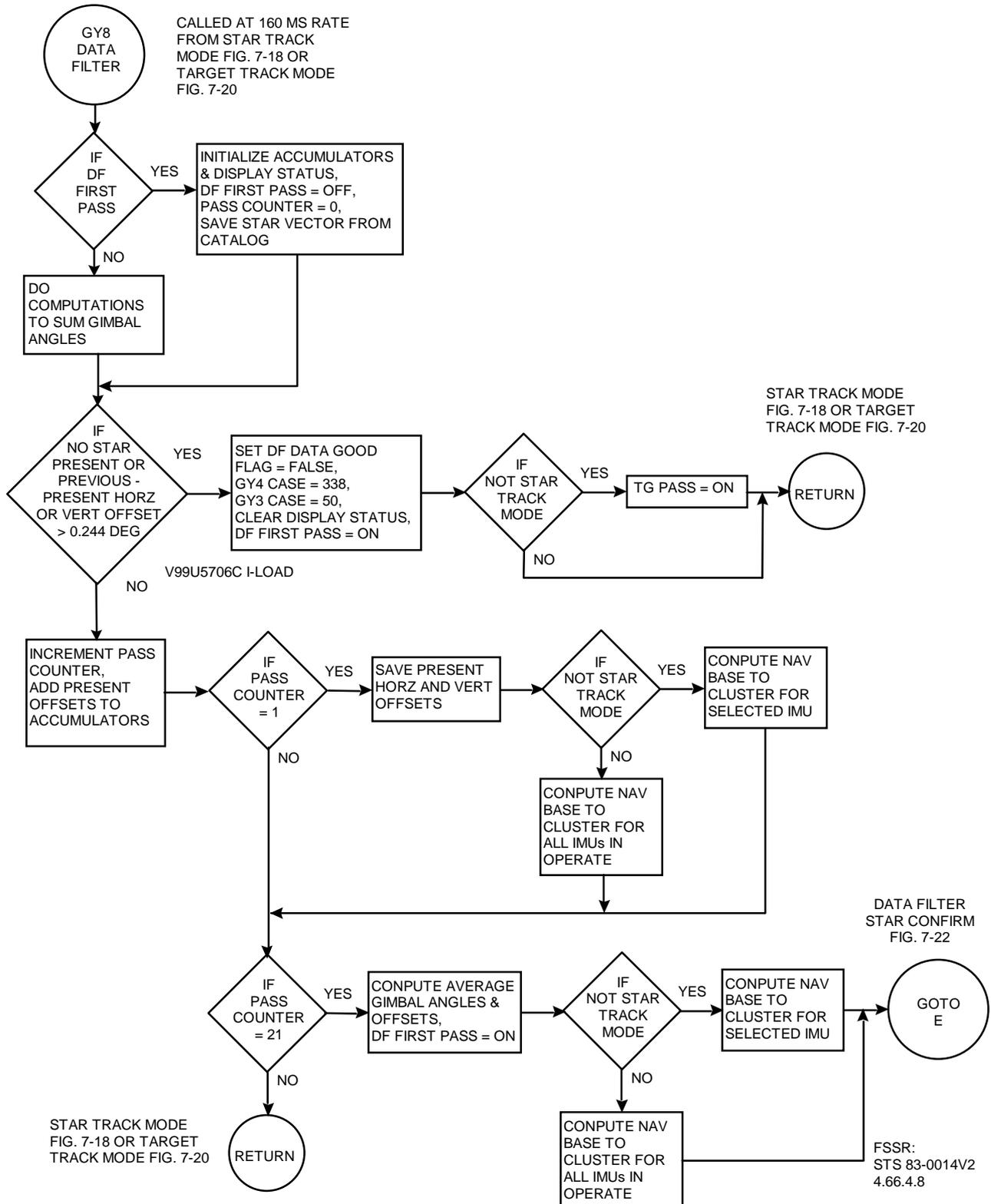


Figure 7-21 Data Filter

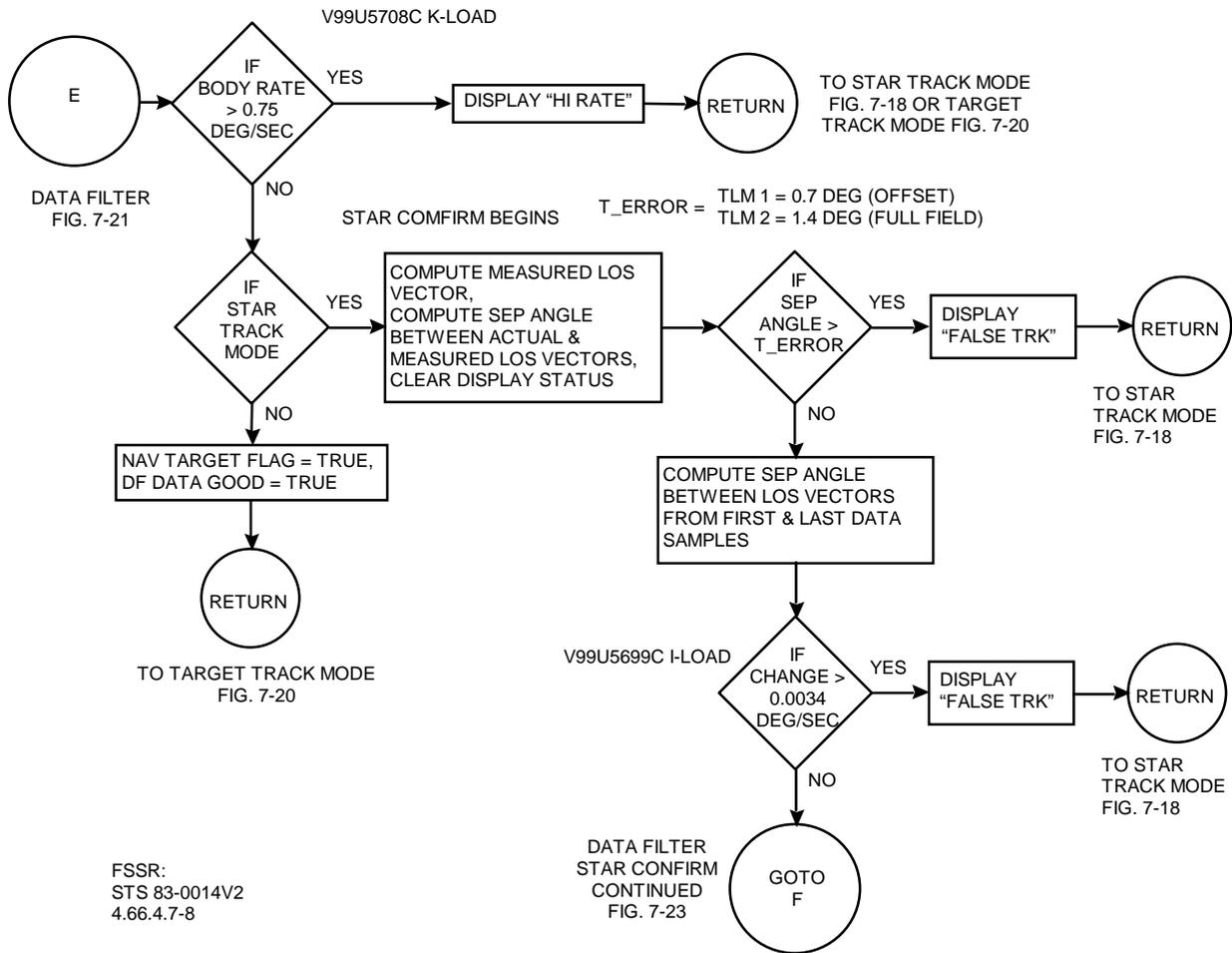


Figure 7-22 Data Filter Star Confirm

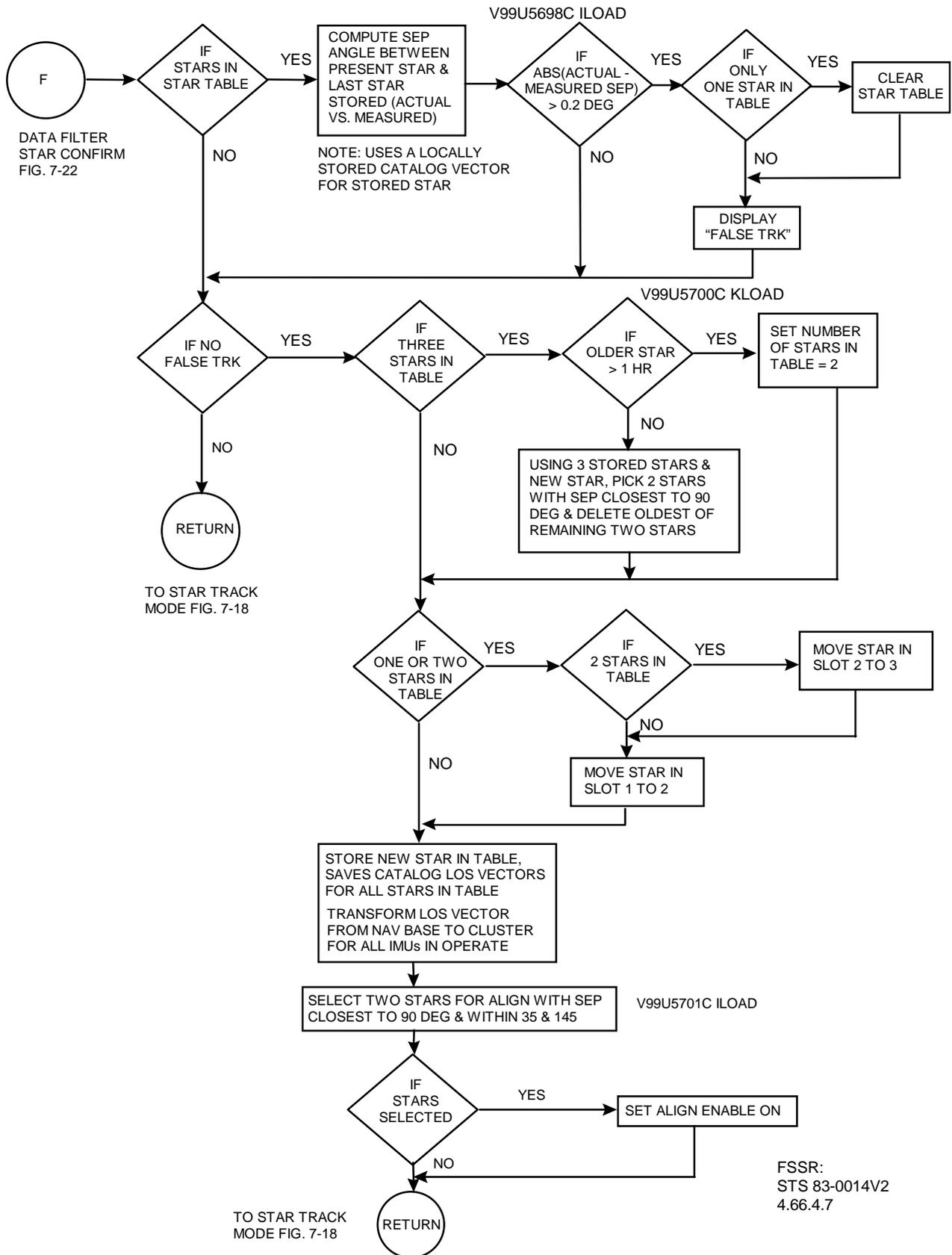


Figure 7-23 Data Filter Star Confirm Continued

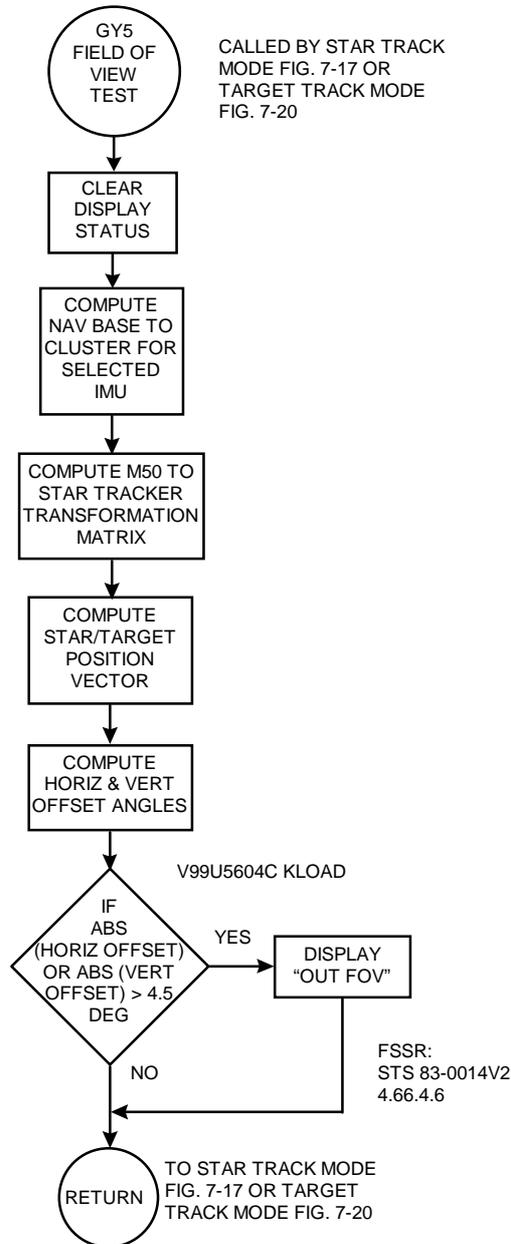


Figure 7-24 Field of View Test

7.9.3 Self-Test History

A data base of the history of self-test results for each star tracker can be found on the web through the GNC home page (http://p51.ksc.nasa.gov/gnc_home/gnc.htm).

7.10 COMMAND/DATA FLOW

The flow of input control commands and output tracker data is controlled by the MDM-generated message out, message in, and word discrete signals. The operation of the data transfer is as follows:

- a. The message-out discrete controls data transmission from the MDM to the ST.
- b. The message-in discrete controls data transmission from the ST to the MDM.
- c. The word discrete enables turn-on and turn-off of the individual words transmitted between the MDM and ST.
- d. Transmission of word discrettes and subsequent data is initiated when the message discrete is switched to a logic 1 state, and shall be ended when the message discrete is switched to a logic 0 state. When a multiple word message is transferred, the message discrete remains in the logic 1 state during the entire multiple word message.
- e. Multiple word transfers always take place beginning with the first word. The number of words transferred is a function of the number of word discrettes issued from the MDM while the message discrete is true. Regardless of the duration of the message-in discrete, the start of the next message-in discrete initiates the data transfer beginning with the first word.

7.10.1 Data Transmission

- a. The data are transmitted using pulse code modulation (PCM).
- b. The data code is a Manchester II. A logic 1 is transmitted as a bipolar coded signal; i.e., a positive pulse followed by a negative pulse.
- c. The data rate shall be one megabit per second, plus or minus 0.1 percent average per word.
- d. The clock is generated by the transmitting device and used in the fabrication of the Manchester II code format. When receiving data, the receiver derives the clock from the Manchester II code format as received.

7.10.2 ST Interfaces

The ST interfaces with the GPC via the serial digital multiplex-demultiplex input and output data channel. The digital input/output words are 16 bits in size, plus 3 bits for synchronization and 1 bit for odd parity. Generally, throughout the ST SOP, the synchronization and parity bits are disregarded, and the most significant bit (MSB) in the word is designated as bit 0 with the least significant bit (LSB) being bit 15. The true condition for a given parameter is indicated by a

logic 1. The contents, format, and bit values of the single command word to the ST and the three data words from the ST are shown in tables 7-I, 7-III, 7-IV and 7-V.

TABLE 7-I.- COMMAND WORD (GPC TO ST)

| Bit number | Assignment |
|------------|--------------------------|
| 0 | Offset scan mode command |
| 1 | Self-test command |
| 2**** | Break track command |
| 3*** | Shutter open command |
| 4* | Threshold set 1 - MSB |
| 5* | Threshold set 2 - LSB |
| 6** | Offset mode horiz |
| 7 | Offset mode horiz |
| 8 | Offset mode horiz |
| 9 | Offset mode horiz |
| 10 | Offset mode horiz |
| 11** | Offset mode vert |
| 12 | Offset mode vert |
| 13 | Offset mode vert |
| 14 | Offset mode vert |
| 15 | Offset mode vert |

- * 0, 0 = Sensitivity threshold equivalent to 3.0 magnitude
0, 1 = Sensitivity threshold equivalent to 2.4 magnitude
1, 0 = Sensitivity threshold equivalent to 2.0 magnitude
1, 1 = Sensitivity threshold equivalent to 1.0 magnitude

** Refer to table 7-II for command values.

*** When the shutter has been manually opened, there is no protection (neither hardware nor software) for the ST from bright objects.

**** ST responds to a change of state from 0 to 1. ST SOP must reset bit to zero after outputting 1.

TABLE 7-II.- STAR TRACKER OFFSET COMMAND

| Bit values | Angular values (in degrees) |
|--------------------|--------------------------------|
| 6 7 8 9 10 (H) | |
| 11 12 13 14 15 (V) | |
| 00000 | -4.75000 |
| 00001 | -4.44355 |
| 00010 | -4.13710 |
| 00011 | -3.83065 |
| 00100 | -3.52420 |
| 00101 | -3.21775 |
| 00110 | -2.91130 |
| 00111 | -2.60485 |
| 01000 | -2.29840 |
| 01001 | -1.99195 |
| 01010 | -1.68550 |
| 01011 | -1.37905 |
| 01100 | -1.07260 |
| 01101 | -0.76615 |
| 01110 | -0.45970 |
| 01111 | -0.15325 |
| 10000 | +0.15325 |
| 10001 | +0.45970 |
| 10010 | +0.76615 |
| 10011 | +1.07260 |
| 10100 | +1.37905 |
| 10101 | +1.68550 |
| 10110 | +1.99195 |
| 10111 | +2.29840 |
| 11000 | +2.60485 |
| 11001 | +2.91130 |
| 11010 | +3.21775 |
| 11011 | +3.52420 |
| 11100 | +3.83065 |
| 11101 | +4.13710 |
| 11110 | +4.44355 |
| 11111 | +4.75000 |

Note that the S/W shall select the bit value that is closest to the required angle. If the angle is exactly midway between bit values, the bit value in the direction of increasing angle magnitude shall be selected. Angles exceeding $\pm 4.75^\circ$ shall be limited to $\pm 4.75^\circ$ respectively, and the corresponding bit value shall be selected.

TABLE 7-III.- DATA WORD 1 (ST TO GPC)

| Bit number | Assignment |
|------------|-------------------------------------|
| 0* | Tracker good (not failed) |
| 1 | Target suppressed |
| 2** | Transmission word good (not failed) |
| 3 | Optics shutter closed |
| 4 | Self-test mode engaged |
| 5 | Star present |
| 6*** | Always 1 |
| 7 | Always 0 |
| 8-15**** | Star magnitude (MSB-LSB) |

* Word 1, bit 0 tracker good is a combination of the following bits and states:

Word 2, bit 15 = 1 (Pressurization failure)
or
Word 3, bit 12 = 1 (Self-test angle error)
or
Word 3, bit 13 = 1 (Self-test magnitude error)
or
Word 3, bit 14 = 0 (Power supply good)
or
[Word 3, bit 15 = 1 (Bright object sensor BOS alert off)
and
Word 1, bit 4 = 1] (Self-test mode engaged)

If any of the above conditions and states exist, the tracker good flag shall be set to 0 (failed). Normal ST operation occurs when none of the above conditions and states occur. For normal operation, the tracker good flag shall be set to 1.

BOS alert off 1 = No alert present
 0 = Alert present

Power supply good 1 = Good
 0 = Failed

Tracker good 1 = Good
 0 = Failed

Transmission word good 1 = Good
 0 = Failed

** Word 1, bit 2 transmission word good is a negation of the 'or' combination of the following bits:

Word 2, bit 12, bit count error
Word 2, bit 13, Manchester not valid
Word 2, bit 14, parity error

***This note explains a configuration difference that currently exists in the ST hardware. Bits 6 and 7 are not always 1 and 0, respectively. The values of these bits depend upon which serial number tracker is being monitored.

Originally, the ST hardware was designed to measure a wide range of object brightness. Measurement accuracy was increased by having a high and low sensitivity on the video circuitry. Subsequent evaluation of operational requirements indicated that the decreased sensitivity was not required since there was no requirement to measure object brightness at levels above -1 magnitude. Therefore, the two bits that control this sensitivity selection, were set permanently to 1 and 0.

After the initial hardware production, it was decided to use the telemetry space available to relay dynamic information. So, for serial numbers 8 and up, a hardware mod was made and on these trackers the aforementioned discretes now provide information on the temperature of the IDT. The ST IDT design is such that corrections must be applied to the measured H and V angles based on the temperature of the IDT. Four temperature regions have been defined. For either pair of discretes, say V71X5527B and V71X5528B for the -Y tracker, a binary "00" indicates the lowest temperature region and "11" the highest. In flight we may see any of the four possible bit combinations.

These temperature data are of no real operational value to the flight controller and they are only available on serial numbers 8 and up.

**** Data word 1, bits 8 through 15 represent star intensity in unitless numbers. The numerical value of intensity for each star is recorded in the star table.

TABLE 7-IV - DATA WORD 2 (ST TO GPC)

| Bit number | Assignment |
|------------|---|
| 0 | Horizontal star/target position - sign |
| 1 | Horizontal star/target position - 2.6° |
| 2 | Horizontal star/target position - 1.3° |
| 3 | Horizontal star/target position - 0.65° |
| 4 | Horizontal star/target position - 0.325° |
| 5 | Horizontal star/target position - 0.1625° |
| 6 | Horizontal star/target position - 0.08125° |
| 7 | Horizontal star/target position - 0.040625° |
| 8 | Horizontal star/target position - 0.0203125° |
| 9 | Horizontal star/target position - 0.01015625° |
| 10 | Horizontal star/target position - 0.005078125° |
| 11 | Horizontal star/target position - 0.0025390625° |
| 12 | Bit count error |
| 13 | Manchester code not valid |
| 14 | Parity error |
| 15 | Pressurization fail |

TABLE 7-V - DATA WORD 3 (ST TO GPC)

| Bit number | Assignment |
|------------|---|
| 0 | Vertical star/target position - sign |
| 1 | Vertical star/target position - 2.6° |
| 2 | Vertical star/target position - 1.3° |
| 3 | Vertical star/target position - 0.65° |
| 4 | Vertical star/target position - 0.325° |
| 5 | Vertical star/target position - 0.1625° |
| 6 | Vertical star/target position - 0.08125° |
| 7 | Vertical star/target position - 0.040625° |
| 8 | Vertical star/target position - 0.0203125° |
| 9 | Vertical star/target position - 0.01015625° |
| 10 | Vertical star/target position - 0.005078125° |
| 11 | Vertical star/target position - 0.0025390625° |
| 12 | Self-test angle error |
| 13 | Self-test magnitude error |
| 14 | Power supply good |
| 15 | Bright object alert off |

7.11 BITE TECHNIQUE

7.11.1 Description

The BITE being implemented simulates actual ST operation by the use of an artificial star (AS) and an artificial bright object (ABO). The circuitry peculiar to the BITE system consists of the LED's and drivers for the AS and the ABO, the self-test mode logic, and the BITE comparators.

The two BITE operations are accomplished as follows:

- a. The AS is an LED which generates an image that is reflected by a prism into the FOV of the ST. This light source collimated by the ST lens simulates a star. Part of this collimated energy is reflected by the protective window and focused by the ST lens onto the IDT cathode.
- b. The ABO is an LED that has been made an integral part of the BOS. When the ABO is activated, the light energy is sufficient to appear as a bright object to the BOS. The BOS then activates the circuitry to close the shutter in front of the IDT.

7.11.2 Self-Test Operation

BITE operation is initiated when a self-test command is received by the ST interface adapter unit (IAU). At the beginning of the next tracker retrace blanking interval, the test cycle starts with an offset search scan centered about predetermined coordinates, and simultaneously the ABO is turned on, activating the circuitry that closes the optics shutter. When the image of the AS is encountered, the track mode is initiated and the AS is tracked in the conventional manner of a real star. If a difference exists between the detected data and the preset values, this difference appears in the form of a nonzero output of the BITE comparators. For the AS, the position and magnitude signals must compare to preset reference values. For the ABO, the BOA and optics shutter closed (OSC) signals must be present.

7.11.3 Other Circuit Monitors

In addition to the circuitry tested only during the self-test mode, the +5 V and ± 15 V power supplies are continuously monitored for an out-of-tolerance condition. Also, each time the IAU receives a message from the MDM, it completes its own function test of the transmission word failure logic and transmits to the MDM any problems which were encountered during the reception of commands from the MDM. Finally, a continuous monitor of the ST pressurized case is provided to the IAU from which a GO/NO-GO pressure signal is sent.

7.11.4 Unchecked Circuitry

The only circuitry left unchecked by BITE is the threshold command circuitry. The threshold command circuitry could be indirectly checked by alternately setting the commandable threshold above and below that of the AS and noting, respectively, the absence and presence of the star-present signal.

The offset mode circuitry can be partially checked by commanding the offset mode during self-test. If correct offset coordinates are given, a successful test result will be obtained. If coordinates remote from the position of the AS are commanded, a failed condition will result.

Some circuitry is only partially tested during the BITE operation. These circuits include the PROM power switching network, the calibration PROM's, the timing and control logic, the threshold generator, the FOV discriminator, and the temperature sensor discriminator networks.

Because the coordinates of the AS are fixed, only one address of one V and one H PROM pair are checked during the BITE test. Hence, only half of the PROM power switching network is checked and only four out of eight PROM's are tested. Of the 4 PROM's tested, only 1 of 512

addresses is checked. However, the basic functions of the calibration circuitry are tested, and it is believed that most meaningful failures in this circuitry will be checked by this scheme.

A complete check of the calibration circuitry would be too cumbersome to implement in the volume allotted each tracker.

The threshold generator would be partially tested by a 0,0 command from the Orbiter MDM. The 0,0 level corresponds to the most sensitive range of the video detector. The magnitude of the AS should easily exceed this level of detection.

The FOV discriminator is tested only insofar as its operation does not interfere with the BITE cycle, since the AS cannot move out of the FOV and exercise the FOV discriminator.

The tube temperature sensor and discriminator can only be tested for the temperature of the tube at the time the self-test command is given. The particular temperature range which is tested cannot be readily predicted. However, failure of the tube temperature sensor circuitry and probably that of the PROM calibration network could cause a BITE failure to be indicated. Such a failure indication is dependent on final system characteristics, the IDT temperature during self-test, and BITE tolerance established.

7.12 FAILURE DETECTION

7.12.1 Failure Criterion

The failure criterion for BITE position failure system is based upon the ST error budget. If the indicated position of the BITE AS falls outside of a tolerance band of the nominal position stored in the BITE comparators, or the BITE ABO falls outside of tolerance band, a failure will be indicated.

The failure criterion for the BITE SMAG failure detection is a tolerance around the predetermined artificial star magnitude which the magnitude indication is not to exceed. This tolerance is 2.26 M. Any failure or combination of failures or drifts which result in the indicated position or magnitude of the AS exceeding the BITE error tolerance will cause the BITE system to indicate a system failure.

7.12.2 Failure Types

There are several categories or types of failures which can occur in the ST. These are: optical and mechanical, BITE system, IDT tube and focusing and deflection components, and other ST electronic subsystems.

7.12.2.1 Optical and Mechanical

The types of optical failures which can occur include the following:

- a. Window degradation consisting of distortion, cracks, or actual breakage. These defects may not be detected by the BITE unless severe enough to significantly affect AS reflectance off

the window. Position (P) and/or magnitude (M) BITE will be out of tolerance and BITE will indicate a system failure.

- b. Lens degradation consisting of transmission loss or distortion. These defects would be detected if severe enough. M and/or P BITE will be out of tolerance, and BITE will indicate a system failure.

Mechanical failures refer mainly to abnormal shifts in the alignment of the entire ST and/or its optical components. These shifts may be due to abnormally high temperatures, mechanical stresses, or vibrations. The BITE system will detect alignment shifts of the entire ST if they contribute a significant part of 146 arc sec in position alignment of 2.26 in SMAG. If the position is out of tolerance, P BITE will go high.

7.12.2.2 BITE System

- a. AS LED and driver

The types of failure or drift in this part of the BITE system which would trigger a failure indication include drift in the light output of the LED caused by either a drift in LED characteristics or a drift in driver output. The failure indication will be identified by AS magnitude out of tolerance. M BITE will be high.

- b. ABO LED and driver

The type of failure most probable to trigger a failure indicated by BITE would be loss of the LED light output, or a severe degradation in its output. The failure indication will be identified by inability to respond to bright object sensing.

7.12.2.3 IDT Tube and Focus and Deflection Coils

7.12.2.3.1 IDT tube.-

- a. Photocathode degradation

Photocathode aging is a small part of the error budget and normally would not degrade ST performance enough to trigger a failure indication. However, if abnormal photocathode degradation should occur, it will affect the SMAG performance directly, usually causing a fainter than nominal AS M. Such degradation would be detected by the M BITE SMAG detector.

- b. Multiplier degradation

If dynode degradation causes a gain loss (although the gain may show an increase at times) beyond the predicted rate used for the error budget, it may cause the AS magnitude to decrease below the preset tolerance level of the M BITE SMAG failure detector.

- c. Catastrophic failure

Catastrophic failures of the IDT include such items as the following:

- (1) Lead and conductor breakage
- (2) Fracture of the tube causing the tube to become gassy

Catastrophic failures such as those described above will be detected by the P and M BITE SMAG failure detectors.

7.12.2.3.2 Focus field circuit.-

a. Focus coils

The focus coil may fail open or shorted. If the failure is an open coil, the electron image is lost and detected by the P and M BITE failure detectors. If the failure is a shorted coil, the position errors increase or there may be no position indication at all depending on number of turns shorted.

Magnitude errors will also increase and the output may be lost. This will also be detected by the P and M BITE failure detectors. The alignment of the focus coil may change from vibration, shock, or from temperature stresses. If this alignment is significant, a position and/or magnitude error will be indicated during the self-test mode.

b. Focus current regulator

The focus current regulator may fail in one of the following conditions:

- (1) No output
- (2) Unregulated or poorly regulated output
- (3) Output level changed

The first type of failure will have the same effect as an open coil. This type of failure is catastrophic and will be detected by the P BITE and M BITE failure detectors. The second and third type of failure will cause changes in the focal position. If greater than ± 146 sec, the P BITE failure detector will be activated. Any changes in focus current will also change the SMAG output and could activate the P BITE detector.

7.12.2.3.3 Deflection circuits.-

a. Deflection coils

Drifts in the coil resistances are compensated by the fact that the coils are driven by the current sources. Therefore, the only electrical failures or instabilities in the coils that will affect the performance of the ST are shorts and opens in the coil windings. An open coil is catastrophic and will be detected by the P and M BITE failure detectors. A partial failure such as shorting of a section of coil or a section of coil winding will decrease the search deflection reflecting in a smaller FOV in that direction. This type of failure may be detected by the P and/or M BITE failure detector if in such a direction as to affect the search and acquisition of the AS in its preset coordinates.

Deflection coil misalignment, if excessive, may be detected by the P and/or M BITE failure detectors.

7.12.3 Other Electronic Subsystems

There are, in general, two types of failures which can occur in the various electronics subsystems. First, there are hard or catastrophic failures in which a failure occurs causing an entire functional module to become inoperative which in turn causes a significant malfunction in the ST. Second, there is the soft type of failure which may be nothing more than a drift in a component that causes an error in the indicated star position or magnitude (M). These errors may be large enough to exceed the individual BITE error tolerance and trigger the failure indication.

The hard failures are analyzed herein from a reliability standpoint and the failure rates and mean-time-between failures (MTBF) are determined for each electronic subsystem. Failures, drift, or hard failures in the power supply are identifiable to the module level via the power monitor system in the ST. All other failures, due to the end-to-end BITE checkout, cannot be identified to the board level using the BITE test method. To identify the failure to board level would necessitate use of ground support equipment (GSE) to isolate the failure. The failure prevalence as a percentage of the total system failures is given for each subsystem in table 7-VI. This table gives an indication of which parts of the system are most prone to failures although not necessarily drifts. Most hard failures will be detected by the BITE system and those that will not are indicated in table 7-VI.

Soft failures are more difficult to analyze and require a circuit-by-circuit analysis and even system analysis to determine the effect on the system errors. However, a few general comments can be made on soft failures. First, most of the circuitry in the system is of feedback design and, therefore, immune to most solid state component drift. Second, those component drifts to which the circuits are susceptible are due to input and feedback elements and zener diode reference voltage shifts. Finally, offset voltages in operational amplifiers could be a problem if they became excessive.

7.13 SYSTEM FAILURE RATE

7.13.1 Basis for Failure Rate

A prediction of the failure rate per tracker was prepared on the basis of redline schematics available at this time. The prediction was made on parts count using failure rates from (MIL) handbook 217B. The environmental conditions used to establish the failure rates were space flight, ambient temperature of 25° C, and junction temperature of 35° C. Table 7-VII shows the ST/IMU alignment error budget, and table 7-VIII shows the requirements summary.

TABLE 7-VI.- STAR TRACKER SUBSYSTEM

Failure prevalence

| Assembly/module | Prevalence (percent) | Circuitry checked by BITE system |
|--|----------------------|----------------------------------|
| Image dissector tube | 28.6 | Yes |
| Video preamplifier | 0.87 | Yes |
| Digital | 8.3 | |
| • RTB generator | 0.61 | Yes |
| • Break track logic | 0.34 | No |
| • Position sample logic | 0.48 | Yes |
| • Loop sample logic | 0.21 | Yes |
| • Threshold gen. 2 | 0.22 | Partially |
| • Threshold gen. 1 | 0.52 | Partially |
| • Video level det. | 0.26 | Yes |
| • Video filter DC restore | 0.88 | Yes |
| • Peak det. video sample and hold | 0.49 | Yes |
| • Power supply monitor | 0.32 | Yes |
| • Tube temp | 0.16 | Yes |
| • Shutter motor drive and BITE driver | 1.97 | Partially |
| • Press. failure ind. | 0.18 | Yes |
| • Focus current reg. | 0.48 | Yes |
| • Focus current reg. | 0.34 | Yes |
| • Ref. voltage reg. | 0.53 | Yes |
| • Search delay correction gen. | 0.28 | Yes |
| Deflection circuits | 7.8 | |
| • Analog switching | 0.18 | Yes |
| • Staircase gen. | 0.66 | Yes |
| • Staircase gen. | 0.55 | Yes |
| • Offset mode DAC | 0.18 | Partially |
| • Offset mode DAC | 0.46 | Partially |
| • Triangle gen. | 0.57 | Yes |
| • Triangle gen. | 0.64 | Yes |
| • Horizontal defl. ampl. | 0.42 | Yes |
| • Vertical defl. ampl. | 0.42 | Yes |
| • V and H edge sample and hold | 1.26 | Yes |
| • H summing amp. | 0.47 | Yes |
| • V summing amp. | 0.30 | Yes |
| • H and V loop sample and hold output filter | 0.46 | Yes |
| • FOV discriminator | 0.46 | Partially |
| • M/A gain change | 0.77 | Partially |

TABLE 7-VI.- Concluded

| Assembly/module | Prevalence (percent) | Circuitry checked by BITE system |
|---|----------------------|----------------------------------|
| Compensation | 37.5 | |
| • Timing circuit digital comp network | 0.88 | Yes |
| • Tube temp. range comp/encoder | 0.43 | Partially |
| • PROM comp | 13.40 | Partially |
| • PROM comp | 12.97 | Partially |
| • Analog to digital conv. | 1.40 | Yes |
| • PROM switching network | 0.12 | Partially |
| • Add subtract logic and parallel adder | 0.83 | Yes |
| • M/A error function | 0.32 | Yes |
| • Serial output register/BITE comp | 2.29 | Yes |
| • Serial output register/BITE comp | 2.31 | Yes |
| • Serial output register/BITE comp | 0.57 | Yes |
| • Serial output register/BITE comp | 2.30 | Yes |
| Bright object sensor-star tracker | 0.40 | Yes |
| High voltage power supply | 1.33 | Yes |
| • -400 volt regulator | 0.37 | Yes |
| • +1800 volt regulator | 0.17 | Yes |
| • Power supply | 0.79 | Yes |
| Low voltage power supply | 2.25 | Yes |
| Interface adaptor unit | 13.04 | Yes |
| Lens assembly | 0.07 | Yes |

TABLE 7-VII.- ST/IMU ALIGNMENT ERROR BUDGET

| | |
|---|-------------|
| Installation measurement errors (IMU bench installation) | |
| Star tracker | 14 arc sec |
| IMU | 14 arc sec |
| LRU reinstallation errors | |
| Star tracker | 20 arc sec |
| IMU | 20 arc sec |
| Navigation base thermal bending (Star tracker to IMU) | <15 arc sec |
| IMU stable member errors | |
| Pitch to outer roll gimbal | 30 arc sec |
| Case to outer roll gimbal | 28 arc sec |
| resolver errors | 44 arc sec |
| Star tracker error (LOS with respect to case) | 42 arc sec |
| TOTAL ALIGNMENT ERROR (RSS 1-sigma/axis) | |
| All source errors 1-sigma | 82 arc sec |
| 3-sigma error for ST only | 145 arc sec |

TABLE 7-VIII.- STAR TRACKER REQUIREMENTS SUMMARY

| | |
|--|--|
| Number of trackers | 2 |
| Look angle limits | $\geq 29^\circ$ from the Sun; $\geq 19^\circ$ from sunlit Earth horizon |
| Bright source sensor | |
| 1. FOV | 60° dia. |
| 2. Shutter close trip level | -12 magnitude |
| Position accuracy, at a tracking rate of 0.07 deg/sec relative to ST mounting at the ST digital output, 1- σ (degraded accuracy up to 0.05 deg/sec tracking rate) | 60 arc sec |
| Total FOV, square | 10 x 10 deg |
| Offset scan reduced FOV, square | 1.0 x 1.0 deg |
| Warmup time | 15 minutes |
| Star tracking | |
| 1. Threshold brightness | +3 magnitude (S-20) |
| 2. Brightness resolution | 1 bit of 256, saturates at -1 magnitude |
| 3. Acquisition time search, acquire and track | |
| a. Total FOV | 12 sec |
| b. Offset FOV | 1.2 sec |
| Target tracking | |
| 1. Target brightness magnitude | +3 to -7 magnitude |
| 2. Target size | 8.0 arc min max |
| Orbiter maneuver rates during data snap | < 0.2 deg/sec; degraded accuracy up to 0.5 deg/sec |
| Angular difference between LOS's to a selected pair of stars | > 35 and < 45 deg |
| Magnitude accuracy | ± 0.6 magnitude |
| Door opening times | |
| 1. Two motors | 6 sec |
| 2. One motor | 12 sec |
| Target suppress function | |
| 1. FOV | 10 x 10 deg |
| 2. Shutter close and latch level | -8 magnitude (S-20) |

7.13.2 Results

The failure rate per tracker is estimated to be 10.4863 per 10^6 hr with an MTBF of 95,363 hr. This estimated value of MTBF is significantly greater than the required value of 10,000 hr minimum. The IDT at 3.0 per 10^6 hr and PROM's at 2.792 per 10^6 hr are the most significant contributors to the system failure rate.

7.13.3 Failure Rate Contributed by BITE

Since the BITE technique is simply (1) computing the position of a fixed source; and (2) verifying the shutter closure due to an ABO; the additional circuitry is limited to turning on the sources and comparing the position and intensity of the source to a reference. Therefore, the additional circuitry only contributed 0.4782 per 10^6 hr or 5.0 percent to the system failure rate. The probability of failure detection of each failure type is one minus the unreliability of the BITE system (1 - 0.050). This gives the probability of detection by the BITE system of each failure type of 95 percent.

7.13.4 Bright Object Damage

Degradation in IDT tracker performance from exposure to bright sources may be caused by (1) a loss of photocathode sensitivity, (2) a loss of gain in the dynodes, or (3) a combination of the above. The level of performance degradation and the amount of damage depends on (1) source geometry, (2) brightness level, (3) duration of exposure, and (4) individual tube characteristics. Ball System Engineering Report #2614-232, "Effects of Excessive Light on Image Dissector Tube Performance", gives a general indication of the consequences of exposing IDT trackers to bright sources.

The Ball Engineering report states that the Sun generates more than 700 times the photocathode maximum power flux density, resulting in permanent damage to the photocathode. It is important to note that dynodes are more sensitive to high energy resulting from sun exposure than the photocathode. Sun exposure will instantly destroy the dynode and the star tracker will no longer be operable.

In addition to the effects of sun exposure, the Ball Report discusses the effects of exposure to the Moon and sun lit Earth. In general, the photocathode should not be permanently damaged from exposure to the Moon or Earth. However, exposure to the Moon or bright Earth for more than a very brief time will result in permanent degradation of the tube dynodes (reduced tracker sensitivity). During nominal operation, the target suppress threshold will be reached and the shutter closed before the image of the Moon, Earth or Sun fills the field of view of the tracker. This will protect the dynode from being damaged.

In summary, if the Moon, Earth or Sun is in the star tracker field of view, the shutter should not be commanded open if at all possible, to preclude loss of tracker sensitivity (gain loss in the dynodes) or permanent photocathode damage.

7.14 STAR TRACKER I-LOADS AND K-LOADS

Many parameter values are I-Loaded or K-Loaded into GPC memory prior to each flight. Of special concern when developing and using the IMU alignment procedures are those parameter values associated with the ST SOP for collecting and evaluating star LOS data. Since most alignments during a mission are made using data on specific stars obtained while the vehicle is holding at a specific attitude, with the crew being the final data filter, some of the I-Loaded acceptance criteria are more lenient than will be the case for later flights. This leniency is primarily to allow for some tolerance on the system's performance as predicted by mathematical analysis. Without the tolerance, the software may prevent a sighting or discard sighting data without allowing the crew the prerogative of further evaluation.

Key ST limits (LIM) and tolerances (TOL) I-Loads or K-Loads are as follows:

- a. ST output back difference (TOL 6), 0.244° (I-Load, V99U5706C)

An ST sighting is comprised of 21 marks on the stars that are averaged. After each of the 21 marks other than the first mark, the software does a reasonability check to see if the last mark is within TOL 6 of the next to last mark. Based on hardware performance specs and analyses, TOL 6 could be 0.122° . To be ultraconservative, the analytical estimate was doubled.

- b. Angular body rate limit (TOL 8), 0.75 deg/sec (K-Load, V99U5708C)

If the measured Orbiter rate of angular rotation in any axis exceeds the I-loaded value during a sighting, the sighting will be discarded. The ST will track with rates up to 0.2 deg/sec with guaranteed accuracy and at rates up to 0.5 deg/sec with slightly degraded accuracy. Approximate error will be 0.004° (15 arc sec) for each additional 0.3 deg/sec rotation. Sightings used for alignments will be done at attitude hold with 0.2 deg/sec or smaller rates. The value is placed at 0.75° to allow sightings at higher attitude maneuver rates between alignments.

- c. Star pair separation angle difference (TOL 10), 0.2° (0.003490659 rad) (I-Load, V99U5698C)

If during ST SOP interrogation of the sighting data, the difference between the measured star pair separation angle and actual separation angle is greater than TOL 10, the sighting is discarded. If only one other star is in the star table, that star's sighting data are also discarded. Based on 3-sigma ST operation, the value could be limited to 0.089. The limit was opened for STS-1 to 0.2 in order to provide a margin for error on the analysis and to obtain data on as many sightings as possible. The actual acceptance of data for IMU alignments will be by the crewman using 0.09° as the criterion. This is shown as ANG ERR on the SPEC 22 display.

- d. Star LOS rate limit (TOL 12), 0.0034 deg/sec (3.11×10^{-7}) (I-Load, V99U5699C)

This limit is set to preclude accepting data on debris, ice crystals, satellites, or celestial bodies other than stars. The ST requires 3.2 sec to take 21 sighting marks on an object being tracked. If during this time, there is a shift in the inertial position of the object, the sighting data are discarded.

- e. Old data age limit (TOL 13), 60 min (K-Load, V99U5700C)

If data on a new star are obtained after the star table (SPEC 22) is full, the ST SOP will eliminate the oldest star whose data are older than this limit before applying other criteria to eliminate one of the stars already in the table. In future flights, this limit will probably be set equal to the time required to complete one orbit.

f. Optimum star pair separation angle (TOL 14), 35° to 145° (0.819) (I-Load, V99U5701C)

The ST SOP will automatically select a pair of stars from the SPEC 22 star table for IMU alignment only if the separation angle between their LOS's is within the tolerance given ($|\cos 145^\circ| = |\cos 35^\circ| = 0.819$). This tolerance does not affect the crew's capability to select any pair of stars for an alignment. The separation deadband was set at $90^\circ \pm 55^\circ$ for Orbital flight test (OFT) to provide for sighting on many more star pairs than a more nearly optimum value of $90^\circ \pm 5^\circ$ would allow.

g. Offset scan mode star position limit (T LIM 1), 0.7° (not a I-Load or K-Load)

In the offset scan mode, the ST looks for a star in a 1° square, centered where the SOP thinks the star should be. T LIM 1 is a software limit that will accept the star in this mode only if it is located within a circle, centered where the SOP thinks the star should be, with a radius equal to the limit. The value is set at 0.7° to cover the entire 1° square; i.e., if the ST can find a star, the data will be processed. The need for this limit is being reviewed for possible elimination on later flights.

h. Full scan mode star position limit (T LIM 2), 1.4° (not a I-Load or K-Load)

If the required star is not obtained in the offset mode, the ST scans the full FOV. If it locks onto anything in the T LIM 2 radius of where an eligible star should be, the ST SOP will process sighting data.

T LIM 2 must be large enough to allow for 3-sigma equipment operation including maximum IMU drift between alignments. However, the limit must be small enough to prevent sighting on the wrong star. Based on current alignment schedules and equipment specifications, the value could be 1° or less. Since the crew will provide the final filter of star data based on other criteria ($\text{ANG ERR} = 0.09^\circ$), this limit was set at 1.4° to provide a pad on the math.

Note: The above data is presented for general information, as I-load values may be changed when warranted.

MEDS Dedicated Displays Systems Brief

Table of Contents

| | | |
|-----------|--|----|
| 8 | MEDS Dedicated Displays..... | 1 |
| 8.1 | Overview..... | 1 |
| 8.1.1 | A Brief History of Shuttle Dedicated Displays..... | 1 |
| 8.1.2 | The MEDS Interface..... | 1 |
| 8.1.3 | The Display Suite..... | 4 |
| 8.1.4 | First Order Hold..... | 4 |
| 8.2 | The Primary Flight Display..... | 5 |
| 8.2.1 | Major Mode & FCS Configuration..... | 7 |
| 8.2.2 | Trajectory..... | 9 |
| 8.2.3 | G-Meter..... | 9 |
| 8.2.4 | ADI..... | 10 |
| 8.2.4.1 | Display Features..... | 10 |
| 8.2.4.1.1 | Attitude..... | 11 |
| 8.2.4.1.2 | Rates..... | 14 |
| 8.2.4.1.3 | Errors..... | 15 |
| 8.2.4.1.4 | OFF Flag & Invalid Data..... | 15 |
| 8.2.4.1.5 | Theta Limits Bracket..... | 15 |
| 8.2.4.2 | ADI Processing..... | 16 |
| 8.2.4.2.1 | Nominal ADI Processing..... | 16 |
| 8.2.4.2.2 | Mission Aborts..... | 23 |
| 8.2.4.2.3 | BFS ADI Processing..... | 24 |
| 8.2.5 | HSI..... | 25 |
| 8.2.5.1 | Display Features..... | 25 |
| 8.2.5.1.1 | Lubber Line..... | 25 |
| 8.2.5.1.2 | Compass Card..... | 25 |
| 8.2.5.1.3 | Course Arrow..... | 26 |
| 8.2.5.1.4 | Course Deviation..... | 27 |
| 8.2.5.1.5 | Glide Slope..... | 27 |
| 8.2.5.1.6 | Bearing Pointers..... | 28 |
| 8.2.5.1.7 | Bearing & Glide Slope Flags..... | 29 |
| 8.2.5.1.8 | Range..... | 29 |
| 8.2.5.1.9 | Delta Azimuth and HAC Turn Angle..... | 29 |
| 8.2.5.2 | HSI Processing..... | 30 |
| 8.2.5.2.1 | Entry and GRTLS..... | 30 |
| 8.2.5.2.2 | RTLS Abort (GRTLS Phase)..... | 32 |
| 8.2.5.2.3 | BFS HSI Display (MM's 304, 305, 602, and 603)..... | 32 |
| 8.2.5.2.4 | Ascent HSI..... | 33 |
| 8.2.6 | AMI..... | 35 |
| 8.2.6.1 | Display Features..... | 40 |
| 8.2.6.1.1 | ALPHA..... | 40 |
| 8.2.6.1.2 | M/VEL..... | 40 |
| 8.2.6.1.3 | EAS..... | 40 |
| 8.2.6.1.4 | AMI Flags..... | 41 |
| 8.2.6.1.5 | AMI Controls..... | 41 |
| 8.2.6.2 | AMI Processing..... | 42 |
| 8.2.6.2.1 | Data flow..... | 42 |
| 8.2.6.2.2 | Nominal Data Sequence..... | 43 |
| 8.2.6.2.3 | GRTLS Abort Data Processing..... | 43 |
| 8.2.6.2.4 | Invalid Data..... | 43 |
| 8.2.6.2.5 | BFS AMI Processing (Entry)..... | 43 |

| | | |
|-----------|--------------------------------------|-----|
| 8.2.6.2.6 | Ascent AMI | 44 |
| 8.2.7 | AVVI | 45 |
| 8.2.7.1 | Display Features | 48 |
| 8.2.7.1.1 | Altitude Acceleration..... | 48 |
| 8.2.7.1.2 | ALT RATE | 48 |
| 8.2.7.1.3 | ALT | 49 |
| 8.2.7.1.4 | AVVI Controls..... | 49 |
| 8.2.7.2 | AVVI Processing | 49 |
| 8.2.7.2.1 | Data Flow | 49 |
| 8.2.7.2.2 | Nominal Data Sequence | 50 |
| 8.2.7.2.3 | Abort Processing | 50 |
| 8.2.7.2.4 | Invalid Data | 50 |
| 8.2.7.2.5 | BFS AVVI Processing (Entry) | 50 |
| 8.2.7.2.6 | Ascent AVVI | 51 |
| 8.2.8 | Switches..... | 51 |
| 8.2.8.1 | ADI Attitude, Error, & Rate..... | 51 |
| 8.2.8.2 | ADI Sense | 56 |
| 8.2.8.3 | ADI Attitude Reference | 58 |
| 8.2.8.4 | HSI Select Source..... | 58 |
| 8.2.8.5 | HSI Select Mode | 60 |
| 8.2.8.6 | Air Data | 67 |
| 8.2.8.7 | Radar Altimeter | 67 |
| 8.2.8.8 | Default Switch State..... | 68 |
| 8.2.9 | Flight Phase Details | 69 |
| 8.2.9.1 | OPS 9..... | 74 |
| 8.2.9.2 | OPS 1..... | 75 |
| 8.2.9.3 | OPS 6..... | 83 |
| 8.2.9.4 | OPS 2 & 8 | 88 |
| 8.2.9.5 | OPS 3..... | 89 |
| 8.2.9.6 | BFS | 91 |
| 8.3 | SPI | 98 |
| 8.3.1 | SPI Controls | 104 |
| 8.3.2 | SPI Display Processing..... | 104 |
| 8.3.2.1 | Data Flow | 104 |
| 8.3.2.2 | Nominal Data Sequence | 105 |
| 8.3.2.3 | Abort Processing..... | 105 |
| 8.3.2.4 | BFS SPI Processing | 105 |
| 8.4 | RCS Command Lights..... | 105 |
| 8.5 | FCS Mode Pushbutton Indicators | 107 |
| 8.6 | Documents & Resources..... | 108 |

List of Figures

| | |
|--|----|
| Figure 8 - 1 MEDS Forward Cockpit | 2 |
| Figure 8 - 2 MEDS Archetecture | 2 |
| Figure 8 - 3 Multifunction Display Unit | 3 |
| Figure 8 - 4 MEDS Menu Tree..... | 3 |
| Figure 8 - 5 MDU's CDR 1 and 2..... | 4 |
| Figure 8 - 6 First Order Hold Technique | 5 |
| Figure 8 - 7 A/E PFD Generic Format..... | 6 |
| Figure 8 - 8 Generic GSI and ALT ACCEL Meter | 7 |
| Figure 8 - 9 Trajectory Data | 7 |
| Figure 8 - 10 Major Mode & FCS Examples | 9 |
| Figure 8 - 11 G-Meter..... | 10 |
| Figure 8 - 12 Attitude Determination Indicator | 11 |
| Figure 8 - 13 Attitude Ball Axis & Markings | 12 |
| Figure 8 - 14 Roll..... | 13 |
| Figure 8 - 15 Pitch..... | 13 |
| Figure 8 - 16 Yaw..... | 14 |
| Figure 8 - 17 VERT SIT | 16 |
| Figure 8 - 18 ADI Attitude Data Flow | 19 |
| Figure 8 - 19 ADI Rate Data Flow | 20 |
| Figure 8 - 20 Events Establishing a New Command Attitude | 21 |
| Figure 8 - 21 Universal Pointing Display..... | 22 |
| Figure 8 - 22 ADI Error Data Flow | 23 |
| Figure 8 - 23 Horizontal Situation Indicator | 25 |
| Figure 8 - 24 Heading Geometry | 26 |
| Figure 8 - 25 HSI Course Geometry | 26 |
| Figure 8 - 26 Course deviation geometry | 27 |
| Figure 8 - 27 Glide Slope Indicator | 28 |
| Figure 8 - 28 Bearing & Range Geometry | 29 |
| Figure 8 - 29 HSI Data Flow | 31 |
| Figure 8 - 30 Ascent HSI..... | 34 |
| Figure 8 - 31 Alpha/Mach Indicator..... | 35 |
| Figure 8 - 32 Alpha Tape Graphical Detail | 38 |
| Figure 8 - 33 Velocity Tape Graphical Detail 1 | 39 |
| Figure 8 - 34 Velocity Tape Graphical Detail 2 | 39 |
| Figure 8 - 35 Alpha..... | 40 |
| Figure 8 - 36 Horizontal Situation Display..... | 42 |
| Figure 8 - 37 Data flow to AMI | 43 |
| Figure 8 - 38 Altitude/Vertical Velocity Indicator | 45 |
| Figure 8 - 39 AVVI Altitude Tape | 46 |
| Figure 8 - 40 AVVI Altitude Rate..... | 47 |
| Figure 8 - 41 Altitude Acceleration Meter | 48 |
| Figure 8 - 42 Altitude Acceleration Graphical Details | 48 |
| Figure 8 - 43 ADI Attitude Switch..... | 51 |
| Figure 8 - 44 Inertial Frame - All Mission Phases | 52 |
| Figure 8 - 45 LVIY Frame at Two Points | 52 |
| Figure 8 - 46 LVLH frame | 53 |
| Figure 8 - 47 ADI Rate & Error Switches | 55 |
| Figure 8 - 48 Sense Switch..... | 56 |
| Figure 8 - 49 +X, -X, and -Z Sense | 57 |
| Figure 8 - 50 Comparison of Fwd/Aft ADI Attitude Indications | 57 |
| Figure 8 - 51 HSI Select Source Switches..... | 59 |
| Figure 8 - 52 HSI SELECT Mode Switch | 60 |
| Figure 8 - 53 Flight path geometry - top view | 62 |

| | |
|--|-----|
| Figure 8 - 54 Flight path geometry - side view..... | 63 |
| Figure 8 - 55 Entry Display Mode | 64 |
| Figure 8 - 56 TAEM Display Mode..... | 65 |
| Figure 8 - 57 A/L Display Mode | 66 |
| Figure 8 - 58 Air Data Source Switch..... | 67 |
| Figure 8 - 59 Radar Altimeter Switch | 68 |
| Figure 8 - 60 OPS 9 | 74 |
| Figure 8 - 61 MM 101 | 75 |
| Figure 8 - 62 MM 102..... | 76 |
| Figure 8 - 63 MM 103..... | 77 |
| Figure 8 - 64 MM 103 TAL | 78 |
| Figure 8 - 65 MM 104..... | 79 |
| Figure 8 - 66 MM 104 ATO | 80 |
| Figure 8 - 67 MM 105..... | 81 |
| Figure 8 - 68 MM 105 AOA | 82 |
| Figure 8 - 69 MM 106..... | 83 |
| Figure 8 - 70 MM 601 Pre-PPA..... | 84 |
| Figure 8 - 71 MM 601 Post-PPA | 85 |
| Figure 8 - 72 MM 602..... | 86 |
| Figure 8 - 73 MM 603..... | 87 |
| Figure 8 - 74 Orbit PFD..... | 88 |
| Figure 8 - 75 MM 304..... | 89 |
| Figure 8 - 76 MM 305..... | 90 |
| Figure 8 - 77 BFS MM 102 & 103 | 92 |
| Figure 8 - 78 BFS MM 104..... | 93 |
| Figure 8 - 79 BFS MM 105, 106, & 301-303..... | 94 |
| Figure 8 - 80 BFS MM 601 - Pre PPA..... | 95 |
| Figure 8 - 81 BFS MM 601 - Post PPA..... | 96 |
| Figure 8 - 82 BFS MM 602..... | 97 |
| Figure 8 - 83 BFS MM 603..... | 98 |
| Figure 8 - 84 Surface Position Indicator..... | 99 |
| Figure 8 - 85 GNC SYS SUMM 1 | 100 |
| Figure 8 - 86 SPI vs. Elevon Travel | 101 |
| Figure 8 - 87 SPI vs. Body Flap Travel..... | 101 |
| Figure 8 - 88 SPI vs. Rudder Travel | 102 |
| Figure 8 - 89 SPI vs. Aileron Travel..... | 103 |
| Figure 8 - 90 Top View of Open Speedbrake | 104 |
| Figure 8 - 91 SPI Data Flow..... | 105 |
| Figure 8 - 92 RCS Command Lights..... | 106 |
| Figure 8 - 93 RCS Command Light Functional Diagram | 107 |
| Figure 8 - 94 FCS Mode PBI's (panel F2) | 107 |

List of Tables

| | |
|--|-----|
| Table 8 - 1 Major Mode & FCS Configuration Labels | 8 |
| Table 8 - 2 ADI Display Sequence | 17 |
| Table 8 - 3 BFS ADI Sequencing | 18 |
| Table 8 - 4 BFS Rate & Error Scaling | 24 |
| Table 8 - 5 Table 4 BFS HSI Switch Processing | 33 |
| Table 8 - 6 Max L/D Values | 36 |
| Table 8 - 7 MM 304 & 305 Alpha Limits | 36 |
| Table 8 - 8 MM 602 & 603 Alpha Limits | 37 |
| Table 8 - 9 AMI Displays vs. Controls | 42 |
| Table 8 - 10 AMI First Order Hold Processing | 44 |
| Table 8 - 12 ADI Rate & Error Switch Definitions | 55 |
| Table 8 - 13 Fwd & Aft ADI Error & Rate Needle Differences | 58 |
| Table 8 - 14 HSI Data Sources | 60 |
| Table 8 - 15 HSI Function | 60 |
| Table 8 - 16 Switch Default Postions | 68 |
| Table 8 - 17 BFS ADI Attitude Switch States | 69 |
| Table 8 - 18 PASS Display Component Availability | 70 |
| Table 8 - 19 BFS Display Component Availability | 71 |
| Table 8 - 20 RCS Command Light Function | 106 |
| Table 8 - 21 FCS Mode PBI Light Logic | 108 |
| Table 8 - 22 Major Mode | 109 |

SECTION 8

MEDS Dedicated Displays

8.1 Overview

This systems brief describes the function of the GNC dedicated displays. The displays include the Attitude Director Indicator (ADI), the Horizontal Situation Indicator (HSI), the Alpha/Mach Indicator (AMI), the Altitude/Vertical Velocity Indicator (AVVI), the Surface Position Indicator (SPI), FCS Mode switches, and the Reaction Control System (RCS) Activity Lights.

8.1.1 A Brief History of Shuttle Dedicated Displays

The current shuttle fleet utilizes an electronic glass cockpit to provide information to the pilot. Originally, the majority of these displays consisted of dedicated "steam gauge" devices that required digital to analog conversion of data from the General Purpose Computers (GPC's). Because of the difficulty in maintaining the steam gauge technology, the hardware was replaced by a software-driven electronic suite of displays, the Multifunction Electronic Display System (MEDS). In addition to the logistical advantages, the MEDS-based system also provides the flexibility to configure the piloting displays to meet the task at hand, and allows specialized functions to be incorporated without display hardware changes.

MEDS is considered a DPS subsystem, and is the responsibility of the DPS officer in the MCC. GNC is a prime user of the system, and is the MCC point of contact for dedicated display software. To this end, this systems brief will describe the function of the GNC dedicated displays as provided by the MEDS, and describe how the displays are used in each phase of flight. At the end of the brief is a list of applicable documents that should be consulted for details on requirements and specifications. For the reader interested in a system-level description of the MEDS, reference DPS Systems Brief 2 MEDS.

8.1.2 The MEDS Interface

Figure 8 - 1 shows the current MEDS-based Orbiter forward flight deck. Nine panels known as Multifunction Display Units (MDU's) form the core of the MEDS interface. A system diagram is provided in Figure 8 - 2.

The MDU is the most visible component of MEDS, and is comprised of a plasma display, bezel keypad, power switch and illumination intensity controls. The Integrated Display Processor (IDP) provides the interface between the MDU and GPC. In MEDS, the IDP replaced the Display Driver Unit (DDU) used to drive the old steam gauges. A second hardware interface is provided by the Analog/Digital Converter (ADC). The ADC processes hard-wired parameters (such as OMS Pc) that once drove the analog tape displays. In the case of GNC dedicated displays, the SPI continues to be driven by FF 1 and FF 2 MDM data, only now it is processed through an ADC. The software driving the dedicated displays is co-located in both the IDP and the MDU. The dedicated display is drawn at the MDU, and the information driven by software in the IDP based on GPC input. The ADI and HSI are drawn at the MDU by dedicated software, while the AMI/AVVI and SPI utilize utility tape library functions.

At the lower edge of the MDU is a complement of six bezel keys (or edge keys). The edge keys are used to select displays by navigating a series of menus, and are detailed in Figure 8 - 3. A breakdown of dedicated displays menus is provided in Figure 8 - 4.

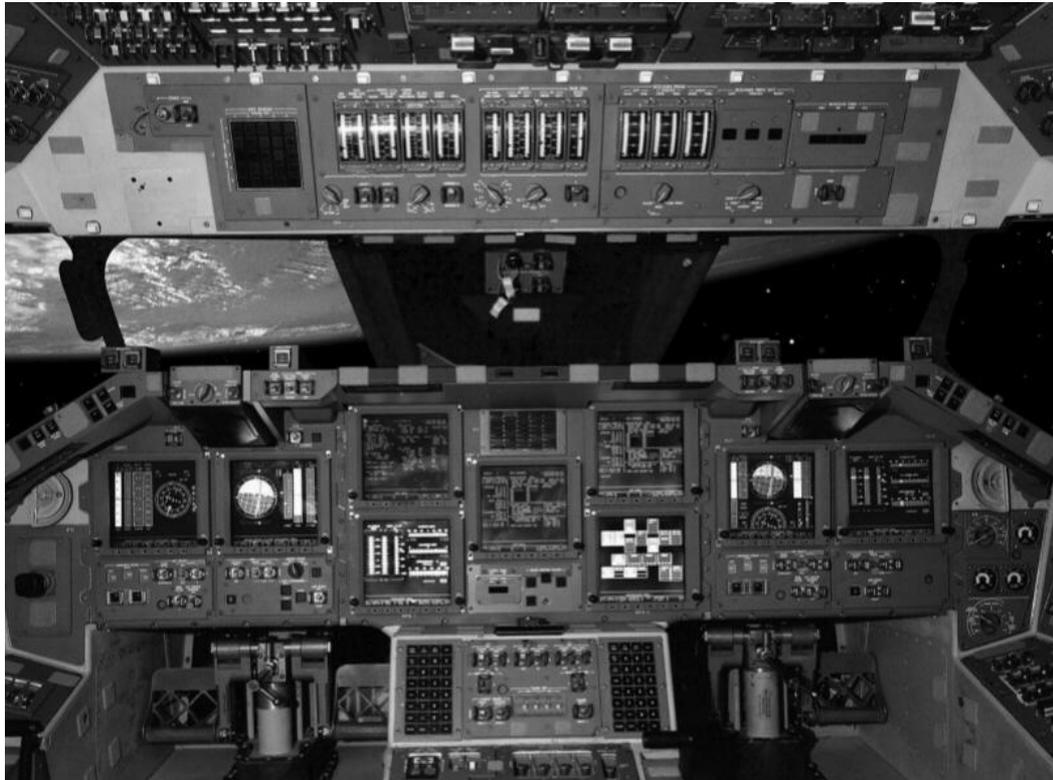


Figure 8 - 1 MEDS Forward Cockpit

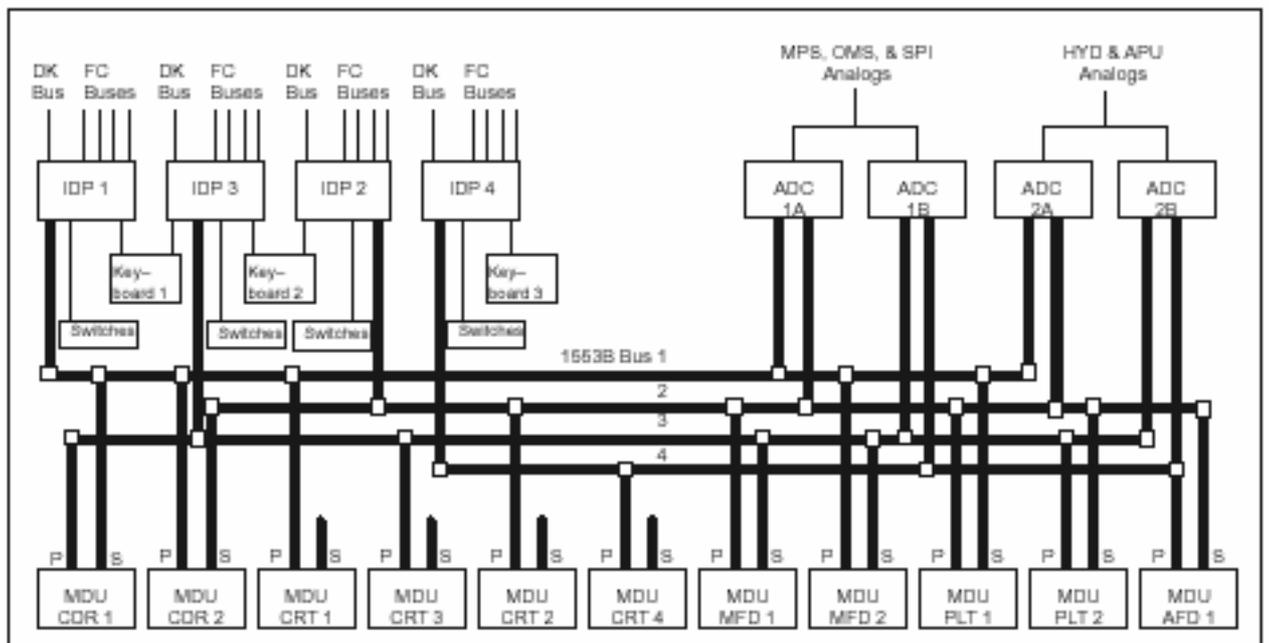


Figure 8 - 2 MEDS Architecture

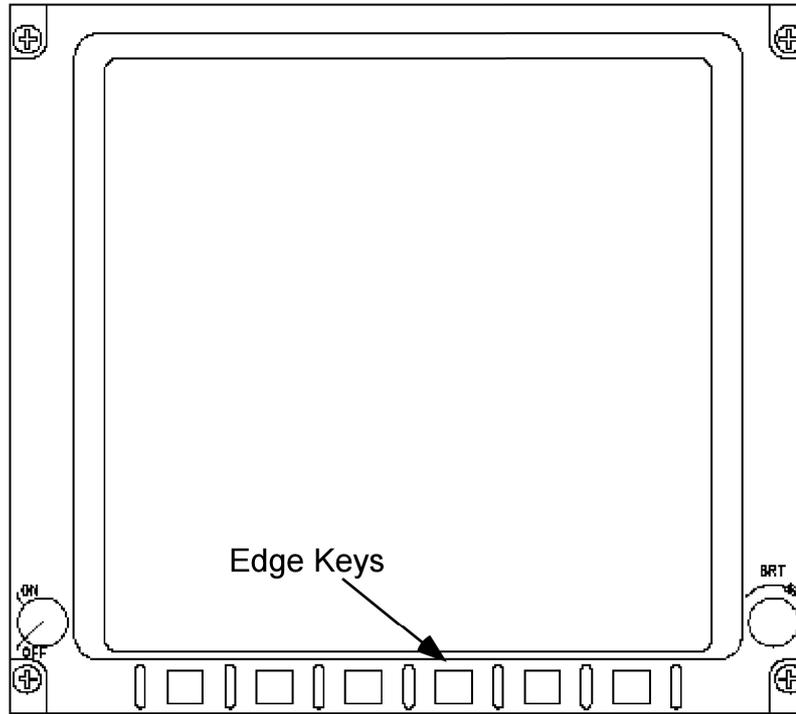


Figure 8 - 3 Multifunction Display Unit

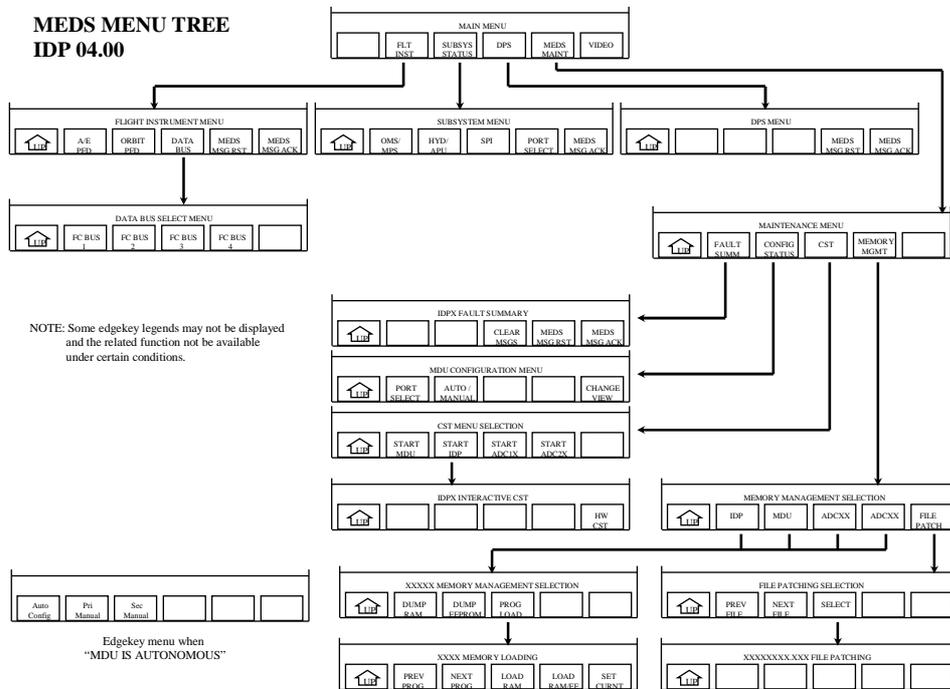


Figure 8 - 4 MEDS Menu Tree

8.1.3 The Display Suite

The suite of GNC related MEDS displays is a composite ascent/entry display consisting of ADI, HSI, AMI, AVVI, a vehicle acceleration meter (G-meter), an altitude acceleration meter, an orbit display consisting of the ADI only, and a SPI. The RCS activity lights were not included in the MEDS upgrade and remain a hardware based system.

Figure 8 - 5 illustrates the CDR 1 and 2 MDU's. PLT 1 and 2 are mirror images of CDR 1 and 2, with the exception there are no RCS activity lights. A comprehensive representation of the forward and aft flight deck stations is provided in Space Shuttle Systems Handbook drawing 20.1.

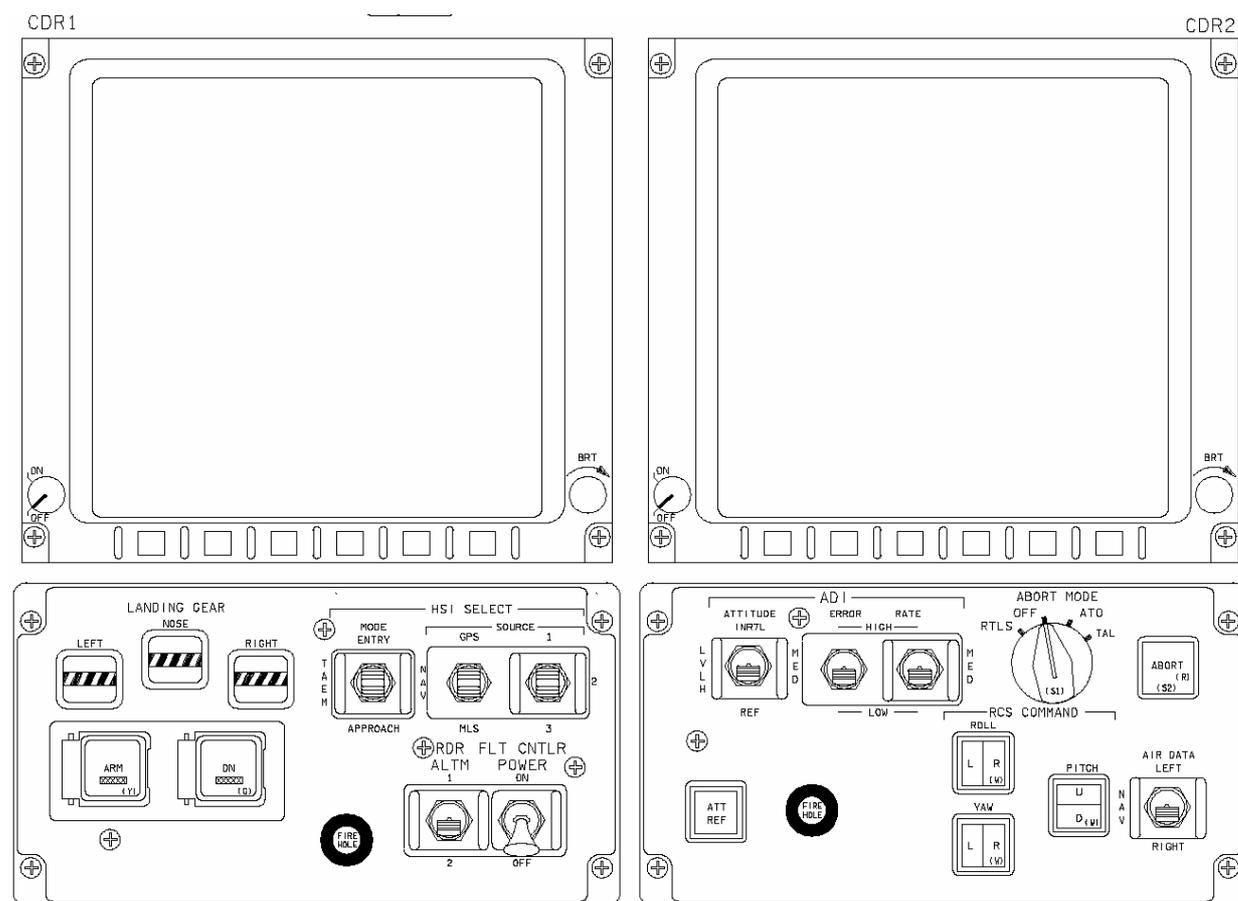


Figure 8 - 5 MDU's CDR 1 and 2

8.1.4 First Order Hold

A number of dedicated display parameters are calculated at a slower rate than the rate required for display. First order hold (FOH) is a simple technique used in the dedicated display processor to minimize display jerkiness in these cases. Illustrated in Figure 8 - 6, the technique consists of iterating at the desired display update rate along the slope defined by the two most recent data points.

The three most recent parameter updates are X_{n-3} , X_{n-2} , and X_{n-1} ; and X is the update yet to occur. The output rate is six times as fast as the update rate. It is clear from this example that the displayed parameter is smoother with FOH than without it. Without FOH, the output would be a step function between updates. There are examples where FOH is not smoother, e.g., equivalent airspeed (EAS), where slope changes sign frequently, but the basic update rate is fast enough in most cases to prevent this from occurring.

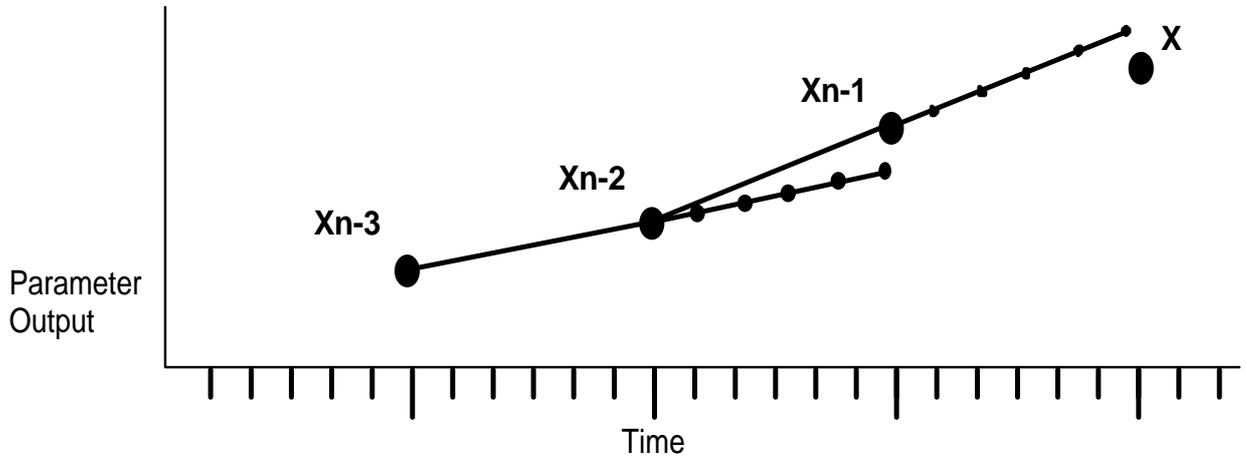


Figure 8 - 6 First Order Hold Technique

8.2 The Primary Flight Display

Beginning with OI-30, the ADI, HSI, AMI, and AVVI were consolidated into a single format, the primary Flight Display (PFD). Functions change with flight phase, and some functions are removed when no longer appropriate. In this section, the individual instruments will be described first, followed by an integrated description of the composite display as it appears in different flight phases. As an initial reference, a general layout of the display is provided in Figure 8 - 7, Figure 8 - 8, and Figure 8 - 9.

Common traits of each format are an ADI, an HSI, an AMI, an AVVI, a G-meter, an FCS configuration status, and major mode. Specialized readouts consist of an altitude acceleration meter, glide slope indicator, and trajectory information.

Note: The displays presented in this document are not to scale, and are intended only to give the reader a sense of the “look and feel” of the displays. Effort has been made to ensure the technical accuracy of the data in the primary flight display representations, but there may be inaccuracies, e.g. range to HAC center may not correspond to the realistic HAC center at the displayed range to the runway. For exact placement of display fields, the reader should consult the IDP requirements document listed at the end of this brief. For accurate data representation, the reader should consult an appropriate technical briefing or document.

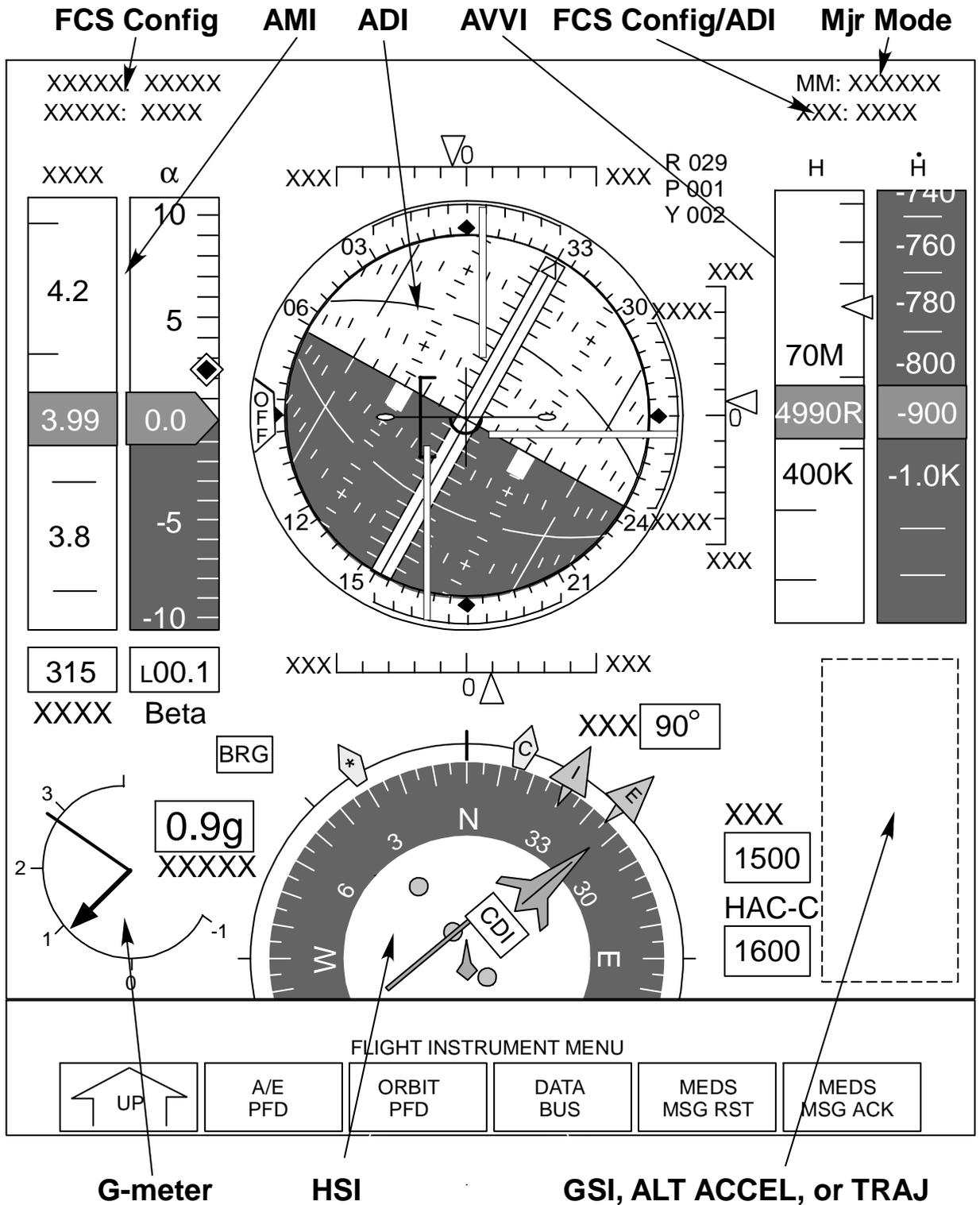


Figure 8 - 7 A/E PFD Generic Format

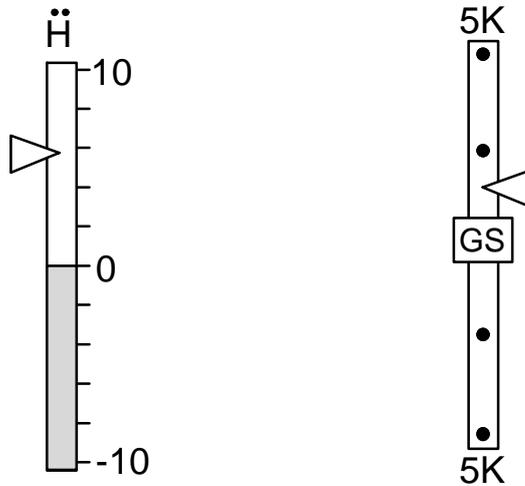


Figure 8 - 8 Generic GSI and ALT ACCEL Meter

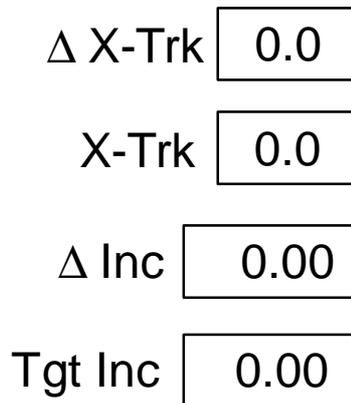


Figure 8 - 9 Trajectory Data

8.2.1 Major Mode & FCS Configuration

In all but OPS 2, the current major mode is identified in the upper right hand corner of the display. If an abort has been declared, an indicator will verify the abort mode selected (R for RTLS, T for TAL, AOA for AOA, ATO for ATO, and CA for contingency aborts).

The FCS configuration fields change with major mode, and consists of DAP/throttle mode on the left and ADI attitude/SB mode below the major mode on the right. During powered flight (MM 101-103 & MM 601), the left side fields show DAP mode (AUTO or CSS) and Throttle mode (AUTO or MAN). If CSS or MAN are selected, a yellow box is drawn around the indicators. The right side field shows the position of the ADI attitude switch (INRTL, LVLH, or REF) for the CDR if the display is on MDU's CDR 1 or 2, and for the PLT if on MDU's PLT 1 or 2. Post-MECO (MM 104-106), the DAP mode will indicate AUTO or INRTL, while the throttle field is blanked at MECO Confirm in MM 103. The right side ADI field is unchanged in MM 104-106, and continues to show the CDR or PLT ADI attitude selected.

In OPS 2, both left and right side fields are blanked.

In MM 301-303, the left side field will show DAP: AUTO or INRTL as in post-MECO OPS 1. The lower field on the left remains blank. The right side field below the major mode shows the applicable CDR or PLT ADI attitude selected (INRTL, LVLH, or REF).

In glided flight (MM 304, 305, 602, 603), the left side indicators change to DAP pitch mode (AUTO or CSS) in the upper field, and DAP R/Y mode (AUTO or CSS) in the lower field. The right side field below major mode shows the SB mode (AUTO or MAN). A yellow box is drawn around the field if MAN SB is selected at any time or if CSS is selected prior to M = 1. The availability of each indicator is provided in Table 8 - 1. To illustrate, Figure 8 - 10 provides major mode and FCS configuration examples.

Table 8 - 1 Major Mode & FCS Configuration Labels

| Indicator | | Major mode | | | | | | | | | | | | |
|---------------------------------------|---------------------------------|------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| | | 102 | 103 | 104 | 105 | 106 | 301 | 302 | 303 | 304 | 305 | 601 | 602 | 603 |
| DAP | ASC DAP AUTO/CSS | X | X | | | | | | | | | X | | |
| | TRANS DAP AUTO/INRTL | | | X | X | X | X | X | X | | | | | |
| | AEROJET DAP P AUTO/CSS | | | | | | | | | X | X | | X | X |
| | AEROJET DAP R/Y AUTO/CSS | | | | | | | | | X | X | | X | X |
| SBTC | ASC DAP Throttle AUTO/MAN | X | X | | | | | | | | | X | | |
| | AEROJET DAP SB AUTO/MAN | | | | | | | | | X | X | | X | X |
| ADI Attitude Switch Position | | X | X | X | X | X | X | X | X | | | X | | |
| Major Mode | | X | X | X | X | X | X | X | X | X | X | X | X | X |
| Abort Modes | | X | X | X | X | | | | | X | X | X | X | X |

| | |
|-------------|------------|
| DAP: CSS | MM: 601CA |
| Throt: MAN | ATT: LVLH |
| DAP: AUTO | MM: 104AOA |
| | ATT: LVLH |
| Pitch: AUTO | MM: 304 |
| R/Y: AUTO | SB: AUTO |
| DAP: AUTO | MM: 601R |
| Throt: AUTO | ATT: LVLH |

Figure 8 - 10 Major Mode & FCS Examples

8.2.2 Trajectory

Considered part of the HSI, trajectory information includes cross track deviation in nautical miles (Δ X-Trk), target cross track range in nautical miles (X-Trk), target inclination deviation in degrees (Δ Inc), target inclination in degrees (Tgt Inc), and Azimuth from landing site (Δ Az) in degrees.

X-Trk is the lateral off-set distance from targeted plane or course in nautical miles, and supplements the course deviation indicator (CDI) described later. The value is positive for positions north of the target insertion plane, and negative for positions south of the target plane. X-Trk is available for nominal uphill, TAL, and ATO powered flight (MM 102 and 103).

In MM103 TAL, X-Trk represents the lateral offset the current inertial velocity vector is making with the TAL runway. If the inertial velocity vector is currently pointing north of the runway, the value is positive. If pointing south of the runway-bearing pointer, the value is negative.

Δ Inc is the difference between the current & targeted inclination (deg), and supplements the "I" bearing pointer. Δ Inc, like X-Trk, is available for nominal uphill, TAL, ATO, and AOA powered flight (MM 102 and 103).

Δ X-Trk is X-Trk minus the radius of the targeted TAL cross-range circle in nautical miles, showing the distance the vehicle must steer to meet TAL cross-range targeting. It is only available for TAL abort, and shows zero if the current ground track/plane is inside the TAL cross-range target. It supplements the course arrow, "I" bearing pointer, & "R" bearing pointer.

For RTLS powered flight (MM 601), X-Trk, Δ Inc, and Δ X-Trk are blanked both pre and post-powered pitch around (PPA).

Δ Az (in degrees), and below and to the right, range to runway (in nautical miles, with SPEC 50 runway selection shown) is available for nominal entry, TAL, RTLS, and ECAL aborts. The Δ Az field changes to the HAC turn angle (HTA) after HAC intercept, and a range to HAC center (HAC-C) is also provided. The Δ Az box flashes when roll reversal limits are met or exceeded (10.5 or 17 degrees).

Tgt Inc indicates the value of the targeted inclination in degrees. It is only available for ATO (MM 103), and serves to indicate to the crew whether the IY plane was redefined at ATO declaration.

8.2.3 G-Meter

Illustrated in Figure 8 - 11, a vehicle acceleration meter (or G-meter) is provided in PASS MM 102, 103, 304, 305, 601, 602, & 603. The function of the meter is different in powered and glided flight, with the label changing to reflect what is being displayed. In powered flight, the meter reflects IMU-derived

vehicle acceleration and is labelled "Accel". During glided flight, the meter displays the RM selected AA N_z value and labelled " N_z ". Both are represented by a digital value and a green needle. A magenta target N_z line is provided during MM 602 showing the target N_z value in glided flight.

The BFS provides a G-meter during glided flight only (MM 304, 305, 602, & 603),.

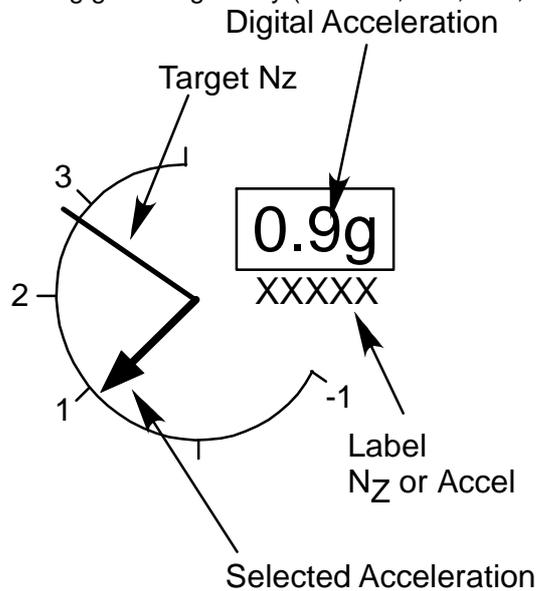


Figure 8 - 11 G-Meter

8.2.4 ADI

8.2.4.1 Display Features

Illustrated in Figure 8 - 12 Attitude Determination Indicator, the ADI provides attitude, attitude error, and rate information.

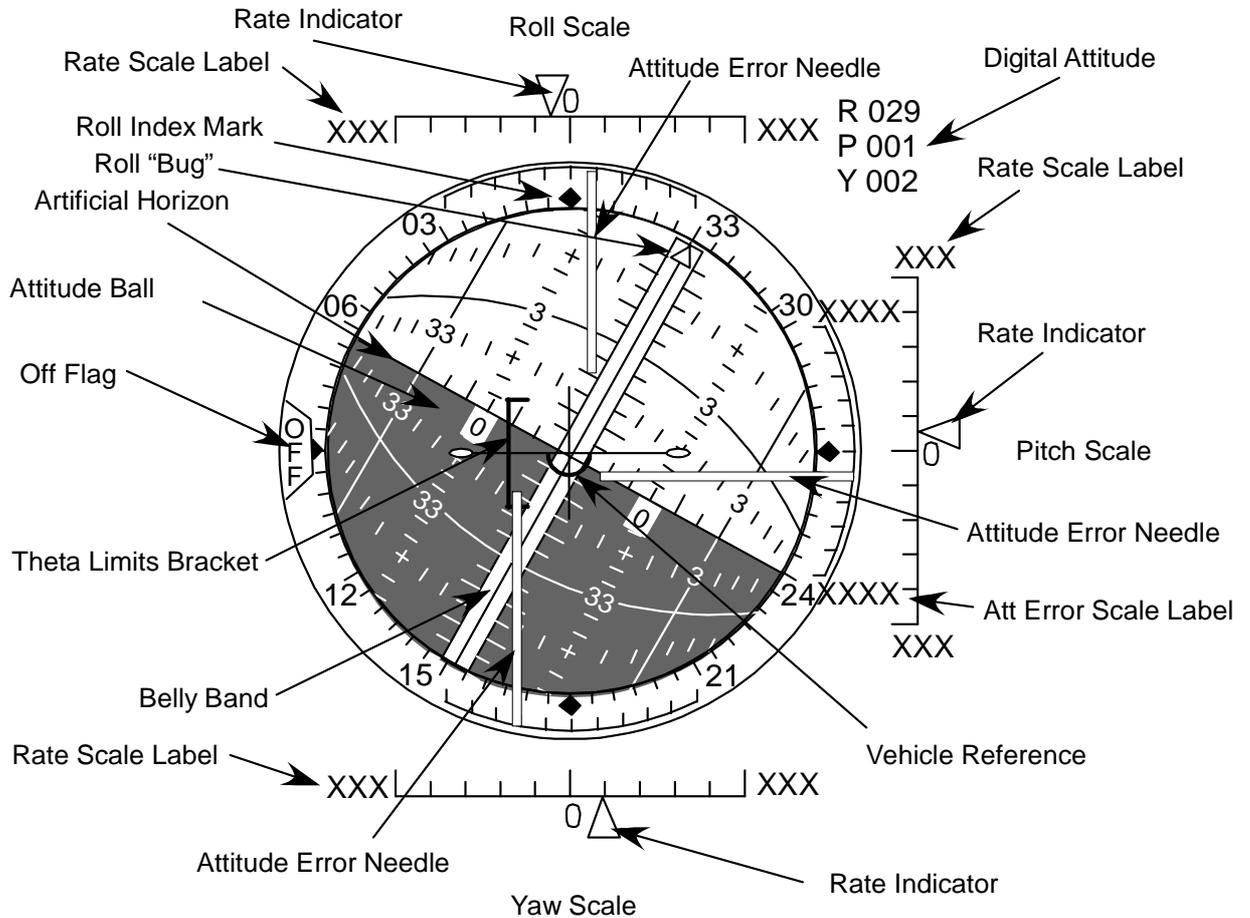


Figure 8 - 12 Attitude Determination Indicator

8.2.4.1.1 Attitude

The Orbiter attitude is displayed by means of a software-simulated enclosed ball and digital readouts. This ball is gimballed to provide three degrees of freedom, allowing it to be positioned in response to software-generated commands corresponding to the Orbiter roll, pitch, and yaw attitude. The numbers on the ball and around the case are angle magnitudes, with the trailing zero deleted for simplicity. Figure 8 - 13 shows the ADI ball as viewed from the top, front, and side.

The digital readout is expressed in Orbiter Roll, Pitch, and Yaw corresponding to the ADI Attitude switch setting (INRTL, LVLH, or REF) which is discussed later. Note that the RPY order is not the same as the Euler sequence used to generate attitude maneuvers, which is PYR. The order used on the displays was chosen to remain consistent with how attitude information is documented in the FDF documentation.

A vehicle symbol is provided fixed to the center of the ADI window as a reference.

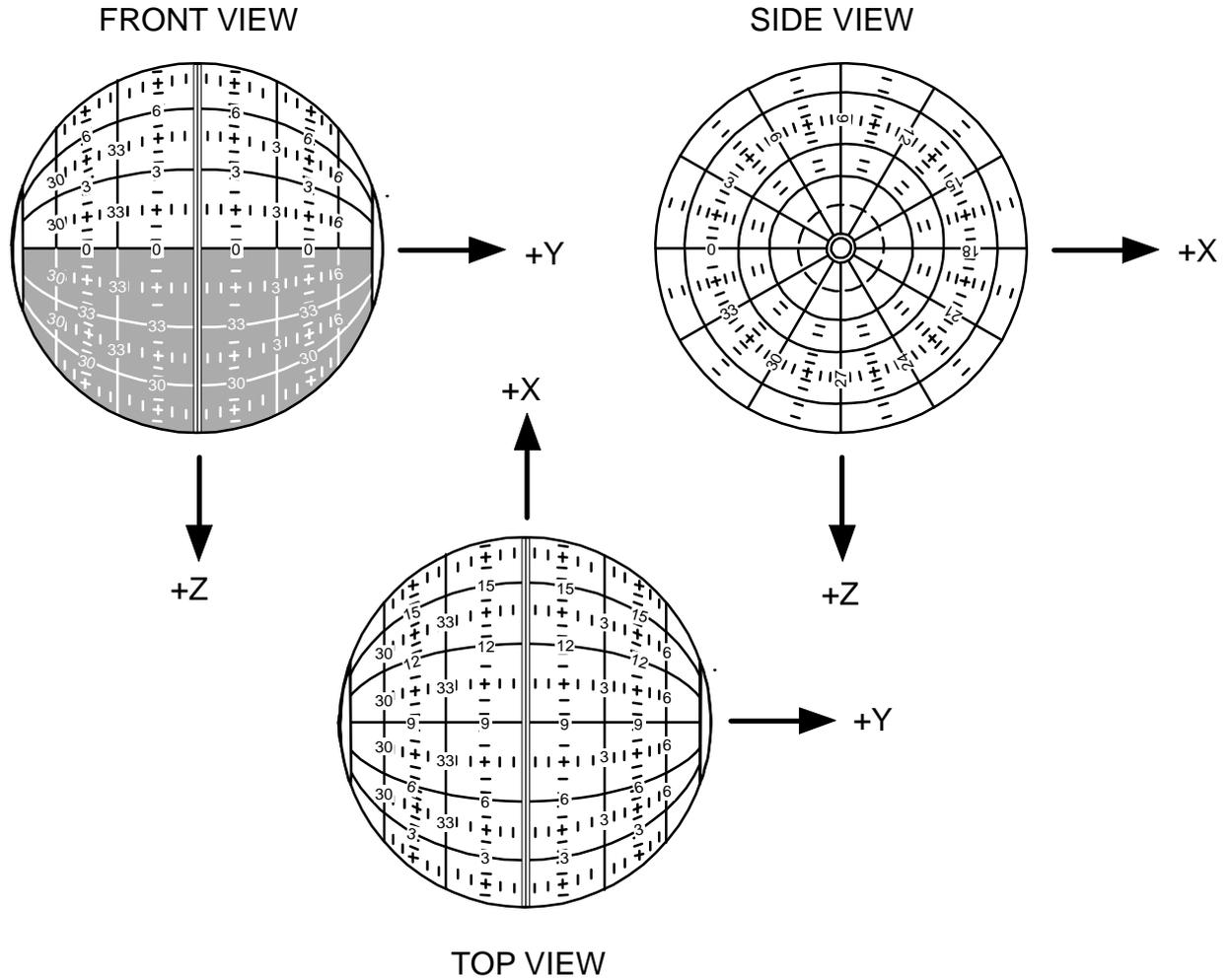


Figure 8 - 13 Attitude Ball Axis & Markings

Roll - The roll markings are not on the ball itself but are fixed on the display (MDU). The scale runs counterclockwise, 0° to 360°, with 90° intervals marked by black diamonds (roll index marks). The roll angle is read where the roll arrow (bug) intersects the roll scale. This bug is driven with respect to the MDU, and will move along the surrounding roll scale regardless of the orientation of the ball. The ADI ball responds to a positive Orbiter roll (roll right) by rotating counterclockwise; this is also true regardless of the initial orientation of the ball (Figure 8 - 14).

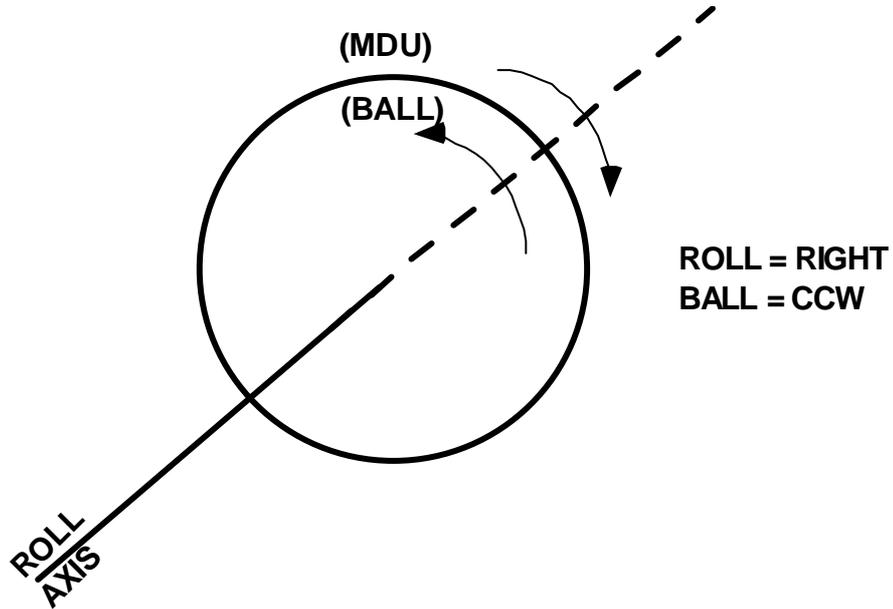


Figure 8 - 14 Roll

Pitch - The ADI is marked for pitch angle with great circles every 30° from 0° to 360°. Pitch attitude is read in the pitch/yaw index at the center of the display. ADI ball response to a positive Orbiter pitch (pitch up) is to rotate downward (top to bottom) regardless of the initial ball orientation (Figure 8 - 15).

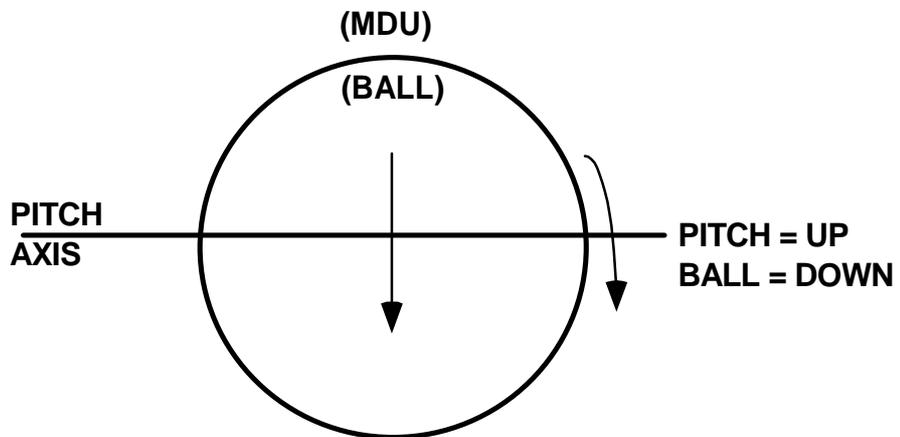


Figure 8 - 15 Pitch

Yaw - The yaw markings include the 0° yaw line (belly band) and minor circles at 30°, 60°, 300°, and 330°. The yaw scale ranges from 0° to 90° and 270° to 360° on both the front and the back of the ADI; hence the viewed yaw angle cannot exceed ±90°. This poses no constraint on ADI total attitude capability since a combination of pitch and roll can be produced to replace any yaw angle outside this range. For example, a roll/pitch/yaw of 0/0/180 can be, and on the ADI will be, read as 180/180/0. The corresponding ADI ball motion for Orbiter positive yaw rotation (yaw right) is a right-to-left rotation, as shown in Figure 8 - 16.

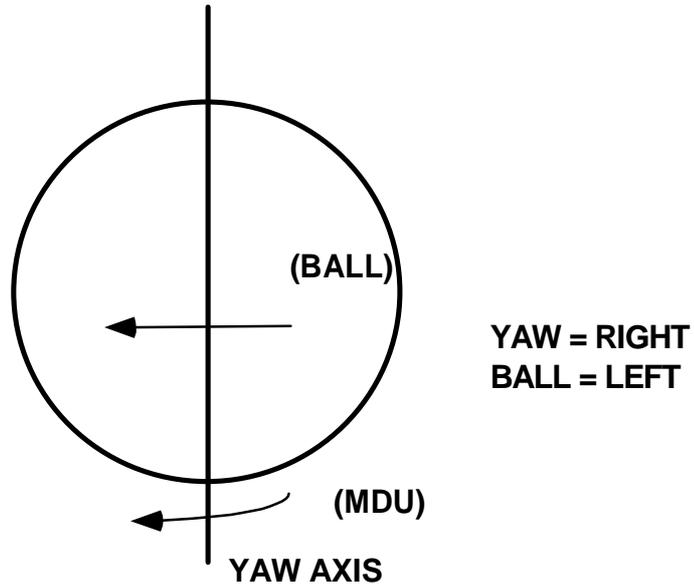


Figure 8 - 16 Yaw

The ball rotates in the opposite sense from the causative Orbiter rotation. This makes the ADI ball a fly-to indicator; that is, an observed attitude change may be corrected (nulled) by moving the RHC in the same sense as the ball. For example, an Orbiter yaw attitude change to the left would be observed as an ADI ball rotation to the right, and could be nulled by an RHC yaw right command. A simple way to remember how the ball moves in response to Orbiter rotations is to think of the zero pitch line (where black and white meet) as a horizon line. For example, assuming the horizon were in view and the crewmember were heads up looking at it, a + pitch Orbiter rotation would result in the horizon dropping down (top-to-bottom).

The ADI's are gimballed and pointed to provide attitude in the pitch/yaw/roll Euler sequence. This implementation results in a coalignment of the ADI pitch and roll axes whenever ADI yaw is at 90° or 270°. With the pitch and roll axes approximately coaligned, the ADI ball cannot respond to pitch changes without drastically changing roll. The attitude processing software protects against large pitch and roll excursions in the region of 270° or 90° ± 1.7° yaw by freezing roll and pitch. The ADI ball is a pitch, yaw, roll Euler-ordered device but, as will be seen later, the ADI's can also be driven in a yaw/pitch/roll sequence; however, in this case the yaw axis must be held at null (zero degrees).

8.2.4.1.2 Rates

The ADI has three pointers which provide a continuous indication of vehicle body rates in all supported major modes, except in MM 305 or 603 with the ADI RATE Switch is in the Medium (MED) position, when the ADI Rate pointers display the Time to Heading Alignment Cone (HAC) Intercept, Altitude Error, and "Crosstrack Error". Roll, pitch, and yaw rates are indicated by the top, right, and bottom pointers, respectively, when the ADI displays vehicle body rates. Time to HAC Intercept, Altitude Error, and "Crosstrack Error" are indicated by the top, right, and bottom pointers, respectively, when in MM 305/603 and the ADI RATE Switch is in the Medium (MED) position. Graduation marks on either side of the null (center) mark allow the pilot to determine the rate magnitude based on full scale range, which varies with the rate switch setting (see para. 8.2.8.1). Rate scale labels correspond to the setting of the ADI rate switch.

For positive Orbiter vehicle rates (roll = right, pitch = up, yaw = right), the corresponding ADI rate pointer deflections are roll = left, pitch = down, yaw = left. Therefore, the rate pointers are also fly-to indicators; i.e., the RHC must be moved in the same sense as the pointer in order to null the rate.

8.2.4.1.3 Errors

The ADI has three needles which provide a continuous indication of vehicle attitude errors. These needles extend in front of the ADI ball from the top (roll), right (pitch), and bottom (yaw). Each needle has a background scale with graduation marks to allow the crew to deduce the magnitude of the attitude error. The scaling and definition of these errors varies with the parameter, the mission phase, and the error switch setting (see para. 8.2.8.1). Attitude error scales correspond to the ADI error switch setting. During TAEM (MM 305 and 603) and N_z hold (MM 602), the Pitch Error scale is in g's, and indicated by a "g" next to the scale value.

During powered flight (either ascent or OMS/RCS burns), the errors are guidance thrust vector errors. This means that during engine burns, the errors reflect TVC trim errors, and during RCS burns the errors reflect the guidance VGO error. Note that if bad IMU velocity data is processed by the User Parameter Propogator (UPP), the error needles will be incorrect.

Orbiter roll right, pitch up, and yaw right errors will result in error needle deflections of roll = left, pitch = down, and yaw = left. Therefore, the attitude error needles are also fly-to indicators; i.e., the RHC must be moved in the same sense as the needle deflection in order to null the error.

8.2.4.1.4 OFF Flag & Invalid Data

The OFF flag is a remnant from when the ADI was a mechanical device powered by the DDU. Today, it is used to indicate whether valid GPC data is driving the ADI software. Presence of the OFF flag can still be used to determine if a flight critical bus problem exists at the GPC driving the IDP hosting the ADI, and is used in DPS malfunction procedures.

For invalid or missing data in any or all axes during ascent or entry phase, the ADI will continue to be displayed but the OFF flag will be present.

For on-orbit, the same is true for invalid data, but the ADI's will freeze in all axes if IMU data are missing (CF or DG bad).

8.2.4.1.5 Theta Limits Bracket

A green limit bracket was added with OI-30 for MM 305 and 603. It is available while $M < 2$ if either 1) air data is inhibited to G&C, 2) no ADTA/probes are available, or 3) an air data dilemma exists (AD DG flag set to 0). The upper and lower bracket represent the theta high and low limits, respectively, as determined by guidance. The limits are the same as the theta limit indicators available on the VERT SIT display (Figure 8 - 17).

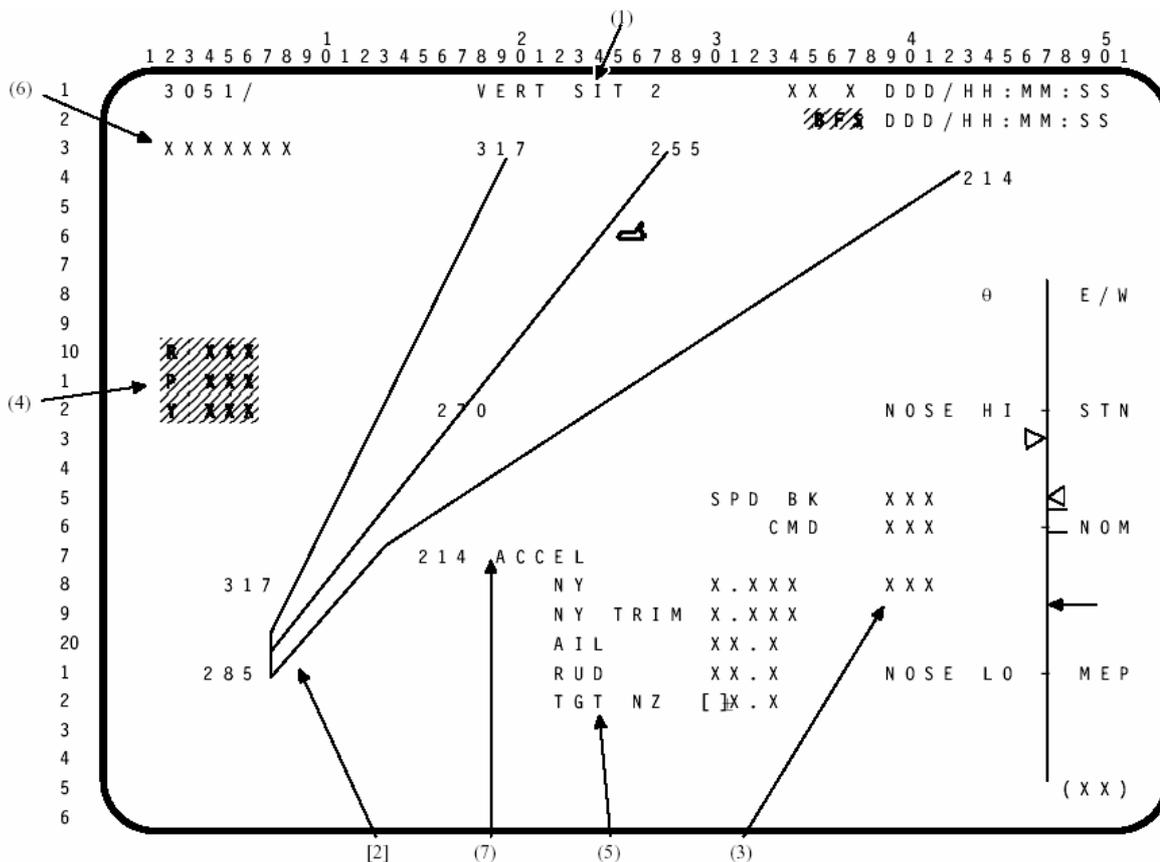


Figure 8 - 17 VERT SIT

8.2.4.2 ADI Processing

Like all dedicated displays, the ADI display is driven by GPC software. This section describes the flexibility of the ADI, explaining what each parameter on the display means for each mission phase. Also discussed will be what the ADI display does during mission aborts, during BFS control, and for invalid data conditions.

The following discussion will be largely based on the data presented in Table 8 - 2 and Table 8 - 3. These tables provide a summary of ADI display parameters versus mission phase.

8.2.4.2.1 Nominal ADI Processing

For all of OPS 1 (ascent phase and transition-to-orbit) and OPS 3 (transition and entry phase), only the two forward ADI's are driven. During OPS 2 and OPS 8, all three ADI's are driven. Software ADI processing is done in two software modules, the attitude processor (ATT PROC) and the ADI processor. The attitude processor develops the ADI ball driver functions as well as certain error signals. The ADI processor develops rate driver functions and the remainder of the error signals.

Attitude - During OPS 1 (after IMU release), OPS 2, and OPS 3 transition (MM's 301-303), the ADI's will display attitude in the reference system selected by the ADI Attitude switch (INRTL, LVLH, or REF). For the INRTL and REF positions of the Attitude switch, the frames are fixed in space (INRTL), the attitude processor maintains these in the form of quaternions¹ relating body axes to the respective coordinate systems. For LVLH, the quaternion relates body axes to LVLH (or

¹ Quaternions are another method of performing rotations from one reference frame to another in space.

LVIY during ascent) and thus requires updating based on the Orbiter position and velocity vectors.

During MM's 304 (entry) and 305 terminal area energy management ((TAEM) and approach and landing (A/L)), the attitude processor ignores the ADI Attitude switches and develops both forward displays as topodetic pitch and roll, with yaw driven to zero. Therefore, the ADI is a "wings-level" display.

The software protects against the large attitude jumps at the $\pm 90^\circ$ pitch or yaw singularity points associated with yaw/ pitch/roll or pitch/yaw/roll Euler sequences, respectively. This protection consists of freezing the ADI in the two vulnerable axes, and while it can always be seen, this freezing will not normally be visible on the ADI display long enough to be important. However, at liftoff in the LVLH position (LVIY frame), the ADI is at the $+90^\circ$ pitch singularity for an extended period, so a larger deadband ($\pm 1.7^\circ$) is employed to cause the ADI to remain frozen in roll until after pitch maneuver initiate (yaw is pinned to zero anyway). The frozen roll value is I-loaded and is the LVIY roll angle which is anticipated when the Orbiter breaks out of the 1.7° deadband in pitch. In a typical flight, however, the ADI's are set to REF prelaunch then reposition to LVLH after roll program complete (about 30 seconds MET).

Table 8 - 2 ADI Display Sequence

| Major Mode | 101-103 | | 104-106, 301-303 | 201& 202 | 304 | 305 | | |
|------------|--|--------------------------|-----------------------|------------------------|-------|---|-------------------------------------|---------------------------|
| Phase | Ascent | | Transition | Orbit | Entry | TAEM | A/L | Touchdown |
| Attitude | INRTL | Inertial Reference Frame | | | Roll | Topodetic Roll | | |
| | LVLH | LVIY | LVLH | | Pitch | Topodetic Pitch | | |
| | REF | Stored Reference Frame | | | Yaw | Fixed at Zero | | |
| Errors | Launch Guidance Command Errors | | Total Attitude Errors | | Roll | Roll Total Attitude Error ($\Phi - \Phi_{CMD}$) | | |
| | | | | | Pitch | Angle of Attack (α) Error | Normal Acceleration Error (N_z) | Slapdown Pitch Rate Error |
| | | | | | Yaw | Estimated Sideslip (β) | | |
| Rates | Selecte RGA Roll, Pitch, & Yaw Rates (Body Axis) | | | IMU Derived Body Rates | Roll | Stability Roll Axis Rate | | |
| | | | | | Pitch | Selected RGA Pitch Rate | | |
| | | | | | Yaw | Stability Yaw Axis Rate | | |

Table 8 - 3 BFS ADI Sequencing

| Major Mode | 101-103 | | 104-106, 301-303 | 201& 202 | 304 | 305 | | |
|------------|--|--------------------------|-----------------------|------------------------|-------|---|-------------------------------------|---------------------------|
| Phase | Ascent | | Transition | Orbit | Entry | TAEM | A/L | Touchdown |
| Attitude | INRTL | Inertial Reference Frame | | | Roll | Topodetic Roll | | |
| | LVLH | LVIY | LVLH | | Pitch | Topodetic Pitch | | |
| | REF | Stored Reference Frame | | | Yaw | Fixed at Zero | | |
| Errors | Launch Guidance Command Errors | | Total Attitude Errors | | Roll | Roll Total Attitude Error ($\Phi - \Phi_{CMD}$) | | |
| | | | | | Pitch | Angle of Attack (α) Error | Normal Acceleration Error (N_z) | Slapdown Pitch Rate Error |
| | | | | | Yaw | Estimated Sideslip (β) | | |
| Rates | Selecte RGA Roll, Pitch, & Yaw Rates (Body Axis) | | | IMU Derived Body Rates | Roll | Stability Roll Axis Rate | | |
| | | | | | Pitch | Selected RGA Pitch Rate | | |
| | | | | | Yaw | Stability Yaw Axis Rate | | |

Illustrated in Figure 8 - 18, the ADI attitude knowledge originates in the inertial measurement unit (IMU). The IMU attitude redundancy management (RM) selects a single IMU as the attitude source, and ATT PROC uses the gimbal angles from this IMU to calculate the current body attitude in the form of Q_{B_I} , the quaternion relating body axes to the INRTL coordinate system. Rate gyro assemblies (RGA) rates are integrated between IMU update cycles in order to smooth the ATT PROC output during the ascent and entry phases, since the IMU RM attitude update rate is only 1.04 Hz. (In fact, if no IMU is considered valid, ATT PROC will continue to provide total body attitude based entirely on RGA rates.) During on-orbit processing, the IMU RM output is rapid enough (6.25 Hz) that no rate-integrated supplement is required. FOH in the ADI processor increases the ADI attitude update rate to 12.5 Hz.

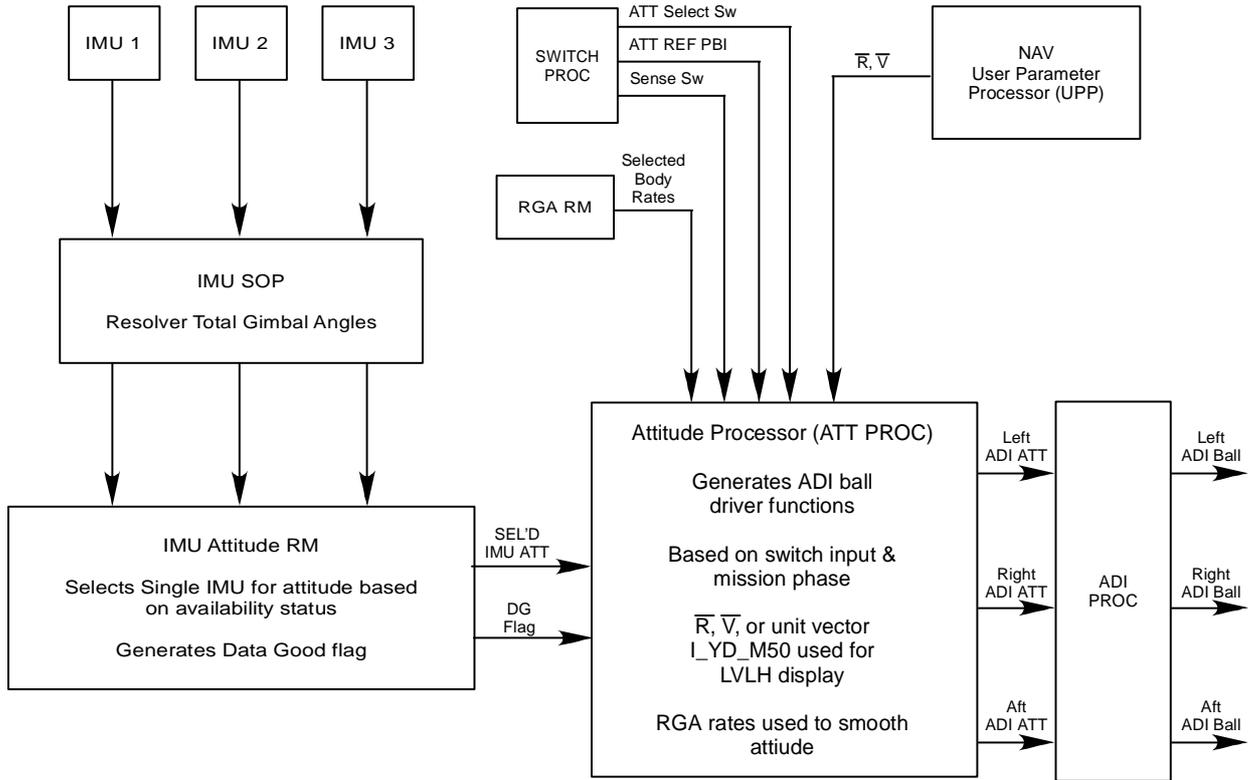


Figure 8 - 18 ADI Attitude Data Flow

Rates - ADI rates, like the attitudes, have the same meaning throughout ascent and transition-to-orbit; they are the RM-selected roll, pitch, and yaw body rates from the RGA's. During on-orbit, the rate pointers are driven using the IMU-derived body rates from the on-orbit DAP. For entry, TAEM, and A/L the ADI pitch, roll, and yaw rate scales/pointers have different meanings depending on the setting of the ADI RATE Switch and conditions in the onboard flight software (See section 8.2.8.1). During entry and post touchdown, the ADI pitch rate scale displays the selected RGA pitch rate (body axis), when the ADI RATE Switch is in either the High or the Low Rate position. In TAEM and A/L while the ADI RATE Switch is in the Medium (MED) position, the pitch rate scale (right scale) displays the guidance calculated altitude error with respect to the spiral HAC, or the needle is blanked depending on which conditions have been met in the onboard flight software. The roll rate scale (top scale) and yaw rate scale (bottom scale) display the stability roll axis rate and yaw axis rate, respectively, in entry, TAEM, and A/L, when the ADI RATE Switch is in either the HIGH or the Low position. (Roll and yaw stability axes are not co-aligned with roll and yaw body axes in aerodynamic flight; pitch stability and body axes are co-aligned. The stability axis rates are desirable for display because control is achieved about stability axes.). While the ADI RATE Switch is in the MED position, the roll rate scale displays the stability roll axis rate in entry; displays the HAC Initiation predictor or blanks the needle in TAEM; blanks the needle in A/L; and displays the stability roll axis rate after touchdown. With the ADI RATE Switch in the MED position, the yaw rate scale displays the stability yaw axis rate in entry; displays the guidance computed heading error, the guidance computed radial error, or the guidance computed y-position error in TAEM; displays the guidance computed y-position error in runway coordinates or blanks the needle in A/L; and displays the stability yaw axis rate after touchdown. Figure 8-31 shows a simplified view of the ADI rate processing.

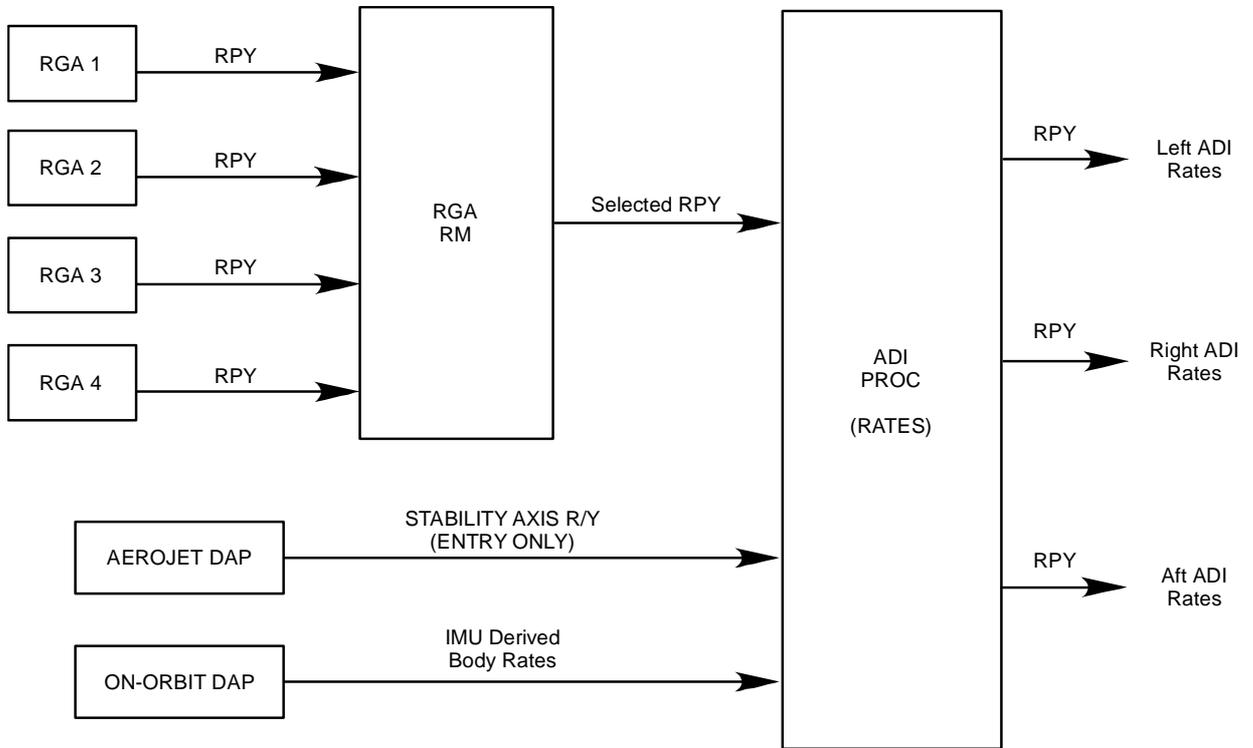


Figure 8 - 19 ADI Rate Data Flow

Errors - Throughout the ascent, on-orbit, and transition phases, the ADI error needles represent total attitude in body axes. Although these errors are computed in different software areas using different schemes, their meaning is basically the same up until MM 304: actual body attitude differenced with desired or commanded body attitude. During the entry phase (MM's 304, 305), the error needles assume meanings appropriate to the flight regime. The following discussion summarizes the error needle sequencing.

During first stage ascent (MM 102), the ADI errors are driven by the difference between the actual body attitude (M50 to body quaternion from the attitude processor) and the desired body attitude (guidance control (GC)) steer quaternion, based on the boost polynomial. In auto control, the ADI errors are the same errors as sent to the thrust vector control (TVC) DAP. In manual, the ADI errors include load relief lateral acceleration N_y and normal acceleration N_z feedback.

In second stage ascent (MM 103), the ADI errors are again derived from the actual and desired body attitude difference, but now the desired body attitude is based on desired thrust turning rate rather than the boost polynomial. In auto control, these errors are the same ones sent to the TVC DAP; in manual, they are not the same because, as in first stage, the DAP errors are shaped by RHC and trim commands. The manual case ADI errors are considered adequate for crew control during second stage ascent because there is no N_y/N_z load relief to add confusion.

During the transition phases (MM 104-106 and MM 301-303), the ADI error needles are still defined as total body attitude errors, only they are calculated in the attitude processor rather than GC steer. They are derived by forming an attitude error quaternion between the actual (IMU-derived) and commanded quaternions (from the TRANS DAP), using a small angle approximation technique to derive roll, pitch, and yaw errors from the error quaternion. The error needles will be dynamic throughout the transition phases because there is always a commanded body attitude. In general, the commanded attitude during transition is defined as the existing vehicle attitude at certain events; subsequent errors are deviations from that initial, fixed attitude. Exceptions are during delta-V burns (orbital maneuvering

system (OMS, RCS) and auto rotation (RCS) when the commanded attitude will be updated throughout the maneuver. The timeline in Figure 8 - 20 summarizes the events (Δ) which establish new commanded attitudes; $\Delta - \Delta$ means the commanded attitude is updated between events.

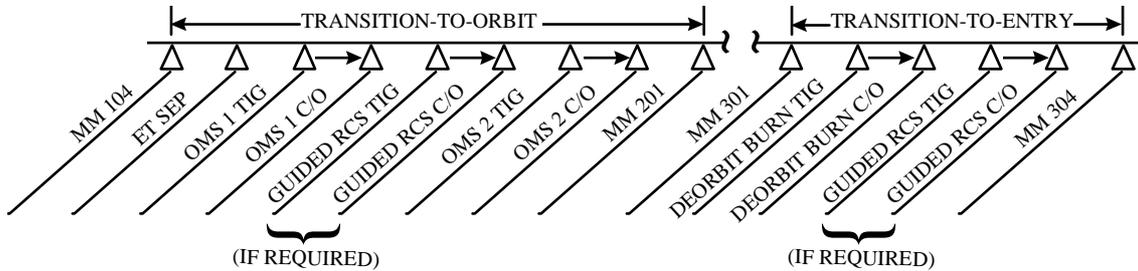


Figure 8 - 20 Events Establishing a New Command Attitude

During on-orbit activities (OPS 2), the ADI errors are calculated in either the ATT PROC or the on-orbit DAP, depending on the selection made by the crew using the Universal Pointing cathode ray tube (CRT) display (Figure 8 - 21, ignore numbered arrows). Item 23 (ATT ERR TOT) selects the ATT PROC, which uses the technique described in the transition phases to derive attitude errors from the actual and commanded body quaternions. This is the normal on-orbit configuration for attitude errors, providing new commanded attitudes at certain events (e.g., burn ignition) by setting commanded = actual attitude at those times.

Item 24 (ATT ERR DAP) selects a module of the on-orbit DAP (RCS ERRORS), which computes attitude errors for display during automatic rotation maneuvers. The three types of auto rotations are crew-selectable on the same display (TRACKING, item 19; ROTATION, item 20; and MANEUVER, item 18). Typically, the ADI errors would start at zero as the initial desired attitude is set equal to actual attitude, then would reflect any attitude differences with the incremental desired attitude as the maneuver progressed.

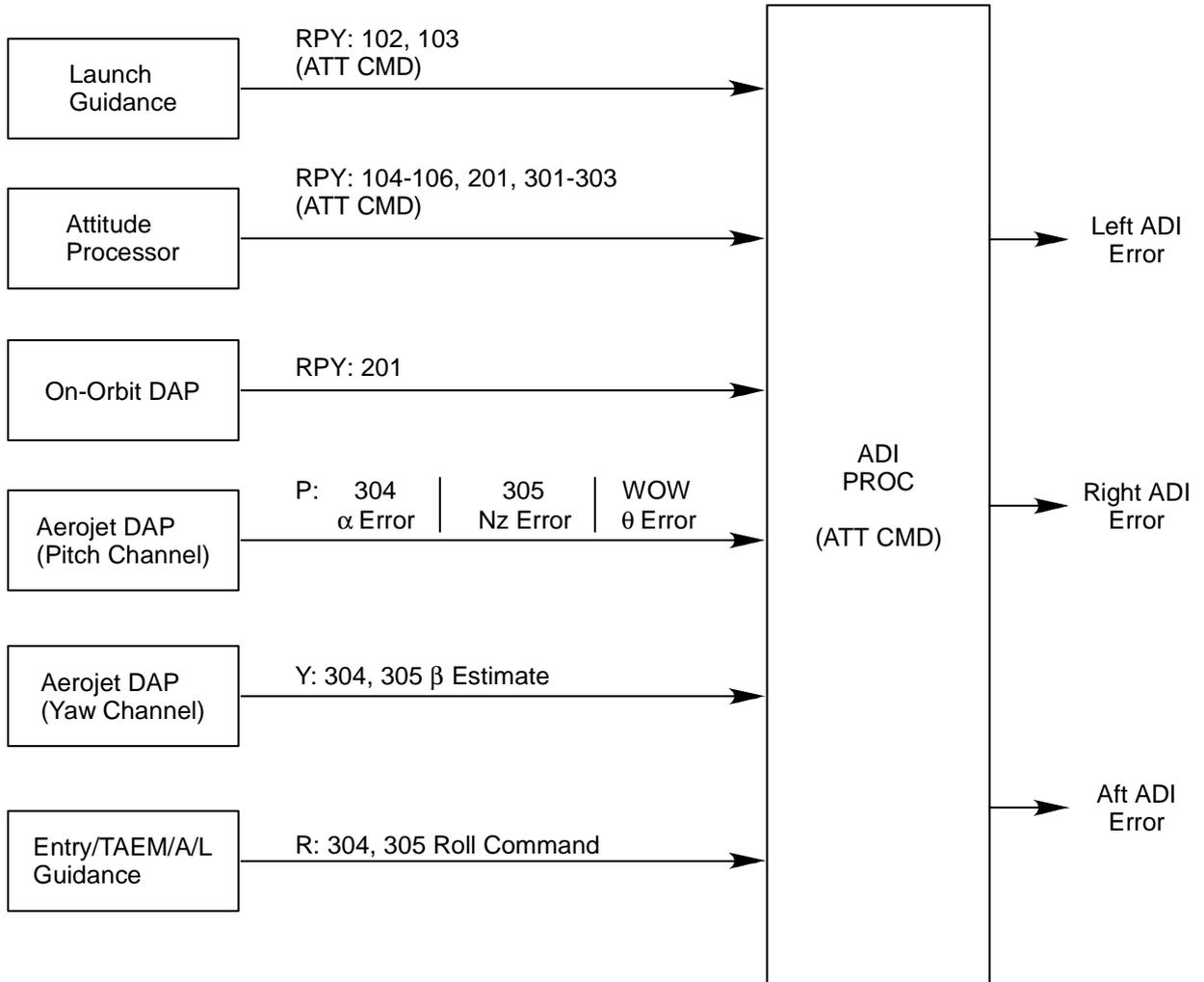


Figure 8 - 22 ADI Error Data Flow

8.2.4.2.2 Mission Aborts

The ADI during an abort-to-orbit (ATO) or an abort-once-around (AOA) is the same as described for a normal mission, per major mode.

For RTLS aborts, the ADI attitudes will be in accordance with the ADI Attitude switches (INRTL, LVLH, or REF) until MM 602, and at which time the reference will be topodetic pitch/roll, yaw pinned at 0. For RTLS errors, the roll needle will always be the total body roll attitude error (like the nominal case: body roll attitude differenced with the guidance limited roll command). In RTLS, the ADI error needles will be blanked until powered explicit (PEG) has converged. Pitch error will be the same in MM 602 as in MM 304 (alpha error; except in N_z hold displays $N_{z,ERR}$, but without the "g" label) and the same in MM 603 as in MM 305 before and after touchdown normal acceleration error ($N_{z,ERR}$ before; slapdown pitch rate error after). The yaw error is the same throughout MM's 602, 603 as in the nominal case (β_{EST} , from ATT PROC for $q\text{-bar} < 20$, from aerojet DAP after $q\text{-bar} 20$).

8.2.4.2.3 BFS ADI Processing

During BFS operations, only the forward ADI's are driven. Only the left-side ADI Attitude switch and ATT REF pushbutton is processed. Table 8 - 4 summarizes the following:

Attitude - For ascent phase (MM's 102, 103), RTLS MM 601, and transition phases (MM's 104-106; 301-303), the ADI's are driven in all three axes, and attitude is determined by the left ATTITUDE switch and ATT REF pushbutton the same as PASS. During MM's 304-305 and 602-603, the attitude becomes LVLH in pitch/roll, with yaw zero as in the primary system.

The attitude OFF flag is not driven in BFS. The source of attitude knowledge is one of the IMU's, selected by an algorithm based on IMU-to-IMU misalignment. As in the PASS, the selected RGA rates are used to propagate attitude between IMU samples. Also, like PASS, the ADI will provide attitude at a 12.5 Hz update rate.

Rates - The rate needles in BFS will be the RM-selected RGA rates for MM's 101-106, 601, and 301-303. During MM's 304 and 602, roll and yaw rates become stability axis rates, while pitch rate remains selected pitch rate (body axis). The stability axis rates are computed in the descent DAP (roll/yaw channel) using body roll and yaw rates, ground relative velocity, angle of attack, and roll and pitch attitudes. During MM's 305 and 603, the ADI Rate scales display the energy management and HAC initiation cues similar to the PASS flight software. The roll rate indicator displays the Time to HAC tangency; the pitch rate indicator displays the altitude error with respect to the desired altitude; and the yaw rate indicator displays the heading error with respect to HAC tangency, the radial error with respect to HAC tracking, or the lateral deviation with respect to the runway centerline, depending on which guidance phase is active. Reference section 8.2.8.1. for information relating to MM 305 and 603 ADI Rate indicator operations. Note that the ADI Rate Switch setting has no effect on the rate data that is displayed on the rate scales in BFS. All ADI displayed rate data, unless otherwise specified, is displayed at a default full rate scale deflection value of 5 deg/sec, regardless of the setting of the ADI Rate Switch.

Note that BFS flight software does not support the crew HAC initiation and energy management cues provided on the PASS version of the Horizontal Situation Display (SPEC 50).

Errors - The error needles for first and second stage ascent (MM's 102, 103) and RTLS (MM 601) are the same as in the PASS; i.e., total attitude errors from GC steer. In all of the transition phases (MM's 104-106 and 301-303), the error needles represent total attitude errors in body axes, they are derived in the transition DAP in the same way as PASS. In MM's 304-305 and MM's 602-603, the error needles are defined as they were in PASS, except that they are all blanked at 2000 ft altitude when BFS guidance terminates.

Table 8 - 4 BFS Rate & Error Scaling

| Item | | 101-106, 301-303 | 304 | 305 |
|-------|-------|---------------------|------------|--|
| Error | Roll | 5 deg | 5 deg | 5 deg |
| | Pitch | 5 deg | 5 deg | 5 deg/1.25 G's |
| | Yaw | 5 deg | 5 deg | 5 deg |
| Rate | Roll | 5 deg/sec | 5 deg/sec | See Table 8 - 11 ADI Rate & Error Switch Definitions |
| | Pitch | 5 deg/ sec | 5 deg/ sec | |
| | Yaw | 5 deg/ sec | 5 deg/ sec | |

8.2.5 HSI

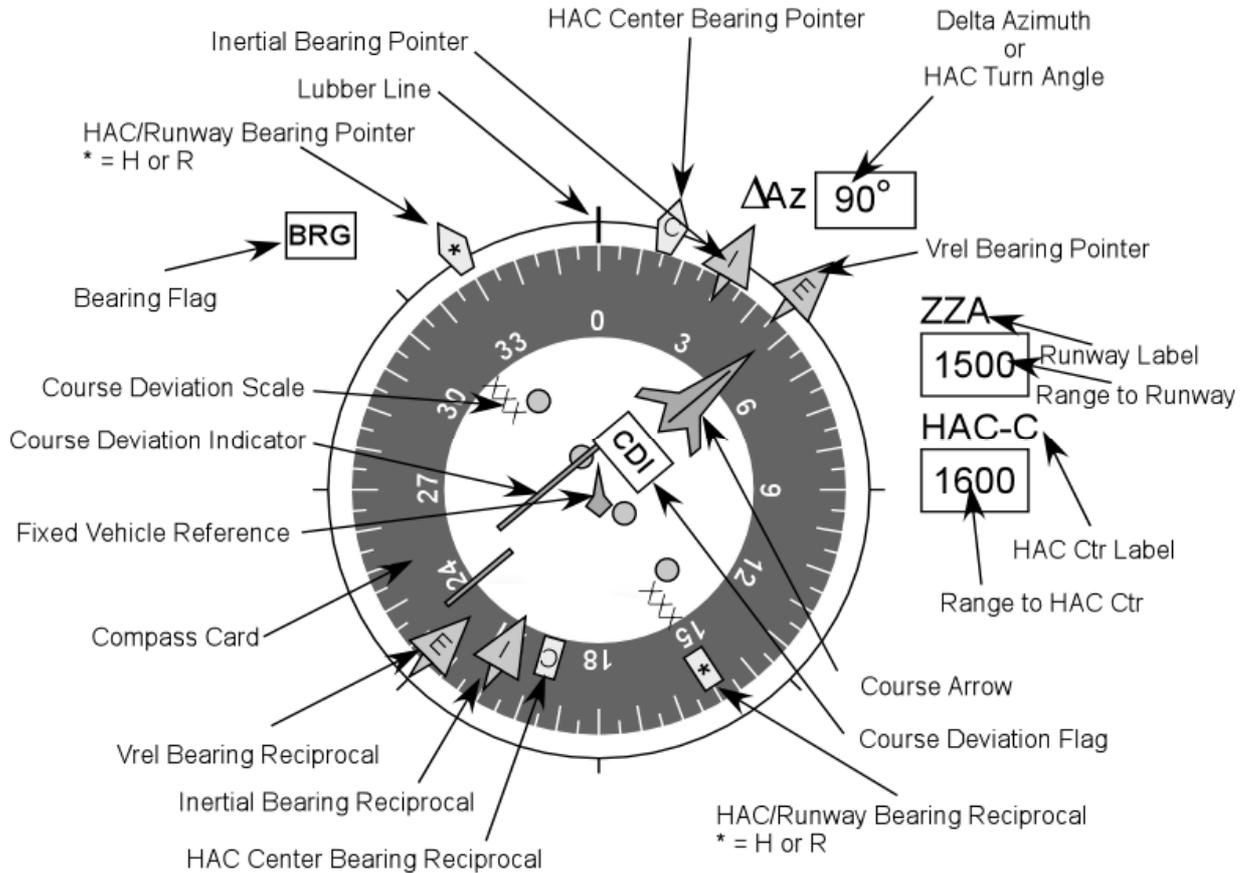


Figure 8 - 23 Horizontal Situation Indicator

8.2.5.1 Display Features

8.2.5.1.1 Lubber Line

The lubber line represents the nose of the orbiter and is consistent with the fixed vehicle reference in the center of the HSI.

8.2.5.1.2 Compass Card

The compass card replaces N, E, S, and W with 0, 9, 18, and 27 in MM 102/103, and references the target insertion plane course as the baseline for 0 on the display. For MM 304/305 and MM 601-603, the card reflects magnetic heading indicated by N, E, S, and W being present on the compass card. In powered flight, the compass card is reverse drawn while inverted, thus will always reflect the correct heading information, regardless of whether the vehicle is heads-up or heads-down

When displaying magnetic heading, the HSI shows the angle between magnetic north and vehicle direction measured clockwise from magnetic north (Figure 8 - 24). For vehicle direction, the software uses the relative velocity vector during entry (MM 304) and the body X-axis during TAEM and A/L (MM 305). Heading is displayed by the compass card and is read under the fixed lubber line at the top of the indicator dial. The compass card will be positioned at zero degrees when the heading input is zero; when the heading input is increased, the compass card rotates counterclockwise. The heading displayed in Figure 8 - 23 is 0°.

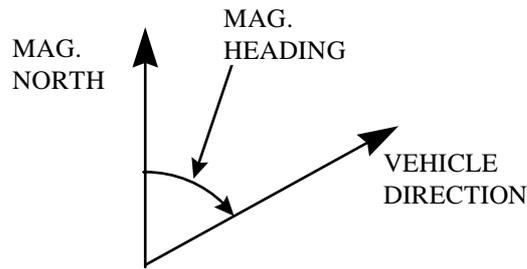
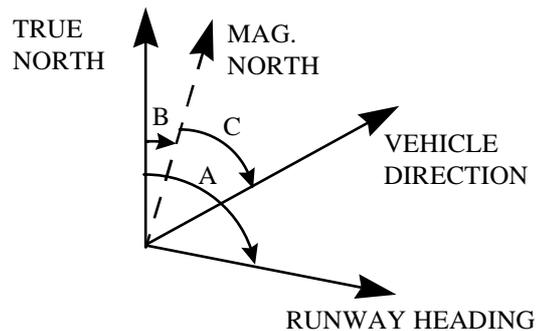


Figure 8 - 24 Heading Geometry

8.2.5.1.3 Course Arrow

The word "course" is defined as a direction to be followed, measured with respect to a geographic reference system. Course is indicated on the HSI by the course arrow which rotates around the inside edge of the compass card. The scale reading at the tip of the magenta pointer is the selected course. The course reading in Figure 8 - 23 is approximately 47°. In special cases, the pointer will turn from magenta to white.



HSI COURSE
REPRESENTS ANGLE BETWEEN
MAG. NORTH AND RUNWAY
DRIVEN BY ANGLE DIFFERENCES
A-B-C WRT CASE

Figure 8 - 25 HSI Course Geometry

The course arrow is used in glided flight (MM 304 & 305 or 602 & 603) to indicate the selected runway heading with respect to true north. As the compass card rotates, the course arrow will rotate with the card. This provides a continuous reference to the selected runway (Figure 8 - 25).

The course arrow is driven with respect to the MDU rather than the compass card. Therefore, a course input of zero results in the pointer being positioned at the lubber line regardless of compass card position. In order to position the course arrow correctly with respect to the compass card scale, the software must subtract the vehicle magnetic heading from the runway azimuth angle (corrected to magnetic north). As this subtraction is done continuously, the course pointer will appear to rotate with the compass card, remaining at the same scale position. An increase in the angle defining runway course would result in a clockwise rotation of the course arrow.

In powered flight, the arrow is pinned at 0 unless TAL or ATO abort is declared with variable IY active, then the course arrow represents the updated target inclination. If TAL abort is declared, the arrow is redefined to point at a tangent of a cross-range circle around the TAL site. If ATO and variable IY is active, the arrow is redefined to the new target inclination. If variable IY is not active, the arrow stays on

0. The arrow is blanked for RTLS aborts in MM 601, and turns from magenta to white when powered ECAL abort yaw steering is active.

8.2.5.1.4 Course Deviation

The Course Deviation Indicator (CDI) shows the lateral off-set from the targeted plane or course in nautical miles. It is auto re-scaled at 50, 10, & 1nm and labeled accordingly, with one mile scaling only occurring after 6+30 MET. The CDI bar & scale dots are available during nominal powered flight MM 102-104, and in TAEM and A/L MM 305 and 603. They are blanked in TAL powered flight (MM 103), but displayed in MM 104 with the CDI flag present (indicating it is static). The CDI is blanked during powered RTLS (MM 601). During entry (MM 304 and 602), the CDI dots are displayed but without scaling (indicated by the absence of labels). During TAEM (MM 305 and 603), the CDI represents the angular displacement from the runway centerline, and the scale and digital set to 10 (deg). At HSI approach and land, the scale and digital changes to 2.5 (deg).

In glided flight, the course deviation is an angular measurement of the vehicle from the extended runway centerline. On the HSI, course deviation is represented by the deflection of the deviation bar from the course pointer line. Full scale on the course deviation scale is 10° during TAEM and 2.5° during A/L. When the course deviation input is zero, the deviation bar will be aligned with the ends of the course pointer. The CDI is a fly-to indicator for flying the vehicle to the extended runway centerline, so with the pointer in the top half of the compass card, an increase in the course deviation to the left (right) will cause the bar to deflect to the right (left). Software processing also ensures that the CDI remains fly-to even when the Orbiter is heading away from the runway.

Figure 8 - 26 shows the course deviation geometry. If the Shuttle were at point A (B), it would have to fly right (left) to reach the extended runway centerline, and the corresponding course deviation bar deflection would be to the right (left). The reference point at the end of the runway is the microwave landing system (MLS) station. Note that the sense of the CDI deflection is a function of vehicle position rather than vehicle heading.

The course deviation flag is displayed when invalid navaid data is present, for instance due to commfault of TACAN, GPS, or MLS.

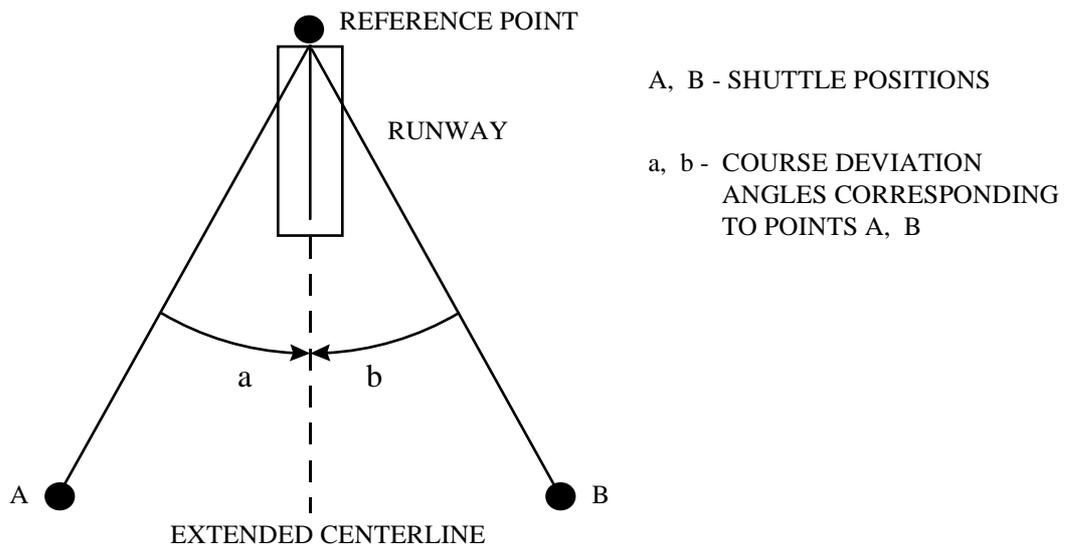


Figure 8 - 26 Course deviation geometry

8.2.5.1.5 Glide Slope

The glide slope deviation is the distance of the vehicle above or below the desired glide slope and is indicated by the deflection of the glide slope indicator (GSI) pointer located to the right of the HSI. The

pointer is a fly-to indicator, so an increase in glide slope deviation above (below) the desired slope results in deflection of the pointer downward (upward). In Figure 8 - 27, the pointer shows the vehicle to be below the desired glide slope by about 2000 ft.

The glidescope deviation is the difference between the Orbiter altitude and a reference altitude computed in the same fashion as the guidance reference altitude. However, also included in the glide slope reference altitude equation are factors for a heavy Orbiter and for high winds.

The GSI computation is available above 1500 ft in TAEM MM 305 and 603. The pointer is blanked prior to MM 305 or 603, and the GS flag is displayed below 1500 ft.

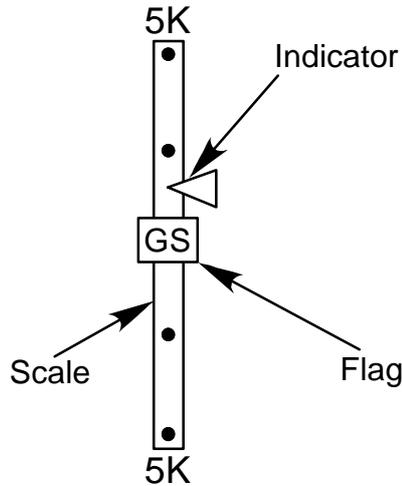


Figure 8 - 27 Glide Slope Indicator

8.2.5.1.6 Bearing Pointers

The V_{REL} Bearing Pointer, labelled "E", indicates the direction of the Earth-relative velocity vector relative to the vehicle nose (lubber line) & course (compass card/course arrow). It works in concert with the beta digital readout. Beta & the pointer are fly-to indicators, with beta showing an "L" or "R" prefix indicating the direction to null.

In MM 102-104 and 601, the Inertial Bearing Pointer, labelled "I", indicates the direction of the inertial velocity vector relative to nose & course. It is not displayed in glided or orbital flight.

The Runway/HAC Bearing Pointer indicates "R" for bearing to runway in powered flight and "H" for bearing to HAC in gliding flight. It appears at RTLS/TAL abort select & entry (MM 304/305 and 602/603).

The HAC Center Bearing Pointer, labelled "C", indicates bearing to the HAC center during TAEM (MM 305 and 603).

In glided flight, the bearing pointers display bearings relative to the compass card. These bearings are angles between the direction to true or magnetic north and to various reference points as viewed from the Shuttle. In order for the bearing pointers to be valid, the compass card must be positioned in accordance with vehicle heading.

When the bearing inputs are zero, the pointers will be at the top lubber line regardless of compass card position. Like the course arrow, the bearing pointer drive commands are developed by subtracting the vehicle heading from the calculated bearing values. This allows the pointers to be driven with respect to the HSI case (the MDU) but still be at the correct index point on the compass card scale. When the bearing inputs are increased, the pointers will rotate clockwise about the compass card. There will be no reversal of pointer movement when it passes through 360° in either direction.

The bearing reciprocals will always be 180° from (opposite) the pointers. Figure 8 - 28 provides an example of the bearing geometry.

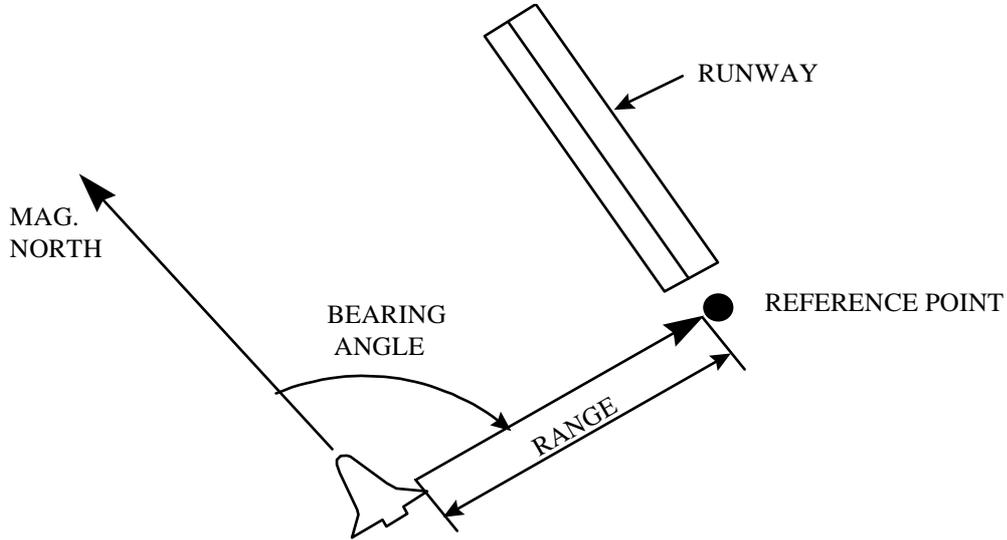


Figure 8 - 28 Bearing & Range Geometry

8.2.5.1.7 Bearing & Glide Slope Flags

The Bearing and Glide Slope flags indicate the absence of valid data, typically due to commfaults or loss of commlock.

As noted in section 8.2.5.1.5, the GSI computation is not performed during entry or below 1500 ft during A/L, and the GSI flag is displayed.

The bearing flag (BRG) is displayed when valid TACAN, GPS, or MLS data is not available due to commfault, lack of comm-lock, or invalid station.

8.2.5.1.8 Range

The range to the SPEC 50 primary runway is displayed in nautical miles. When the HSI Select Mode switch (discussed in section 8.2.8.5) is in ENTRY, the values will be displayed to the 1/10th of a nautical mile in TAEM or A/L (no decimal). Range to the landing site is displayed in powered flight at TAL MM 103, RTLS MM 601, and after ECAL/BDA abort selection. In glided flight, the range to the runway is displayed at MM 304 and 602, and both range to the runway and HAC center (HAC-C) are displayed at MM 305 and 603. A detailed explanation of range is discussed later in section 8.2.8.5.

8.2.5.1.9 Delta Azimuth and HAC Turn Angle

As ΔAZ this field displays the angle between the vehicle flight path and the HAC tangent in degrees (see discussion in section 8.2.2). The outline box will flash red when roll reversal limits are met or exceeded (10.5° or 17°). During TAEM post-HAC intercept, the field changes to HAC Turn Angle (HTA).

8.2.5.2 HSI Processing

8.2.5.2.1 Entry and GRTLS

The HSI's, like the other Shuttle dedicated displays, are software-generated. The HSI processing is complex enough that a separate software module (area NAV) is dedicated to it. Both HSI's are driven during entry (MM's 304, 305), RTLS aborts (MM's 602, 603), and BFS control (backup MM's 304, 305).

Figure 8 - 29 provides a general view of HSI data flow through the PASS. Area NAV uses Shuttle position and velocity from TACAN, GPS, air data, MLS, or the NAV user state plus runway position data from the site lookup table to calculate Shuttle position with respect to the runway. Then, based on the switch position (ENTRY, TAEM, or A/L), the HSI display parameters are calculated. Validity is based on TACAN, GPS, or MLS CF/lock-on indicators, air data good (DG) flag (ADTA RM), and TACAN site validity (site lookup). Using all of this and honoring the HSI Source and Mode switches, Area NAV computes all HSI parameters except for the selected course. This data is output at 1.04 Hz to the HSI processor, which computes the eighth parameter (selected course), forms a command word for each HSI parameter², applies FOH's as required, and forms the HSI control and test words for each transmission. Note: The test words are not used by MEDS software, although they are still generated by the GPC's. A control word is constructed from the various data validity flags and determines whether display parameters will be displayed by the IDP, for instance whether to drive a bearing flag or blank the glide slope indicator.

Entry HSI processing begins on transition to MM 304 (or MM 602 for RTLS aborts) and continues through rollout, and is dependent on crew switch positions. The crew determines for each HSI the source of the data to be displayed (TACAN (GPS), NAV, or MLS), the string if TACAN, GPS, or MLS (1, 2, or 3), and the entry phase to which the data apply (Entry, TAEM, or A/L). The software will cycle the display mode from ENTRY to TAEM to A/L automatically if the MODE switch is left in the ENTRY position.

² Note: This process includes subtracting vehicle magnetic heading from the course, PRI BRG, and SEC BRG to allow the compass card and bearing pointers to be driven with respect to the HSI case; this was discussed in section 8.2.5.1.2.

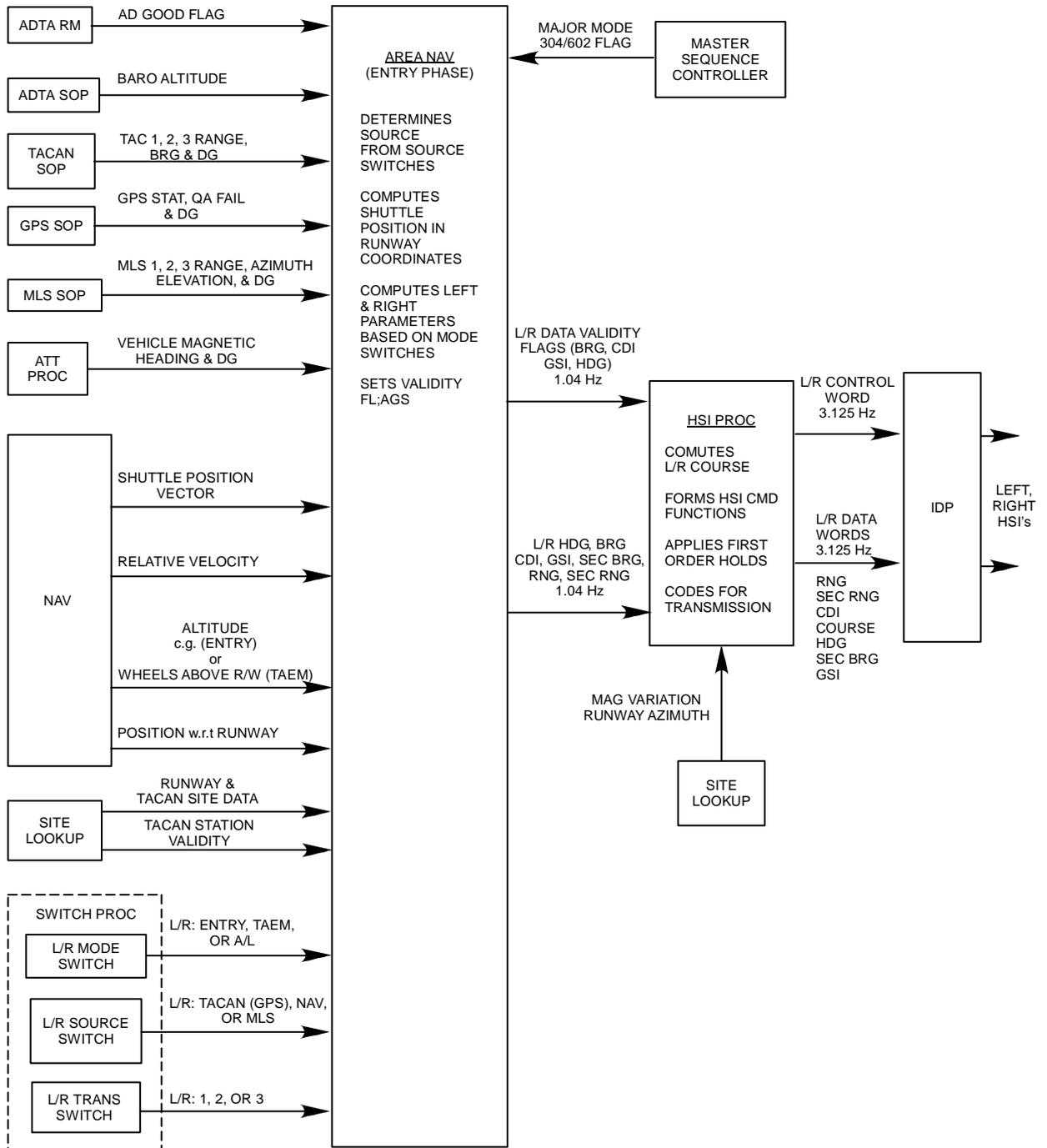


Figure 8 - 29 HSI Data Flow

HSI display peculiarities - Although the crew determines the HSI configuration at any given time during the entry sequence, there are a few instances where software decisions will cause a jump or discontinuity on the HSI display.

- a. When the Shuttle is below 1500 ft, MLS elevation is set to zero to prevent invalid elevation data during landing rollout. This event will be visible by the GS flag.
- b. The CDI becomes active at transition to TAEM; the CDI could jump.

Area NAV determines whether BRG, CDI, and GSI flags will be displayed and in what combinations. It does this based on data validity indicators from each source.

- a. TACAN, GPS, MLS - The TACAN range and bearing, GPS state, and the MLS range, azimuth, and elevation data from each line replacement unit LRU (1, 2, and 3) each has a validity indicator which is the OR of CF, loss of lock conditions, or DG flags. CF is set if the data coming into the GPC is not being updated. Lock-on is a discrete from the TACAN or MLS indicating station lock in the onboard receiver for that parameter. GPS provides a data good flag and QA fail to indicate acceptable data. The data good flag cannot be overridden, whereas the QA fail can.
- b. BARO altitude - The validity indicator for BARO altitude is the air data RM DG flag, reset whenever a dilemma exists.
- c. Heading - There is no longer a check on heading validity via the IMU attitude DG flag from IMU RM.

In general for MEDS, a red outline box is used to indicate invalid or missing data. However, the HSI parameters freeze in response to invalid data, with the exception of the GSI and range digitals, which blank with a red box. These responses occur because the IDP checks the HSI control word and does not update those parameters whose validity flags are set.

8.2.5.2.2 RTLS Abort (GRTLS Phase)

Entry HSI processing will begin automatically when MM 602 is called and will continue through MM 603. The Mode and Source switches function just as they would in a nominal entry sequence, and the entry, TAEM, and A/L definitions in 8.2.5.2.1 are still valid.

8.2.5.2.3 BFS HSI Display (MM's 304, 305, 602, and 603)

During entry the BFS has its own Area NAV software module, which is independent of the PASS Area NAV but parallels it in design. That is, it is designed to use data sources independent of the BFS auto guidance to perform the following functions.

- a. Monitor and assess auto guidance (BFS)
- b. Provide minimum information necessary to control the vehicle manually during entry

The same parameters and flags are displayed on the HSI's under BFS control as under PASS control.

Differences from PASS - There are some significant HSI display processing differences from PASS when in BFS:

- a. Both HSI's are driven with the same set of commands (no right/left computations are done in BFS Area NAV).
- b. Switch effectivity - In BFS, the HSI switch processing is very limited (see Table 8 - 5).
- c. In BFS, the HSI display parameter definitions change automatically with the entry guidance transitions (Entry, TAEM, A/L) rather than with the HSI Mode switch (which is not sampled). The parameter definitions, however, are the same as given in section 8.2.8.5, for the Entry, TAEM, and A/L positions of the Mode switch.

Table 8 - 5 Table 4 BFS HSI Switch Processing

| Switch | BFS Effectivity |
|---|--|
| Left/Right Mode (Entry, TAEM, A/L) | Not sampled in any position. |
| Left/Right Source (TACAN (GPS), NAV, MLS) | Left switch only. <u>TACAN</u> - TACAN range & azimuth used from single TACAN source; baro altitude for ADS <u>GPS</u> - GPS state (if 3 string GPS) <u>NAV</u> - BFS NAV state vector used <u>MLS</u> - If selected, results in invalid data (BRG, CDI, and GS flags) |
| Left/Right Source (1, 2, 3) | Not sampled in any position. |

8.2.5.2.4 Ascent HSI

During MM's 102, 103, and RTLS 601, the HSI provides information relative to the target insertion orbit (TIO). The definitions of heading, course, course deviation, and bearing are provided below; GS deviation, and range are not displayed. The ascent HSI is shown in Figure 8 - 30.

Heading references the target insertion plane, with 0 representing the pre-flight planned target insertion plane. The course arrow is target inclination, and is pinned at 0 unless TAL or ATO with variable IY active. If TAL, the arrow is redefined to point at a tangent to a cross-range circle around the TAL site, and reflects the nominal or minimum crossrange, as appropriate. If ATO and variable IY is active, the course arrow indicates the new target inclination course (if redefined), with 0 still representing the pre-launch target insertion plane course. If variable IY steering is not active, the course arrow remains pegged at 0.

The CDI provides the lateral error from the targeted plane in nautical miles. It is auto re-scaled at 50,10, & 1nm and labeled. One mile scaling only occurs after 6+30 MET. The CDI is blanked for RTLS, ECAL/BDA, and TAL. If ATO is selected with IY steering active, the CDI is changed to white along with the course arrow.

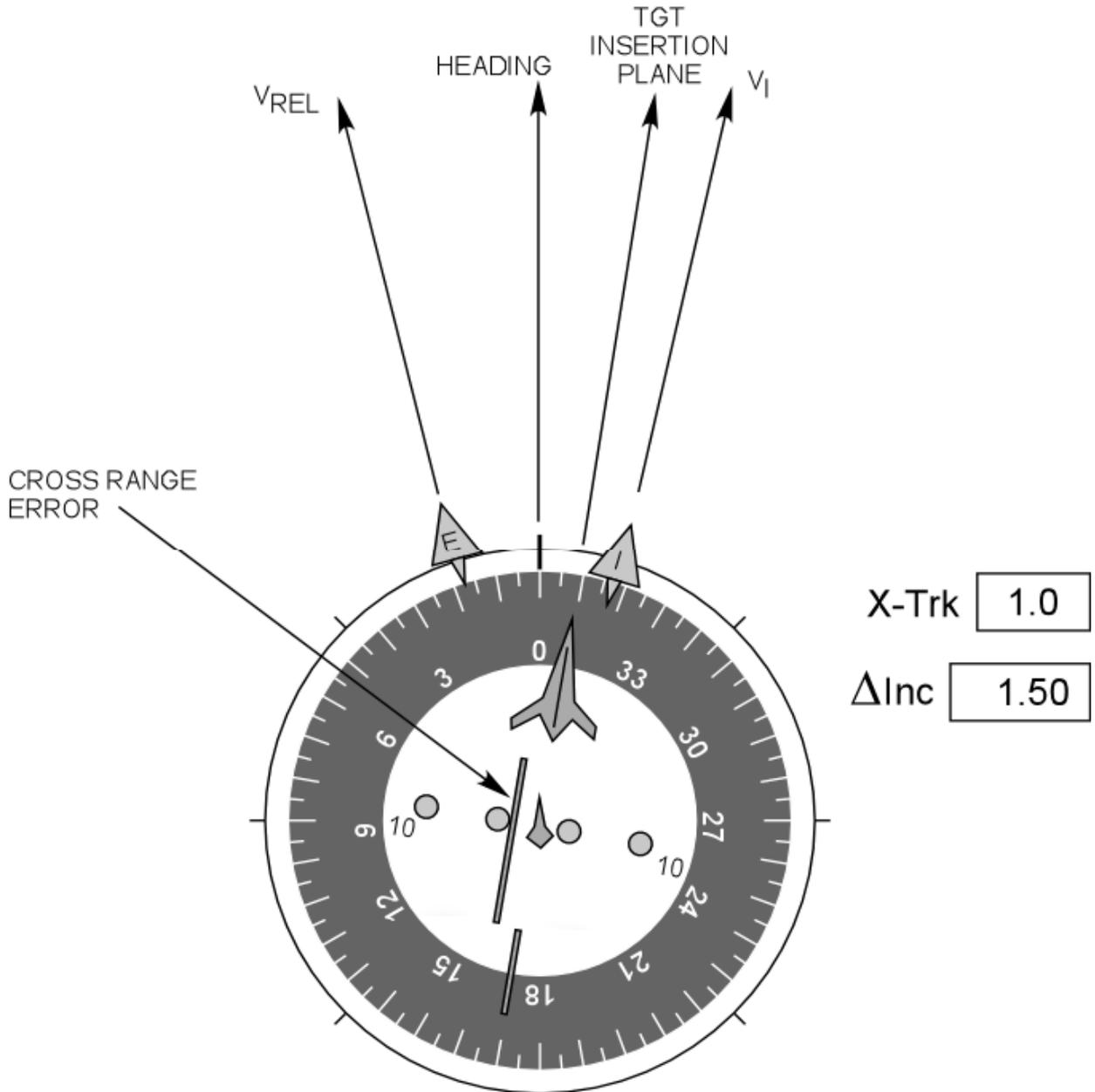


Figure 8 - 30 Ascent HSI

The ascent NAV user parameter processing (UPP) computes the unit vector normal to the target insertion plane in M50 coordinates (I_{YD_M50}) and forwards it to both RTLS UPP and the ATT PROC. During MM 102 and 103 the ascent NAV UPP also computes headings with respect to the target insertion orbit plane (TIOP) for the inertial and relative velocity vectors (HDG V_I , HDG V_{REL}) and sends them to the HSI processor to drive the inertial and V_{REL} bearing pointers, respectively.

During MM 601 (powered RTLS), the RTLS UPP computes these same two headings and also the runway heading with respect to TIOP.

The ATT PROC uses I_{YD_M50} to compute the vehicle yaw angle with respect to the TIOP, and also computes estimated sideslip (angle between body +X axis and the relative velocity vector). It sends these and the heads up/down indicator to the HSI processor.

The basic update rate for the source parameters during ascent is 1.04 Hz; however, the rate is increased for the UPP parameters to 3.125 Hz during the guidance fine countdown to main engine cutoff (MECO). The HSI processor applies an FOH to all parameters to smooth the values to 3.125 Hz.

When engaged, the BFS drives the HSI exactly as the PASS does during ascent MM's 102, 103 and RTLS MM 601.

8.2.6 AMI

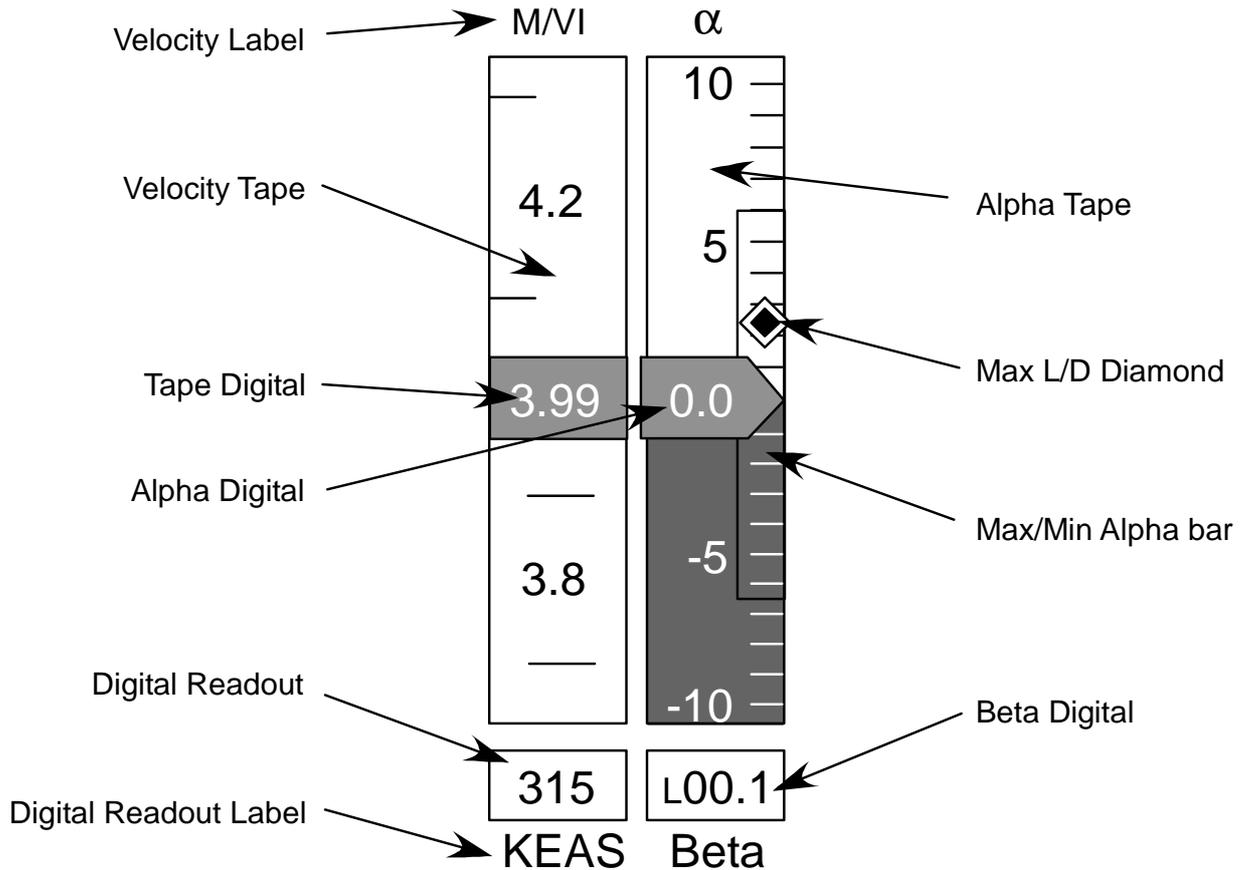


Figure 8 - 31 Alpha/Mach Indicator

Illustrated in Figure 8 - 31, these tapes provide NAV derived and barometric angle of attack (α) in degrees, and velocity in Knots Equivalent Airspeed (KEAS) or decimal Mach number. The source of the data is determined by the Air Data Source Select switch (para 8.2.8.6). Both tapes scroll up and down within their respective windows, with the tape digital value provided in a white-on-black background format in a box at the tape's center. The tape displays the Mach value for all major modes except MM 305 and 603. The digital readout below the tape shows KEAS except in MM 305 and 603. For MM 305 and 603, the digital readout and tape values will swap at $M < 0.9$. In other words, once on the HAC, the tape values change to KEAS, with Mach number provided in the digital readout below. The digital readout label will also change from "KEAS" to "M". Likewise, the label above the velocity tape changes with major mode and RTLS flyback status. In MM 101, 102, 304, 305, 602, and 603, the label will be "M/VR" indicating relative velocity is displayed. In MM 103 and 104, "M/VI" is displayed indicating inertial velocity. During Powered RTLS (MM 601), prior to PPA, the label will be "M/VI" indicating inertial velocity, and after PPA "M/VR" indicating relative velocity. A detail of the velocity tape graphics is provided in Figure 8 - 33 and Figure 8 - 34.

Invalid data is indicated when a tape is replaced by a blank red box.

In MM 102/103 and 601, a beta value (in degrees) is provided below the alpha tape. “L” and “R” indicate the yaw steering required to null the beta value, and used in conjunction with the “E” bearing pointer. Note: Due to a coding limitation, there is a brief window during RTLS flyback (between $V_{REL} = 0$ and alpha = 90 degrees) when the beta sense will not be consistent with the “E” bearing pointer, and the two indicators will appear to be in conflict. This is due to vehicle attitude and relative motion, but will be corrected once vehicle acceleration is high enough to cause alpha to decrease from 90 degrees. Both the “E” pointer and Beta digital blank at altitude > 200K or MET > 2:30, whichever occurs first. Resolution of the digital is one decimal place.

Alpha is shown in a scrolling window with a fixed pointer indicating the current angle of attack in one degree increments. The digital decimal value is provided as part of the pointer using white numbers on a black background. Positive alpha values are black numbers on a white background, and negative values are white numbers on a black background. A Max L/D diamond indicates the optimum alpha value for maximum lift over drag flying techniques, and is displayed when $M < 3.0$. The values for the diamond’s location are linearly interpolated from a table of values provided in Table 8 - 6. A detail of the alpha tape graphics is provided in Figure 8 - 32.

Table 8 - 6 Max L/D Values

| Mach | Alpha (Max L/D) |
|------|-----------------|
| 3.0 | 17.0 |
| 2.0 | 15.0 |
| 1.0 | 12.0 |
| 0.95 | 10.5 |

A green maximum and minimum alpha bar is provided in both PASS and BFS MM 304, 305, 602, and 603, with the exception of contingency aborts. Note: When using low alpha techniques during aborts, the alpha bar may not be visible. The values of the bar are derived by linear interpolation of tables of values in Table 8 - 7 and Table 8 - 8.

Table 8 - 7 MM 304 & 305 Alpha Limits

| Mach | Alpha | |
|------|-------|------|
| | Max | Min |
| 0.0 | 20.0 | -4.0 |
| 0.2 | 20.0 | -4.0 |
| 0.5 | 20.0 | 0.0 |
| 0.6 | 20.0 | 0.7 |
| 0.8 | 15.0 | 2.0 |
| 1.1 | 15.8 | 4.0 |
| 2.0 | 18.3 | 4.0 |
| 2.5 | 19.6 | 6.0 |
| 3.0 | 21.0 | 7.6 |
| 3.2 | 21.4 | 8.2 |
| 3.5 | 22.0 | 12.0 |
| 5.0 | 28.0 | 16.0 |
| 8.0 | 40.0 | 28.9 |

| | | |
|------|------|------|
| 9.6 | 44.0 | 33.0 |
| 11.4 | 44.0 | 36.0 |
| 27.0 | 44.0 | 36.0 |

Table 8 - 8 MM 602 & 603 Alpha Limits

| Mach | Alpha | |
|------|-------|------|
| | Max | Min |
| 0.0 | 20.0 | -4.0 |
| 0.2 | 20.0 | -4.0 |
| 0.5 | 20.0 | 0.0 |
| 0.6 | 20.0 | 0.7 |
| 0.8 | 15.0 | 2.0 |
| 1.1 | 15.8 | 4.0 |
| 2.0 | 18.3 | 4.0 |
| 2.5 | 19.6 | 6.0 |
| 3.0 | 21.0 | 8.2 |
| 3.4 | 21.0 | 10.0 |
| 4.0 | 21.0 | 10.0 |
| 5.1 | 50.0 | 10.0 |
| 6.0 | 53.0 | 10.0 |
| 8.0 | 53.0 | 10.0 |

The BFS only supports the velocity and alpha tapes in glided flight (MM 304, 305, 602, 603), although the Beta digital is available until MECO.

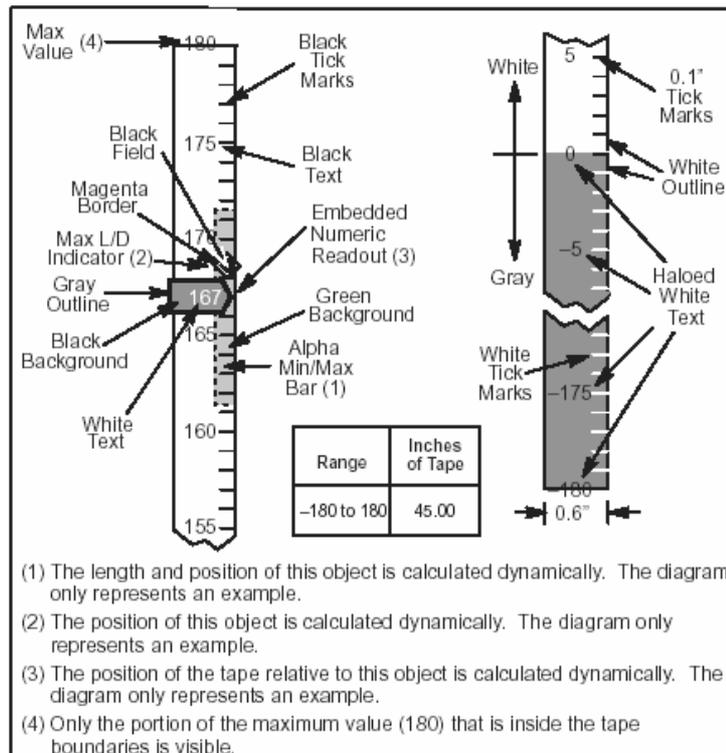


Figure 8 - 32 Alpha Tape Graphical Detail

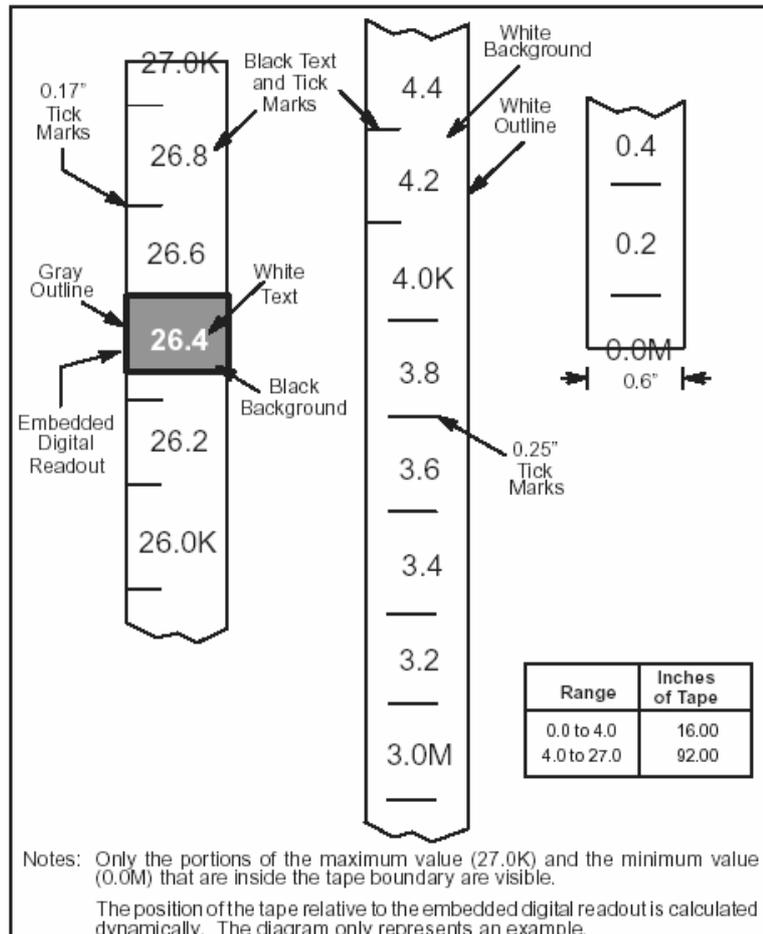


Figure 8 - 33 Velocity Tape Graphical Detail 1

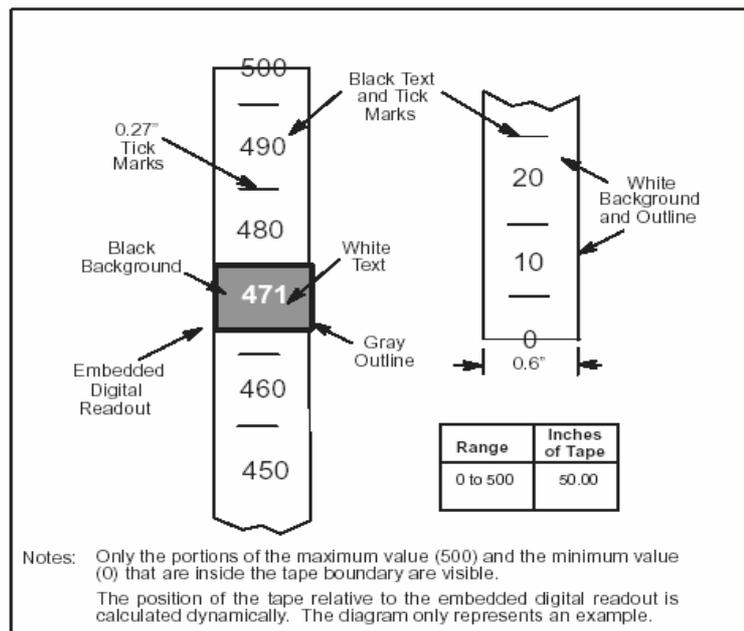


Figure 8 - 34 Velocity Tape Graphical Detail 2

The purpose of the AMI is to display three essential flight parameters to the crew during the ascent, RTLS, and entry phases:

Angle of attack (ALPHA)

Mach/velocity (M/VEL)

Equivalent airspeed (EAS)

Mach and EAS are displayed simultaneously, one on the tape, and the other in the digital field below the tape. The value driving the tape/digital depends on the phase of flight, and is discussed earlier in this section. The AMI's are driven in MM's 102, 103, 601-603, 304, 305, and GNC OPS 8 (on-orbit dedicated display checkout). The information displayed on the AMI's is determined by both crew switching and software configuration. The two AMI's are driven independently but can have the same data source.

8.2.6.1 Display Features

8.2.6.1.1 ALPHA

The leftmost scale (ALPHA) is for the display of vehicle angle of attack. The vehicle angle of attack is defined as the angle between the vehicle +X axis and the wind-relative velocity vector (negative wind vector) (Figure 8 - 35).

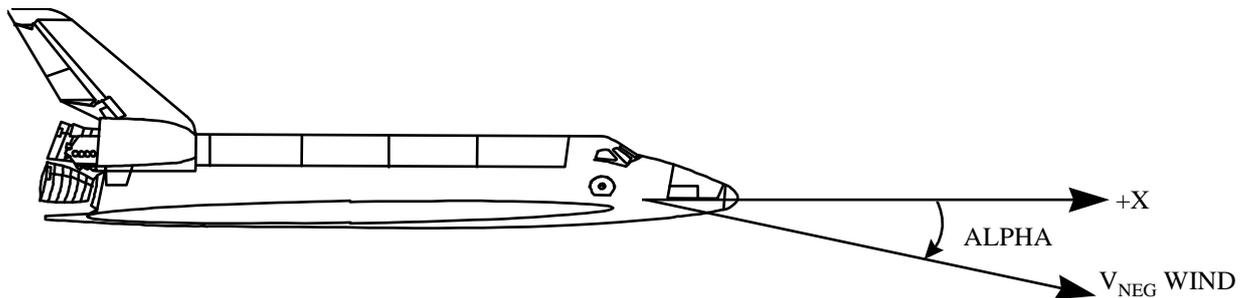


Figure 8 - 35 Alpha

ALPHA is displayed using a combination moving scale and moving pointer. For ALPHA between -4° and $+28^\circ$, the scale remains stationary and the pointer moves to the correct reading. For ALPHA less than -4° or greater than $+28^\circ$, the pointer stops (at -4° or $+28^\circ$) and the scale moves so that the correct reading is adjacent to the pointer. The ALPHA tape ranges from -18° to $+60^\circ$ with no scale changes. Figure 8-50 shows an angle of attack of 12. Note that the negative scale numbers (below the 0) have no minus signs; the actual tape has black markings on a white background on the negative side, and white markings on a black background on the positive side. ALPHA is displayed during MM's 102, 103, 601, 602, 603, 304, and 305.

8.2.6.1.2 M/VEL

The M/VEL scale is for the display of Mach number or velocity. Mach number is the ratio of vehicle airspeed to the speed of sound in the same medium. Relative velocity in this case is the vehicle airspeed. Figure 8 - 1 has a velocity reading of Mach 3.99. M/VEL displays V_{REL} in MM's 102, 304, 305, 602, and 603 and V_1 in MM's 103 and 601.

8.2.6.1.3 EAS

The EAS scale is used for the display of equivalent airspeed. The sequence of airspeed calculation follows:

Indicated airspeed (IAS) = instrument-derived reading, corrected for error

Calibrated airspeed (CAS) = IAS, corrected for position error and line lag

Equivalent airspeed (EAS) = CAS, corrected for compressibility

True airspeed (TAS) = EAS, corrected for density

The EAS scale is a moving tape, with EAS read at the central window. The tape range is 0 to 500 knots, and scaling is 1 in. per 10 knots. EAS is displayed in MM's 102, 103, 601, 602, 603, 304, 305.

8.2.6.1.4 AMI Flags

AMI flags, once part of the old steam gauge AVVI, are not used in MEDS. Invalid data is indicated by a red outline box.

8.2.6.1.5 AMI Controls

The AMI is controlled through the Air Data Source switch and the Horizontal Situation Display (HSD).

The Air Data Source switch is described in section 8.2.8.6.

The Horizontal Situation SPEC processed through entry, TAEM, and A/L, provides the crew with another control relating to the AMI display. The ADTA to G&C item entries (AUT, INH, FOR) are located on the HORIZ SIT display (figure 8-54). These commands affect the AMI display only when the AIR DATA switch is in the NAV position. The LEFT or RIGHT position will override these commands for the associated AMI.

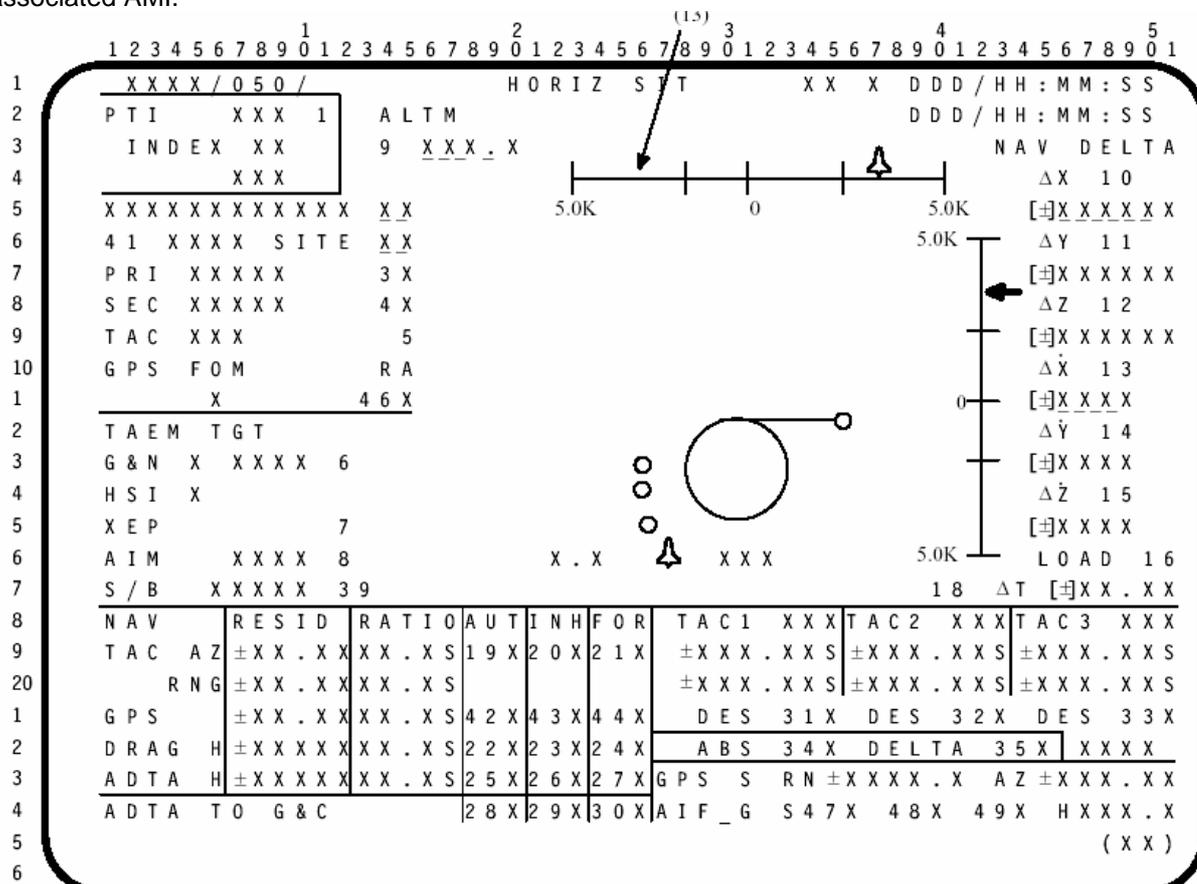


Figure 8 - 36 Horizontal Situation Display

AUT - When AUT (item 28) is active and the AIR DATA switch is in NAV, the ADTA SOP will automatically enable the center air data calculations at Mach 2.5. Assuming the probes are deployed and the data are valid, ALPHA, M/VEL, and EAS will change from NAV-derived parameters to center air data at Mach 2.5.

INH - The inhibit (INH) entry (item 29) inhibits the use of center air data. If INH is active prior to Mach 2.5, the automatic transition to center air data will not occur. If active after transition, INH causes a return on the AMI to NAV-derived parameters. If transition has occurred and the probe data are valid, INH is the only way the crew has to see NAV-derived data on the AMI.

FOR - The FOR entry (item 30) forces the use of center air data before auto transition (Mach 2.5), but does not override the lockout of center air data due to an RM dilemma after Mach 2.5.

Table 8 - 9 summarizes the effect of the controls on the display. Remember that the AMI can display the same ALPHA, MACH, and EAS parameters that are being provided to the GNC users only when the AIR DATA switch is in the NAV position. Even then there are two cases where the AMI and user parameters will differ:

Mach jump region - Between Mach 1.1 and Mach 1.6 (transition to subsonic), the ADTA SOP sends substitute parameters (Mach jump constants) to users to avoid the transients in that region. The AMI does not receive the Mach jump constants, and will continue to display dynamic air data (center output).

Default parameters - The second case where AMI and user air data parameters differ with the AIR DATA switch in NAV is when the ADTA RM DG flag is reset (RM dilemma). In this case the ADTA SOP will output default constants to users until the dilemma is resolved. The AMI does not receive the default parameters, and will display instead the NAV-derived parameters (rather than center air data).

Table 8 - 9 AMI Displays vs. Controls

| Air Data Switch | | Prior to Probe Deploy | Probe Deployed, before M 2.5 | After Mach 2.5 |
|-----------------|----------|-----------------------|------------------------------|---|
| | | α , Mach/EAS | α , Mach/EAS | α , Mach/EAS |
| Left/Right | | Red Box | Left/Right probe data | Left/Right probe data |
| NAV | AUTO/FOR | NAVDAD | NAVDAD | Center AD unless DLMA, then NAVDAD or DEFAD (after M 1.5) |
| | INH | NAVDAD | NAVDAD | NAVDAD or DEFAD (after M 1.5) |

8.2.6.2 AMI Processing

8.2.6.2.1 Data flow

Figure 8 - 37 is a simplified view of the data processing required for the AMI display. Left and right air data probes provide localized angle-of-attack, static, and total pressures along with total temperature to the left and right ADTA pairs, which compensate the data and then throughput it in serial form to the GPC's. The air data SOP and RM together form a complete right and left data set for the left/right air data calculations and a center set of pressures and temperatures for the center calculations most representative of the actual air data environment.

The left, right, and center calculated parameters for the AMI include ALPHA, Mach, and EAS. The left and right parameters are sent directly to the AMI processor. The center parameters become an option, along with software-derived parameters, to send to GNC users, including the AMI processor when the AIR DATA switch is in the NAV position. The logic defined in the previous section determines whether the AMI processor receives center air data or software parameters for driving the AMI's when the AIR DATA switch is in NAV.

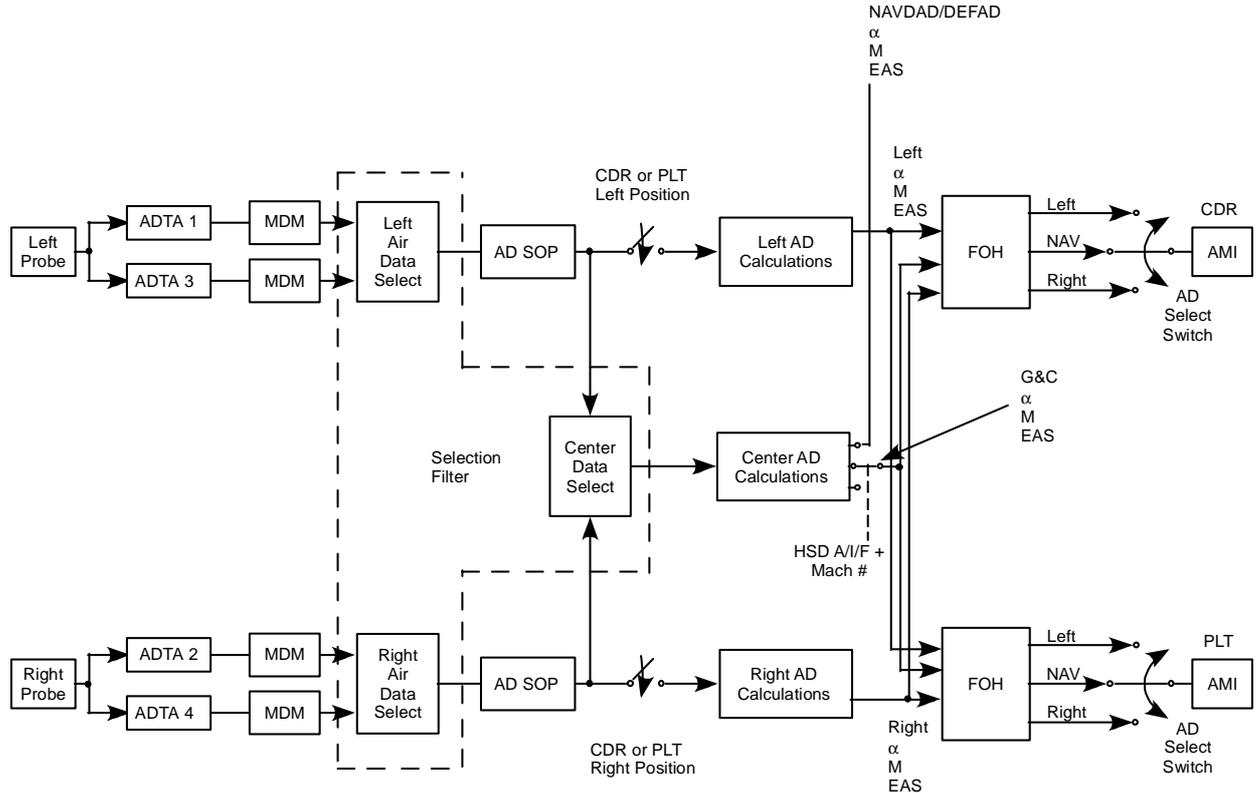


Figure 8 - 37 Data flow to AMI

The AMI processor, applies FOH to ALPHA, Mach, and ACCEL and forwards them at 160 ms intervals (6.25 Hz) to the left and right IDP's.

8.2.6.2.2 Nominal Data Sequence

The nominal data sequence was summarized in the previous section, along with processing differences between the AMI display and the software GNC air data users. In summary, AMI processing is done throughout MM's 304 and 305.

8.2.6.2.3 GRTLs Abort Data Processing

RTLs AMI processing during MM's 602 and 603 is identical to that done in MM's 304 and 305, respectively.

8.2.6.2.4 Invalid Data

Invalid data is represented by a red outline in place of the AMI. The red box will be present if air data is not available with the Air Data Source switch in Left or Right, or if the last ADTA is comfaulted. MEDS will also drive the red box if it detect invalid data.

8.2.6.2.5 BFS AMI Processing (Entry)

As currently designed, BFS AMI processing for entry has the same features as the primary software concerning parameter definition, data sources, and data sequencing, but is simplified in crew controls (display options) and is only available in MM 304, 305, 602, and 603..

Parameter definition - ALPHA, M/VEL, and EAS, are the same physical quantities as in the PASS.

Data sources - The display parameters are derived from probe data or from NAV software, as in the PASS, but for probe data only the center air data is available (no left and right probe data calculations).

Data sequence - Like PASS, the BFS AMI's can display NAV-derived data prior to probe deploy, and probe or NAV-derived data after probe deploy.

Crew controls - The BFS AMI displays respond only to the left AIR DATA switch.

(1)LEFT/RIGHT position - Both AMI's display center air data, averaged from a left and a right ADTA.

(2)NAV position - Both AMI's display NAV-derived data above Mach 2.5 and user data below Mach 2.5. User data are determined by the HORIZ SIT SPEC AUT, INH, or FOR item entry. These item entries have the same function as in the PASS.

8.2.6.2.6 Ascent AMI

Both AMI's are driven during ascent and powered RTLS with NAV-derived parameters. The left and right AIR DATA switches are not sampled and have no effect on the displays.

ALPHA - NAV-derived angle of attack from the attitude processor (MM 102, 103, 601).

M/VEL - NAV-derived INRTL velocity vector magnitude from the ascent or RTLS UPP (MM's 102, 103, 601).

EAS - NAV-derived EAS from the ascent on RTLS UPP (MM 102, 103, 601).

After the MM 102 transition (liftoff) the NAV-derived parameters are assumed always to be valid.

FOH is applied to the AMI parameters according to Table 8 - 10.

Table 8 - 10 AMI First Order Hold Processing

| Parameter Source Rate (Hz) | | | | |
|----------------------------|--------|--------|--------|------------|
| | MM 102 | MM 103 | MM 601 | FOH to -Hz |
| Alpha | 6.25 | 6.25 | 6.25 | N/A |
| Mach | 6.25 | 1.04 | 3.125 | 6.25 |
| EAS | 1.04 | 1.04 | 1.04 | N/A |

The AMI is not processed by the BFS during ascent, and the AMI is not available after BFS is engaged. However, the BFS ASCENT TRAJ provides the data.

8.2.7 AVVI

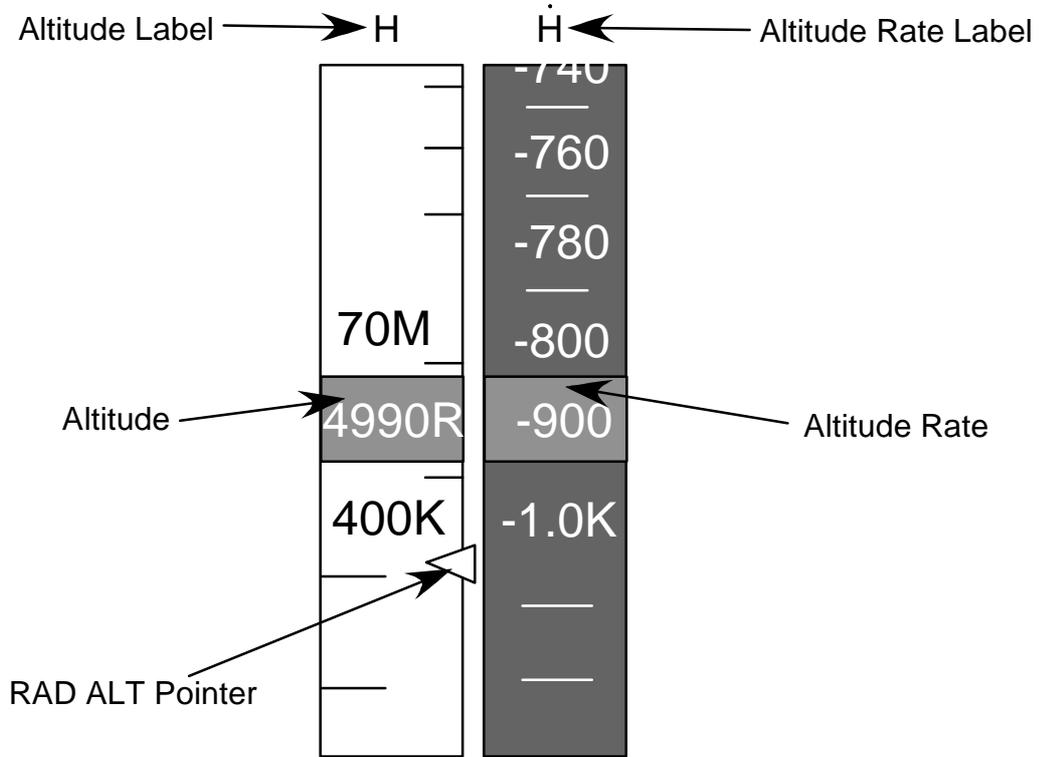


Figure 8 - 38 Altitude/Vertical Velocity Indicator

The purpose of the AVVI is to display four essential flight parameters to the crew during ascent, RTLS, and entry.

Altitude acceleration (entry only)

Altitude rate (ascent and entry)

Barometric altitude (ascent and entry)

Radar altitude (entry only)

The AVVI's are driven only during OPS 3 (MM 304, 305), OPS 6 (MM's 602, 603), and GNC OPS 8 (on-orbit dedicated display checkout). The information displayed on the AVVI's is determined by both crew switching and software configuration. The AVVI's are driven independently but can have the same data source. The Altitude/Vertical Velocity Indicator (AVVI) (Figure 8 - 38) provides NAV-derived or barometric altitude (determined by position of the air data source switch) in feet (ft), and NAV-derived and barometric altitude rate in feet/second (fps). Illustrated in Figure 8 - 39, the tape will display NAV altitude, but a radar altitude pointer will point to the radar altitude on the NAV tape, and the embedded digital will indicate valid radar altitude with an "R". Calibrated barometric data is selected in TAEM by positioning the AIR DATA switch to select the left or right probe. Starting at ground level (0 feet), the altitude tape numerical value increments every 50 feet, with 10 foot marks along the center of the tape up to 200 feet. From 200 to 1000 feet, the values increment every 200 feet, with marks every 100 feet. Beginning at 2000 feet altitude, the tape values transition to thousands of feet (indicated by a following "K" at 5000 feet), with numbers every 1000 feet, and marks every 500 feet up to 30K feet. From 30K to 100K, numbers increment every 5K feet, with marks every 1K feet. Beginning at 100K, the marks are moved to the left side of the tape, and numbers incremented every 50K feet up to 400K. Above 400K feet altitude, the scale transitions to nautical miles (nmi), indicated by a following "M". Altitude numerical value is then incremented every 5 nmi beginning at 70 nmi, with marks along the right side of the tape every nautical mile up to 165 nm. Numbers and marks are black on a white background. During Entry

and below 2000 feet, the tape background transitions from white to yellow. Invalid data is indicated when a tape is replaced by a blank red box.

Invalid data is indicated when a tape is replaced by a blank red box.

In Figure 8 - 40, the altitude rate is divided into black-on-white background for positive values (climbing), and white-on-black for negative values (descending). Numerical increments are specified every 20 fps, and marks used every 10 fps up to 1000 fps. Thereafter, numerical increments are specified every 500 feet, followed by a "K". One hundred (100) foot increments are specified by horizontal marks in the center of the tape. Digital values for each tape are provided in centered windows with white numbers on a black background.

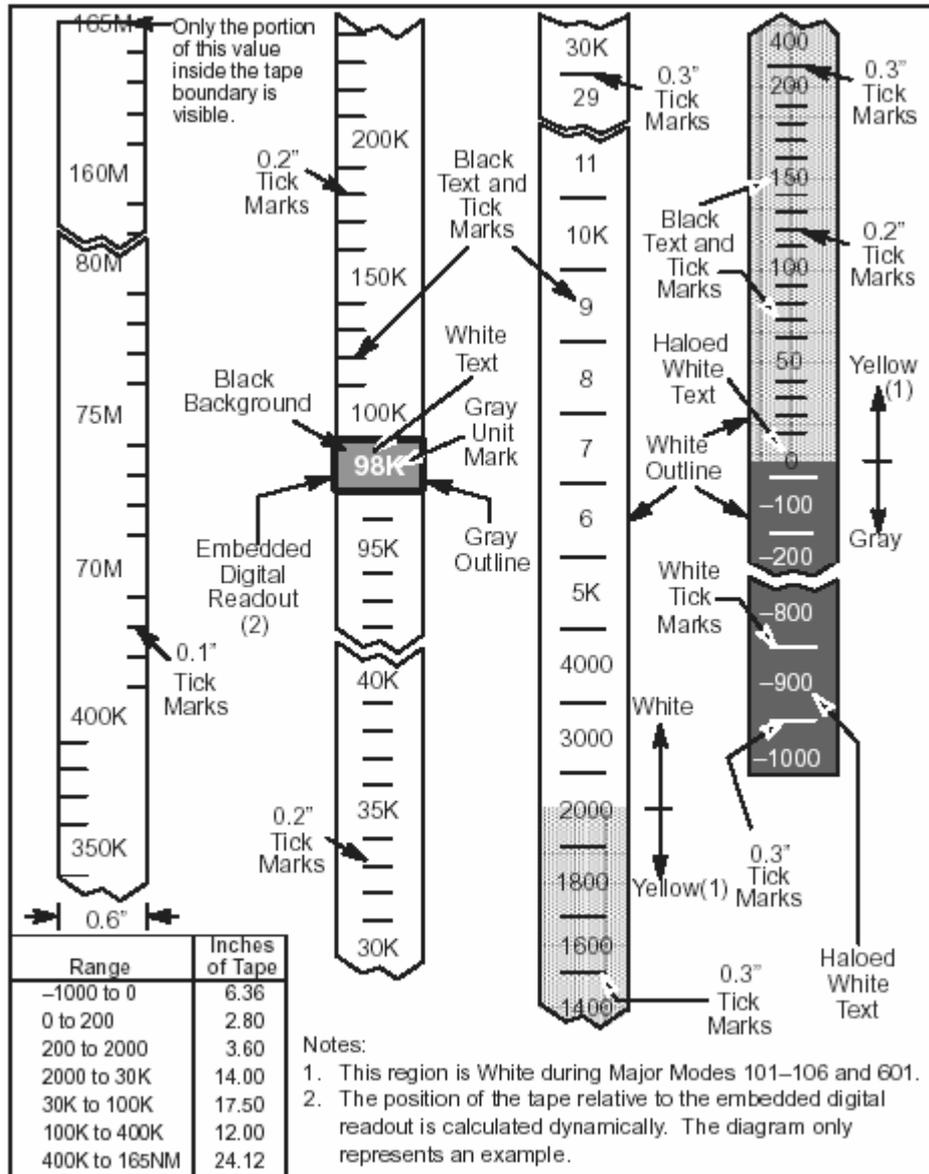


Figure 8 - 39 AVVI Altitude Tape

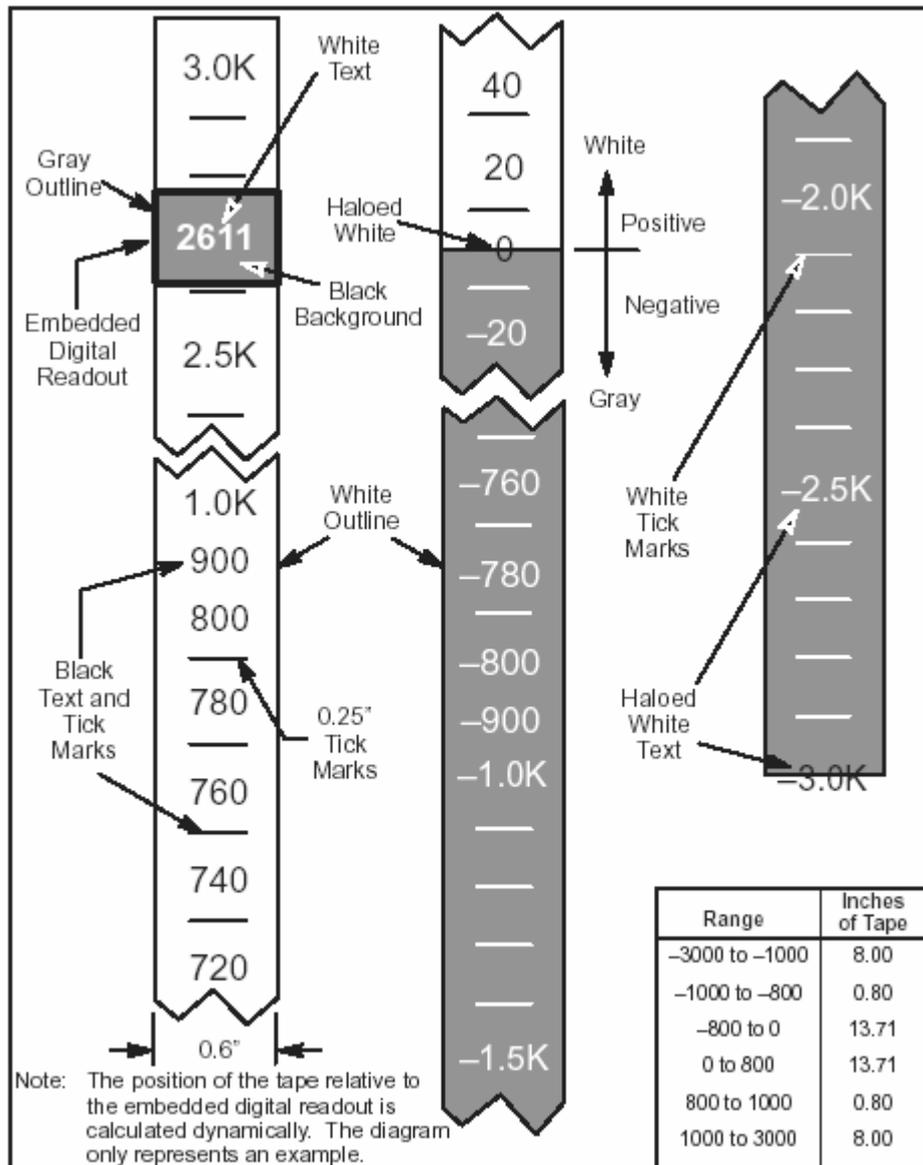


Figure 8 - 40 AVVI Altitude Rate

8.2.7.1 Display Features

8.2.7.1.1 Altitude Acceleration

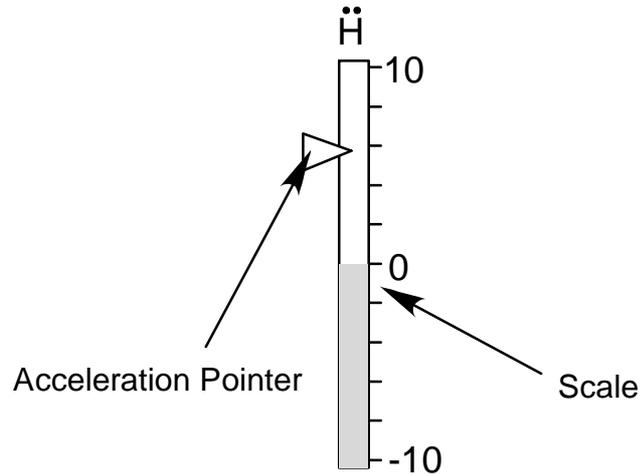


Figure 8 - 41 Altitude Acceleration Meter

Considered a part of the AVVI, the altitude acceleration meter is available in PASS MM 304 and 602. A pointer slides vertically on a scale in fps^2 . Maximum range of the scale is $\pm 10 \text{fps}^2$. A green pointer indicates the current NAV-derived altitude acceleration value. Graphical details are provided in Figure 8 - 42 Altitude Acceleration Graphical Details.

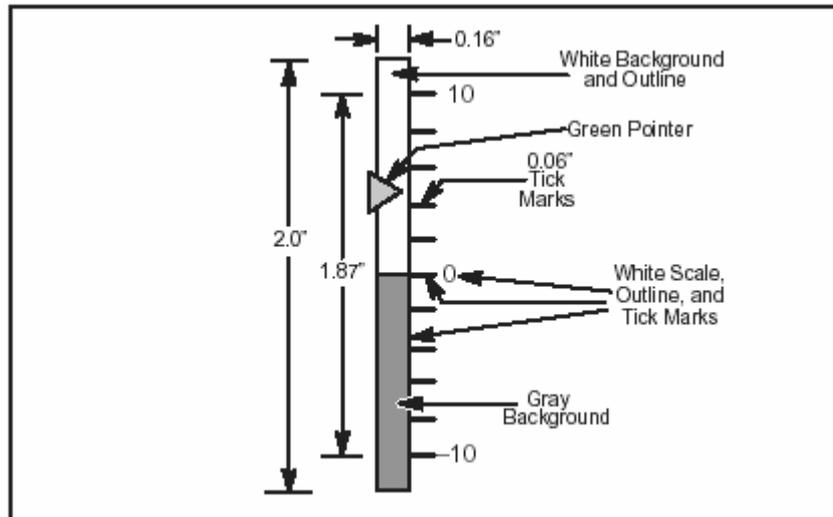


Figure 8 - 42 Altitude Acceleration Graphical Details

8.2.7.1.2 ALT RATE

The ALT RATE scale displays vehicle altitude rate, which is read where the moving tape intersects the central window.

The scale range is -2940 to +2940 fps with scale changes at -740 fps and +740 fps. The negative and positive regions are color-reversed; negative numbers are white on a black background, the positive numbers, black on white.

8.2.7.1.3 ALT

The ALT scale is a moving tape read against a central window which displays the altitude of the vehicle above the runway (BARO altitude). The scale range is -1100 ft to +165 n. mi., with scale changes at -100 ft, 0 ft, and 200 ft, +30K ft, and +400K ft. The scale is in feet from -1000 to +400K and in nautical miles from +40 to +165 (feet and nautical miles overlap from +40 n. mi. to +61 n. mi.). Below 5K ft when RAD ALT data is available, and "R" will be displayed in the digital window to the right of the value.

8.2.7.1.4 AVVI Controls

The AVVI is controlled through the Air Data Source and RAD ALT select switch discussed in section 8.2.8.6 and 8.2.8.7. The AIR DATA switches discussed in the previous section also allow the crew to select the data source for part of the AVVI display.

The left and right RADAR ALTM switches (Figure 8 - 59) are two-position toggle switches. The sole purpose of this switch is to select a radar altimeter source (RA 1 or RA 2) for RDR ALT on the altitude tape (there is no dedicated radar altimeter tape since the MEDS implementation). Radar altimeter data is displayed on the altitude tape and indicated on by an "R" label, located to the right of the value in the central window, when valid RA data are available.

8.2.7.2 AVVI Processing

8.2.7.2.1 Data Flow

Figure 8-59 is a simplified view of the data flow to the AVVI's. The selected left and right ADTA pressure/temperature sets are provided to the left and right air data calculation modules. Left- or right-side calculations are done if either AIR DATA switch is in LEFT or RIGHT, respectively, and the resulting pressure-corrected altitude and altitude rate are passed to the AVVI processor. The AVVI processor also receives NAV-derived altitude, altitude rate, and altitude acceleration from entry NAV (ENTRY, TAEM, and A/L), as well as RA 1 or 2 altitude from the RA SOP.

The source parameters used by the AVVI processor are determined by the AIR DATA switch. After applying FOH's to ALT ACCEL, ALT RATE, and ALT to upgrade them from 1.04 Hz to 6.25 Hz, the AVVI processor limits and scales the data, then forwards them to the IDP's. Note the absence from AVVI processing of center air data and also of the AUT/INH/FOR item entries.

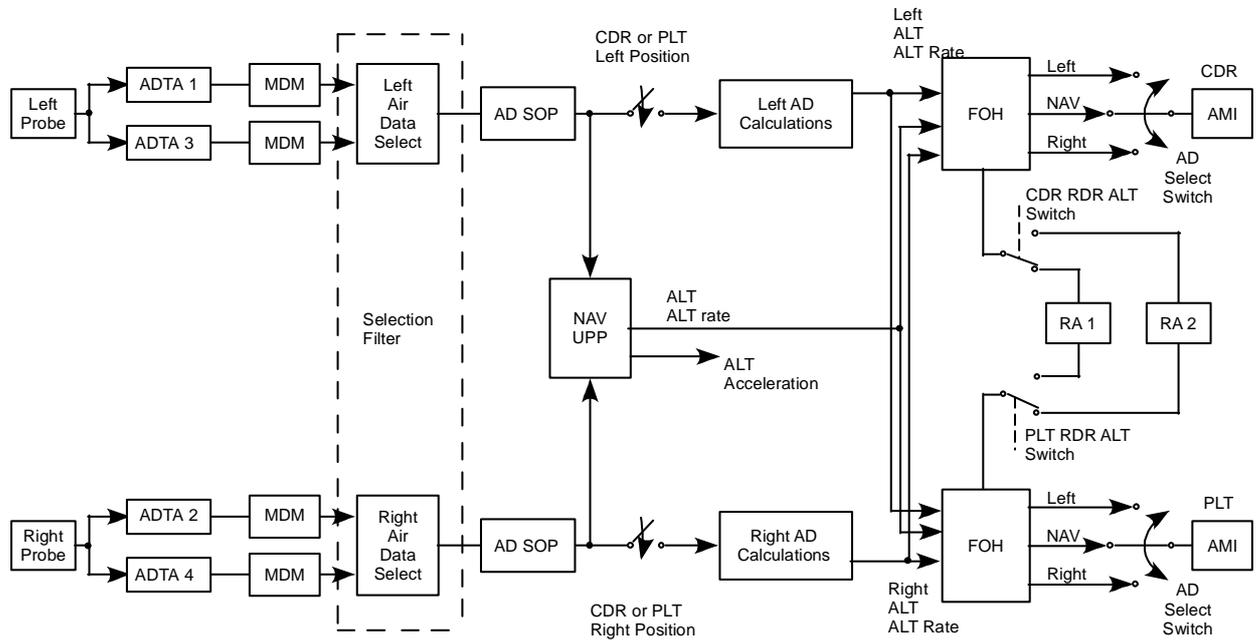


Figure 8-59.- Data flow to AVVI's

8.2.7.2.2 Nominal Data Sequence

There is no automatic AVVI data sequencing; i.e., the parameters do not vary in meaning during entry, TAEM, and A/L. Two minor exceptions are refinements made to the NAV-derived ALT and ALT RATE.

ALT - During entry (MM 304), ALT is the altitude of the NAV base above the runway. During TAEM (MM 305), NAV ALT is the altitude of the rear wheels above the runway altitude, with a correction made for the vehicle pitch angle. During A/L, ALT is the altitude above the extended runway plane, with the same correction for pitch angle.

ALT RATE - During entry, ALT RATE is not corrected for the rotation of the vehicle about the center of gravity (c.g.); during TAEM and A/L, this correction is made.

8.2.7.2.3 Abort Processing

Like the AMI, the AVVI is not processed during OPS 1. During OPS 6 (RTLS), AVVI processing begins at MM 602 and continues through MM 603. This AVVI processing is identical to that done in MM's 304 and 305.

8.2.7.2.4 Invalid Data

The MEDS AVVI has no off flags. Invalid data is represented by red outline boxes in place of the normal display.

8.2.7.2.5 BFS AVVI Processing (Entry)

The BFS AVVI processing during entry is similar to BFS AMI processing, described in the previous section. Summarizing the differences with PASS processing follows:

Parameter definition - ALT, ALT RATE, and ALT ACCEL are the same physical quantities as in the PASS. RDR ALT is driven on both AVVI'S by the selected RA (ref. RA Systems Brief for selection process).

Data sources - The display parameters are derived from probe data or from NAV software, as in the PASS. But for air data, only the average of a left and right ADTA is available; no left and right calculations are done.

Data sequence - Like PASS, the BFS AVVI's can display valid NAV-derived parameters prior to probe deployment, and air data or NAV-derived parameters after probe deploy.

Crew controls - Both AVVI's respond only to the left AIR DATA switch. When the AIR DATA switch is in LEFT or RIGHT, both AVVI's display average left/right ALT and ALT RATE, and NAV-derived ALT ACCEL. In the NAV position, the AVVI's always display NAV-derived data. There is no interface with the HORIZ SIT SPEC. For RDR ALT, the RA select switch is not used since the BFS software selects a RA based on lockon and commfault status (ref. RA Systems Brief).

8.2.7.2.6 Ascent AVVI

ALT and ALT RATE are driven on the AVVI's during ascent and powered RTLS. The left and right AIR DATA switches have no effect on the respective displays.

ALT - NAV-derived altitude with respect to the runway (MM's 103 and 601 only).

ALT RATE - NAV-derived radial component of INRTL velocity (MM 103); HDOT ellipsoid (MM 601).

FOH is applied to smooth the 1.04 Hz input of both parameters into a 6.25 Hz display rate.

The BFS does not process the AVVI's during ascent, but the same information is available on the ASCENT TRAJ displays.

8.2.8 Switches

8.2.8.1 ADI Attitude, Error, & Rate

The attitude switches are three-position toggle switches that select the frame of reference for the attitude display of the corresponding ADI. The two forward attitude switches are effective during the ascent, on-orbit, and transition phases; they have no effect (are overridden by software) during entry major modes ((MM's) 304, 305), and during the latter part of RTLS aborts (MM 602, 603). Only the left side is effective when the BFS is driving the ADI's. The aft ATTITUDE switch is effective only on orbit. The CDR switches will control the ADI on IDP's CDR 1, CDR 2, CRT 1, CRT 3, and MFD 1. The PLT switches will control an ADI on IDP's PLT 1, PLT 2, and MFD 2. The aft switch will control an ADI on IDP AFD 1 and CRT 4.

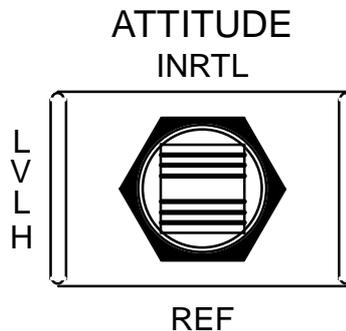


Figure 8 - 43 ADI Attitude Switch

INRTL - With the ATTITUDE switch in this position, the ADI attitude will be referenced to a predefined inertial (INRTL) frame of reference (Figure 8 - 44). Initially defined in the I-load, this frame can be redefined by ground uplink.

The I-load INRTL reference frame will provide a +X sense ADI ball reading of roll 0°, pitch 0°, and yaw 0° when the vehicle is pointed at 0° right ascension, 0° declination and the vehicle +Y

axis is pointed at 90° right ascension. In this case, the vehicle axis system is coaligned with the Mean of 1950 (M50) reference system.

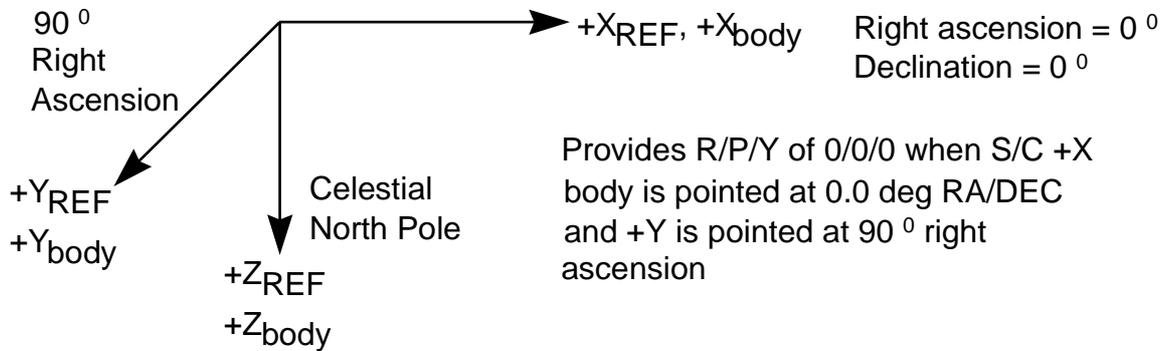


Figure 8 - 44 Inertial Frame - All Mission Phases

LVLH - The local vertical/local horizontal (LVLH) position requests one frame for ascent (MM 101-103) and another for transition (MM 104-106, 301-303) and on-orbit. The ascent reference is called LVIY and is a rotating reference frame defined by Z = down (along negative of M50 position vector), X = downrange (normal to plane formed by M50 position vector and a unit vector normal to the desired insertion plane), and Y completing the right-hand system. Figure 8 - 45 represents the LVIY frame at two points in the trajectory.

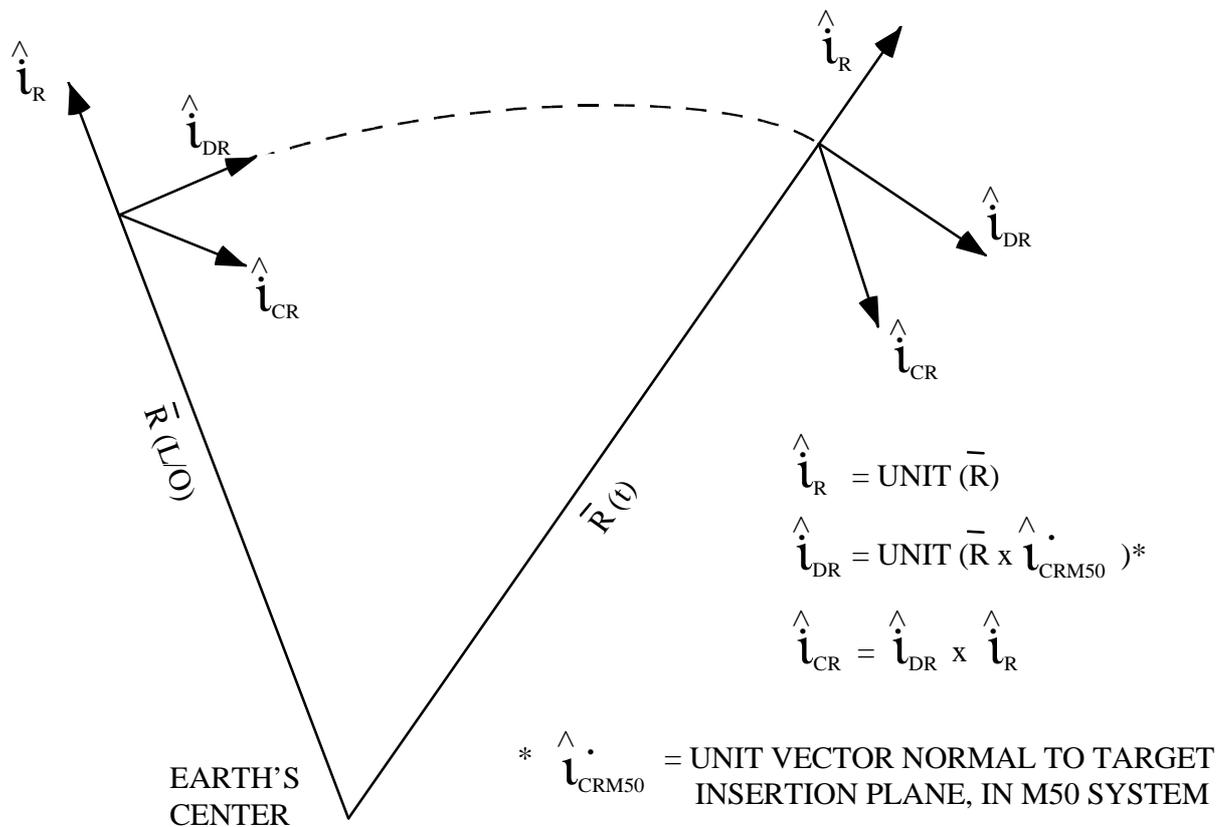


Figure 8 - 45 LVIY Frame at Two Points

For LVIY, the attitude is processed in a yaw/pitch/roll sequence; however, the ADI is driven to zero in yaw so that the ball becomes simply a pitch/roll (wings level) display. The yaw angle is displayed on the HSI display (section 8.2.5) and represents yaw with respect to the desired insertion plane. So, as complicated as LVIY sounds, as far as what is seen on the ADI, it reduces to a simple Earth-referenced pitch/roll display. The reason for going through the LVIY transformation as opposed to LVLH is to obtain the yaw relative to the orbit insertion plane for the HSI display and an ADI ball pitch-roll display that is more useful to the crew.

The LVLH attitude display for the transition and on-orbit ADI display reverts to three-axis drive and a pitch/yaw/roll sequence. The reference frame is defined as +Z toward the Earth center, +Y in the opposite direction of the Orbiter momentum vector, and +X completing the orthogonal set (Figure 8 - 46). The LVLH frame is not a fixed system; it is computed based on state vector and time. With the ATTITUDE switch in LVLH, the ADI can serve as a fly-to indication for maneuvering to the current LVLH attitude. With the Orbiter aligned inplane (+X and +Z body axes in the orbital plane) and holding INRTL attitude, the ADI ball will rotate in pitch at the orbital rotation rate.

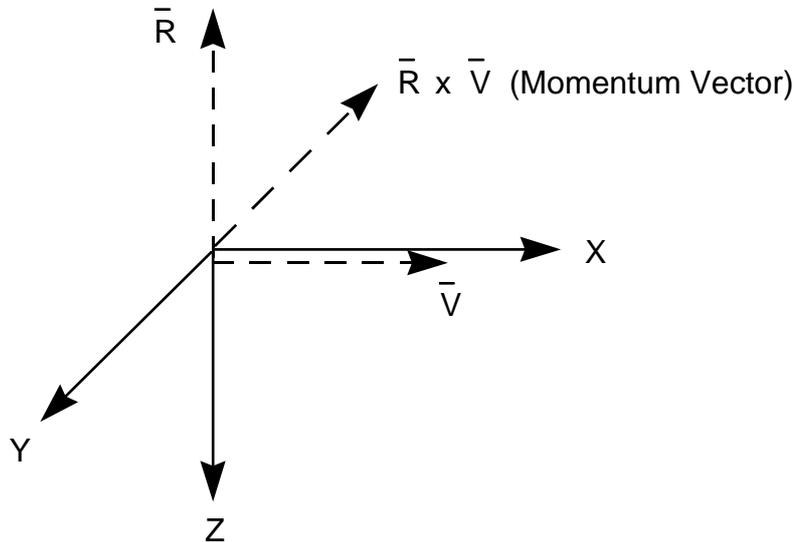


Figure 8 - 46 LVLH frame

REF - Selecting the reference (REF) position causes the ADI attitude to be based on an INRTL reference frame which, like the INRTL frame described earlier, is made equal to the LVIY frame prelaunch and in OPS 3 is initially defined in the I-load but can be redefined in flight by g-mem. The REF frame has an added option, however, in that it also can be redefined by depressing the attitude reference (ATT REF) pushbutton (described in section 8.2.8.3).

Figure 8 - 47 illustrates the ADI Rate and Error switches. The ADI RATE switch determines, for the corresponding ADI, the full deflection values and functions of the roll (top), pitch (right), and yaw (bottom) rate pointers for the Major Modes and switch configurations as defined in Table 8 - 11. Each ADI RATE switch has three positions: HIGH, MED, and LOW. The full scale deflection values and functions for all three rate pointers depend on the position of the ADI RATE switch, the mission phase, and the parameter (the rate can be I-load dependent). For Major Modes 101-106, 601, 801, and 301-303, the ADI RATE pointer full scale deflections are 10 deg/s in HIGH, 5 deg/s in MED, and 1 deg/s in LOW, for all axes, on all ADI's. The deflections are 5, 1, and 0.2 deg/s, respectively, in OPS 2 (as of OI-32), due to the smaller rates usually seen on orbit.

In Major Modes 304 and 602, the ADI RATE Switch provides an I-load dependent full scale deflection for all RATE switch settings, in all axes, on all supported ADI's. In Major Modes 305 and 603, the HIGH and LOW ADI RATE Switch settings provide an I-load dependent full scale deflection value for all axes, on all supported ADI's. In Major Modes 305 and 603 only, the MED (Medium) ADI RATE Switch setting

displays the Time to Heading Alignment Cone (HAC) Intercept, Altitude Error, and "Crosstrack Error" where "Crosstrack Error" can be either Heading Error, Radial Error, or y-position Error depending on the phase of TAEM (to be described later). The HAC Initiation Predictor, Altitude Error, and "Crosstrack Error" indicators provide the Crew with useful cues for anticipation of the Heading Alignment Cone (HAC) and provide insight to vehicle energy during TAEM.

Table 8 - 11 provides a summary of the ADI RATE pointer functions in all major modes in which the ADI is supported, for all axes. Note that HIGH RATE, MED RATE, LOW RATE, HIGH (RATE) in MM 305/603, and LOW (RATE) in MM 305/603 are I-Loaded ADI Error Rate full scale deflections in the onboard flight software. Items marked N/A in Table 8-III.A. indicate that the rate pointers are nominally stowed out of view until a new set of conditions are satisfied in the software.

The following paragraphs define the variables used in Table 8-III.A. and correspond to the variable name used in the onboard flight software:

Time to HAC Intercept (TGO) in seconds is displayed in the roll rate pointer field. The Time to HAC Intercept pointer is initially displayed in view at the left end of the scale for a right HAC or at the right end of the scale for a left HAC. Once the computed time to HAC Intercept is less than full scale I-loaded deflection (10 seconds based on current I-Load), the pointer moves to the other end of the scale. For a right turn onto the HAC, the pointer moves from the right end to the left end of the scale. The pointer moving along the scale provides an anticipatory cue for the initial bank onto the HAC. The ADI RATE scale used to display Time to HAC Intercept, corresponds directly to the horizontal scale located on the Horizontal Situation Display (SPEC 50) during the same phase.

Altitude Err (H_{ERR}) is the guidance computed Altitude Error in feet with respect to a spiral HAC. The magnitude of the full scale deflection for H_{ERR} is dependent on the guidance phase. The scale ranges from -5000 feet to +5000 feet from MM 305/603 initialization until the prefinal phase. At the prefinal phase, the altitude error scale changes from -1000 feet to +1000 feet. Both the 5000 ft. and 1000 ft. full scale deflections are I-Load dependent. The Altitude Error scale is zeroed about the center-most tickmark and is displayed in the "fly-to" sense. The ADI RATE scale used to display the H_{ERR} , corresponds directly to the vertical scale located on the Horizontal Situation Display (SPEC 50) during the same phases.

"Crosstrack Error" is broken down into three distinct subphases, which are all displayed in the yaw rate pointer field under specific conditions: In the S-Turn or Acquisition phase, the Heading Error (DPSAC) is displayed in degrees from HAC tangency (intercept). The magnitude of full scale deflection is I-Load dependent and is currently set equal to +/- 5 degrees, with the scale centered at zero. In the Heading Alignment phase, the yaw rate pointer displays the Radial Error (RERRC) from the spiral HAC. The Radial Error has an I-Load dependent full scale deflection of +/- 5000 feet and is centered about zero. During the prefinal phase and approach and landing guidance, the yaw rate pointer displays the Y-Position Error (Y) in runway coordinates. The Y-Position Error has an I-Loaded full scale deflection of +/- 1000 feet and is centered about zero. When an I-Loaded altitude (H_{BLANK}) is reached (2000 feet), the ADI pointers are blanked until an I-Loaded time delay expires, at which time the pointers revert back to body rates. Body rates are also displayed when $WOWLON=1$. The pointer scale for all crosstrack errors is zeroed at the center-most tickmark and is displayed in the "fly-to" sense. The ADI Rate scale used to display "Crosstrack Error", corresponds directly to the horizontal scale located on the Horizontal Situation Display (SPEC 50) during the same phases. Other onboard flight software variables used in Table 8 - 11 that were not defined above, are defined here for information purposes:

IPHASE is the TAEM Guidance phase indicator. TAEM Guidance is broken down into several subphases defined by a number; IPHASE = 0 is the S-Turn phase, IPHASE - 1 is the Acquisition phase, IPHASE = 2 is the Heading Alignment phase, and IPHASE = 3 is the Prefinal phase.

The DISPLAY OPTION variable determines whether the MED RATE Switch provides crew energy management cues during TAEM (DISPLAY OPTION = ON) or whether the rate pointers provide nominal rate information (DISPLAY OPTIONS = OFF).

TG_END is defined as the TAEM Guidance termination flag (discrete).

ALT_WHEELS is defined as the altitude of the main landing gear in feet.

H_BLANK is defined as an I-Loaded altitude (2000 ft).

WOWLON is defined as Weight On Wheels Longitudinal and indicates that both the main landing gear and nose landing gear have touched down on the runway.

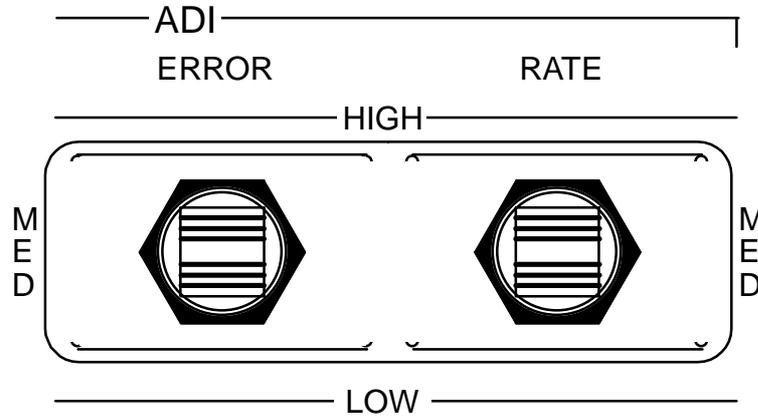


Figure 8 - 47 ADI Rate & Error Switches

Table 8 - 11 ADI Rate & Error Switch Definitions (OI-32 and subs)

| GNC Major Mode | Switch Position | Attitude Rate Full Scale | | | Attitude Error Full Scale | | | |
|--|---------------------------|--------------------------|----------------------|-----------------------------|---------------------------|-------------|-------------|------------------------------|
| | | Roll | Pitch | Yaw | Roll | Pitch | Yaw | |
| Ascent (101-106, 601) On-Orbit (801) Transition (301-303) | High | Deg/s 10 | Deg/s 10 | Deg/s 10 | Deg 10 | Deg 10 | Deg 10 | |
| | Med | 5 | 5 | 5 | 5 | 5 | 5 | |
| | Low | 1 | 1 | 1 | 1 | 1 | 1 | |
| On-Orbit (201, 202) | High | Deg/s 5 | Deg/s 5 | Deg/s 5 | Deg 10 | Deg 10 | Deg 10 | |
| | Med | 1 | 1 | 1 | 5 | 5 | 5 | |
| | Low | 0.2 | 0.2 | 0.2 | 1 | 1 | 1 | |
| Entry (304, 602*) * α transition & α recovery | High | Deg/s 10 | Deg/s 10 | Deg/s 10 | Deg 25 | Deg 5 | Jets 2.5 | |
| | Med | 5 | 5 | 5 | 25 | 2 | 2.5 | |
| | Low | 1 | 1 | 1 | 10 | 1 | 2.5 | |
| MM 602* *Nz Hold | High | Deg/s 10 | Deg/s 10 | Deg/s 10 | Deg 25 | G's 1.25 | Jets 2.5 | |
| | Med | 5 | 5 | 5 | 25 | 1.25 | 2.5 | |
| | Low | 1 | 1 | 1 | 10 | 0.5 | 2.5 | |
| TAEM (MM 305, 603) WOWLON = 0 | High | Deg/s 5 | Deg/s 5 | Deg/s 5 | Deg 25 | G's 1.25 | Jets 2.5 | |
| | Med | 5 | 5 | 5 | 25 | 1.25 | 2.5 | |
| | Low | 5 | 5 | 5 | 10 | 0.5 | 2.5 | |
| HAC Phases (MM 305, 603) | High | 5 Deg/s | 5 Deg/s | 5 Deg/s | | | | |
| | Low | 5 Deg/s | 5 Deg/s | 5 Deg/s | | | | |
| | S Turn & Acquisition | Med | Time to HAC (10 sec) | Altitude Err (\pm 5K ft) | | | | Heading Err (\pm 5 Deg) |
| | Heading Alignment | Med | N/A - Blanked | Altitude Err (\pm 5K ft) | | | | Radial Err (\pm 5 Deg) |
| | Prefinal | Med | N/A - Blanked | Altitude Err (\pm 1K ft) | | | | T-Posn Err (\pm 2.5 K ft) |
| | Alt_Wheels \geq H_Blank | Med | N/A - Blanked | Altitude Err (\pm 1K ft) | | | | T-Posn Err (\pm 2.5 K ft) |
| | Alt_Wheels > H_Blank | Med | N/A - Blanked | N/A - Blanked | | | | N/A - Blanked |
| Alt_Wheels < H_Blank | Med | 5 Deg/s | 5 Deg/s | 5 Deg/s | | | | |
| WOWLON = 1 Rollout (MM 305, 603) | High | Deg/s 5 | Deg/s 5 | Deg/s 5 | Deg 20 | Deg 10 | Deg 2.5 | |
| | Med | 5 | 5 | 5 | 5 | 5 | 2.5 | |
| | Low | 5 | 5 | 5 | 1 | 1 | 2.5 | |

The ADI Error switch determines the full deflection values of the roll, pitch, and yaw error needles. Each switch has three positions: HIGH, MED, and LOW. The error needle full scale definitions vary with the axis and mission phase, as well as with the ADI Error switch position.

When full scale error is 1.25 gs, label on error scale actually says 1.2g due to space limitations and truncation.

Rate scales are labeled for all axes and are in deg/sec unless indicated by a suffix. For example, "5K" indicates the rate scale now serves purpose of being a lateral deviation indicator with max scale deflection of 5000 feet. Only the pitch error scale is labeled and is assumed to denote degrees of error except when a suffix is indicated; i.e., 1.2 g.

The yaw error needle has the same meaning throughout MM's 304 and 305: estimated sideslip (β). For $q\text{-bar} < 20$ psf, the sideslip angle is INRTL β ; i.e., the angle between the +X body axis and the Earth relative velocity. For $q\text{-bar} \geq 20$ psf, estimated beta is based on N_Y and yaw jet effectiveness. Full scale on the display is equivalent to the yaw produced by the side force of 2.5 yaw jets.

8.2.8.2 ADI Sense

This is a two-position toggle switch (Figure 8 - 43) which allows the crew to establish the aft ADI attitude display as either a -X or a -Z control axis reference (the forward ADI's are always referenced to +X control axis). The SENSE switch also determines the control mode of the aft RHC and THC as -X or -Z, so the controllers and ADI display are compatible. The three sets of Orbiter control axes are shown in Figure 8 - 49. A control axis set is simply a right-hand orthogonal coordinate system with respect to which rotational and translational maneuvers are performed.

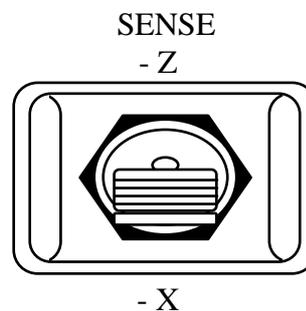


Figure 8 - 48 Sense Switch

As shown in Figure 8 - 49, the -X control set is rotated 180° in roll, 180° in pitch, and 0° in yaw with respect to the +X set. The -X system allows the crewmember to perform payload operations and maneuvers about control axes suited to the task, with the ADI in agreement. The -Z control set is rotated 180° in roll and 90° in pitch with respect to the +X set, lending itself to rendezvous and other overhead window operations.

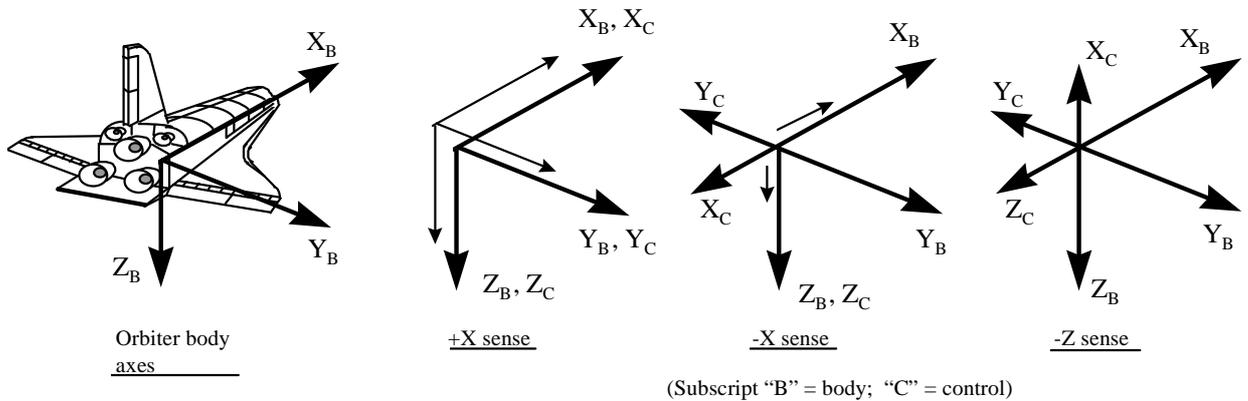


Figure 8 - 49 +X, -X, and -Z Sense

To assist in clarifying what the SENSE switch does, Figure 8 - 50 Comparison of Fwd/Aft ADI Attitude Indications pictures a simplified comparison of forward and aft ADI displays. This example assumes that the ATTITUDE switches are in the same position; which position does not matter.

Another way to think about what the sense switch does would be to imagine the controllers, ADI, and DAP panel physically mounted on the overhead windows with the sense switch in -Z, or on the aft bulkhead windows in -X. This would have the effect of swapping reference frames to the most convenient for flying the vehicle. For instance, pushing the THC in would result in the pilot translating "forward". "Forward", however, means different things depending on which window the pilot is looking out of to fly. Looking out the overhead windows, "forward" is in the Orbiter -Z direction. Looking out the aft windows, "forward" is in the Orbiter -X direction. Left/Right, Up/Down, and Back would be treated in a similar fashion. Rotational inputs with the RHC are also processed based on the sense switch, with the effect being Roll and Yaw will be about the Orbiter Z and X axes, respectively, when the sense switch is in -Z, and about the X and Z axes in -X. The aft DAP panel will also reflect the sense switch, with roll and yaw modes swapping as the sense switch changes from -X to -Z. The sense switch has no effect on the forward cockpit.

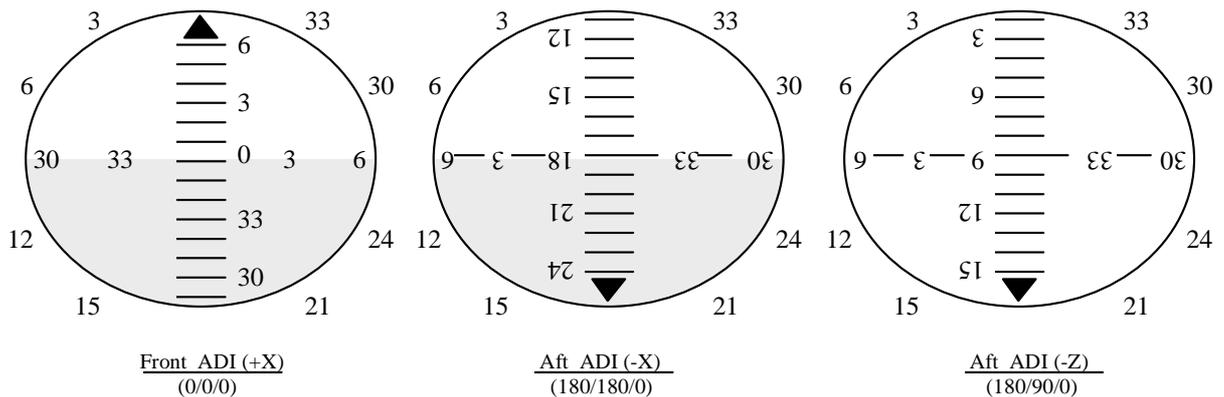


Figure 8 - 50 Comparison of Fwd/Aft ADI Attitude Indications

In Figure 8 - 50, the front ADI reads 0/0/0. The corresponding aft ADI display is shown for the -X and -Z positions of the SENSE switch. The -X sense attitude reads 180/180/0; -Z reads 180/90/0. Changing the position of the SENSE switch will change the aft ADI attitude regardless of the position of the aft ADI ATTITUDE switch. The SENSE switch simply determines which of the two aft control axis sets will be related to the selected REF frame by the aft ADI ball.

The aft ADI rates and errors also reflect the position of the SENSE switch. Table 8 - 12 summarizes the differences in rate and error needle deflections in the aft ADI compared with the forward ADI.

Confirmation of the relation-ships in the table, and also confirmation that the aft ADI error and rate deflections are still fly-to indications, can be made with a little pencil geometry: using a pencil as the Shuttle, execute +roll, +pitch, and +yaw rotations in the +X control axis set, and observe the corresponding rotations in the -X and -Z systems.

Table 8 - 12 Fwd & Aft ADI Error & Rate Needle Differences

| Sense switch | Roll | Pitch | Yaw |
|--------------|---------------------|----------------------|---------------------|
| -X | Reverse of FWD roll | Reverse of FWD pitch | Same |
| -Z | Reverse of FWD yaw | Reverse of FWD pitch | Reverse of FWD roll |

8.2.8.3 ADI Attitude Reference

The attitude reference PBI is used to define the origin of the ADI when the ADI Attitude switch is in REF. Depressing the PBI will define the current attitude (as defined by QBI) as zero, reposition the ADI attitude ball in (0, 0, 0), and zero the Roll, Pitch, and Yaw attitude error needles. All subsequent errors and attitude will be based on the last time the REF PBI was depressed.

Only one reference attitude is supported, so only one station can be used at a time to "REF the ball". Although once performed, all ADI's will reflect the positioned where the PBI was depressed, and in the case of the aft station, where the sense switch was positioned. For example, if the sense switch is in the -X position when the aft station ATT REF PBI is depressed, the aft ADI will reflect a (0, 0, 0) attitude while the forward ADI's will reflect a (0, 0, 180) attitude as shown in Figure 8 - 50. If the aft station is in the -Z sense, depressing the PBI will result in the aft station reflecting (0, 0, 0) while the forward station will reflect (180, 90, 0).

8.2.8.4 HSI Select Source

The HSI SELECT SOURCE switches (Figure 8 - 51) interface with the area NAV software module, which is dedicated to HSI processing. When the left switch is in the TACAN, GPS, or MLS position, Area NAV will use the data from the hardware unit specified by the right switch. The MLS provides range, azimuth, and elevation, which is sufficient information for Area NAV to compute Shuttle position with respect to the runway. TACAN provides only range and azimuth; therefore, Area NAV uses BARO altitude from the air data system to complete its processing when the SOURCE switch is in the TACAN position. Air data BARO altitude is used because it is not dependent upon NAV and guidance processing. Both the BARO altitude and the MLS elevation measurements have backup (substitute) functions in the event they are declared invalid. Table 8 - 13 HSI Data Sources shows the HSI data sources.

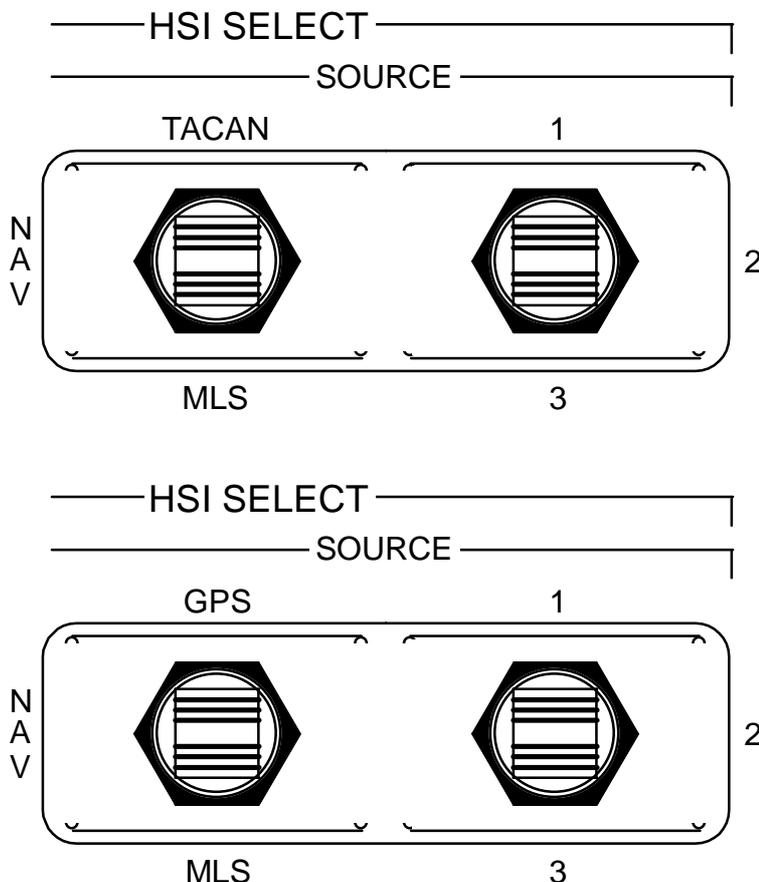


Figure 8 - 51 HSI Select Source Switches

BARO Substitute Function - The Area NAV substitute for BARO altitude depends on the selected position of the MODE switch:

ENTRY - Uses average BARO altitude for entry phase (I-load)

TAEM - Uses average BARO altitude for TAEM phase (I-load)

A/L - Uses estimated BARO altitude based on NAV-derived altitude above runway ($H_{BARO} = H_{WHEELS\ ABOVE\ RUNWAY} + H_{RUNWAY\ ABOVE\ REF\ ELLIPSOID} - H_{MEAN\ SEA\ LEVEL\ ABOVE\ ELLIPSOID}$)

MLS Elevation Substitute Function - If MLS elevation is invalid, Area NAV uses one of two altitude constants:

EL_STEEP = steep GS elevation, used if MLS range shows the Shuttle has not yet crossed the steep/shallow GS intersection.

EL_SHALLOW = shallow GS elevation, used if past the steep/shallow GS intersection.

The TACAN, GPS, or MLS position of the SOURCE switch should be used only when TACAN (GPS) or MLS data are available. TACAN data can be acquired even in Earth orbit, but will be unavailable between 360 and 185K ft, and generally will not be used until $V_{REL} < 9500$ ft/s). The MLS has a range of 20 n. mi. and will normally be selected after the Shuttle is on the HAC.

In the NAV position, the entire HSI display is driven using NAV-derived data (i.e., from the Orbiter state vector). This makes the HSI display dependent on the same sources as NAV software (IMU, selected air data, selected NAVAIDS), but the display will still be independent from guidance targeting parameters

(auto control loop). When the left SOURCE switch (TACAN (GPS), NAV, or MLS) is in the NAV position, the SOURCE switch (1/2/3) is not processed.

Table 8 - 13 HSI Data Sources

| SOURCE Switch Position | X, Y Data Source | Altitude Data Source |
|------------------------|------------------|----------------------|
| TACAN | TACAN | Air Data or NAV |
| NAV | NAV | Nav State |
| MLS | MLS | MLS |
| GPS | GPS | Air Data or NAV |

8.2.8.5 HSI Select Mode

The HSI SELECT Mode switch (Figure 8 - 52) allows the crew to specify the phase (entry, TAEM, or A/L) for which the HSI display will be representative. The switch positions function independently of the actual mission phase in progress, except that if the switch is left in the ENTRY position, the software will automatically cycle the display to TAEM at MM 305 and to A/L at a specified altitude. Like the Source switches, the right and left Mode switches apply only to their associated HSI. Each position of the Mode switch calls for a different set of definitions for the HSI display parameters, as summarized in Table 8 - 14. Note that the parameter changes from one phase to the next are dictated by the different geometric profiles for the phases.

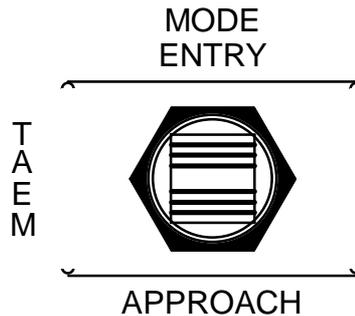


Figure 8 - 52 HSI SELECT Mode Switch

A brief review of the flight path geometry should help to clarify the definitions in Table 8 - 14. Refer to Figure 8 - 53 and Figure 8 - 54. Figure 8 - 53 shows the top view of the Orbiter entering the TAEM region. The heading alignment circles (HAC's) are predefined in the site lookup table for the nominal and minimum energy cases for each runway. Way point 1 (WP-1) is the point of tangency with the HAC (HAC intercept) and varies in location with the approach. Way point 2 (WP-2) is a fixed point on the runway about 1500 ft from runway threshold (actually the intersection of the shallow GS with runway centerline), and is the targeted touchdown point. The nominal entry point (NEP) is the intersection of the two nominal-energy HAC's and coincides with WP-1 for a straight-in approach. The minimum entry point (MEP) is the intersection of the two HAC's for the minimum energy case. The MEP HAC's are shown smaller but in reality are the same size as the normal HAC's (20K ft radius; modifiable by I-load). Figure 8 - 54 shows the side view of TAEM and A/L, with the various GS. The transition from TAEM to A/L occurs approximately at WP-1 (H = 10K ft, EAS = 290 ft/s, Crossrange ≤ 1000 ft, Rng = 5 n. mi.).

Table 8 - 14 HSI Function

| HSI Mode | Bearing () indicates bearing pointer label | Range | Course Deviation (CDI) | Glide Slope Deviation (GSI) | Compass Card (Heading) |
|----------|--|---|------------------------------|-----------------------------|---|
| Entry | (H) Spherical bearing to NEP HAC intercept | Spherical surface range to runway touchdown point via NEP | Pegged at zero (no CDI flag) | Blanked | Magnetic heading of vehicle velocity vector |

| | | | | | (flight path) |
|----------|---|---|---|---|--|
| TAEM | (H) Bearing to NEP HAC intercept (C) Bearing to HAC center | Horizontal distance to runway touchdown point for selected EP (NEP or MEP) | Deviation from runway centerline (full scale $\pm 10^\circ$) | Error from guidance TAEM reference altitude (full scale ± 5000 ft) | Magnetic heading of X body axis |
| Approach | (R) Bearing to runway touchdown point | Horizontal distance to runway touchdown point | Deviation from runway centerline (full scale $\pm 2.5^\circ$) | Error from guidance A/L reference altitude. (full scale ± 1000 ft) Not computed below 1500 ft (GS flag displayed) | Magnetic heading of X body axis |

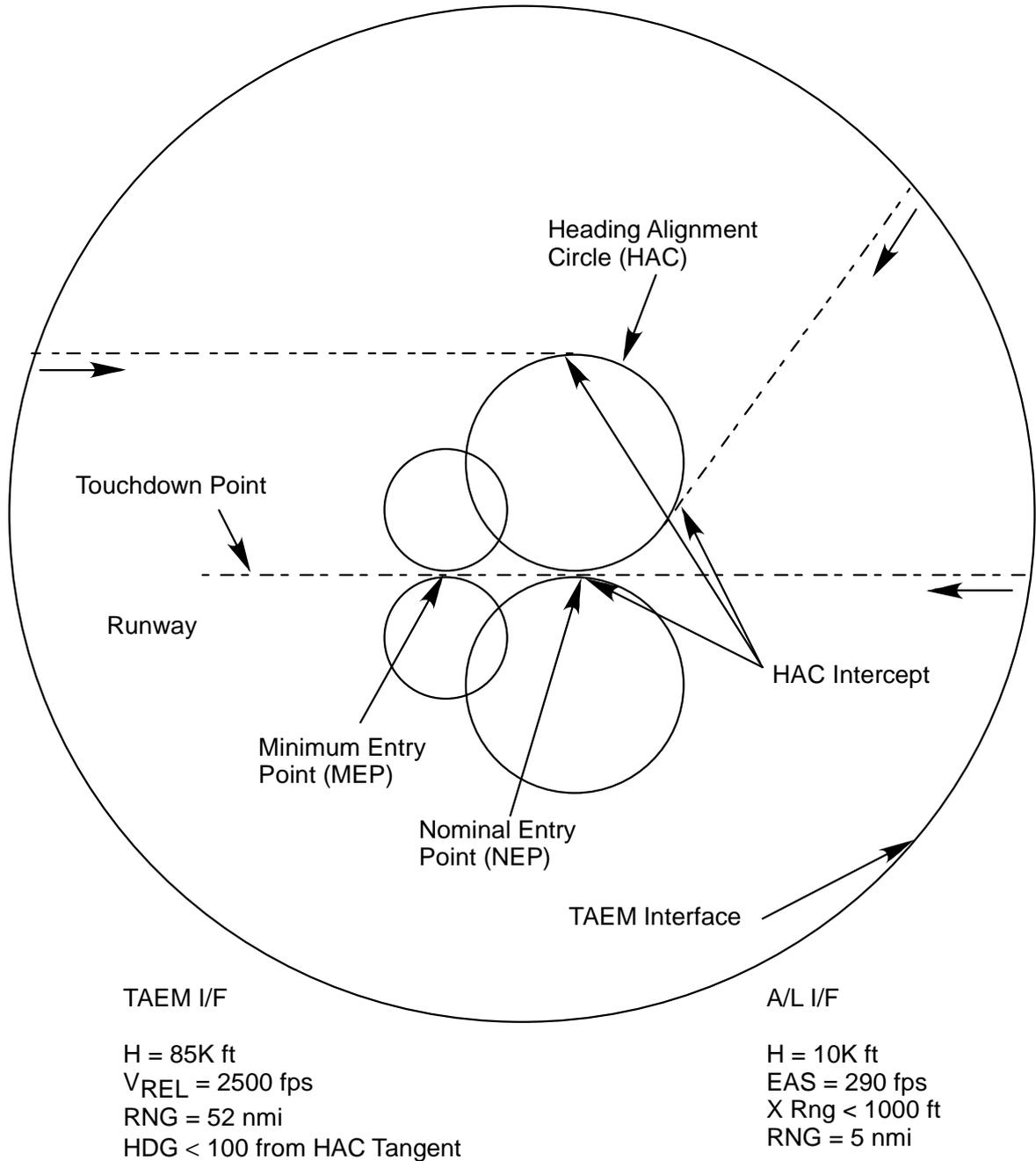


Figure 8 - 53 Flight path geometry - top view

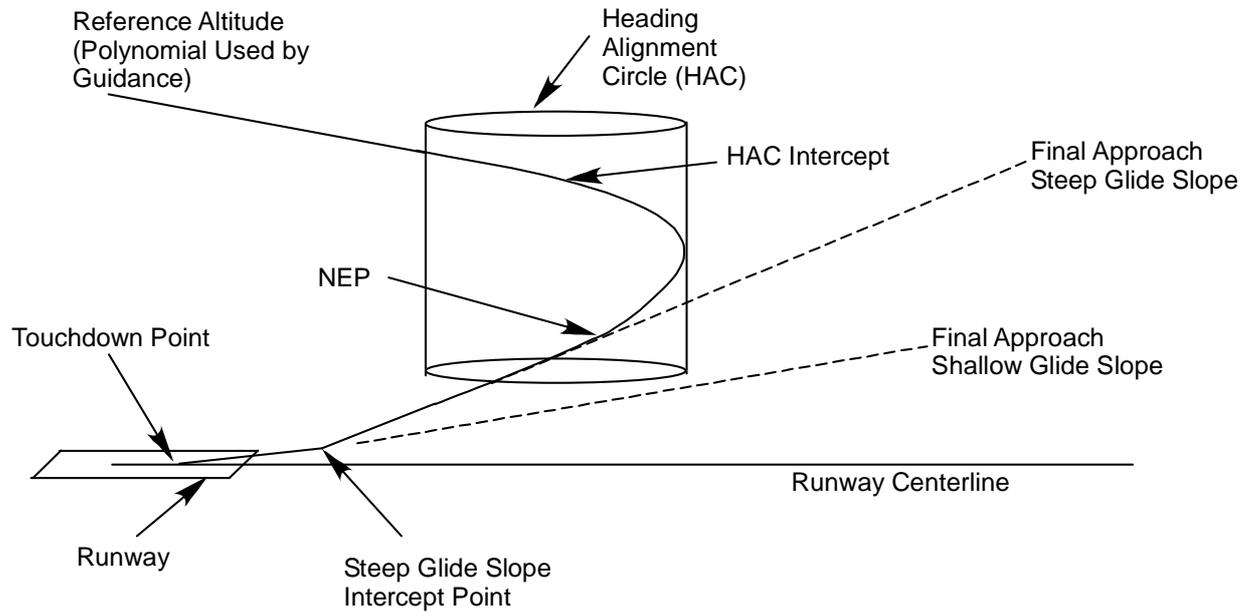


Figure 8 - 54 Flight path geometry - side view

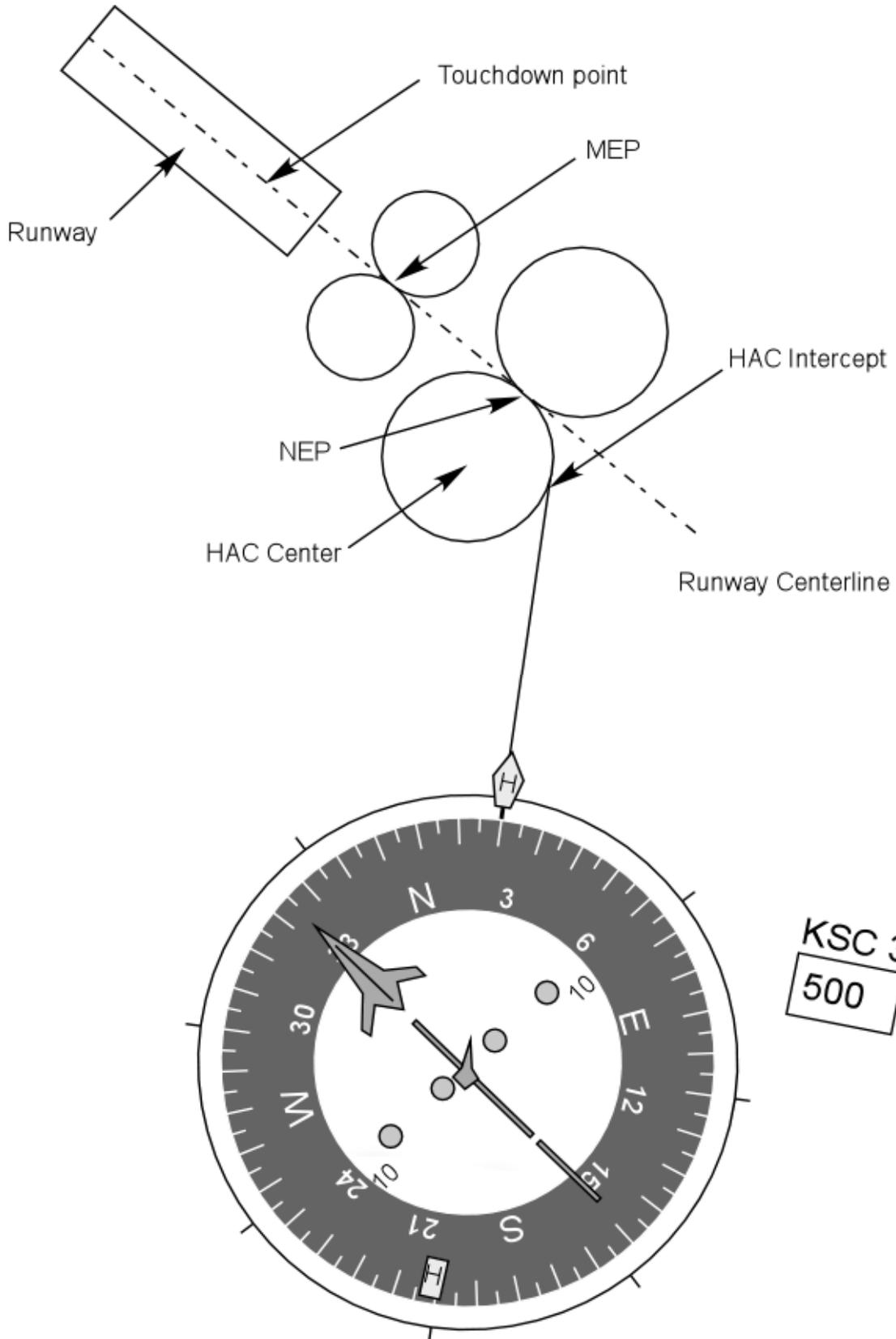


Figure 8 - 55 Entry Display Mode

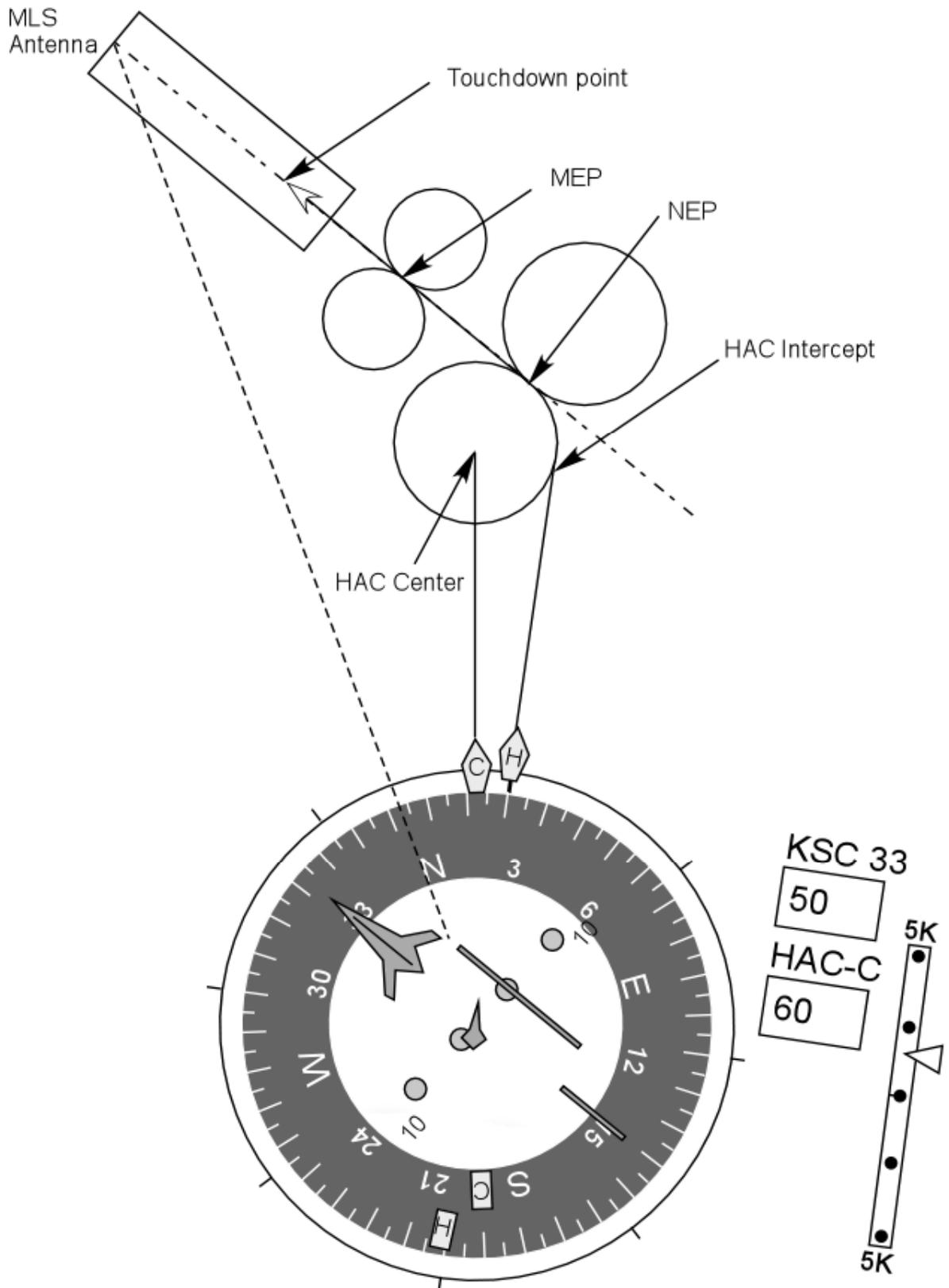


Figure 8 - 56 TAEM Display Mode

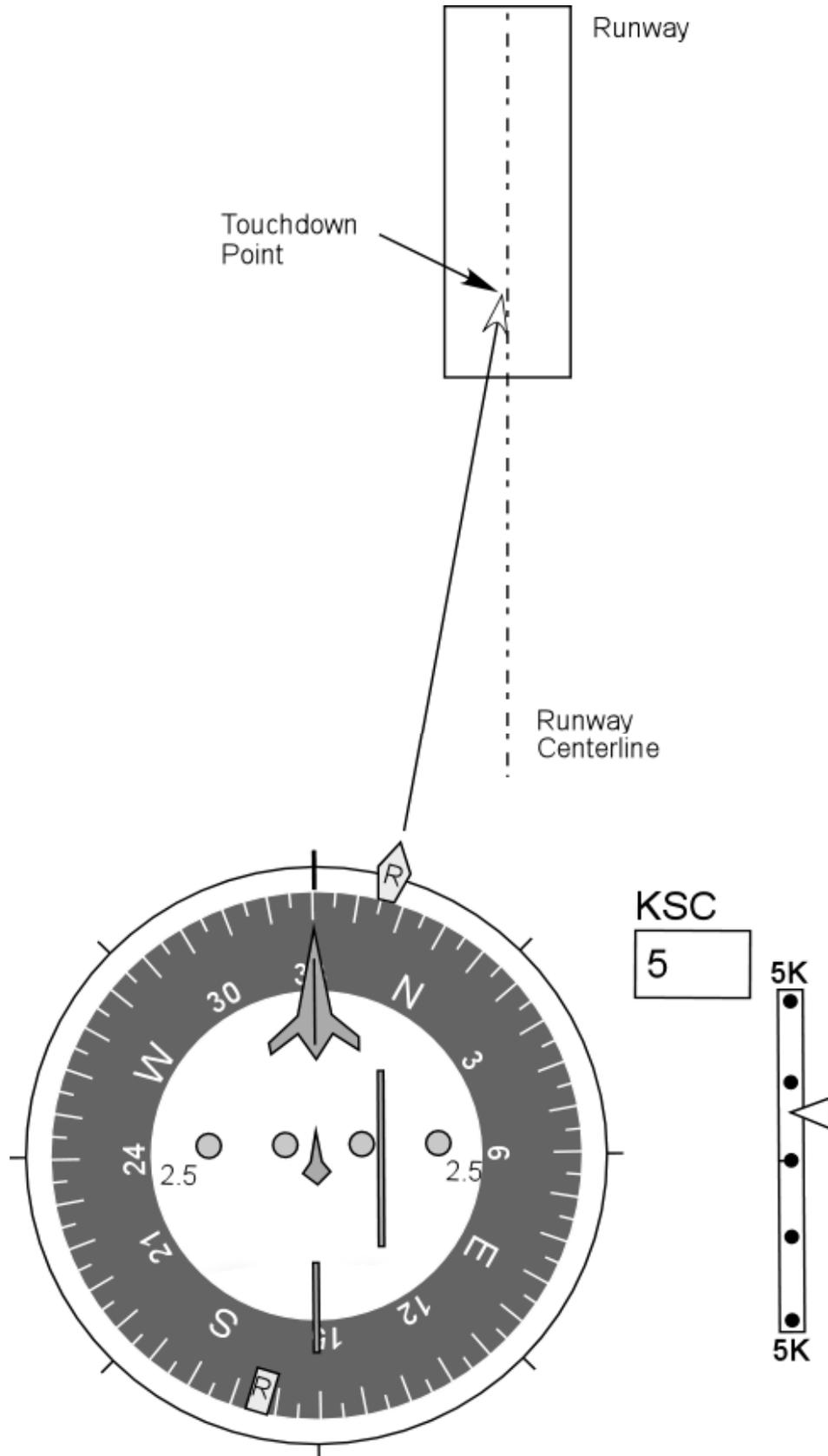


Figure 8 - 57 A/L Display Mode

8.2.8.6 Air Data

The Air Data Source switch allows the CDR and PLT to select the source of data driving the AMI and AVVI. In the NAV position, the tapes are driven by NAV selected parameters. In the Left or Right positions, the tapes are driven by calibrated pressure data from the left and right air data probes, respectively.

The left and right AIR DATA switches (SSSH Drawing 9.1) are three-position toggle switches which allow the crew to determine the data source for the left and right AMI and AVVI displays.

LEFT, RIGHT - With the AIR DATA switch in the LEFT or RIGHT position, ALPHA, M/VEL, and EAS on the adjacent display will be driven by selected data from the left or right air data probe assembly, assuming the probe is deployed and the data are not missing. The air data probes will normally be deployed by the crew at Mach 5.

With the AIR DATA switch (Figure 8 - 58) in the LEFT (or RIGHT) position, the ALT and ALT RATE will be based on left (or right) probe air data, assuming the probes are deployed and the data are not missing. ALT ACCEL will not be affected by the AIR DATA switch as their source is NAV-derived.

NAV - This position of the AIR DATA switch determines that ALPHA, MACH, and EAS on the adjacent AMI will be the same parameters being sent to guidance, flight control, NAV and other software users. ALPHA, MACH, and EAS will be either the center ADTA SOP parameters (selected from left and right probe air data) or NAV-derived parameters, depending on probe status, Mach number, air data validity, and the active ADTA to guidance and control (G&C) item entry on the HORIZ SIT specialist function (SPEC).

With the AIR DATA switch in the NAV position, ALT ACCEL, ALT RATE, and ALT are always NAV-derived. Unlike the AMI display, the AVVI does not depend on the HORIZ SIT SPEC ADTA to G&C item entries; the AVVI never reflects probe air data when the AIR DATA switch is in the NAV position.

There is no switch default logic available in the BFS for the Air Data Source switch, and only the CDR's switch position is monitored. The BFS logic can be described as having two states, in NAV or not in NAV. In the NAV position, selected NAV parameters will drive the AMI and AVVI, and if NAV is not selected, then raw pressure data is used from any available probe/ADTA as determined by the BFS AD HIP (for a more detailed description of the availability logic, reference GNC Systems Brief 4, Advanced Air Data Transducer). In other words, there is no left and right selection in the BFS, only NAV and AD HIP selected values. A consequence of this logic is that a loss of electrical bus CNTL BC3 will leave the switch in an indeterminant state (no contacts selected). This will cause AMI and AVVI data to be driven by BFS NAVDAD until a probe is deployed at M = 5, when air data will be available but not accurate until M = 3.5 when the calibration of the ADTA data becomes valid.

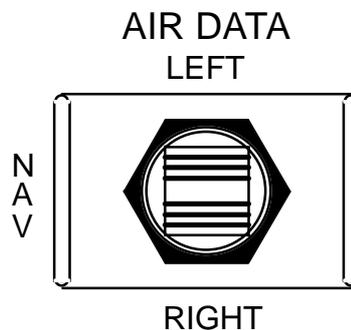


Figure 8 - 58 Air Data Source Switch

8.2.8.7 Radar Altimeter

The Radar Altimeter (RA or RAD ALT) switch selects the RA source (RA 1 or 2) for the AVVI below 5000 feet. A switch is available at both the CDR and PLT stations to drive the respective AVVI. When RAD

ALT data are available, the AVVI altitude tape will display the value and indicate valid RAD ALT data with an “R” next to the digital value.

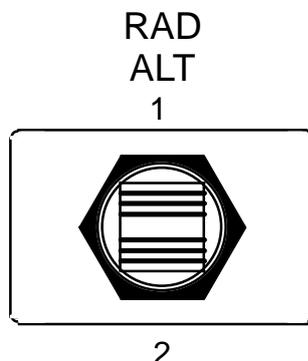


Figure 8 - 59 Radar Altimeter Switch

8.2.8.8 Default Switch State

The switch-processing software in PASS examines the toggle switch inputs for consistency. If a switch has more than one position selected or no position selected, then a default state is assumed for that switch. The default state becomes the true state which is sent to the dedicated display processing software. The default state allows processing to continue even though a switch is failed, using an optimum switch state. The default state also results when the switch data is missing (COMM fault (CF)). The current switch default states are shown in Table 8 - 15. The software annunciates a switch default condition via PASS systems management (SM) ALERT light and tone and message DISPLAY SW L(R,A). The default state of each switch is indicated by a green dot next to the position label. This is often referred to as the “green dot” position.

Table 8 - 15 Switch Default Postions

| Switch | Positions | Default |
|---------------------|-------------------|---------|
| HSI Mode | Entry | X |
| | TAEM | |
| | Approach | |
| HSI Source | TACAN (GPS) | |
| | NAV | X |
| | MLS | |
| | 1 2 3 | X |
| ADI Rate | Low | |
| | Med | X |
| | High | |
| ADI Error | Low | |
| | Med | X |
| | High | |
| Air Data Select | Left | |
| | NAV | X |
| | Right | |
| ADI Attitude Select | INRTL | |
| | LVLH | X |
| | REF | |
| CDR RA Select | 1 | |
| | 2 | X |
| PLT RA Select | 1 | X |
| | 2 | |

The BFS only recognizes the CDR's ADI Attitude Select, HSI Source Select, and Air Data Select switches. The Air Data Select only monitors the NAV contact, and therefore defaults to whatever state that contact defaults to. When "1", BFS drive the AMI and AVVI with NAV derived air data. When "0", BFS drives the tapes with any available ADTA data. For the ADI Attitude Select, only the REF and INRTL contacts are monitored, and special logic defaults to one of the states per Table 8 - 16. The HSI Source Select defaults to NAV if the switch output is 0 (as with CNTL AB2 failed), and the last state for commfaults (MDM FF1). In the BFS, HSI Source output of 0 = NAV, 1 = TACAN/GPS, and 2 = MLS. The MLS position results in no HSI data and the BRG, CDI, and GSI flags displayed.

Table 8 - 16 BFS ADI Attitude Switch States

| REF | INRTL | Output State |
|-----|-------|--------------|
| 0 | 0 | LVLH |
| 1 | 0 | REF |
| 0 | 1 | INRTL |
| 1 | 1 | INRTL |

8.2.9 Flight Phase Details

The following sections detail the MEDS Composite Display for each software major mode. Table 8 - 17 and Table 8 - 18 summarize the availability of the different components in PASS and BFS. The figures provided in this section are not to scale, and are provided only as a representation of the look-and-feel of the displays. For precise information of the design of the graphics, the interested reader should consult the MEDS software requirements listed at the end of this systems brief.

Table 8 - 17 PASS Display Component Availability

| | | OPS 9 | OPS 1 | | | | | | OPS 2 | | OPS 8 | OPS 3 | | | | | OPS 6 | | |
|-------|--------------------------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|
| | | MM 901 | MM 101 | MM 102 | MM 103 | MM 104 | MM 105 | MM 106 | MM 201 | MM 202 | MM 801 | MM 301 | MM 302 | MM 303 | MM 304 | MM 305 | MM 601 | MM 602 | MM 603 |
| ADI | BALL | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | ERROR NEEDLES & SCALE | X [5] | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | RATE NEEDLES & SCALE | X [5] | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | DIGITAL R, P, Y | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | THETA LIMIT BRACKET | - | - | - | - | - | - | - | - | - | - | - | - | - | - | X | - | - | X |
| | OFF FLAG | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | ADI ATT SW | - | X | X | X | X | X | X | - | - | - | X | X | X | - | - | X | - | - |
| HSI | COMPASS CARD | X | X | X | X | X | - | - | - | - | - | - | - | X | X | X | X | X | |
| | COURSE ARROW | X | X | X | X | X | - | - | - | - | - | - | - | X | X | - | X | X | |
| | CDI | X | X | X | X [2] | X [2] | - | - | - | - | - | - | - | X | X | - | X | X | |
| | CDI FLAG | X | X | X | X [2] | X [2] | - | - | - | - | - | - | - | X | X | - | X | X | |
| | GSI | X | - | - | - | - | - | - | - | - | - | - | - | - | X | - | - | X | |
| | GSI FLAG | X | - | - | - | - | - | - | - | - | - | - | - | - | X | - | - | X | |
| | HAC POINTER (PRI BRG) | X [7] | - | - | X [1] | X [1] | - | - | - | - | - | - | - | X | X | X [4] | X | X | |
| | HAC CENTER POINTER | - | - | - | - | - | - | - | - | - | - | - | - | - | X | - | - | X | |
| | RUNWAY POINTER (SEC BRG) | X [6] | - | - | - | - | - | - | - | - | - | - | - | X | X | X [3] | X | X | |
| | VREL POINTER | - | - | X | X | - | - | - | - | - | - | - | - | - | - | - | X | - | - |
| | VI POINTER | - | X | X | X | X | - | - | - | - | - | - | - | - | - | - | - | - | - |
| | PRI R/W RANGE | X [8] | - | - | X [1] | - | - | - | - | - | - | - | - | X | X | X | X | X | |
| | BRG FLAG | X | - | - | X [1] | - | - | - | - | - | - | - | - | X | X | X | X | X | |
| | Δ AZ | - | - | - | X [1] | - | - | - | - | - | - | - | - | X | X | X | X | X | |
| | Δ X-TRK | - | - | - | X [1] | - | - | - | - | - | - | - | - | - | - | - | - | - | |
| | Δ INC | - | - | X | X | X | - | - | - | - | - | - | - | - | - | - | - | - | |
| | TGT INC | - | - | X [9] | X [9] | - | - | - | - | - | - | - | - | - | - | - | - | - | |
| X-TRK | - | - | X | X | X | - | - | - | - | - | - | - | - | - | - | - | - | | |
| AMI | V _i | X [8] | - | - | X | - | - | - | - | - | - | - | - | - | - | X [3] | - | - | |
| | V _{REL} | X [8] | - | X | - | - | - | - | - | - | - | - | - | X | X | X [4] | X | X | |
| | VEL DIGITAL | X [8] | - | X | X | - | - | - | - | - | - | - | - | X | X | X | X | X | |
| | α | X [8] | - | X | X | - | - | - | - | - | - | - | - | X | X | X | X | X | |
| | β | - | - | X | X | - | - | - | - | - | - | - | - | - | - | X | - | - | |

| | | | | | | | | | | | | | | | | | | | |
|------------|------------------------|-------|---|---|--------|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | MAX L/D DIAMOND | - | - | - | - | - | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | MAX/MIN α | - | - | - | - | - | - | - | - | - | - | - | - | - | X | X | - | X | X |
| AVI | H | X [8] | - | X | X | X | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | H DOT | X [8] | - | X | X | X | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | H ACCEL | X [8] | - | - | - | - | - | - | - | - | - | - | - | - | X | - | - | X | - |
| G-METER | N _z | X [8] | - | X | X | - | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | N _z DIGITAL | X [8] | - | X | X | - | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | TARGET N _z | - | - | X | X | - | - | - | - | - | - | - | - | - | X | X | X | X | X |
| MAJOR MODE | MAJOR MODE | - | X | X | X | X | X | X | - | - | - | X | X | X | X | X | X | X | X |
| | ABORT INDICATOR | - | - | X | X | X | - | - | - | - | - | - | - | - | X | X | X | X | X |
| FCS CONFIG | DAP AUTO/ CSS | - | X | X | X | - | - | - | - | - | - | - | - | - | - | - | X | - | - |
| | DAP AUTO/ INRTL | - | - | - | - | X | X | X | - | - | - | X | X | X | - | - | - | - | - |
| | DAP P AUTO/ CSS | - | - | - | - | - | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | DAP R/Y AUTO/ CSS | - | - | - | - | - | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | THROT AUTO/ MAN | - | X | X | X [11] | - | - | - | - | - | - | - | - | - | - | - | X | - | - |
| | SB AUTO/ MAN | - | - | - | - | - | - | - | - | - | - | - | - | - | X | X | - | X | X |

- [1] TAL only
- [2] Blanked for TAL
- [3] RTLS Pre PPA
- [4] RTLS Post PPA
- [5] No Scale
- [6] As secondary bearing pointer
- [7] As primary bearing pointer
- [8] Invalid (red box)
- [9] ATO only
- [10] Except contingency abort
- [11] Blanks at MECO Confirmed

Table 8 - 18 BFS Display Component Availability

| | | OPS 1 | | | | | OPS 3 | | | | | OPS 6 | | |
|-----|-----------------------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|--------|
| | | MM 102 | MM 103 | MM 104 | MM 105 | MM 106 | MM 301 | MM 302 | MM 303 | MM 304 | MM 305 | MM 601 | MM 602 | MM 603 |
| ADI | BALL | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | ERROR NEEDLES & SCALE | X | X | X | X | X | X | X | X | X | X | X | X | X |

| | | | | | | | | | | | | | | |
|------|------------------------------|----------------|-------|-------|---|---|---|---|---|---|---|-------|---|---|
| | RATE NEEDLES & SCALE | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | DIGITAL R, P, Y | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | THETA LIMIT BRACKET | - | - | - | - | - | - | - | - | - | - | - | - | - |
| | OFF FLAG | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | ADI ATT SW | X | X | X | X | X | X | X | X | - | - | X | - | - |
| HSI | COMPASS CARD | X | X | X | - | - | - | - | - | X | X | X | X | X |
| | COURSE ARROW | X | X | X | - | - | - | - | - | X | X | - | X | X |
| | CDI | X | X [2] | X [2] | - | - | - | - | - | X | X | - | X | X |
| | CDI FLAG | X | X [2] | X [2] | - | - | - | - | - | X | X | - | X | X |
| | GSI | - | - | - | - | - | - | - | - | - | X | - | - | X |
| | GSI FLAG | - | - | - | - | - | - | - | - | - | X | - | - | X |
| | HAC POINTER (H) (PRI BRG) | - | X [1] | X [1] | - | - | - | - | - | X | X | X [4] | X | X |
| | HAC CENTER (C) POINTER | - | - | - | - | - | - | - | - | - | X | - | - | X |
| | RUNWAY POINTER (R) (SEC BRG) | - | - | - | - | - | - | - | - | X | X | X [3] | X | X |
| | VREL POINTER (E) | X | X | - | - | - | - | - | - | - | - | X | - | - |
| | VI POINTER (I) | X | X | X | - | - | - | - | - | - | - | - | - | - |
| | PRI R/W RANGE | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | BRG FLAG | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | Δ AZ | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | Δ X-TRK | - | - | - | - | - | - | - | - | - | - | - | - | - |
| | Δ INC | X | X | X | - | - | - | - | - | - | - | - | - | - |
| | TGT INC | X [9] | X [9] | X | - | - | - | - | - | - | - | - | - | - |
| | X-TRK | X | X | X | - | - | - | - | - | - | - | - | - | - |
| AMI | V _I | - | - | - | - | - | - | - | - | - | - | - | - | - |
| | V _{REL} | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | VEL DIGITAL | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | α | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | β | X | X | - | - | - | - | - | - | - | - | X | - | - |
| | MAX L/D DIAMOND | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | MAX/MIN α | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | H | - | - | - | - | - | - | - | - | X | X | X | X | X |
| AVVI | H DOT | - | - | - | - | - | - | - | - | X | X | X | X | X |
| | H ACCEL | - | - | - | - | - | - | - | - | X | - | - | X | - |
| | M E T | N ₂ | - | - | - | - | - | - | - | - | X | X | - | X |

| | | | | | | | | | | | | | | |
|------------|------------------------|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | N ₂ DIGITAL | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | TARGET N ₂ | - | - | - | - | - | - | - | - | X | X | - | X | X |
| MAJOR MODE | MAJOR MODE | X | X | X | X | X | X | X | X | X | X | X | X | X |
| | ABORT INDICATOR | X | X | X | - | - | - | - | - | X | X | X | X | X |
| FCS CONFIG | DAP AUTO/ CSS | X | X | - | - | - | - | - | - | - | - | X | - | - |
| | DAP AUTO/ INRTL | - | - | X | X | X | X | X | X | - | - | - | - | - |
| | DAP P AUTO/ CSS | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | DAP R/Y AUTO/ CSS | - | - | - | - | - | - | - | - | X | X | - | X | X |
| | THROT AUTO/ MAN | - | - | - | - | - | - | - | - | - | - | - | - | - |
| | SB AUTO/ MAN | - | - | - | - | - | - | - | - | X | X | - | X | X |

[1] TAL only

[2] Blanked for TAL

[6] As secondary bearing pointer

[7] As primary bearing pointer

[9] ATO only

8.2.9.1 OPS 9

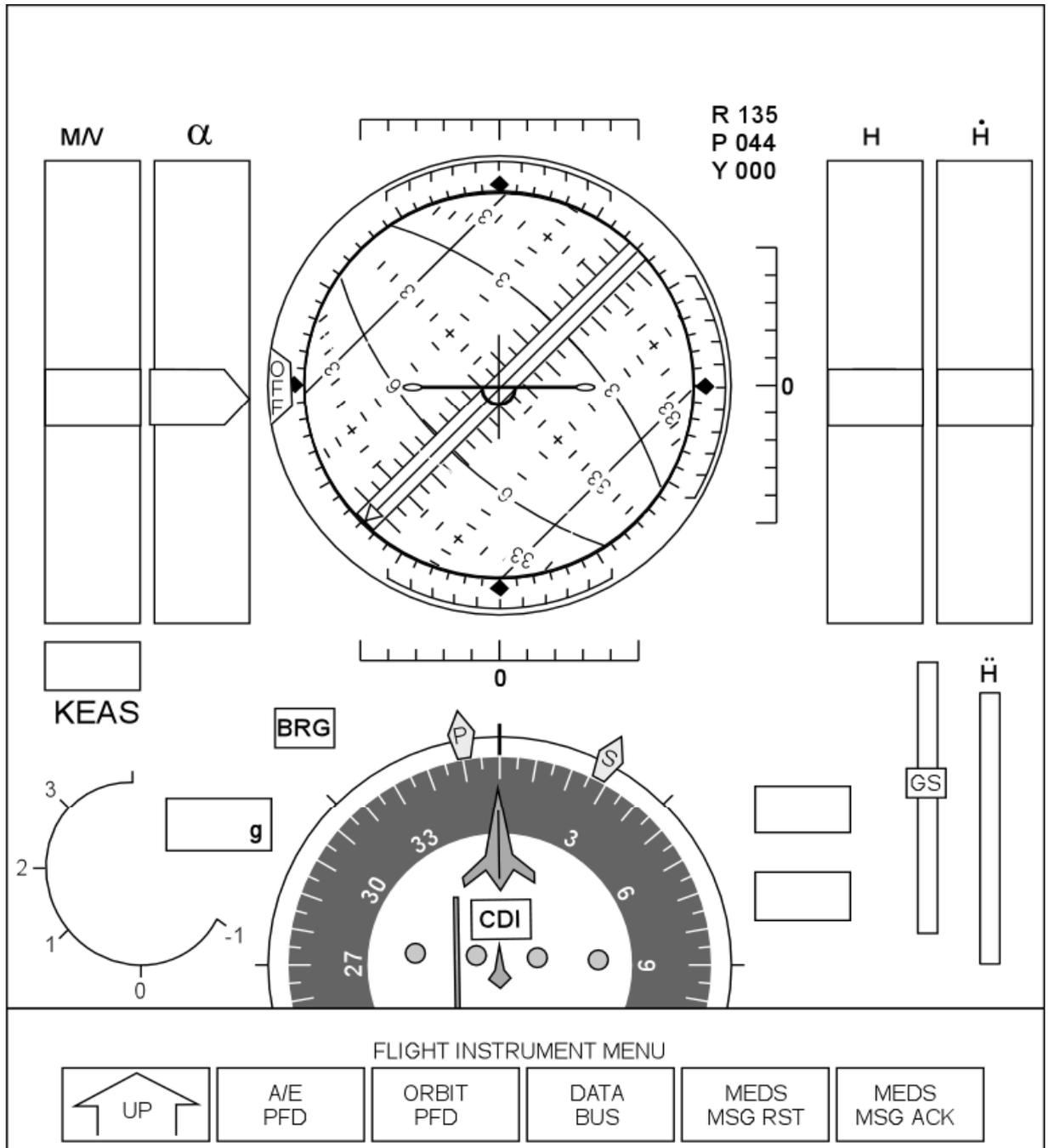


Figure 8 - 60 OPS 9

The OPS 9 display provides no real-time information. The AMI, AVVI, g-meter, and GSI fields are invalid and represented by a red outline box. The ADI and HSI are present, but are not active since there is no ATT PROC or AREA NAV software available. FCS Mode, ADI Attitude switch, and major mode fields are blank.

8.2.9.2 OPS 1

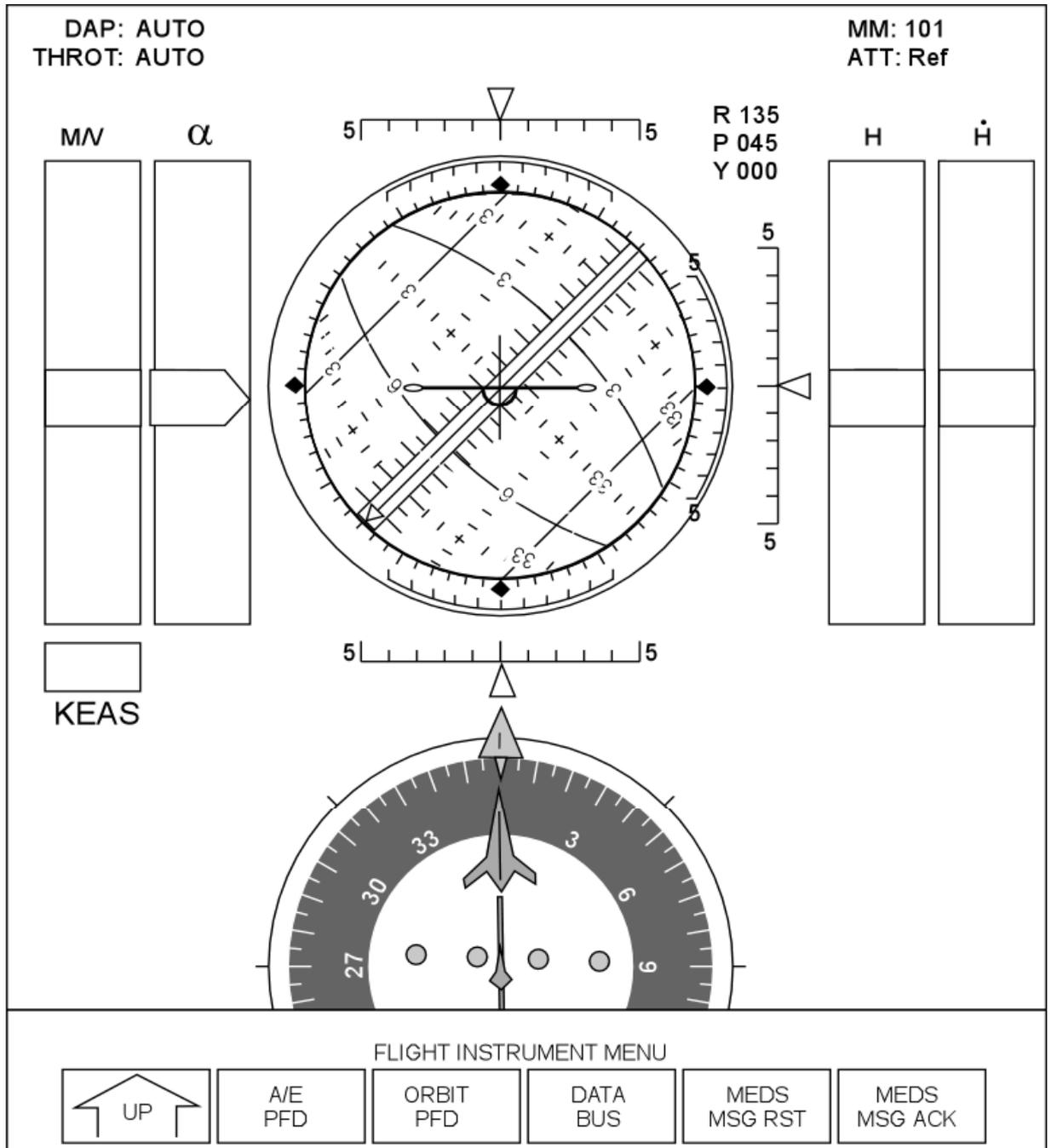


Figure 8 - 61 MM 101

In MM 101, the AMI and AVVI are invalid and represented by red outline boxes. The ADI is fixed in its prelaunch attitude with error needles blanked (stowed). The HSI is not valid until liftoff. FCS Mode, ADI Attitude Switch and major mode are active.

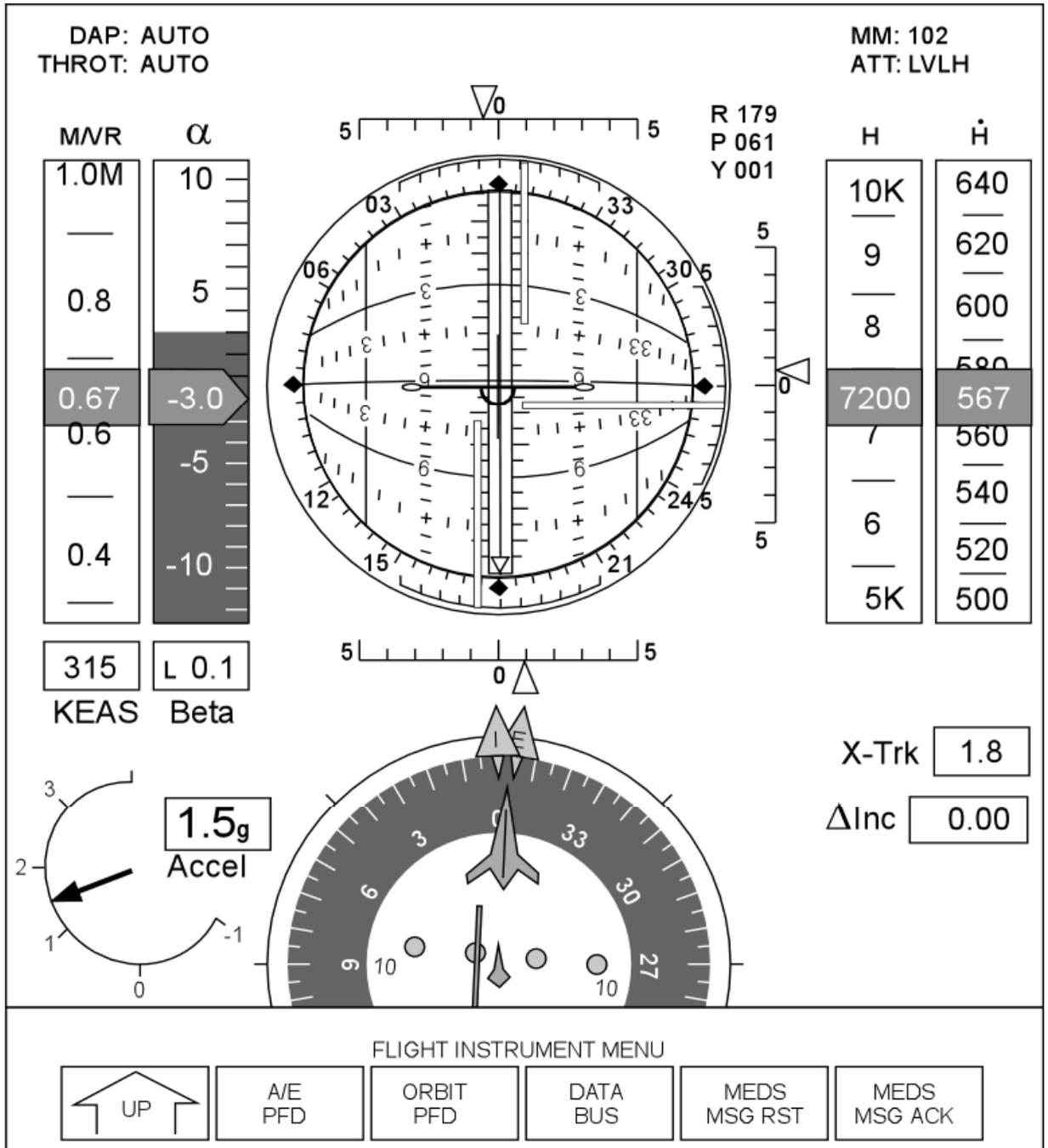


Figure 8 - 62 MM 102

At SRB ignition (MM 102 transition), the AMI, AVVI, g-meter, CDI, and HSI become active. The HSI bearing pointers provide V_1 and V_{REL} . X-Trk and INC digitals are also now driven. ADI error and rate needles are now driven, and the labels reflect the selected ADI ERROR and RATE switch position. The AMI velocity tape is V_{REL} , and digital is KEAS. FCS mode provides Ascent DAP and Throttle selection. If CSS or MAN is selected, a yellow box will highlight the appropriate FCS Mode field. The HSI compass card is relative to inclination, and the N, S, E, and W labels replaced by 0, 180, 90, and 270, respectively. The course arrow indicates the target insertion plane course. The g-meter reflects vehicle X axis acceleration. The ADI errors reflect Ascent DAP TVC trim errors.

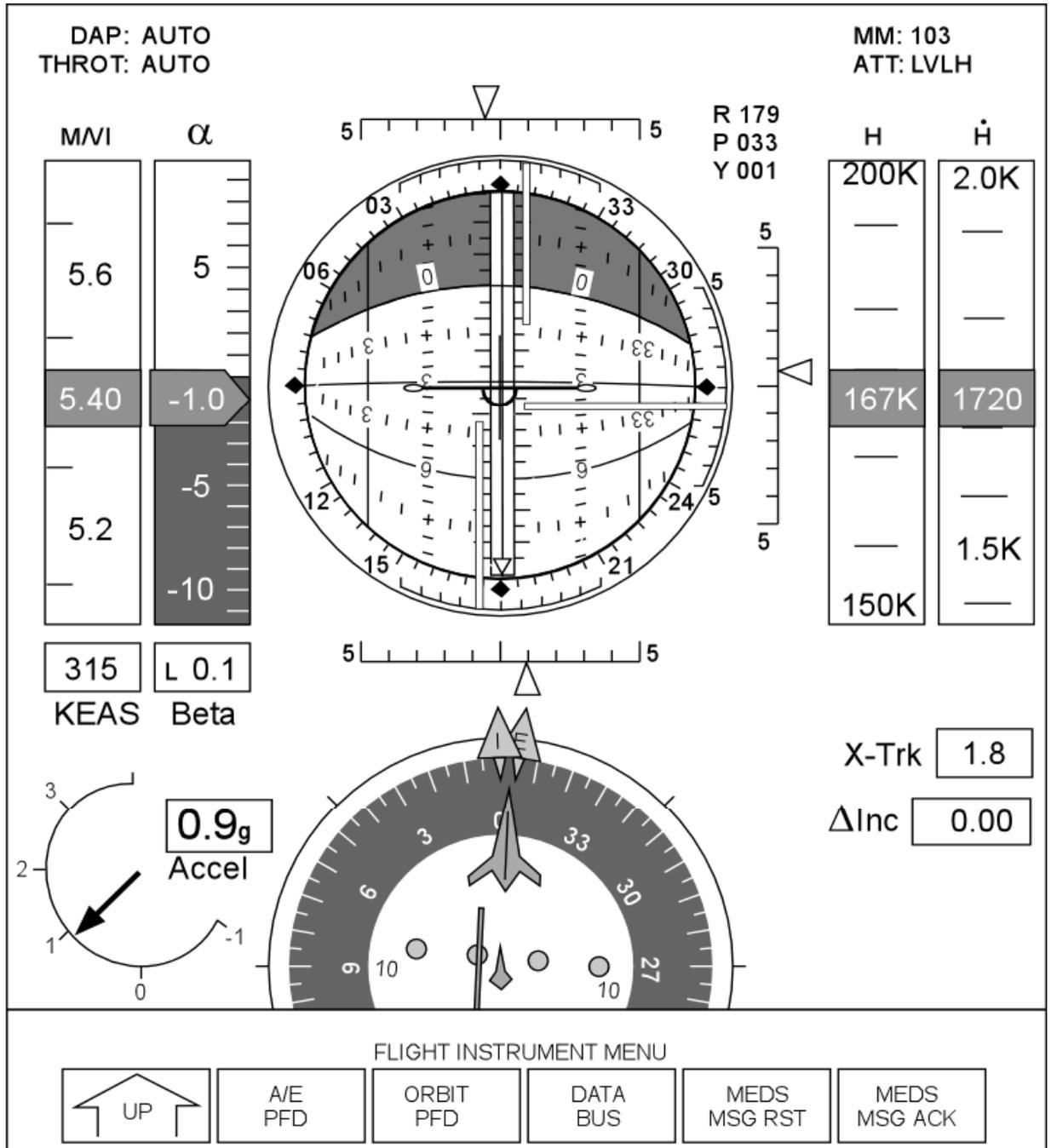


Figure 8 - 63 MM 103

In second stage (MM 103) the display is driven the same as MM 102 except the AMI velocity tape is now V_1 . The throttle field will blank at MECO confirmed.

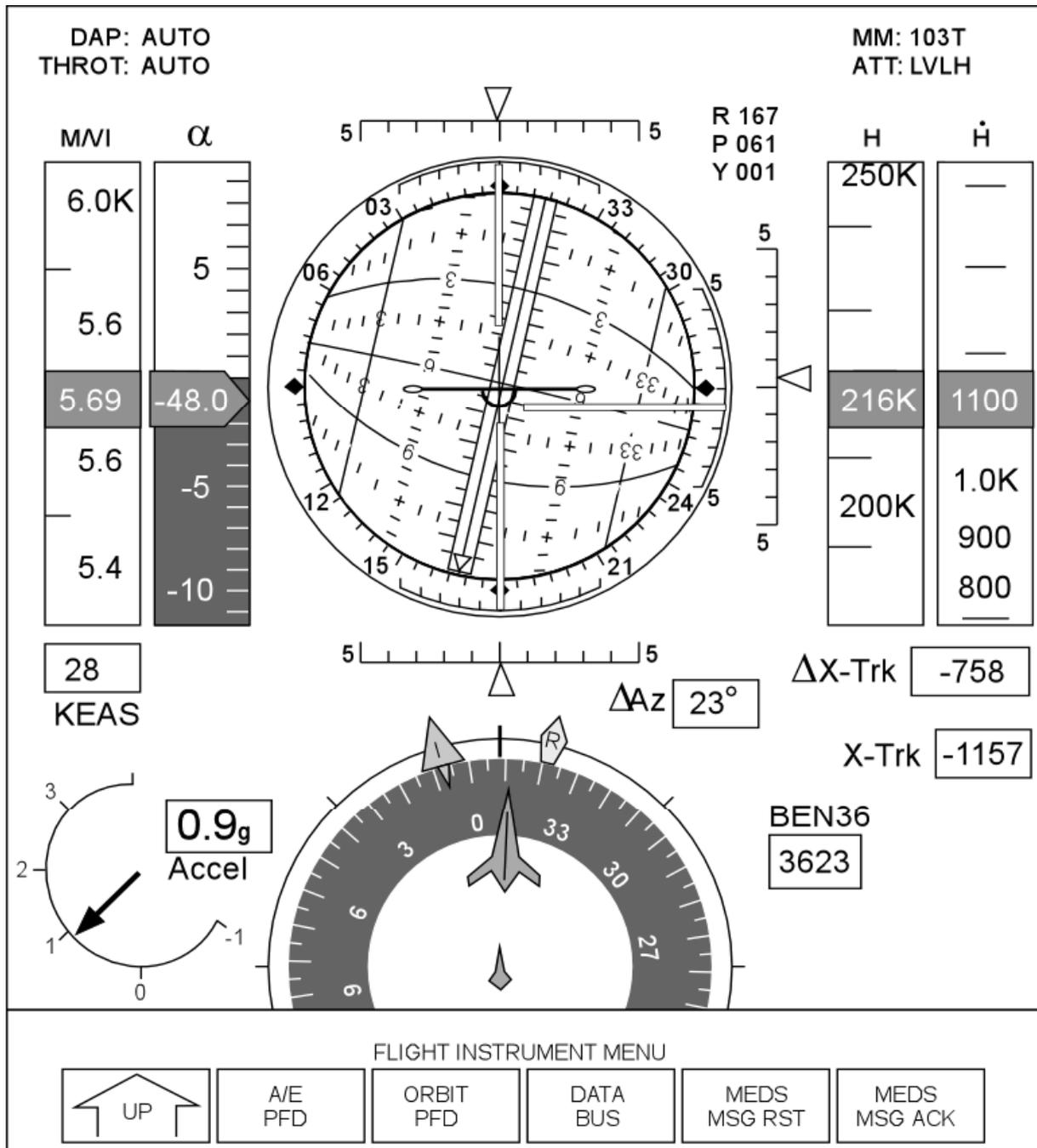


Figure 8 - 64 MM 103 TAL

At TAL selection, range and ΔAz to the primary SPEC 50 TAL site is displayed to the right of the HSI. ΔInc is replaced with $\Delta X-Trk$ in the trajectory field. The V_{REL} pointer is blanked and replaced by the runway pointer. The CDI is also blanked until OPS 3. The compass arrow points to the tangent of a circle around the TAL site.

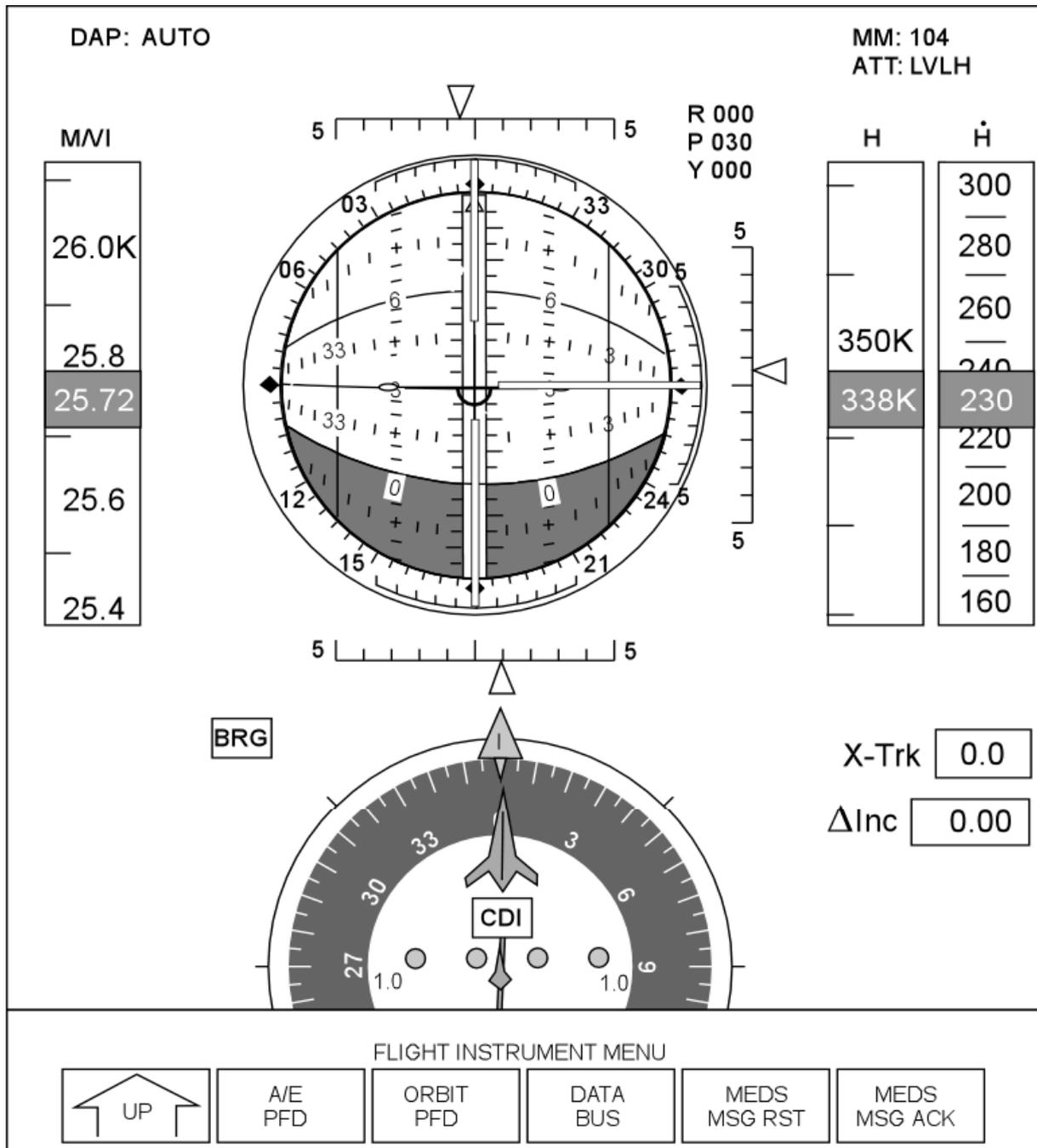


Figure 8 - 65 MM 104

For nominal uphill and AOA, the G-meter, alpha tape, and AMI digitals are blanked at the MM 104 transition. The CDI is displayed, but not driven. The V_1 pointer and compass arrow reflect deviation for the target inclination, which is also reflected in the X-Trk and Inc digitals. The BRG flag is set indicating the Hsi compass card is no longer being driven. The FCS Mode fields only indicate TRANS DAP AUTO or INRTL mode (throttle selection is blanked). For ATO, X-TRK and Δ Inc continue to be provided, but are now referenced to the new target inclination, which is also provided in the trajectory field. Additionally, the Major Mode field will reflect an AOA or ATO abort.

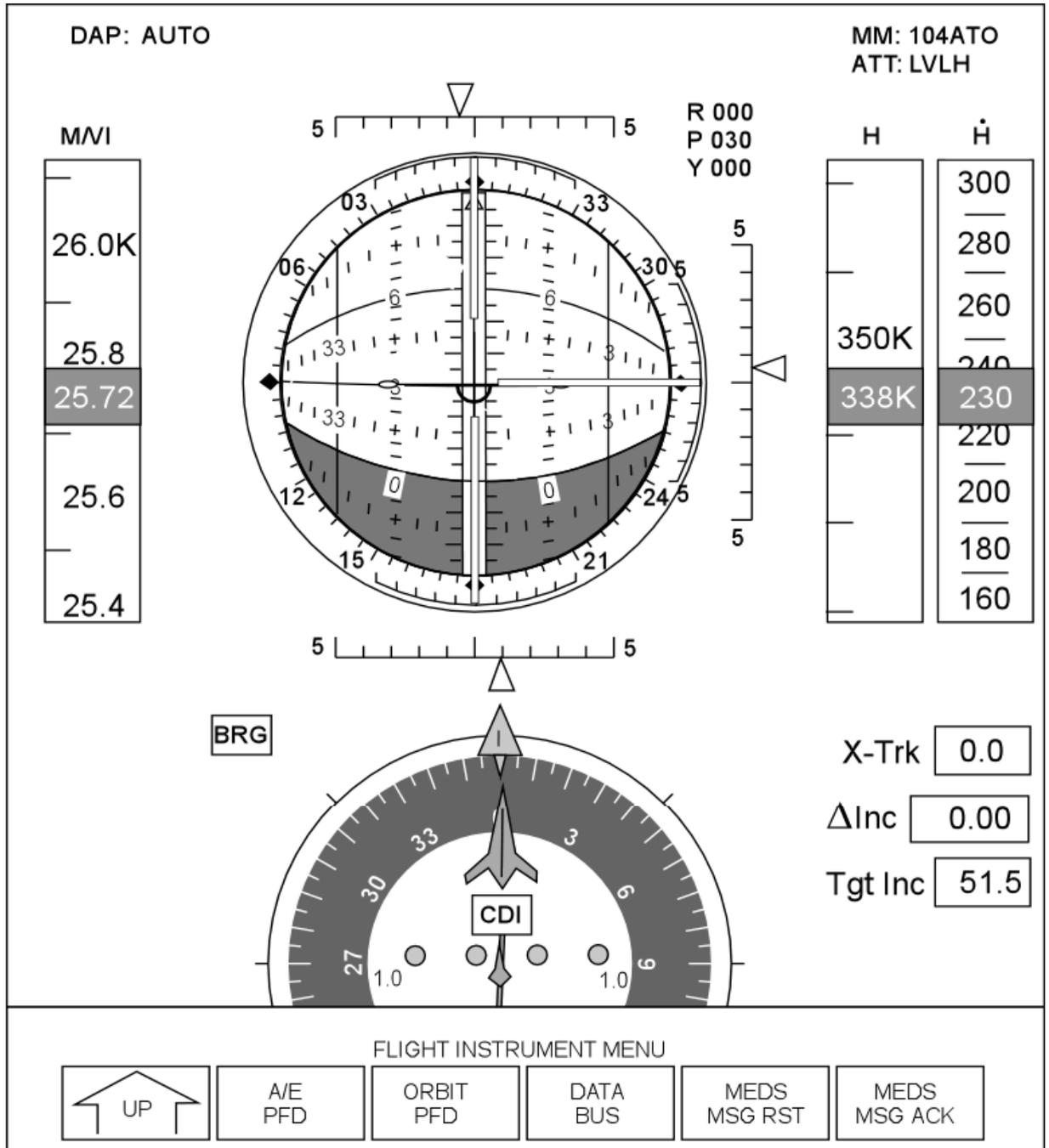


Figure 8 - 66 MM 104 ATO

At ATO abort selection, the new ATO target inclination is displayed in the trajectory field.

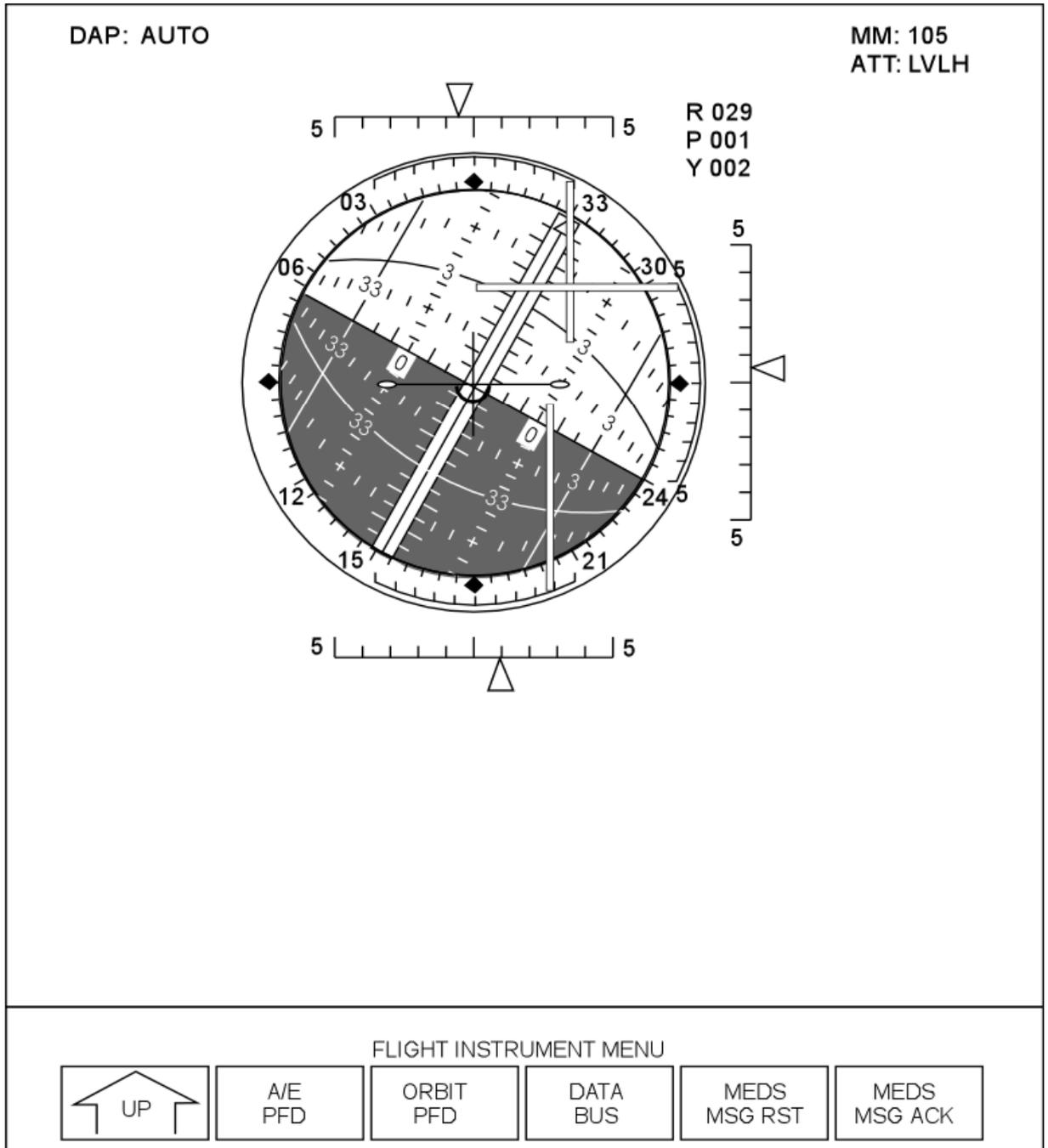


Figure 8 - 67 MM 105

Post OMS-1 (MM 105), the PFD devolves to essentially the orbit display. The exception is the MM 105 still retains the FCS Mode and Major Mode fields. There is no functional difference between MM 106 and MM 105, nor between nominal uphill, AOA, or ATO.

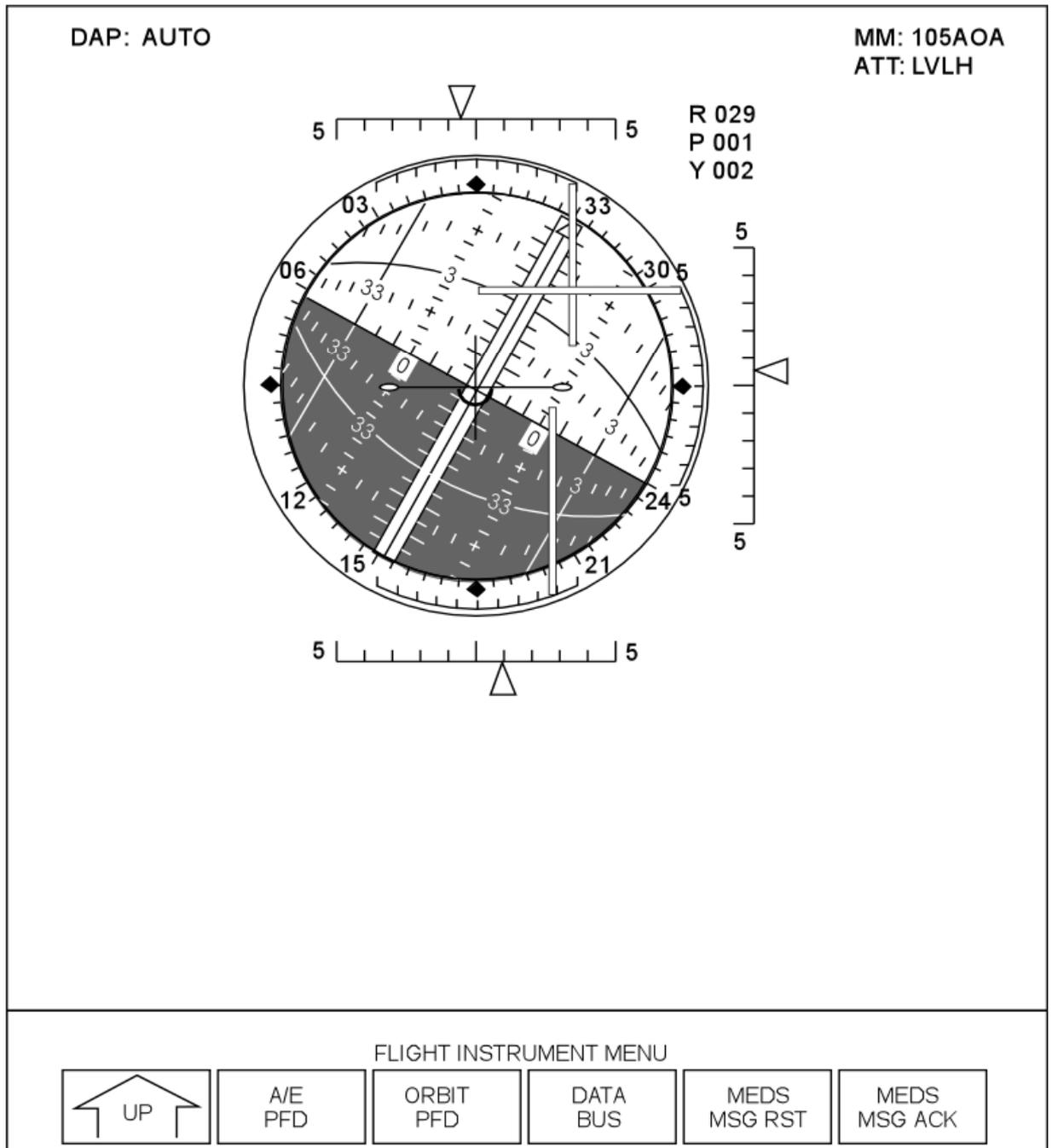


Figure 8 - 68 MM 105 AOA

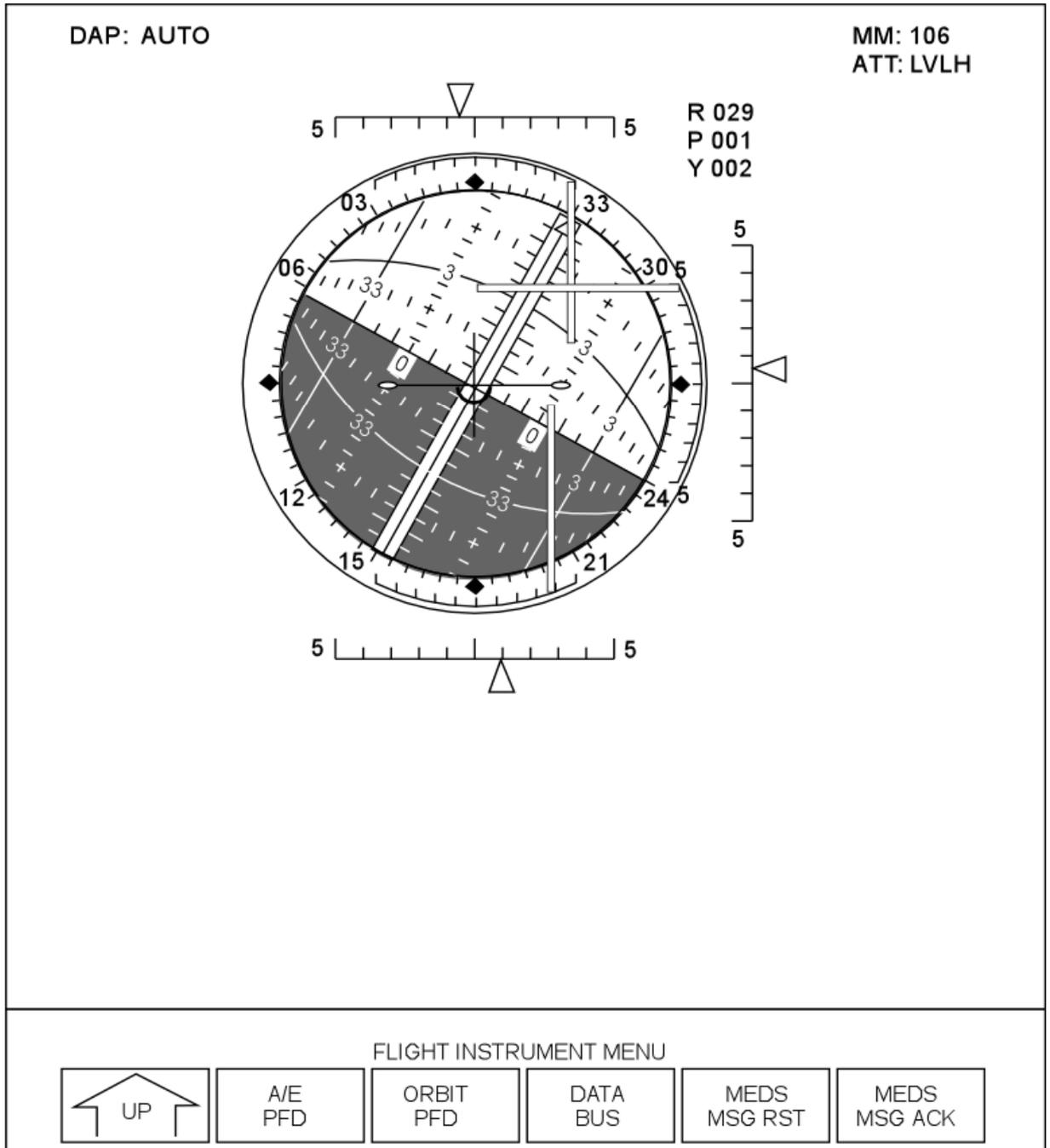


Figure 8 - 69 MM 106

8.2.9.3 OPS 6

The OPS 6 displays support both RTLS/ECAL and contingency abort modes. The Major Mode field will indicate "R" for RTLS, and "CA" for ECAL or contingency abort modes.

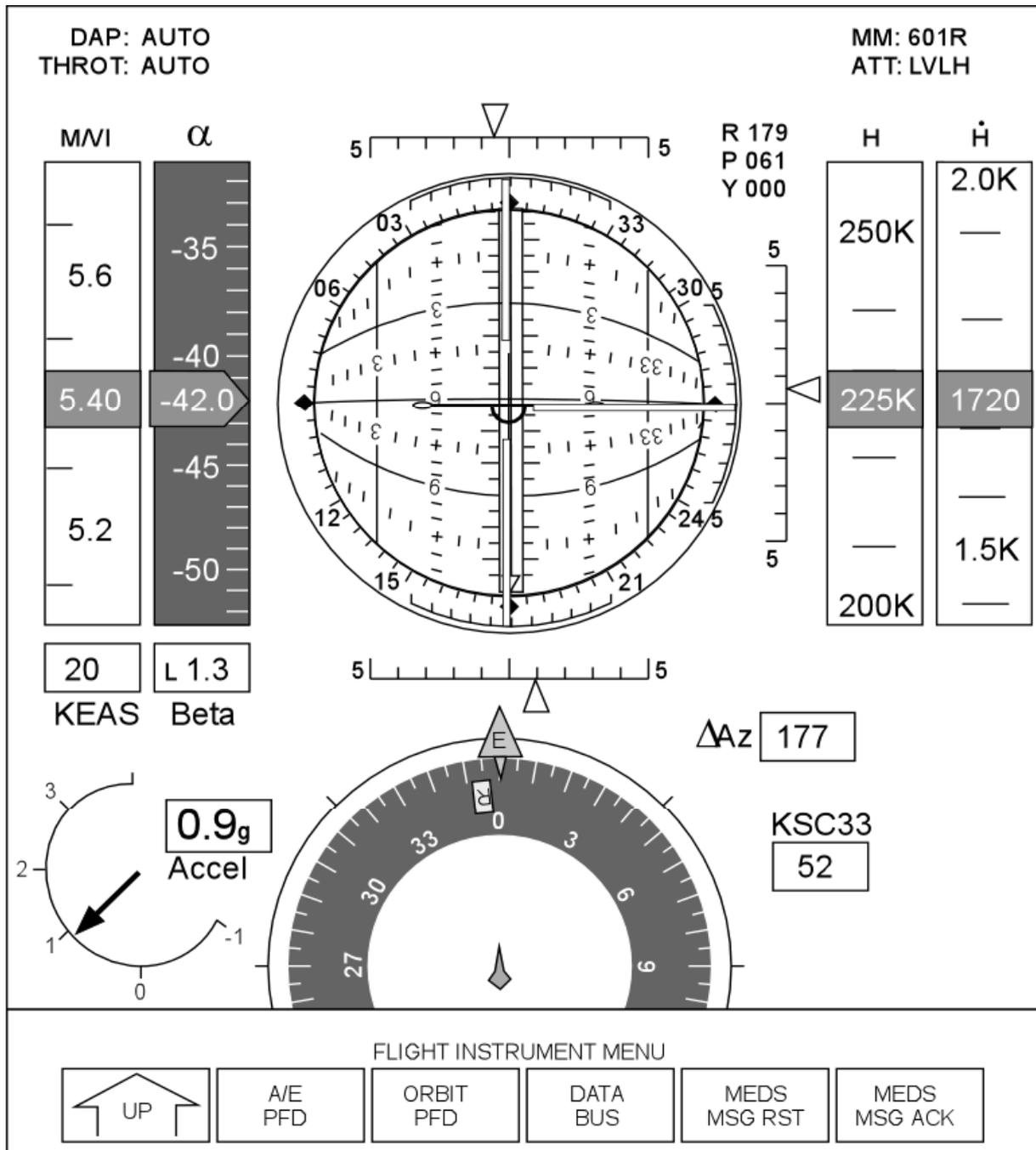


Figure 8 - 70 MM 601 Pre-PPA

After RTLS abort is declared, the HSI V_i bearing pointer is replaced with the runway bearing pointer. During the outbound leg, only the runway pointer reciprocal will be displayed. ΔAz and range to KSC also becomes available. The AMI displays Earth-relative velocity (V_{REL}) and the HSI only provides the V_{REL} pointer ("E"). For RTLS, the course arrow and CDI are blanked, but for ECAL or contingency abort, the course arrow will be displayed as white during active yaw steering.

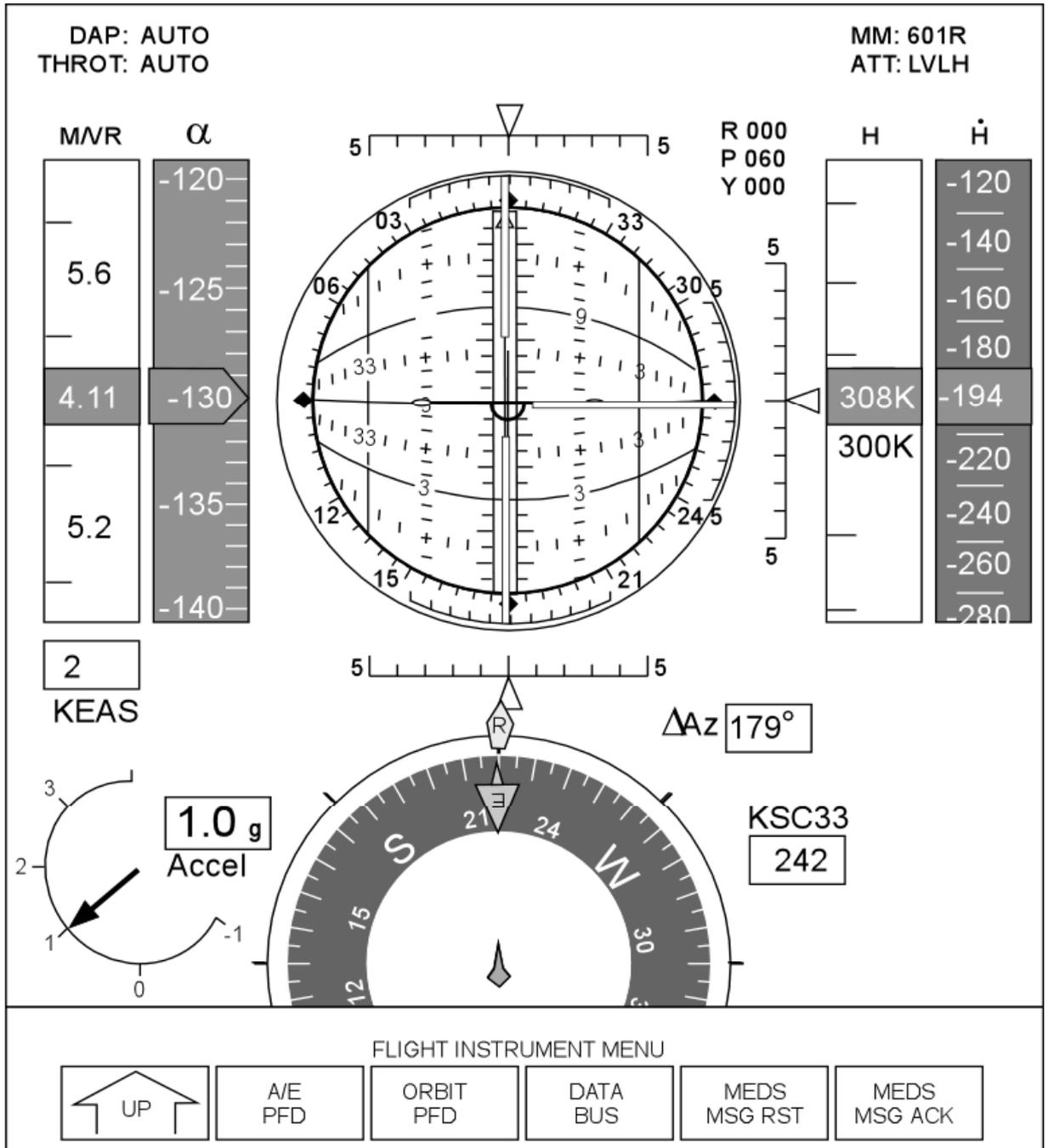


Figure 8 - 71 MM 601 Post-PPA

Post-PPA for RTLS, the AMI changes from inertial to Earth-relative velocity and labelled "M/V R". The runway pointer is now shown since the runway heading is now in front of the vehicle, although the V_{REL} reciprocal will now be shown until flyback is achieved.

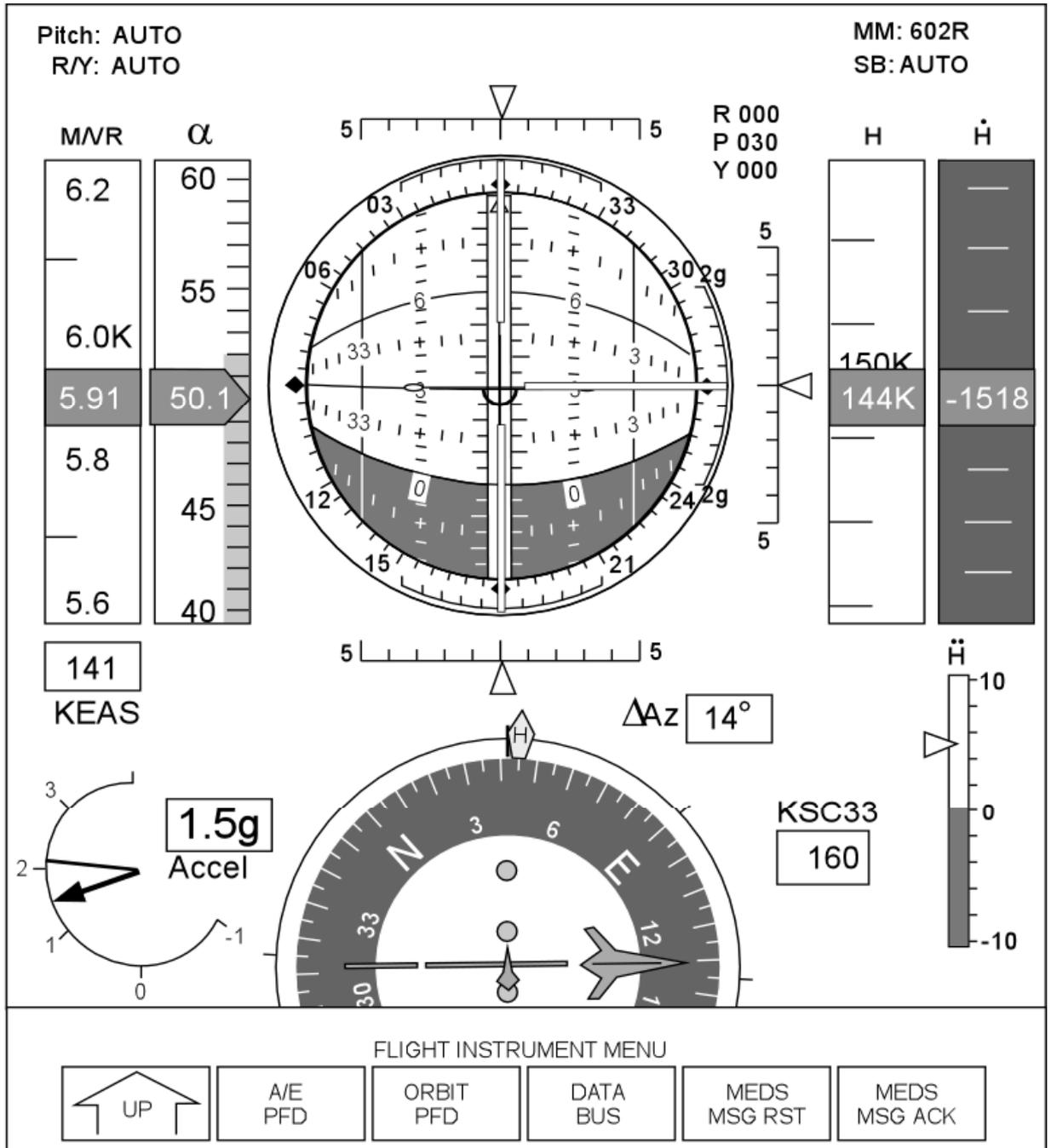


Figure 8 - 72 MM 602

Post-MECO (MM 602), the altitude acceleration meter becomes available. The ADI pitch error scale changes to g's and is labelled accordingly, and the g-meter provides a magenta target N_z line. Once out of N_z hold, the pitch error reverts to degrees. The FCS Mode fields now reflect the Aerojet DAP Pitch and R/Y modes, with yellow boxes highlighting if MAN is selected for either. The ADI Attitude field is replaced with the selected speedbrake mode. An α min/max becomes available in the AMI α tape for RTLS, but is not driven for ECAL or contingency aborts. The HSI course arrow and CDI again become active.

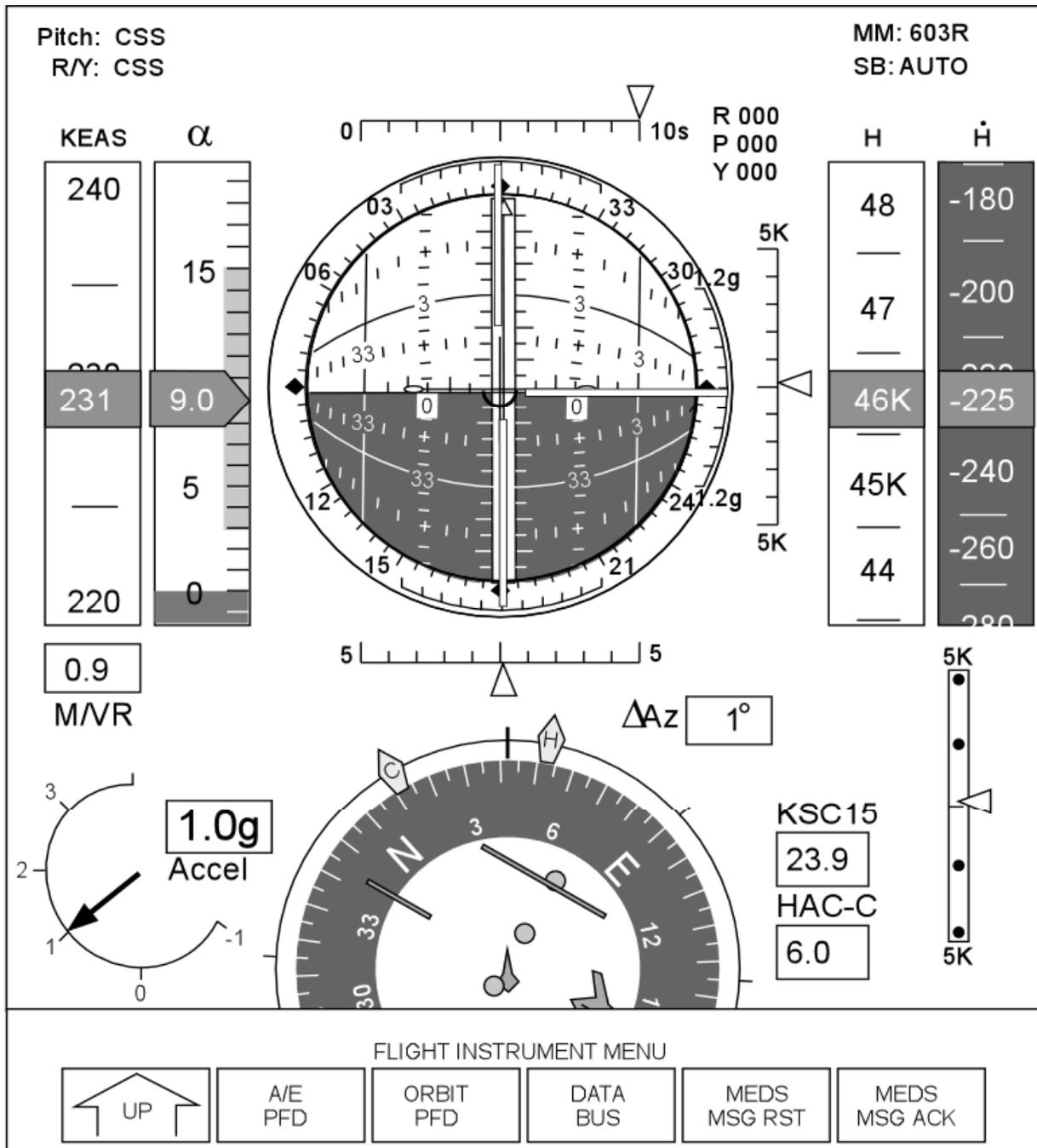


Figure 8 - 73 MM 603

For RTLS or ECAL TAEM (MM 603), the altitude acceleration meter is replaced with the GSI. The GSI label during TAEM is 5K feet, and changes to 1K at A/L. In addition to a range to runway touchdown digital and HAC intercept (H) bearing, a range to HAC center digital and HAC center bearing pointer (C) are also provided. The target N_z line on the g-meter is removed. ADI Pitch Error is again in g's, and indicated by the "g" label. The ADI Rate needle becomes a counter to HAC intercept (provided the ADI Rate switch is in MED).

8.2.9.4 OPS 2 & 8

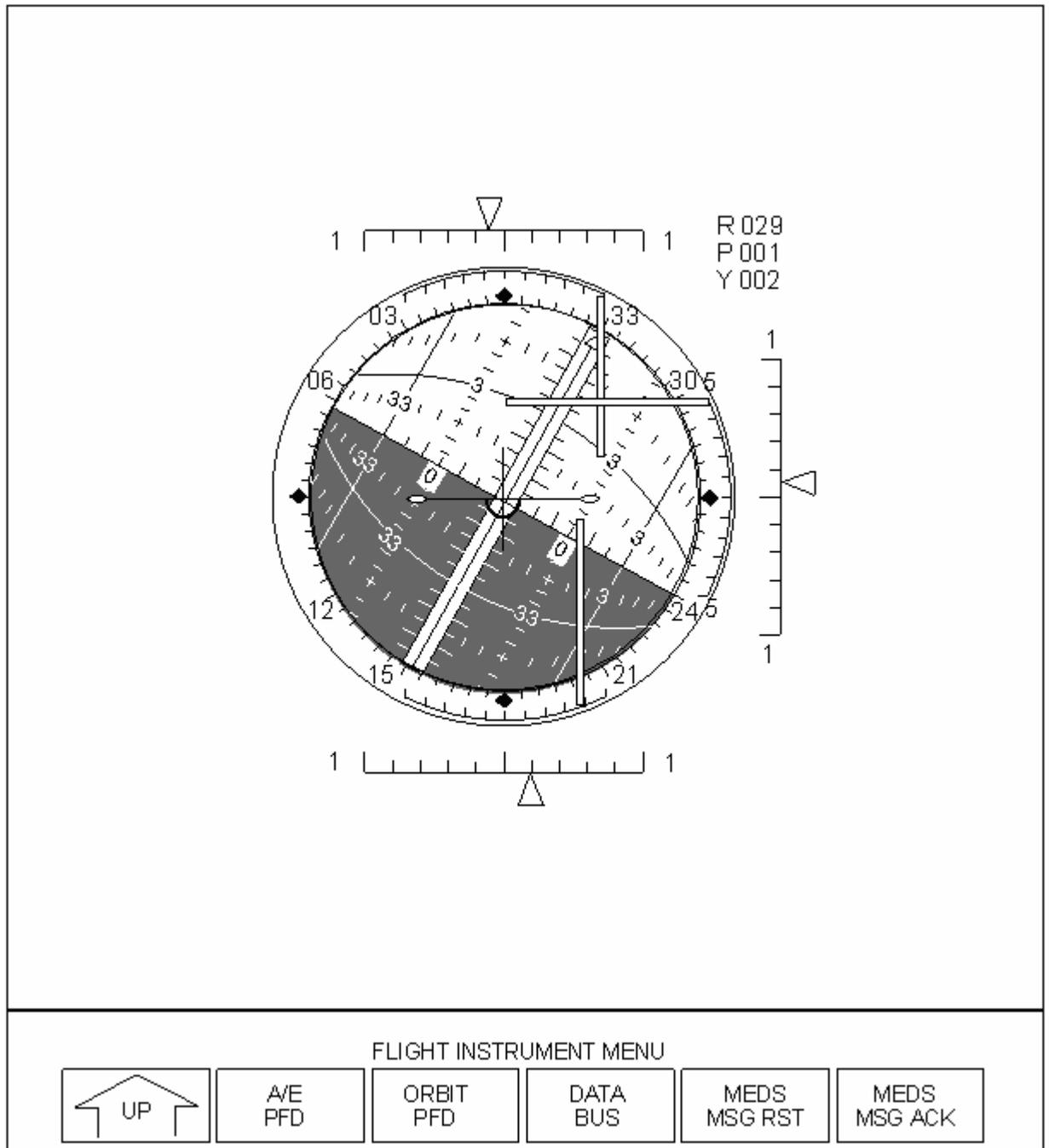


Figure 8 - 74 Orbit PFD (OPS 2 Rate Switch MED setting; OPS 8 Rate Switch LOW setting).

In orbital phases, the PFD consists of only the ADI and digital attitude readouts. The ADI reflects the ADI Attitude, Error, and Rate switches, and reflects IMU-derived body rates.

8.2.9.5 OPS 3

The display for MM 301-303 is the same as MM 105/106.

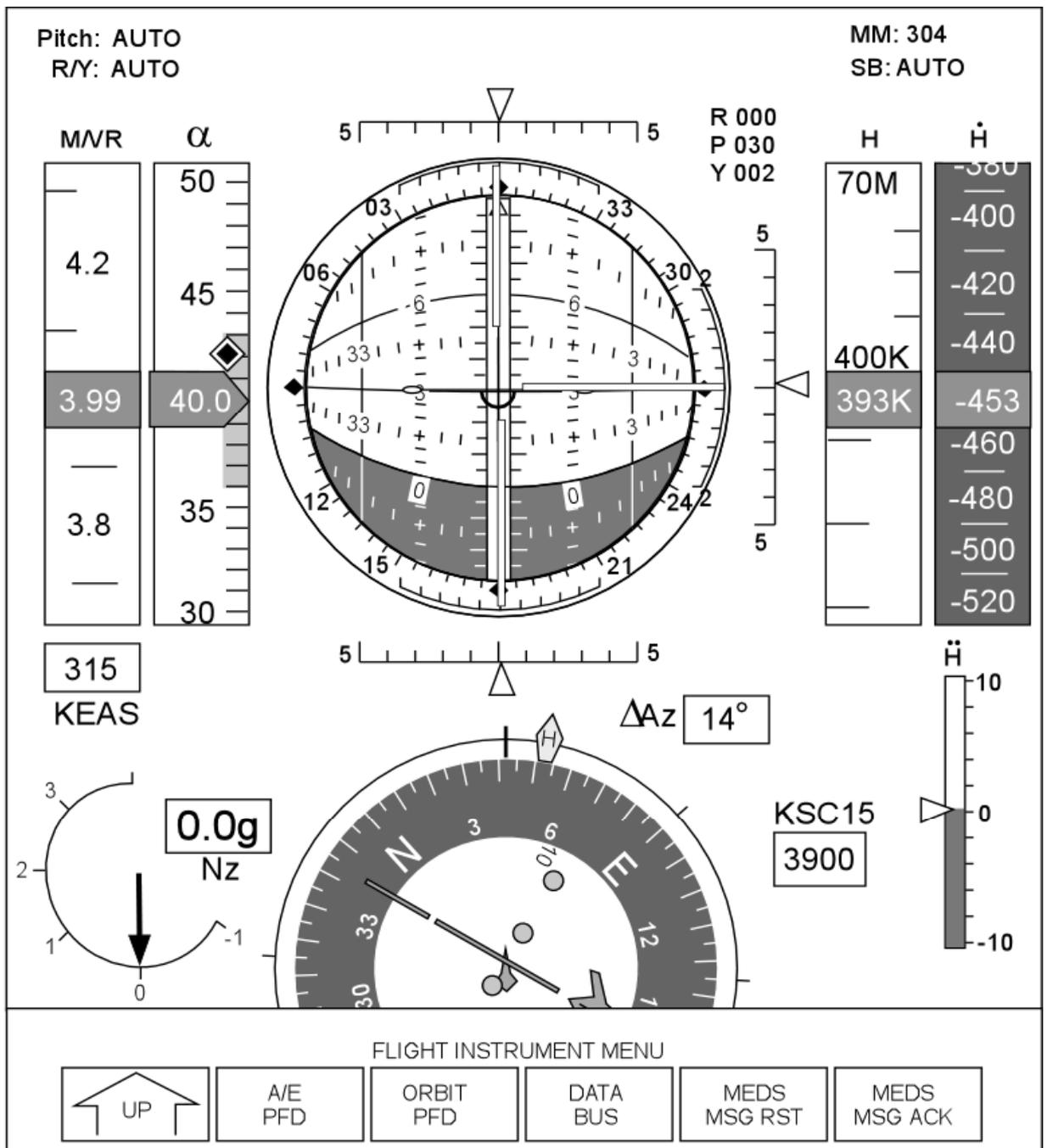


Figure 8 - 75 MM 304

For glided flight, the ADI, AMI, and AVVI are provided the same as powered flight. Velocity on the AMI is Earth-relative (V_{REL}), and the beta digital is blanked. An altitude acceleration meter is provided in MM 304. The HSI provides a HAC intercept bearing pointer, and reflects magnetic heading (as indicated by

the cardinal point labels of N, S, E, and W). The AMI alpha tape provides both a min/max alpha bar, and a max L/D diamond. FCS Mode fields reflect the Aerojet DAP pitch and R/Y modes, with the ADI Attitude field being replaced by the speedbrake mode. If TAL is selected, the major mode field will indicate this with a "T".

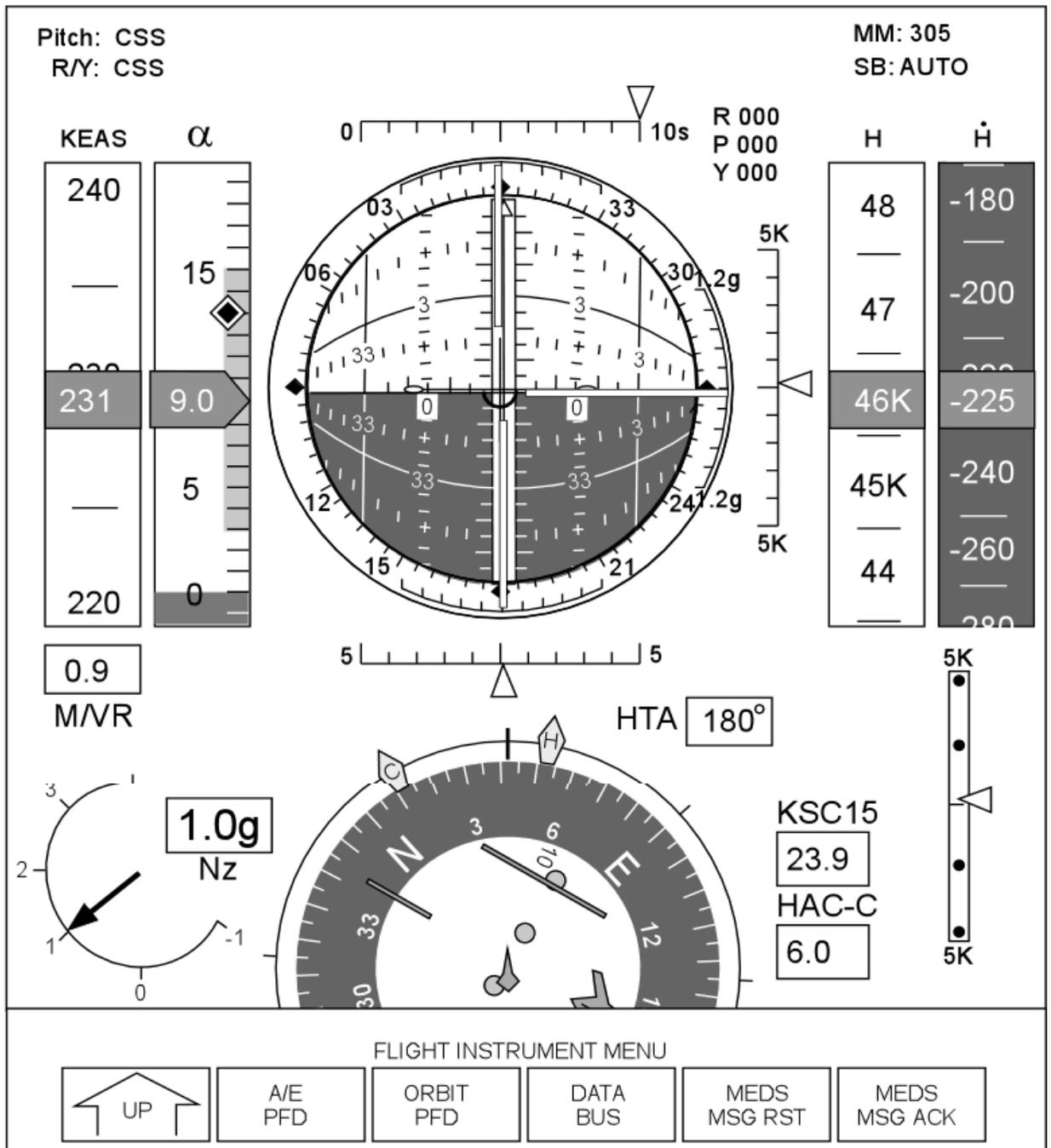


Figure 8 - 76 MM 305

The TAEM display replaces the Altitude Acceleration meter with the GSI. The GSI scale is initially 5K feet, and changes to 1K feet at A/L, and becomes inactive at 1500 feet (GSI flag is displayed). A digital of range to HAC center and HAC center bearing pointer are also provided until A/L. At $M = 0.9$, the scale of the AMI velocity tape changes from Mach number to KEAS, with the alternate velocity is provided by the digital below the tape. The ADI pitch error is expressed in g's, and labelled accordingly. The roll rate needle becomes a 10 second counter to HAC intercept if the ADI Attitude Rate switch is in MED. The CDI scale changes from 10 nm at TAEM to 2.5 nm at A/L.

8.2.9.6 BFS

In general, the BFS Composite display is the same as the PASS, within the ability of the BFS to support the different flight phases. For Post-MECO and glided phases (MM 104-305), the display is the same as in the PASS. For powered flight, however, the AMI, AVVI, and g-meter are blanked. A β digital is provided however, as are X-Trk and Δ Inc digitals. The HSI provides both a V_{REL} and V_I pointer, and a CDI. In the FCS Mode field, the throttle selection is blanked since there is no manual powered flight capability in the BFS.

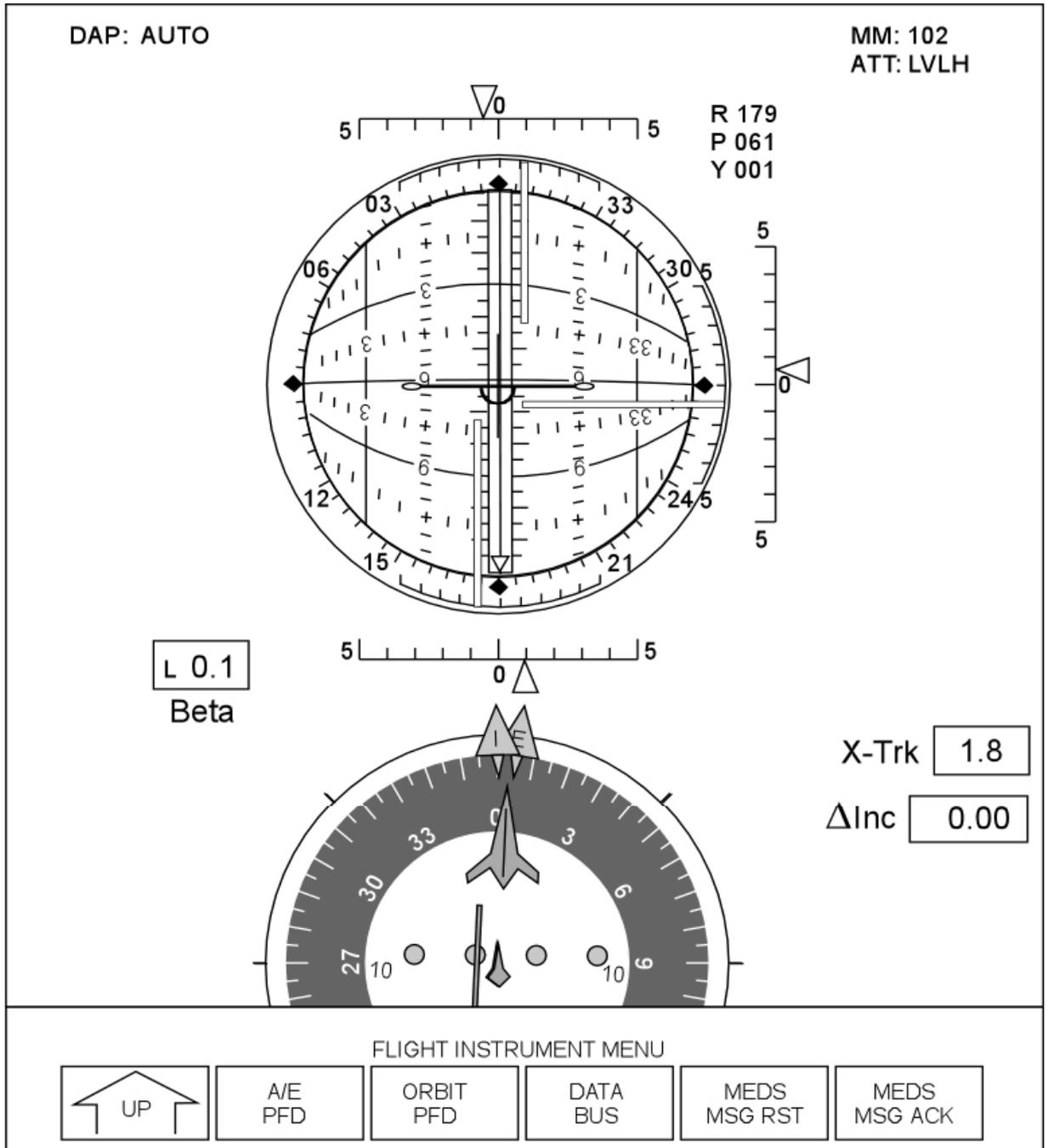


Figure 8 - 77 BFS MM 102 & 103

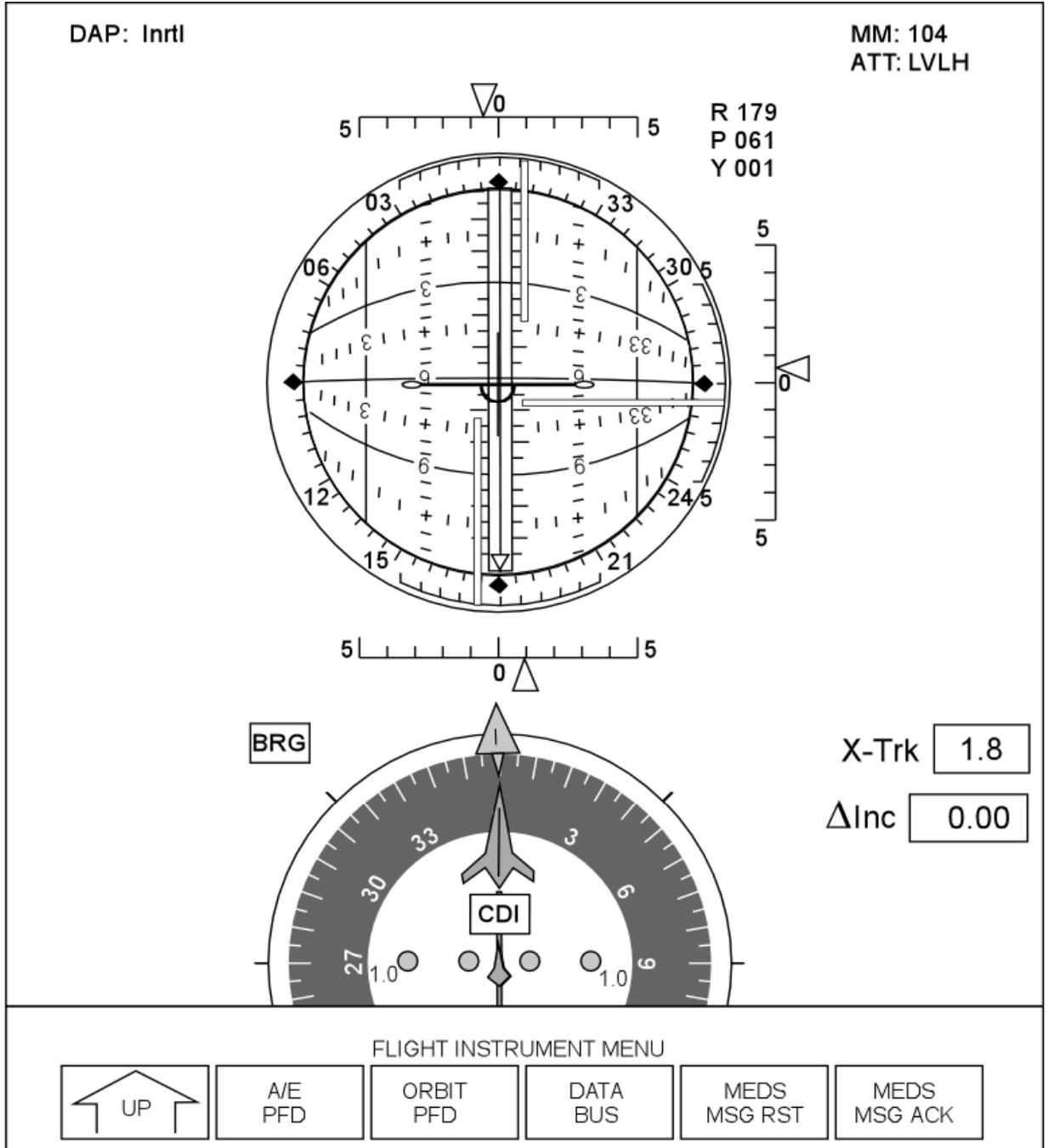


Figure 8 - 78 BFS MM 104

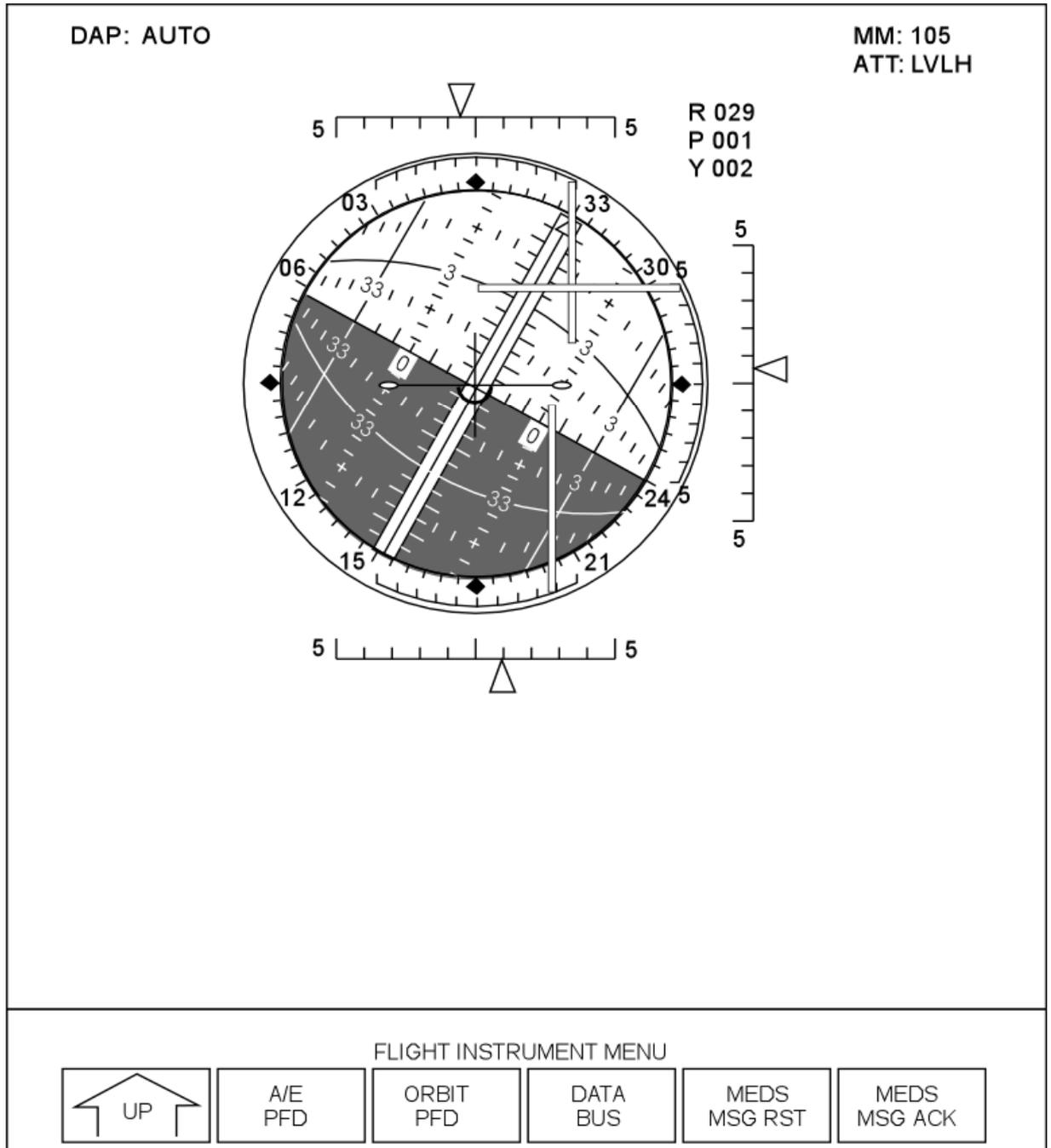


Figure 8 - 79 BFS MM 105, 106, & 301-303

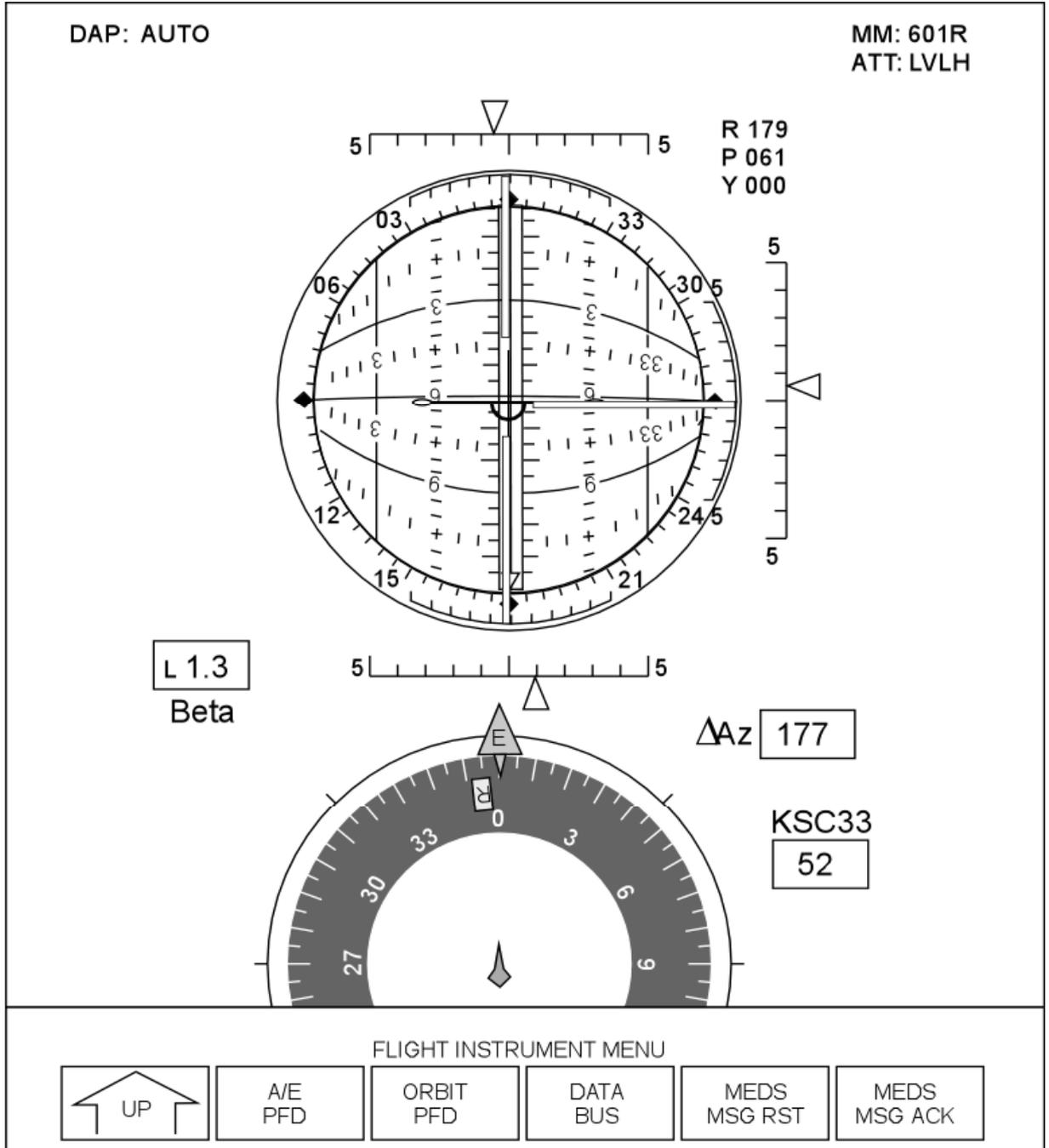


Figure 8 - 80 BFS MM 601 - Pre PPA

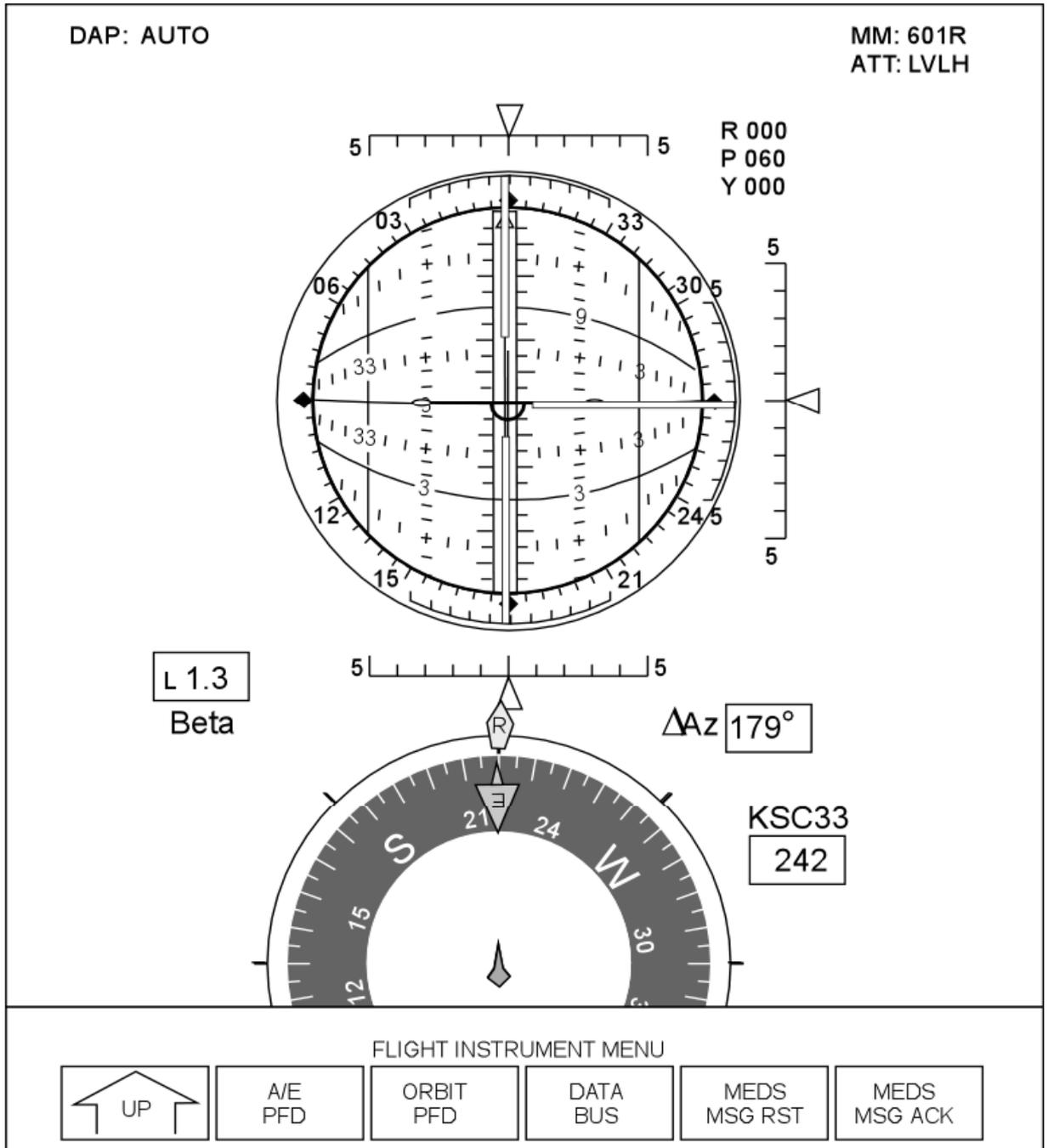


Figure 8 - 81 BFS MM 601 - Post PPA

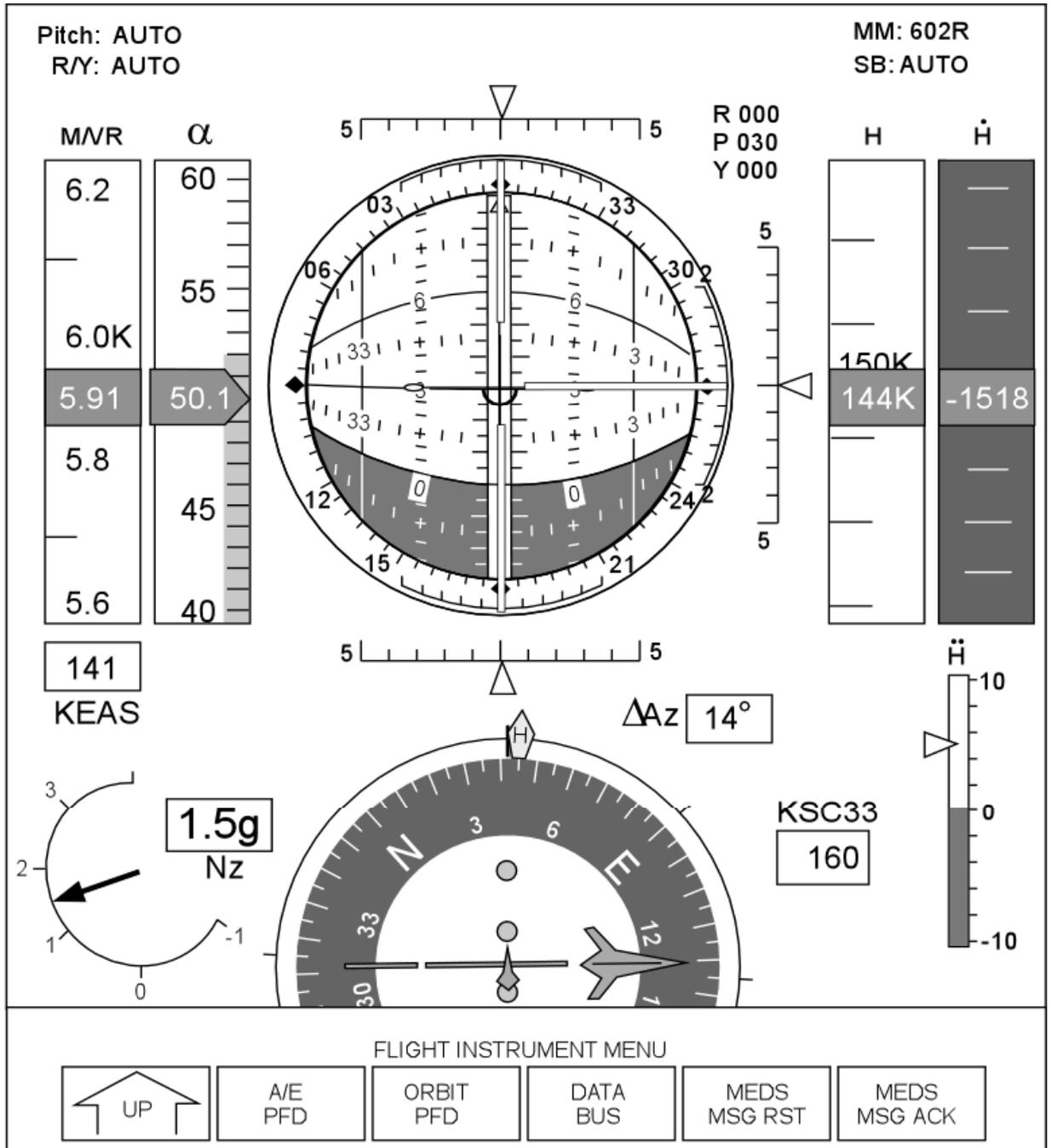


Figure 8 - 82 BFS MM 602

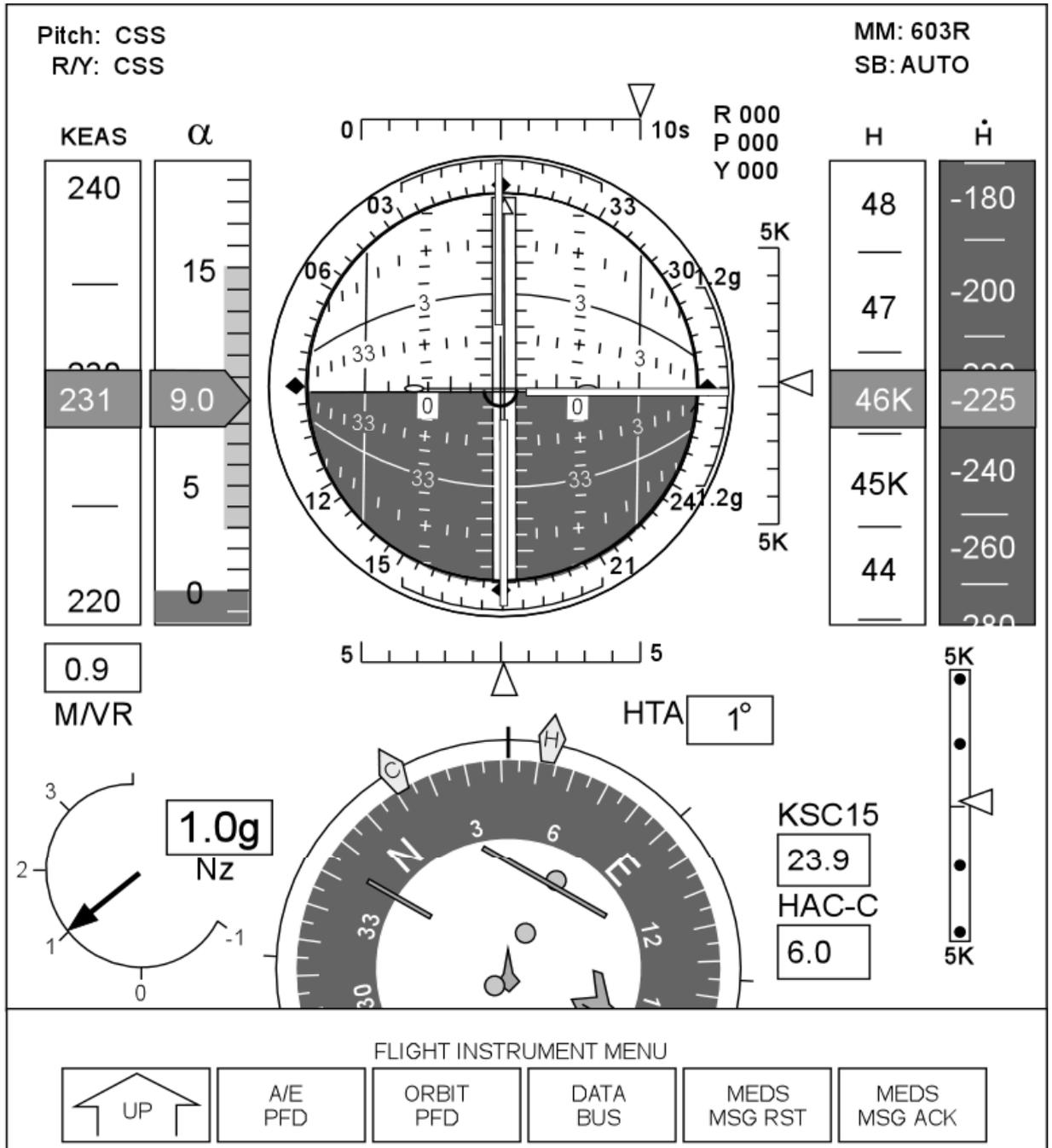


Figure 8 - 83 BFS MM 603

8.3 SPI

The Surface Position Indicator (SPI) provides the crew insight into aerosurface activity. Data is processed through MEDS Analog/Digital Converter One (ADC - 1). Elevon, speedbrake command and

position, and rudder position are channelized through MDM FF1 card 08 outputs, and aileron and body flap positions channelized through MDM FF2 card 08 outputs. Elevon positions are provided at 25 Hz, with the other indications provided at 12.5 Hz.

Elevon and body flap positions are provided in degrees and percent, respectively. Trailing Edge (TE) up is at the top of the display, and TE down at the bottom. A special mark is provided to indicate the body flap trail position at 34.2%. Body flap, elevon, rudder, and speedbrake positions are the RM selected position feedback values. Speedbrake command and position are also displayed in digital readouts.

Selected aileron, elevon, rudder, speedbrake, and body flap position digital values are also available on GNC SYS SUMM (Figure 8 - 85). Note the aileron position provided on the SPI is not the aileron trim, but the combined aileron position. The aileron trim is available on the Entry TRAJ and VERT SIT displays.

Labels and digital values are typically white on a black background, with the following exceptions. All pointers are yellow with the exception of Speedbrake Command, which is cyan. Aileron, rudder, and elevon zero positions are yellow. The labels TE UP, TE DN, R AIL, L AIL, R RUD, and L RUD are green. The scale backgrounds are gray. Invalid parameters (for instance due to commfaults) are indicated by a red box. Invalid digital values and pointers are blanked with scales indicated by red.

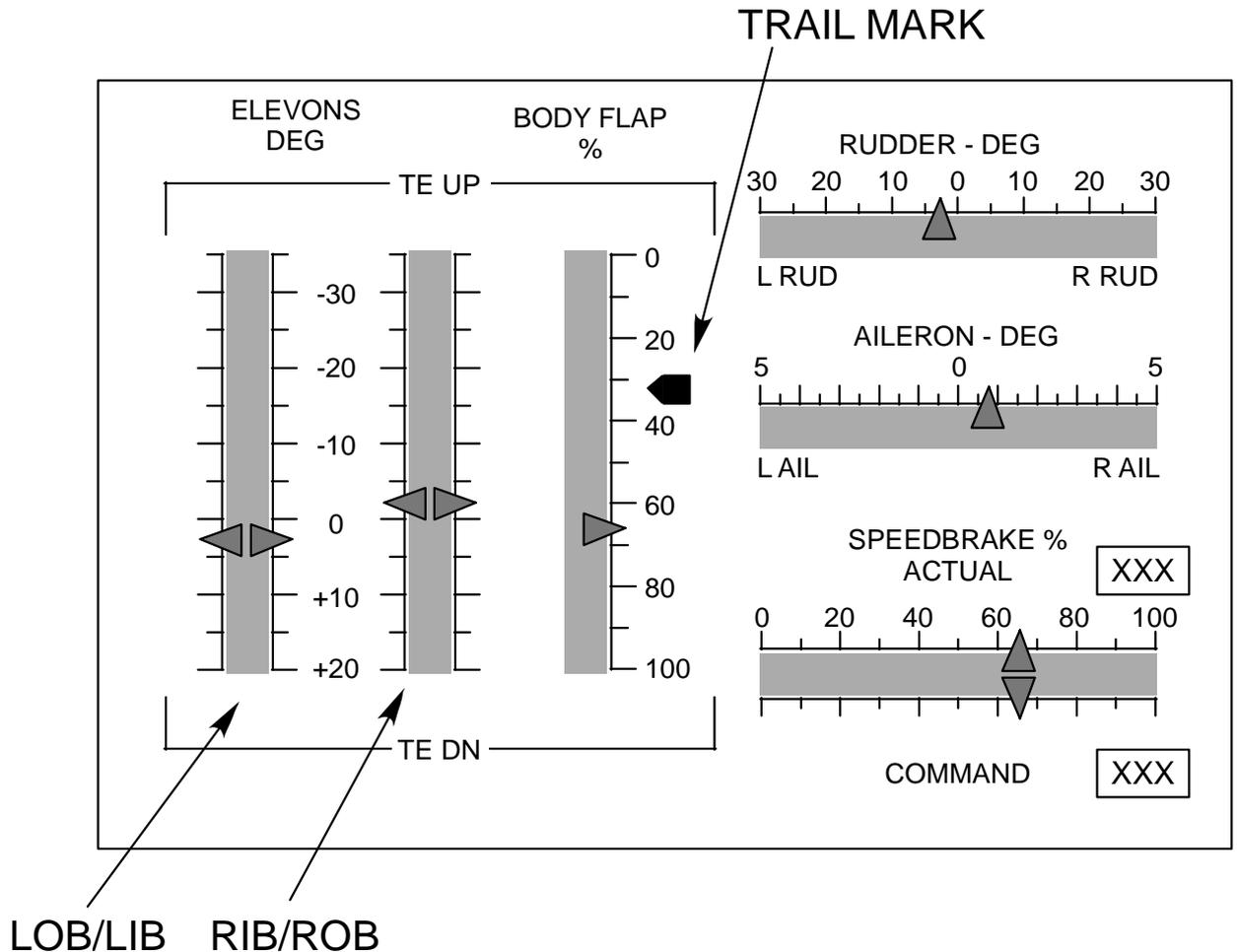


Figure 8 - 84 Surface Position Indicator

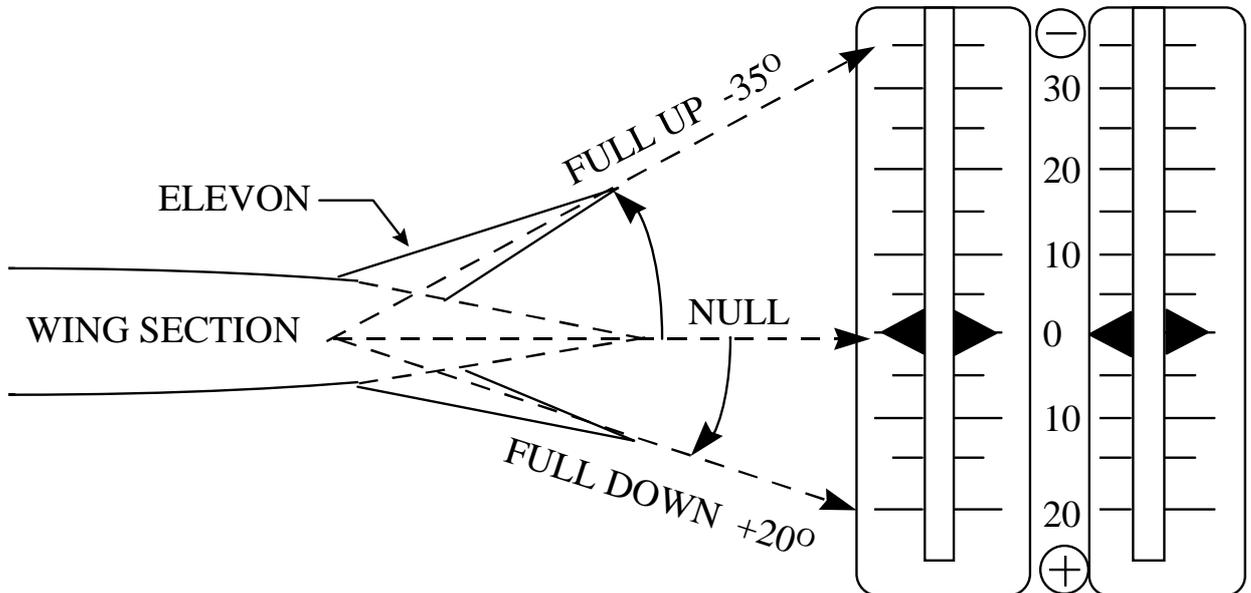


Figure 8 - 86 SPI vs. Elevon Travel

BODY FLAP % - This scale reads body flap position from 0 to 100 percent of software-allowed travel (Figure 8 - 87). 0% corresponds to full-up (-11.7°); 100% corresponds to full-down (+22.5°). The small pointer at 35% is fixed and shows the trail position.

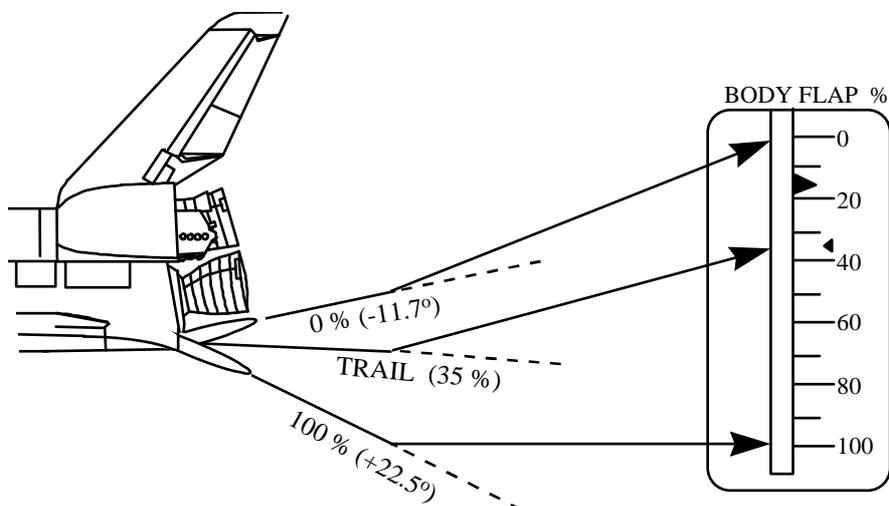


Figure 8 - 87 SPI vs. Body Flap Travel

RUDDER-DEG - Rudder position is displayed as if viewed from the rear of the vehicle (Figure 8 - 88); e.g., deflection to the left of center represents left rudder. The scale is +30° (left) to -30° (right), but software limits the rudder command to $\pm 27.1^\circ$.

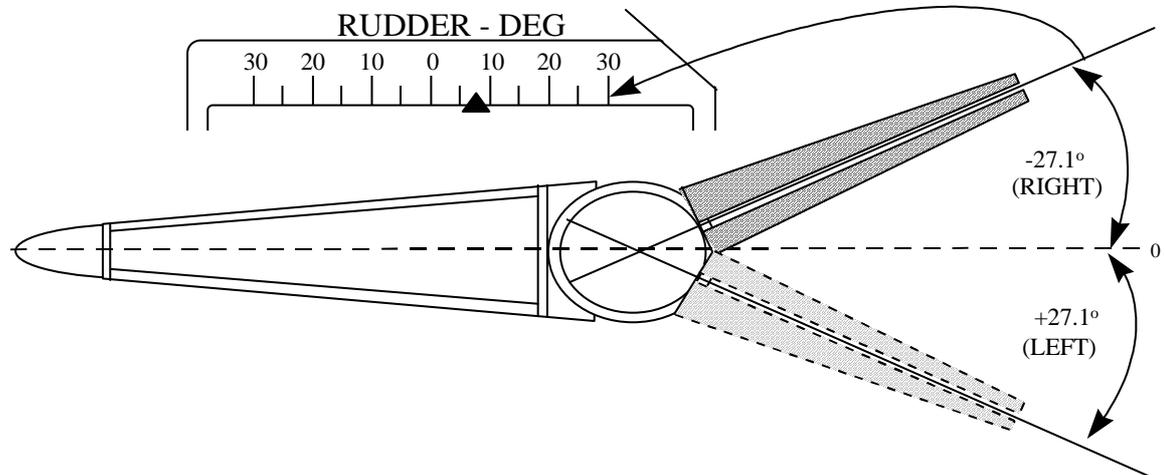


Figure 8 - 88 SPI vs. Rudder Travel

AILERON-DEG - This scale provides the effective aileron function of the elevons in combination:

$$\text{AILERON} = \frac{\text{AVG LEFT ELEV} - \text{AVG RIGHT ELEV}}{2}$$

Deflection of the pointer to the right of center indicates a roll right configuration (left elevons down, right elevons up) and vice versa. The scale is -5° to $+5^\circ$, with -5° at the left side. The aileron command can exceed $\pm 5^\circ$ (maximum $\pm 10^\circ$) in which case the meter saturates at $\pm 5^\circ$. Figure 8 - 89 is an example of -5° aileron; note that the differential is not necessarily with respect to the zero elevon point.

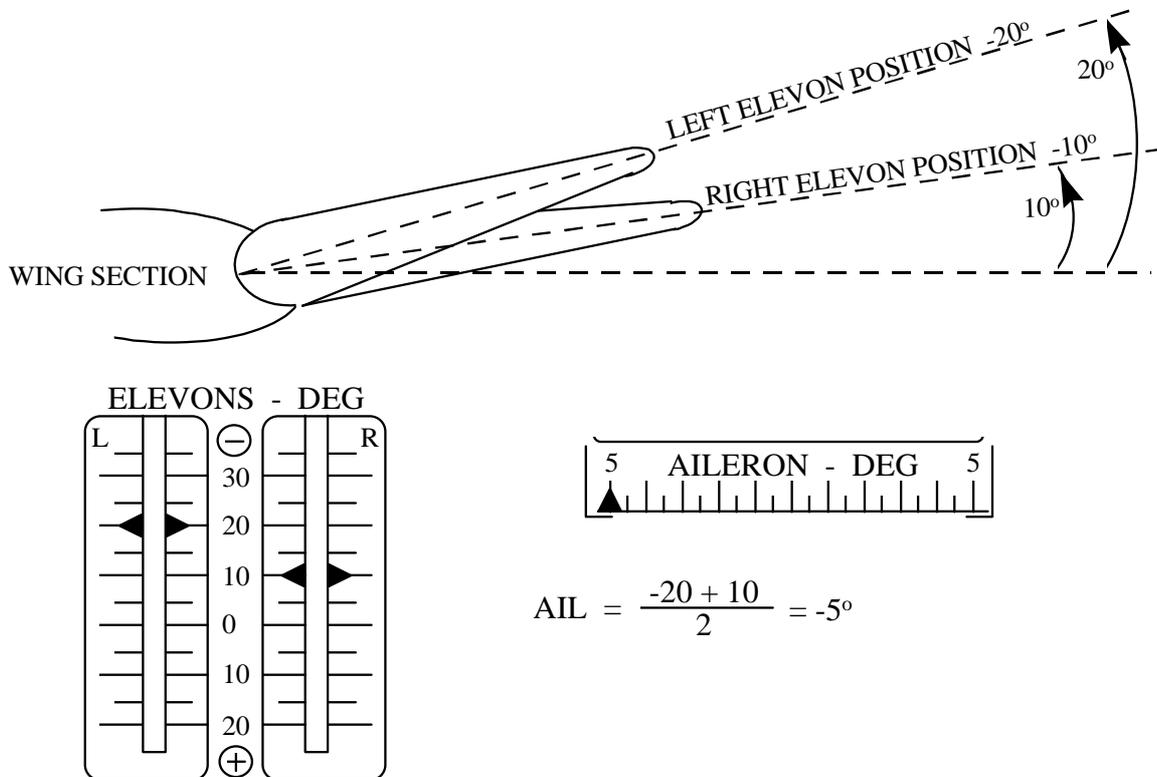


Figure 8 - 89 SPI vs. Aileron Travel

SPEEDBRAKE % - This scale indicates the speedbrake position on the upper scale and speedbrake command on the lower scale. The position ranges 0 to 100 percent; 0% is fully closed; 100% is fully open, which corresponds to 98° with respect to the hinge lines. The small point at 25% is fixed and represents the point at which the speedbrake surfaces and the remainder of the tail form a smooth wedge. Figure 8 - 90 shows the open speedbrake.

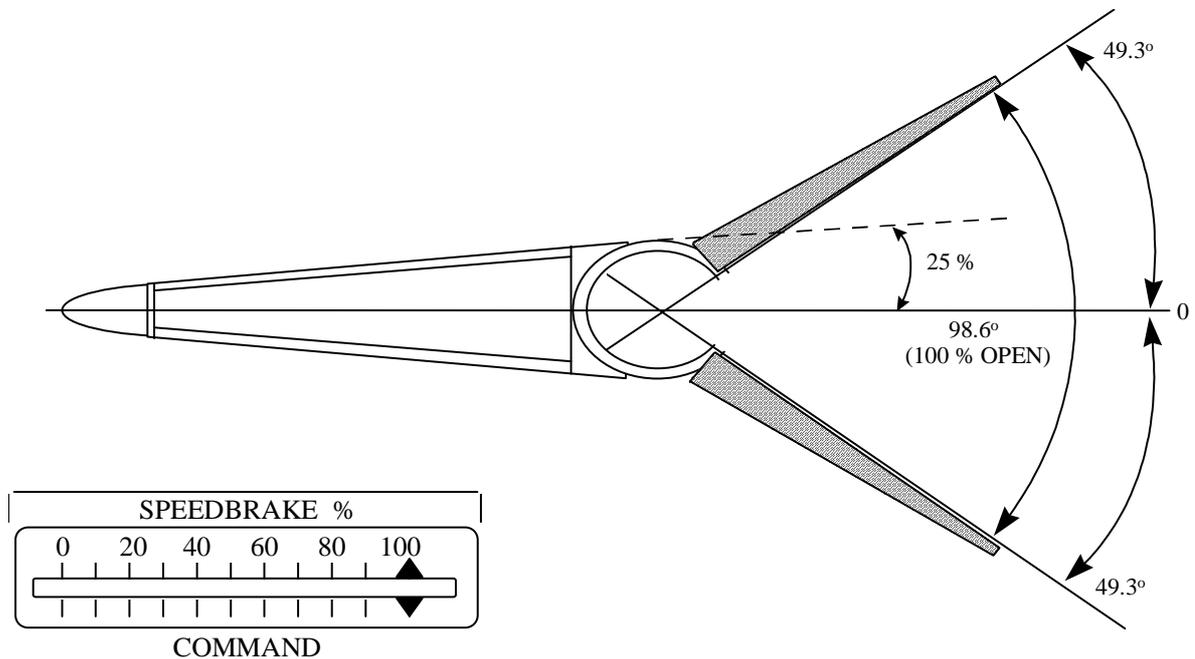


Figure 8 - 90 Top View of Open Speedbrake

The speedbrake command is scaled identically to position and has the same travel limits. It always represents the speedbrake auto guidance command.

8.3.1 SPI Controls

There are no controls which affect SPI processing.

8.3.2 SPI Display Processing

8.3.2.1 Data Flow

Each aerosurface sends four position feedback parameters to RM, as shown in Figure 8 - 91. RM selects one feedback for each surface and forwards it to the appropriate position feedback SOP for compensation and scaling. The respective feedbacks are then sent to the SPI processor, where the aileron function is computed and all parameters are scaled and converted to fixed point. The SPI processor also receives the auto speedbrake command from flight control or guidance software, depending on the phase. No FOH's are required as the input and output data rates are the same (6.25 Hz). The SPI parameters are then sent to MEDS via MDM FF1 and ADC-1.

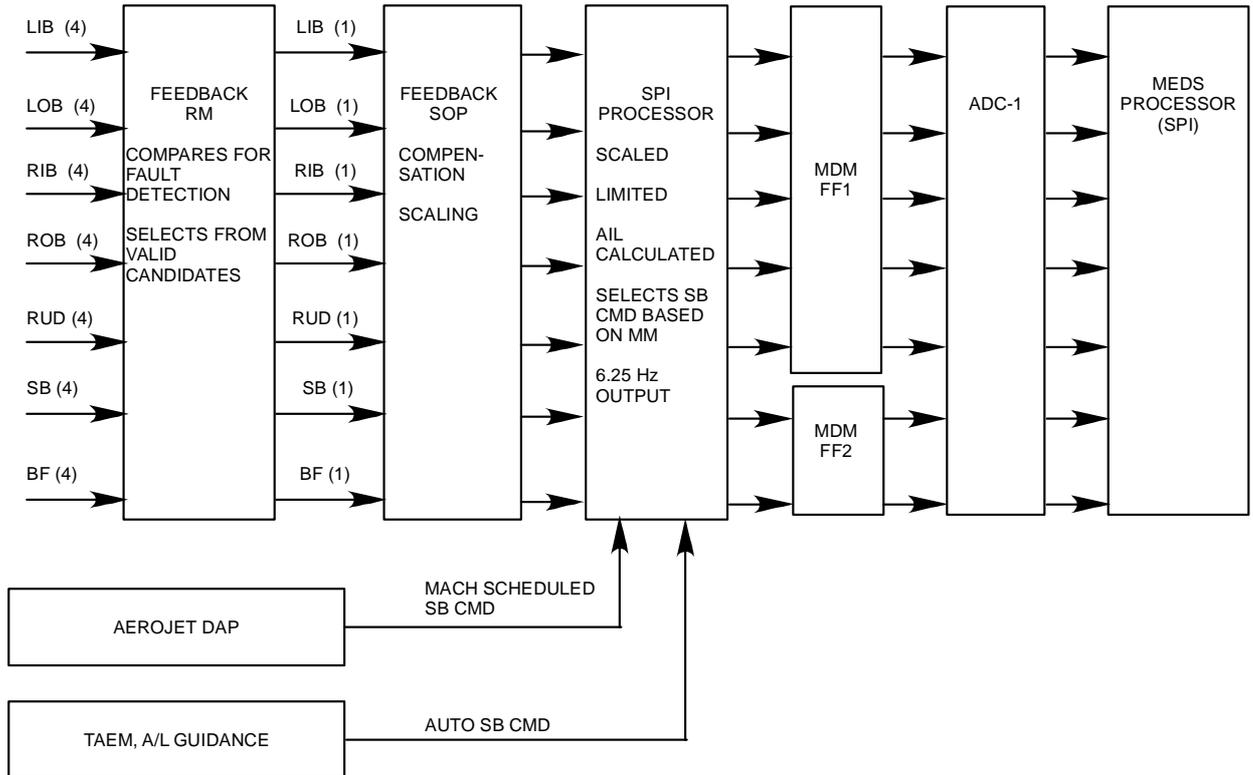


Figure 8 - 91 SPI Data Flow

8.3.2.2 Nominal Data Sequence

SPI processing begins with MM 301 and continues through MM 305. The speedbrake command is driven to zero prior to MM 304. During MM 304, the speedbrake command is the Mach-scheduled auto command from the DAP. During MM 305, it changes to the guidance auto speedbrake command. The SPI is also active during the GNC OPS 8 dedicated display checkout.

8.3.2.3 Abort Processing

The SPI is active during OPS 6. Processing during the GRTLs sequence (MM's 602, 603) is identical to that during the nominal entry sequence (MM's 304, 305), except that the guidance auto speedbrake command is displayed throughout MM 602, 603. (The Mach-scheduled speedbrake command is not used.)

8.3.2.4 BFS SPI Processing

There is no difference in the SPI definitions between BFS and PASS. The aerosurface positions are still the selected values from the four feedback candidates from each surface, and the speedbrake command is still the auto command (Mach-scheduled speedbrake (SB) CMD during entry; auto guidance SB CMD during TAEM). The BFS SPI differs from PASS in that the speedbrake command is driven to zero during A/L (no A/L guidance), requiring a manual speedbrake setting.

8.4 RCS Command Lights

The RCS COMMAND lights (Figure 8 - 92) on panel F6 are active during OPS 2, 8, 3, and RTLS flight phases. Their primary function is to indicate RCS jet commands by axis and direction, and secondary functions in OPS 3 and RTLS are to indicate when more than two yaw jets are commanded and when

the elevator or aileron drive rate is saturated. The RCS commands are stretched to provide increased visibility. Table 8 - 19 summarizes the light functions.

ROLL - During MM's 201, 202, 801, 301-305, 602, and 603 up until the roll jets are no longer commanded (dynamic pressure exceeds 10 lb/ft²), the ROLL L and R lights indicate that left or right roll commands have been issued by the DAP. In OPS 3 or OPS 6 the minimum light-on duration is extended to allow the light to be seen even for minimum impulse firings. When q-bar ≥ 10 lb/ft², the ROLL lights will be quiescent until 50 lb/ft², after which time both lights will be lit whenever more than two yaw jets are commanded on.

PITCH - The PITCH U and D lights indicate up and down pitch jet commands until q-bar = 20 lb/ft² in OPS 1 or q-bar ≥ 40 lb/ft² in OPS 3, after which the pitch jets are no longer used. When q-bar ≥ 50 lb/ft², the PITCH lights, like the ROLL lights, assume a new function: both light whenever the elevon surface drive rate exceeds 20 deg/s (10 deg/s if only one hydraulics system is left).

YAW - The YAW L and R lights will function as yaw jet command indicators throughout entry until the yaw jets are disabled at Mach 1. The YAW lights have no other functions.

Table 8 - 19 RCS Command Light Function

| OPS Mode or Entry Event | RCS Command Light | | |
|--|--|--|------------------------------------|
| | Roll L/R | Pich U/D | Yaw L/R |
| MM 201, 202, 801, 301, 302, 303 | On if left/right roll jet commanded | On if up/down pitch jet commanded | On if left/right yaw jet commanded |
| Q ≥ 10 lb/ft ² | Off | Off | |
| Q ≥ 20 lb/ft ² (OPS 1) or Q ≥ 40 lb/ft ² (OPS 3) | | | |
| Q ≥ 50 lb/ft ² | Both L/R ROLL on if > 2 yaw jets commanded at a time | Both U/D PITCH on if elevon drive rate > 20°/sec (10°/sec if single APU) | |
| M < 1 | Off | | Off |

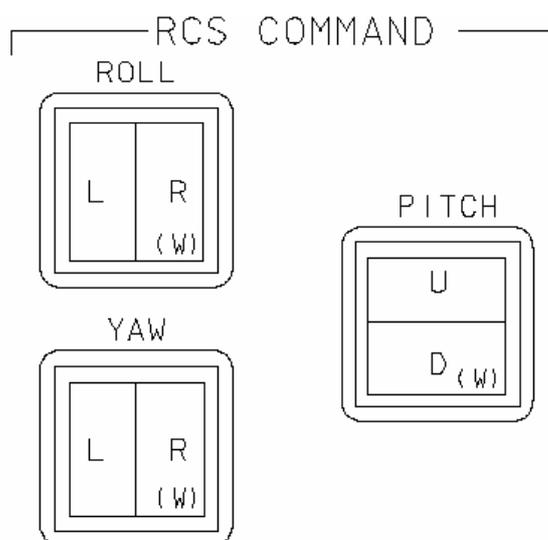


Figure 8 - 92 RCS Command Lights

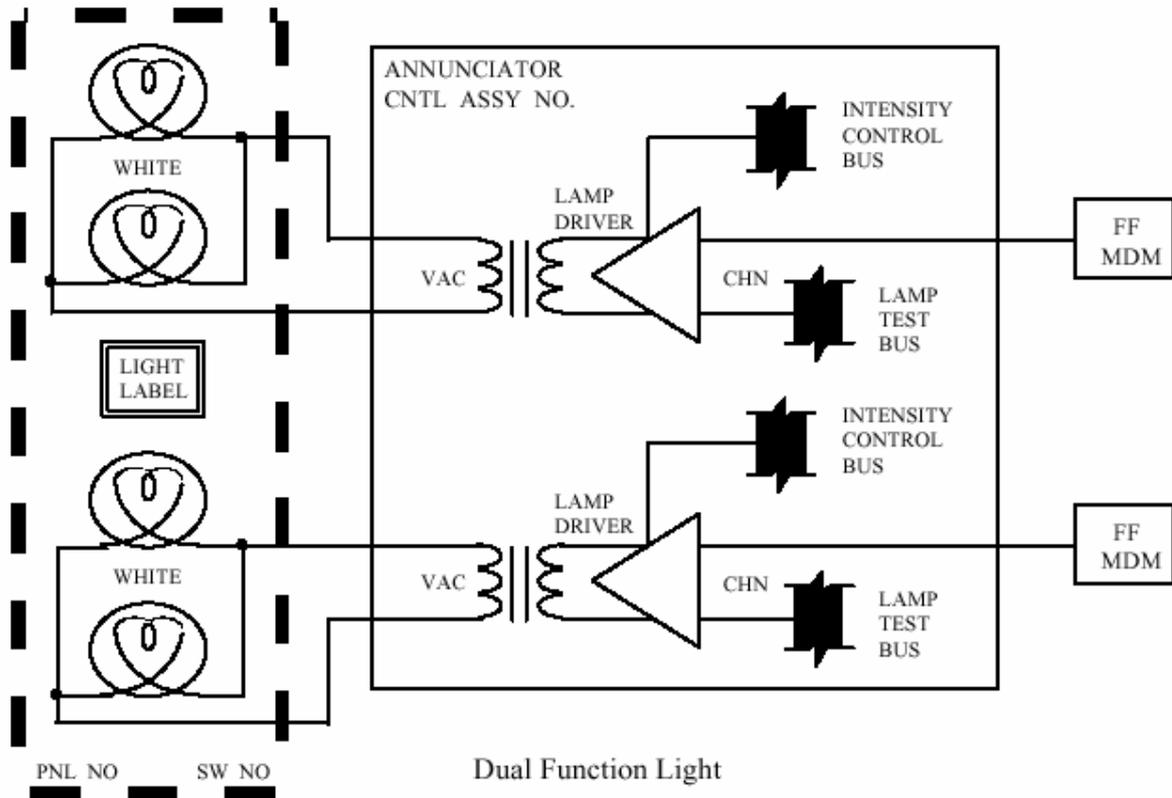


Figure 8 - 93 RCS Command Light Functional Diagram

8.5 FCS Mode Pushbutton Indicators

The FCS mode PBI's (Figure 8 - 94) transmit crew moding requests to the DAP in flight control software and reflect by illumination the effective DAP state. These lights are processed during OPS 1 (ascent), OPS 6 (RTLs), and OPS 3 (entry).

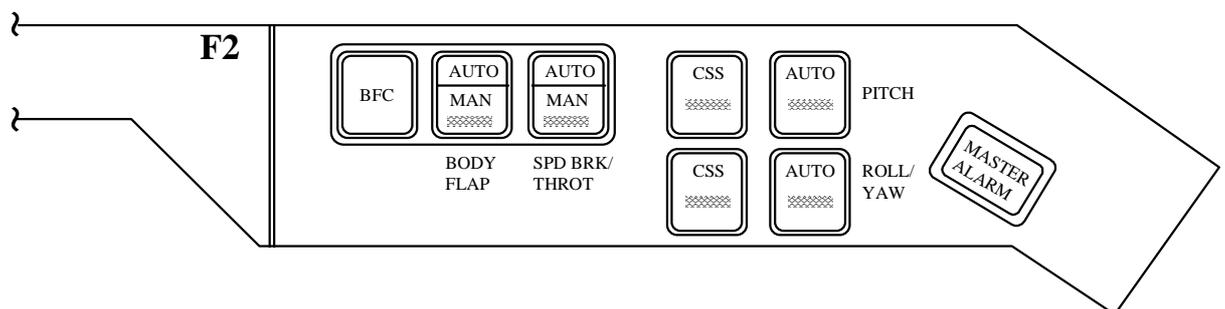


Figure 8 - 94 FCS Mode PBI's (panel F2)

PITCH and ROLL/YAW AUTO, control stick steering (CSS) PBI's - The PITCH and ROLL/YAW PBI's transmit moding requests to the DAP and indicate the effective state of the pitch, roll, and yaw DAP channels by lighting.

AUTO indicates that the control is automatic and that no crew inputs are required for control. For CSS is crew inputs are required but will be smoothed by the DAP (stability augmentation, turn coordination).

SPD BRK/THROT - The SPD BRK/THROT PBI has two separate lights, AUTO and MAN. The lights indicate that the DAP speedbrake channel is in the auto or manual mode. The PBI transmits only the AUT request.

BODY FLAP - The BODY FLAP PBI also has separate AUT and MAN lights, indicating the state of the body flap channel. Like the SPD BRK/THROT PBI, the BODY FLAP PBI transmits only the AUT request.

The left and right PBI light logic is summarized in Table 8 - 20, noting for each PBI the cause of the light and whether one or both lights illuminate.

Table 8 - 20 FCS Mode PBI Light Logic

| Lights | Left/right light logic | | Causes |
|---------------|------------------------|------------------------------|---|
| Pitch AUTO | Both on or both off | AUTO, CSS mutually exclusive | Pitch AUTO PBI depressed |
| Pitch CSS | Both on or both off | | Pitch CSS PBI depressed <u>or</u> RHC out of detent in pitch |
| Roll/Yaw AUTO | Both on or both off | AUTO, CSS mutually exclusive | ROLL/YAW AUTO PBI depressed |
| Roll/Yaw CSS | Both on or both off | | ROLL/YAW CSS PBI depressed <u>or</u> out of detent in roll or yaw |
| SPD BK/THROT | | | |
| AUT | Both on or both off | AUT, MAN mutually exclusive | Neither SBTC out of detent <u>and</u> PBI is then depressed |
| MAN | One on, one off | | Either SBTC out of detent |
| Body Flap | | | |
| AUT | Both on or both off | AUT, MAN mutually exclusive | AUTO PBI depressed and neither slew switch active Either slew switch active or inactive (AUTO/OFF) and neither AUTO PBI depressed |
| MAN | Both on or both off | | |

8.6 Documents & Resources

MEDS Display Requirements

Software Requirements Specification (SRS) For The Integrated Display Processor (IDP) Software Of The Multifunction Electronic Display Subsystem (MEDS)
MG070100A1012E2, Rev. K

Displays and Controls

Level C Functional Subsystem Software Requirements (FSSR) displays and controls

(a) STS 83-0020C, Appendix F

Principal functions

- 4.105 AMI processing
- 4.106 AVVI processing
- 4.168 ADI ascent processing
- 4.169 ADI orbit processing
- 4.103 ADI entry/landing processing
- 4.104 HSI processing
- 4.107 SPI processing
- 4.108 Event light processing
- 4.230 RCS activity light processing

(b) STS 81-0020A, Appendix G - Contains all input/output tables which GNC Interface Definition Document, VP70710043POOL list all parameters by measurement stimulus identification (MSID) and source/destination

(c) JSC-19350, STS Flight Software Initialization Load, volume 1 - Contains all I-load parameters and engineering values for each mission plus principal function and MSID numbers.

(d) Space Shuttle programs Orbiter avionics software OPS detailed design specification (DDS) - The applicable flight software code can be found on software CD's using the three-letter module names below.

AMI/AVVI

- GDJ - Entry/RTLS AMI/AVVI non-ADTA processor
- GDZ - Entry/RTLS AMI AVVI processing of ADTA data
- GDX - AMI/AVVI formatter
- GDY - Ascent AMI/AVVI processor

ADI

- GDA - Ascent ADI D/D
- GDF - Orbit ADI D/D
- GDG - Deorbit ADI processor
- GDE - E/L ADI processor

HSI

- GDW - Ascent HSI processing
- GDO - RTLS HSI 3.125 Hz processing
- GDN - Entry HSI 3.125 Hz processing

SPI

- GDM - Surface position indicator processing

Entry Light

- GDR - Entry event lights processing

RCS Activity Lights

- GFB - Transition DAP RCS processor
- GFF.2 - On-orbit phase plane and jet select block
- GFI - GRTLS DAP priority rate limiting and rudder logic
- GFM - Aerojet DAP 12.5 Hz JET SEL logic

Table 8 - 21 Major Mode is a list of the displays and the applicable MM's.

Table 8 - 21 Major Mode

| Dedicated display | P.F. | O | 1 | 1 | 1 | 1 | 1 | 1 | 6 | 6 | 6 | 2 | 2 | O | 3 | 3 | 3 | 3 | 3 | |
|-------------------|-------|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | P | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | P | 0 | 0 | 0 | 0 | 0 |
| | | S | 1 | 2 | 3 | 4 | 5 | 6 | 1 | 2 | 3 | 1 | 2 | S | 1 | 2 | 3 | 4 | 5 | |
| | | 9 | | | | | | | | | | | | 8 | | | | | | |
| AMI | 4.105 | | | X | X | | | | X | X | X | | | | | | | | X | X |

| | | | | | | | | | | | | | | | | | | |
|--------------------|-------|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Alpha | | | X | X | | | | X | X | X | | | | | | | X | X |
| ACCEL | | | X | X | | | | X | X | X | | | | | | | X | X |
| M/VEL | | | X | X | | | | X | X | X | | | | | | | X | X |
| EAS | | | X | X | | | | X | X | X | | | | | | | X | X |
| AVVI | 4.106 | | X | X | | | | | X | X | | | | | | | X | X |
| ALT ACCEL | | | | | | | | | X | X | | | | | | | X | X |
| ALT rate | | | X | X | | | | X | X | X | | | | | | | X | X |
| ALT | | | | X | | | | X | X | X | | | | | | | X | X |
| RDR ALT | 4.168 | | | | | | | | X | X | | | | | | | X | X |
| ADI ascent | 4.169 | X | X | X | X | X | X | X | | | | | | | | | | |
| ADI orbit | 4.103 | | | | | | | | | | X | X | X | | | | | |
| ADI ENT/LDG | | | | | | | | | X | X | | | | X | X | X | X | X |
| R,P,Y ERR | | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| R,P,Y rate | | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| R,P,Y ATT | | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| HSI | 4.104 | X | X | X | | | | X | X | X | | | | | | | X | X |
| SEL course | | X | X | X | | | | X | X | X | | | | | | | X | X |
| Heading | | X | X | X | | | | X | X | X | | | | | | | X | X |
| PRI BRG | | X | X | X | | | | X | X | X | | | | | | | X | X |
| SEC BRG | | X | X | X | | | | X | X | X | | | | | | | X | X |
| PRI DIST | | X | X | X | | | | X | X | X | | | | | | | X | X |
| SEC DIST | | X | X | X | | | | X | X | X | | | | | | | X | X |
| Course DEV | | X | X | X | | | | X | X | X | | | | | | | X | X |
| Glide slope DEV | | | | | | | | | X | X | | | | | | | | |
| SPI | 4.107 | | X | X | | | | X | X | X | | | | | | | X | X |
| L ORD ELEVX | | | | | | | | | X | X | | | X | X | X | X | X | X |
| L IBD ELEV | | | | | | | | | X | X | | | X | X | X | X | X | X |
| R IBD ELEV | | | | | | | | | X | X | | | X | X | X | X | X | X |
| R ORD ELEV | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Body FLP | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Rudder | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Aileron | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Speedbrake CMD | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Speedbrake POSN | | | | | | | | | X | X | | | X | X | X | X | X | X |
| Event SEQ lights | 4.108 | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X | X |
| RCS command lights | 4.230 | | | | | | | | X | X | X | X | X | X | X | X | X | X |

(3) JSC-08220-3, Master Measurement List - Contains all telemetry (TLM) measurement numbers and data

(4) JSC 16486-5, Shuttle Calibration Data Book - Contains calibration data used in the Mission Control Center (MCC) for all TLM

Attitude Processor

Space Shuttle Orbiter Operational Level C Functional Subsystem Software Requirements
Guidance, Navigation And Control Part C
Flight Control–Ascent Flight Phase Volume 1
STS 83–0008V1–30

Space Shuttle Orbiter Operational Level C Functional Subsystem Software Requirements
Guidance, Navigation And Control Part C
Flight Control Orbit Dap
STS 83-0009-30

Area NAV

Space Shuttle Orbiter Operational Level C Functional Subsystem Software Requirements
Guidance, Navigation And Control Part B
Entry Through Landing Navigation
STS 83-0004-30

BFS

Backup Flight System (BFS)
GN&C Guidance Navigation
Program Requirements Document (PRD)
MG038104-30

Backup Flight System (BFS)
GN&C Hardware Interface
Program Requirements Document (PRD)
MG038106

8.12 FLIGHT SOFTWARE WAIVERS AND OPS NOTES

Several flight software discrepancies which affect the dedicated displays are described in the following paragraphs. These conditions have associated operational notes and waivers which are referenced by an associated discrepancy report (DR) number.

- A. PASS DR 10137 - AVVI ALTITUDE ACCELERATION INCORRECT DURING MM603 TAEM. This waiver and program note explains a condition that affects the altitude, altitude rate, and the vertical acceleration tapes of the AVVI. The Glide User Parameter Processor (UPP) calculates these parameters at 6.2 Hz in MM603. During the TAEM phase (Mach <3.2) of entry, the altitude and altitude rate are in error 5 out of every 6 cycles. This results in an altitude error of up to 75 ft and an altitude rate error of no more than 1 ft/sec. The vertical acceleration is in error during the entire TAEM phase. This is not considered to present an operational concern based on the granularity of the mechanical displays and the display requirements for manual flight control. Furthermore, the user response is reasonable in that it states that the crew can infer vertical acceleration from the rate of change of altitude rate.
- B. PASS DR 102403 - INCORRECT INITIALIZATION IN GDS CAUSES INDEX OUT OF RANGE. This waiver and program note explains a problem with software initialization of two simplex display switches. The affected switches are the AIR DATA switch (on panels F6, F8) and the HSI SELECT SOURCE switch for selecting transceiver source (on panels F6, F8). When switch inputs are invalid on the first pass in MM304 or MM602, the corresponding switch position outputs are set to incorrect values. A simplex switch output is invalid if none or more than one position is indicated. The resultant effect is that the associated data shown on the AVVI, AMI, ADI, and the HSI displays will be unreliable for about 2 seconds. The display switch processing software requires errors on 3 consecutive cycle (»2 sec) before it assigns a display switch to its default position. No user action is required because the problem is corrected when the switch positions are set to their default values. GPC overflow and underflow errors are possible although none have shown up in testing. This problem cannot cause GPC fail-to-sync errors.
- C. BFS DR 101893 - DDU HSI COURSE DEVIATION INDICATOR. This waiver and program note explains a problem with HSI CDI which causes it to swing from one limit to the other and back within one major cycle. (Information driving the CDI is computed at 1 Hz.) The transient condition happens when a deviation extreme is approached very rapidly. This phenomenon results because of an overflow in the extrapolation process that causes the sign bit to change when the deviation is changing very fast. The user response informs the crew to ignore a full range, one cycle swing of the CDI if it happens within 1 second.

SECTION 9

HEAD UP DISPLAY

9.1 GENERAL

The head up display (HUD) system is designed to provide the crew with a display of flight-critical information with minimum transitions from eyes out the window (head up).

The information is provided by the general purpose computer (GPC) and passed to a special processor in the head up display electronics (HUDE) during major modes (MM's) 304, 305, 602, and 603. However, the primary focus is the approach and landing (A/L) phase of MM 305 and MM 603 during which runway-related information can be displayed on the HUD to aid the crew in performing a precise landing.

The HUD will be available in the backup flight system (BFS) for OI-32 and subsequent software.

As a secondary function, and due to increased repeatability, the HUD has replaced the -Z/+X COAS as the main backup device to the star trackers for performing IMU aligns (see GNC SCP 3.4 HUD/COAS CALIBRATION COORDINATION AND ANALYSIS for more details).

9.2 HARDWARE OPERATION

Two HUD systems are installed in each Orbiter. Each system consists of the HUDE and a pilot display unit (PDU). The two systems are identical and independent of each other to the GPC data source level.

9.2.1 HUDE

The HUDE's are 16K x 24 programmable read-only memory (PROM) miniprocessors that are physically located in forward avionics bay 3B. The HUDE's process data, generate display symbols, and provide deflection voltages for symbol positioning on the PDU. An exploded view of the HUDE is shown in figure 9-1.

The data required to support the HUDE's consist of three types: HUD unique, dedicated display data, and 28 V discrete data. All of the data except the 28 V discrete data are acquired from the GPC.

HUD unique data are either gathered by the GPC HUD principal function module, from other modules within the GPC, or are delivered directly to the HUD output buffer by the originating GPC module. Data gathered by the HUD principal function module are processed, scaled, and integerized as required therein for HUD application and then sent to the output buffer. HUD data in the output buffer are then controlled by the input output processor (IOP) for delivery to the HUDE's across the flight-critical (FC) display driver unit (DDU) data buses FC1 to 4.

Dedicated display data used by the HUDE's are acquired directly from the FC1 to 4 DDU buses. The HUDE's and dedicated displays share the FC DDU buses. Data which are common to both the DDU's and the HUDE's are addressed to and received by both. Similarly, data for HUDE use only or DDU use only are addressed to the user and rejected by the nonuser.

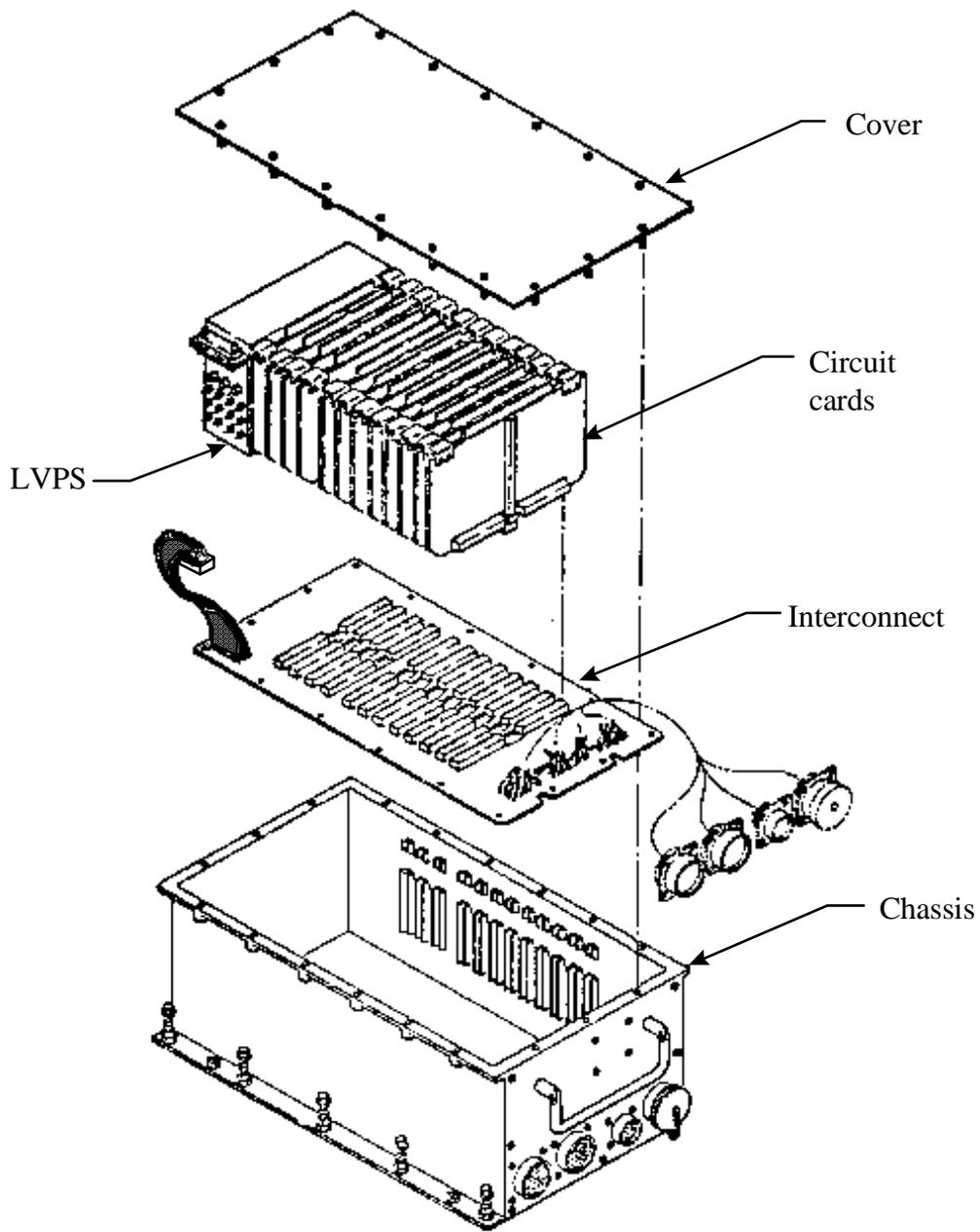


Figure 9-1. HUD exploded view.

The commander's (CDR's) HUDE receives GPC data from FC1 and 2 buses only, and the pilot's (PLT's) HUDE receives GPC data from FC3 and 4 buses only. This distinction should be noted in that crew selection of the flight control FC data bus for HUD connection is not quad redundant. The following table illustrates the available selections.

| CDR Data Bus sw pos | CDR HUD connection | PLT Data Bus sw pos | PLT HUD connection |
|---------------------|--------------------|---------------------|--------------------|
| FC1 | FC1 | FC1 | FC3 |
| FC2 | FC2 | FC2 | FC4 |
| FC3 | FC1 | FC3 | FC3 |
| FC4 | FC2 | FC4 | FC4 |

The 28 V discrete data used by the HUDE's is acquired via hardwire directly from the data source and consists of the landing gear down discrettes.

9.2.2 PDU

The PDU's are located in the instrument panels in front of the CDR and PLT. The PDU's consist of an elongated box assembly which contains a high voltage power source, X- and Y-deflection amplifiers, a small 4-in. exit diameter cathode ray tube (CRT), and a series of mirrors and lenses. Affixed to the top of the box is a dual combiner lens assembly positioned in the line of sight from the pilot's eye through the forward windows. The PDU is shown in figures 9-2 and 9-3.

Symbol and symbol deflection voltages from the HUDE's are used to generate and position the HUD symbology in the CRT. The CRT images are then transmitted through the series of mirrors and lenses which focus them at infinity and project them on the combiner lenses. The PDU optics chassis is shown in figure 9-4. These lenses reflect the images towards the pilot's eyes while passing outside light rays. Since the images are focused at infinity, pilot's head motion does not affect symbol position with respect to the outside light rays.

The dual combiner lens assembly provides increased vertical instantaneous field of view (IFOV) over that obtainable in a single combiner lens installation using the same size optics. This feature is important to the Orbiter HUD since large angle of attack changes are experienced between terminal area energy management (TAEM) and touchdown flight attitudes. The dual combiner assembly accommodates these variations without requiring the pilot to move his head to maintain the HUD symbology in his FOV.

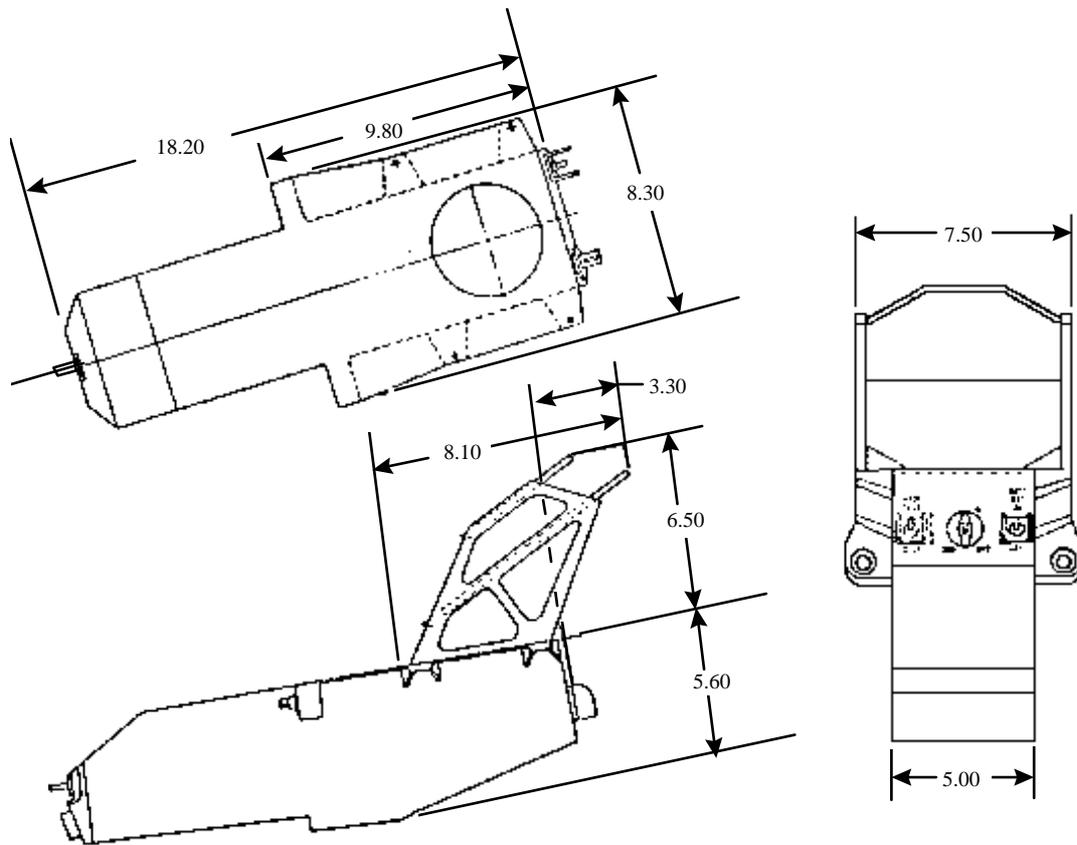


Figure 9-2.- PDU installation

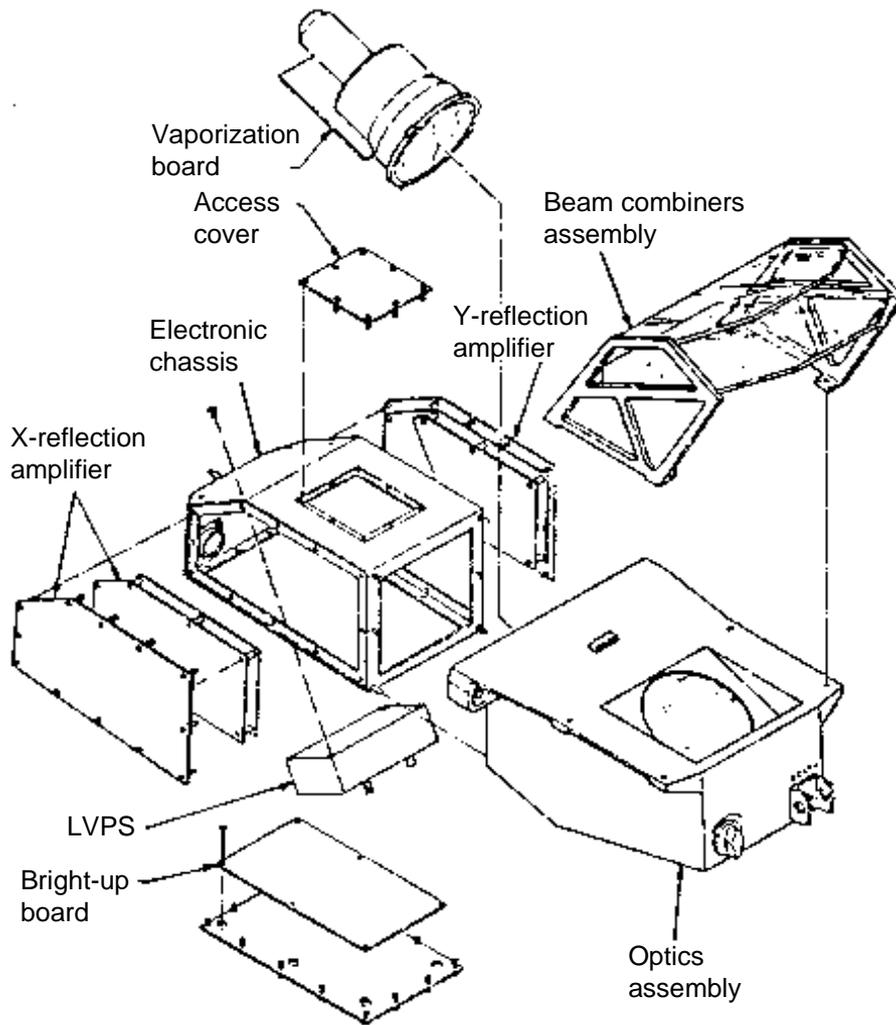


Figure 9-3.- Pilot's display unit.

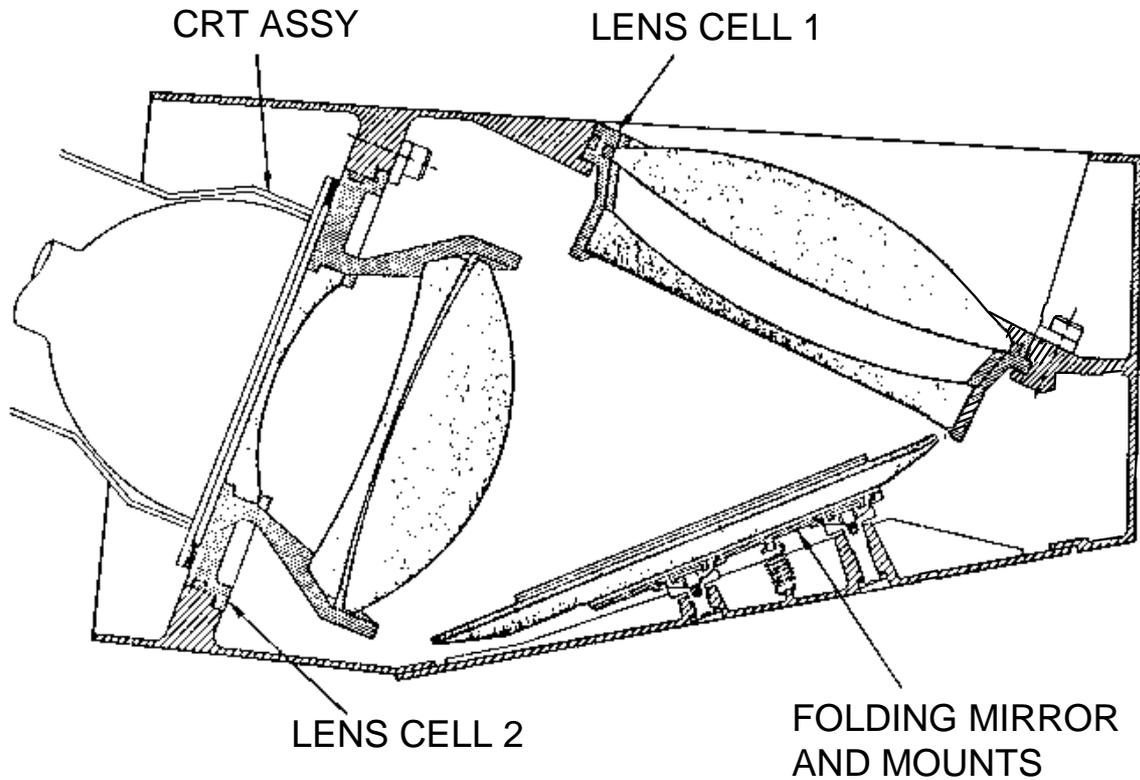


Figure 9-4.- PDU optics chassis.

9.2.3 System Control

9.2.3.1 Powerup

HUD system powerup is provided through a HUD POWER switch mounted inboard of the PDU on panel F3 (see SSSH Section 20). Power for the CDR's HUD is from main A through control AB2. Power for the PLT's HUD is from main C through control BC1.

9.2.3.2 Moding

A three-position MODE switch is located on the front of the PDU. Switch positions are up, TEST; center, NORM; and down, declutter (DCLT). The DCLT position is momentary; the switch is spring-loaded to return from that position.

In the NORM position, automatic sequencing of formats and symbology within formats is provided.

Activation of the MODE switch to the up, TEST, position forces up a test display on the PDU. No GPC is required to support this test format. Proper functioning of the test display will verify HUDE processor and PDU display operation. Repositioning the switch to the center, NORM, position will resume the interrupted format.

The symbology that is displayed in a 5-second (sec) period is shown in figures 9-19 through 9-22 in Section 9.5.3, System Test. The symbology rotates during this 5-sec period. At the end of this period, figure 9-22 remains on the HUD as long as the MODE switch is in TEST. Proper functioning of the test display will verify HUDE processing and PDU display operation.

Activation of the MODE switch to the down DCLT position initiates a symbol blanking routine. This routine selectively removes different symbols from the displayed format. Successive operations of the DCLT switch serially removes display elements in accordance with a preprogrammed logic until only the boresight symbol remains. The next operation of the switch will then reintroduce the existing format in its entirety and reinitialize the declutter logic. Declutter reinitialization can also be accomplished by manually forcing up a new format through the horizontal situation display (HSD) HUD item entry. The declutter logic prior to touchdown (WOWLON flag set) is as follows:

| <u>Declutter level</u> | <u>Result</u> |
|------------------------|---|
| 0 | Nominal |
| 1 | Blanks runway symbol |
| 2 | Blanks airspeed scale, altitude tape, glideslope indication, attitude scales. Forces up digital airspeed and altitude |
| 3 | Only the boresight remains |
| 4 | Reintroduces full up display and resets declutter level to 0 |

After weight on wheels (WOW), declutter logic is as follows:

| <u>Declutter level</u> | <u>Result</u> |
|------------------------|--|
| 0 | Nominal |
| 1 | Blanks attitude scales and horizon |
| 2 | Blanks deceleration rate, digital/ground airspeed, and speedbrake (only the boresight remains) |
| 3 | Reintroduces full up display and resets declutter level to 0 |

9.2.3.3 Brightness

The pilot can select the brightness of the HUD symbology using the rotary brightness control on the front of the PDU. The range of the brightness control is determined by a three-position switch labeled BRIGHT. In the top (MAN DAY) position, the rotary brightness control covers the full range of HUD brightness. In the bottom (MAN NIGHT) position, the rotary control covers the range from 0 to 2 percent of HUD brightness. In the center (AUTO) position, the rotary control sets a relative contrast level between the HUD symbology and the outside light conditions. The chosen contrast level is maintained automatically, using an outside light sensor mounted on the PDU forward of the combiner lens. This sensor is aligned to view the same scene that the pilot sees through the HUD.

9.3 DISPLAY DESCRIPTION

The HUD symbology has been designed to provide the pilot with the information required to accomplish precise and repeatable Orbiter approach and landings. Symbology sequencing is used to avoid display clutter and to provide an orderly progression from the TAEM flight phase to postlanding rollout. The symbology integrates the real-world scene with parametric information thereby enhancing information transmittal.

Both situation and guidance type information are displayed. Data source commonality between the dedicated displays and the HUD has been observed where practical. When data sources differ, the source change has been made to acquire data for the HUD at a higher update rate than used by the dedicated displays.

9.3.1 Formats

Provisions have been made in the GPC-to-HUD system interfaces to provide for a total of seven formats. Currently, two formats have been defined for the ASCENT (OPS 1/6) and ENTRY (OPS 3) configurations. These formats are as follows.

| <u>Format</u> | <u>Function</u> |
|---------------|-------------------|
| 0 | Blank |
| 1 | Approach and land |
| 2 | N/A |
| 3 | N/A |
| 4 | N/A |
| 5 | N/A |
| 6 | Test |

(Note that Display Format 2 used to be a function called Autoland Monitor, but that has been deleted from flight software. The FSSR still retains information on this display format as reference only.)

Display format 1 is presented in figure 9-5 for pre-WOW and in figure 9-6 for after WOW.

Note: This figure is not a valid flight configuration

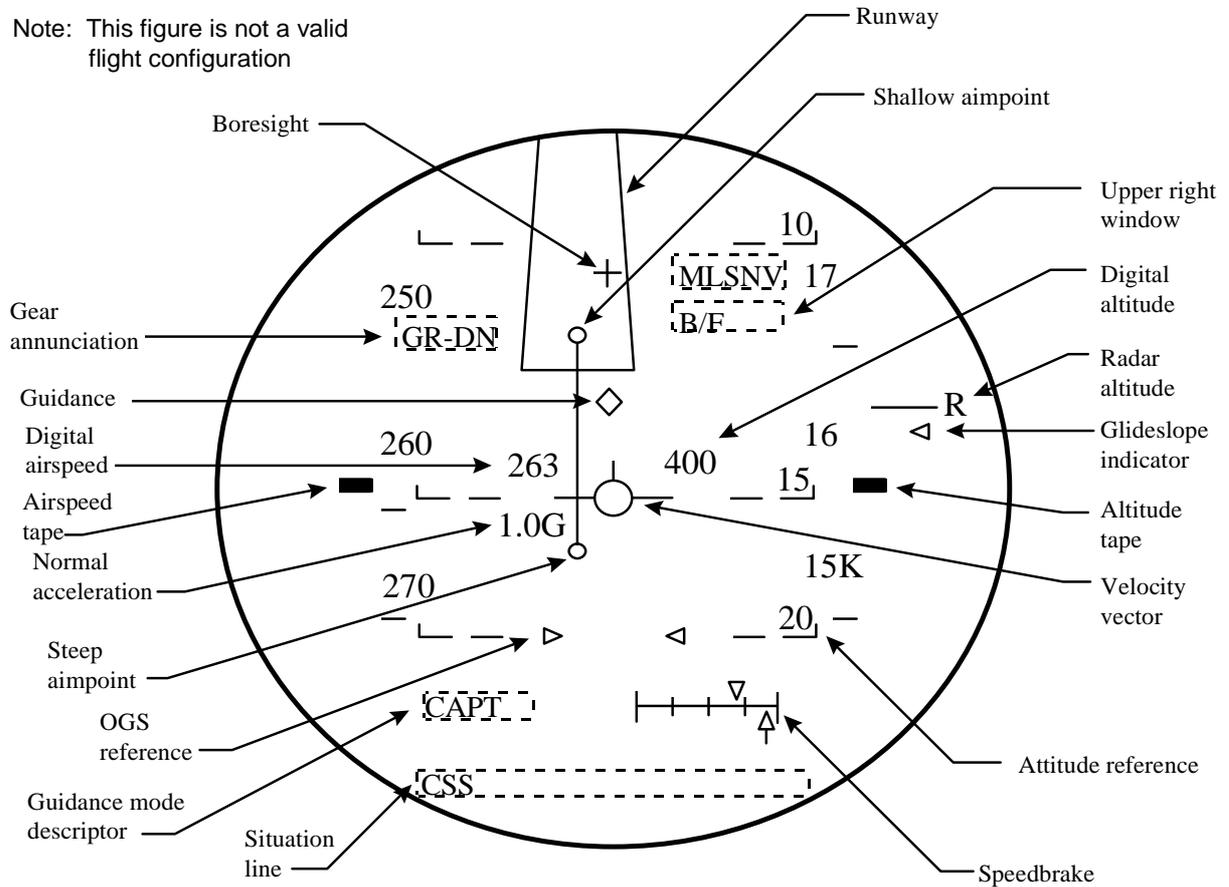


Figure 9-5. Display Format 1 pre-WOW.

Note: This is not a valid flight configuration

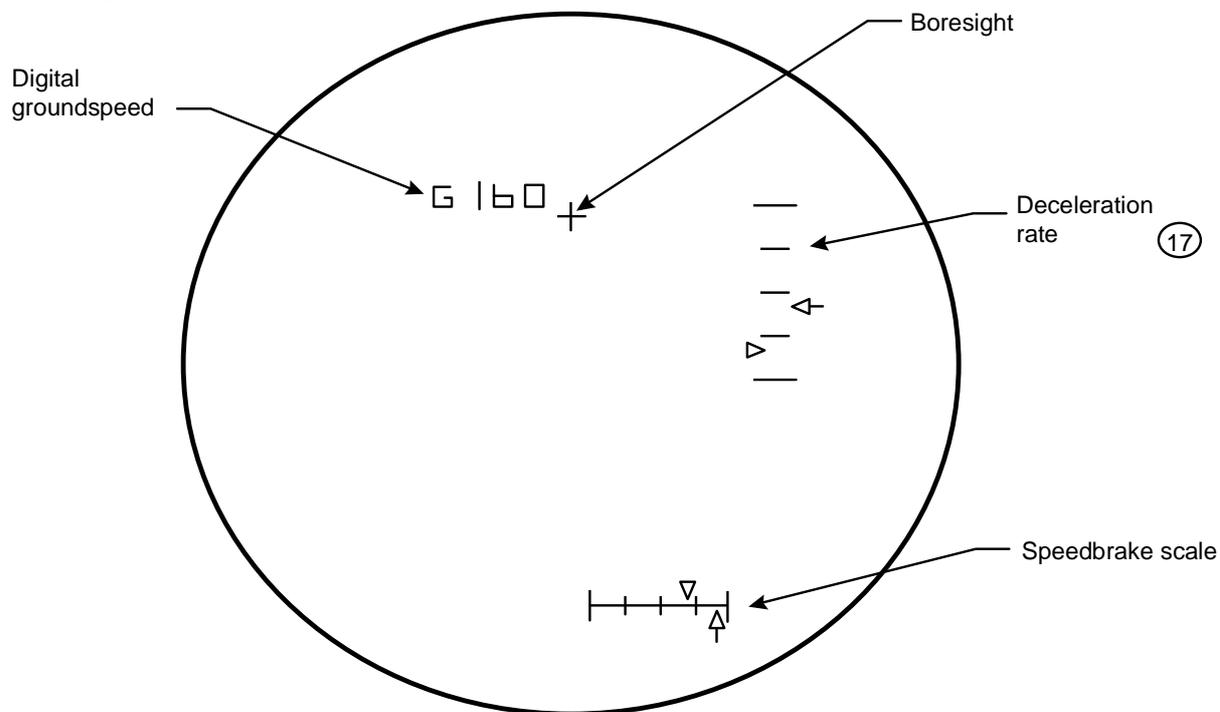


Figure 9-6. Display Format 1 after WOW.

9.3.2 Approach and Land

The symbology displayed in format 1 (see figure 9-7 and figure 9-8) has been designed to provide the pilot with the information required to accomplish precise and repeatable Orbiter A/L's. Symbology sequencing is used to avoid display clutter and to provide an orderly progression from the TAEM flight phase to postlanding rollout. The display integrates the real-world scene with parametric information, thereby enhancing information transmittal. Both situation and guidance type information are displayed. Data source commonality between the dedicated displays and the HUD has been maintained where practical.

The HUD format in figure 9-7 will automatically be displayed at TAEM interface (MM 305 and MM 603) if the HUD power switch is on. The format can also be called up in MM 304 or MM 602 by item entry on the SPEC 50 display (Item 37 + 1 for the CDR HUD; Item 38 + 1 for the PLT HUD).

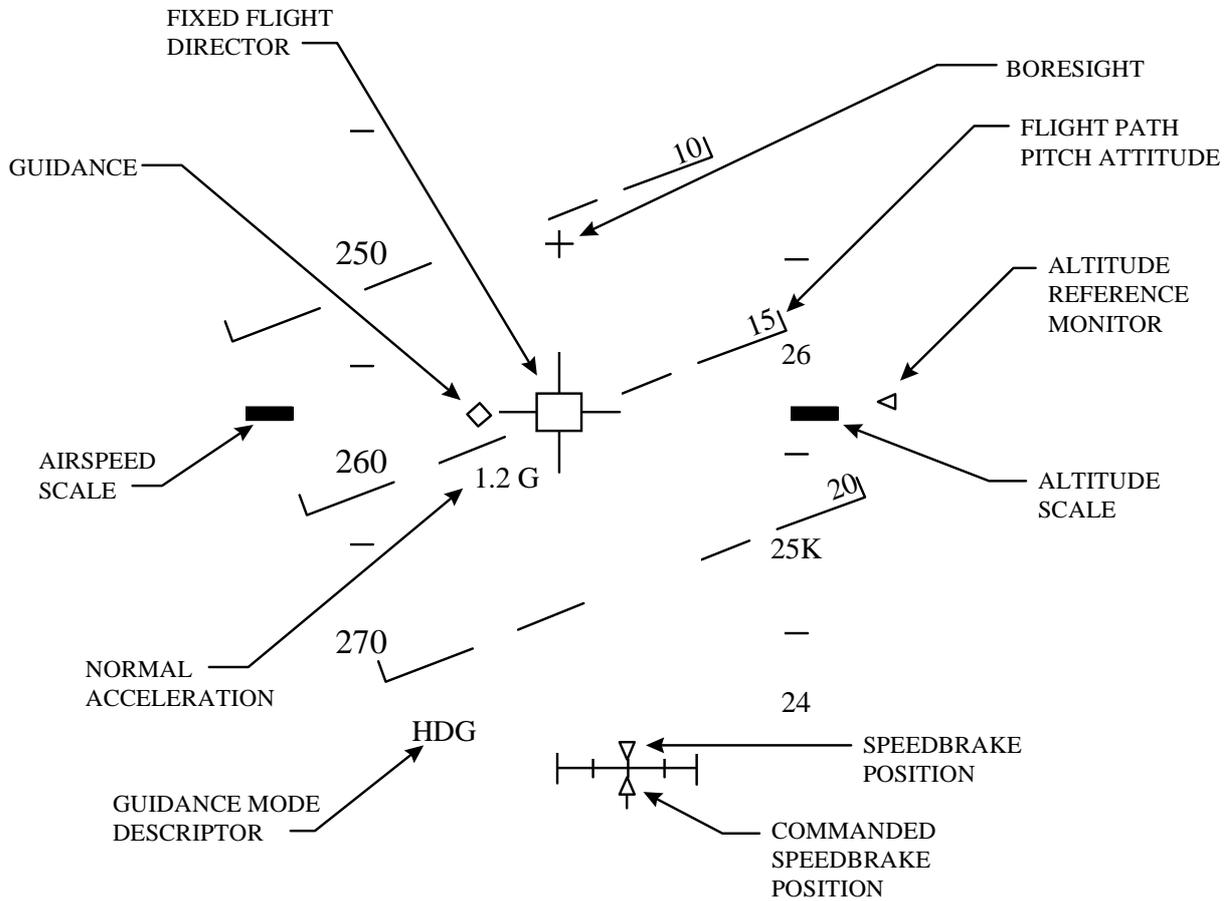


Figure 9-7. Approach and land display (TAEM heading phase).

As the runway for the selected approach enters the HUD FOV, the runway symbology shown in figure 9-8 will be seen to overlay the real runway (assuming navigation (NAV) is correct).

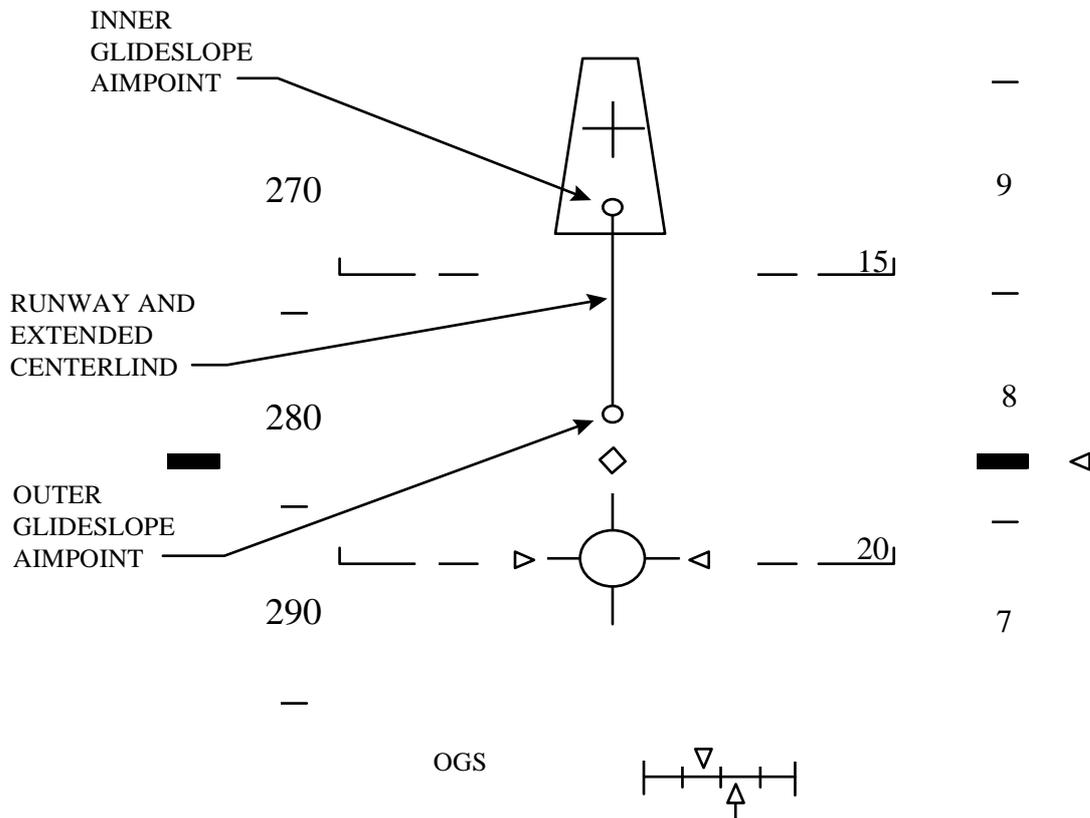


Figure 9-8. HUD field of view, runway symbology.

Activation of the declutter switch modes the display as described in section 9.2.3.2. Figures 9-9 and 9-10 show the results of the first and second activation of the mode switch.

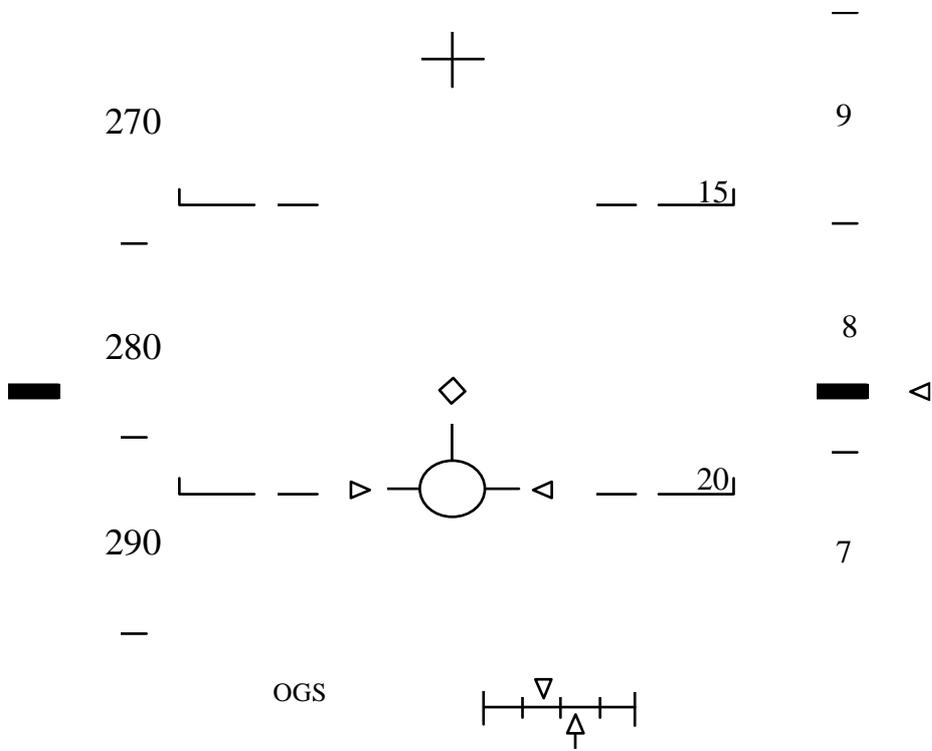


Figure 9-9. First activation of the mode switch.

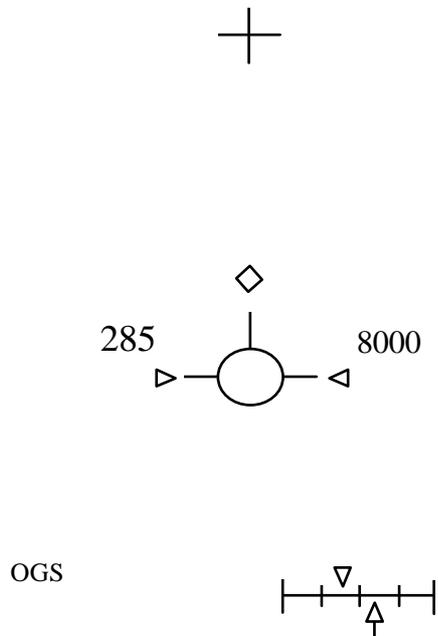


Figure 9-10. Second activation of the mode switch.

9.3.2.1 Boresight

The boresight symbol is a small cross (+). Its position in the display is fixed and depicts an extension of the Orbiter X-axis. The horizontal bar of the symbol remains parallel to and lies on the Orbiter Y-axis at all times. Pitch attitude can be determined by reading the boresight symbol position with respect to the pitch attitude lines. The boresight symbol requires no GPC interface and is never decluttered.

9.3.2.2 Runway Symbology

The runway symbol consists of a four-line drawing of a runway depicting the threshold, sides, and end. The symbol also includes the outer and inner glide slope (OGS, IGS) aim points depicted as small circles and joined together by a line representing the runway and centerline extended. All of the elements of the runway symbol are perspective drawn so as to appear proportionally one to one with the real-world scene whenever that scene is within the HUD FOV. The runway symbol accurately depicts the selected runway length to a maximum of 15,000 ft. That is, if the selected runway length is equal to or less than 15,000 ft, then the runway symbol length will be equal to the actual runway. If the actual runway is greater than 15,000 feet, the runway symbol end line will not overlie the actual runway end line. The maximum length of the runway symbol is 15,000 ft. The runway symbol threshold line will always overlie (no NAV state error) the actual runway threshold line. The runway symbol width is a constant 300 ft. The runway symbol centerline will always (no NAV state error) overlie the actual runway centerline. Any observed position errors between the actual runway scene and the HUD runway symbol are an indication of Orbiter NAV state errors.

The IGS aim point is fixed at 1000 ft. The OGS aim point is I-LOADed at 6500 ft or 7500 ft, depending on crew selection of nominal or closest ground intercept point (ITEM 8) on SPEC 50.

9.3.2.3 Digital Altitude

Digital altitude displays altitude to main wheels with respect to the runway and is forced up at the second declutter level if altitude < 32,767 ft. If the declutter level 1 or 0 display is up, digital altitude is forced up when the altitude is equal to or less than 100 ft.

If an R is displayed after the digits, the altitude is radar. If no R is displayed, the altitude is NAV.

Displayed altitude is truncated at 200 ft (i.e., 8000, 7800, 7600, etc.) to 1000 ft altitude. Altitude is then truncated at 100 ft (1000, 900, 800, etc.) to 400 ft altitude. Truncation is then changed to 10 ft (400, 390, 380, etc.) to 50 ft altitude. Altitude is then truncated at 1 ft from 50 ft to touchdown (50, 49, 48, etc.).

9.3.2.4 Altitude Tape

The altitude tape consists of a fixed rectangle against which a moving scale of altitude of the main gear height above the runway is read. The scale moves in the same direction as the altitude / vertical velocity indicator (AVVI) altitude. Scaling is variable with scaling changes occurring at the same time that changes occur to the AVVI altitude. The displayed altitude is NAV altitude. Numerics are the same as the AVVI with the exception that from 100K to 400K ft they are provided every 10K ft as opposed to every 50K ft on the AVVI. Symbol range is from 0 to 400K ft. The numerical values roll on/off the scale as altitude changes. A smoother has been incorporated above 20,000 ft to increase altitude data rate from 6.25 Hz (on the Orbiter data bus) to 50 Hz (on the HUD). This symbol is displayed until touchdown.

9.3.2.5 Radar Altitude

The radar altitude symbol consists of a double width tick mark (—) and an R character. Radar altitude is read off the altitude scale at the tick mark. Differences between radar altitude and NAV altitude are indicated by the delta between the radar (— R) and NAV altitude. The radar altitude symbol is blanked when radar altitude data is not available. Displayed data depends on the position of the RADAR ALTM switch (1 or 2) on F6/F8. Each HUD is individually selectable.

9.3.2.6 Glide Slope Indicator

The glide slope indicator (GSI) is a triangle facing the altitude scale. The symbol is free to move with respect to the altitude scale. When no error exists between the guidance computed reference trajectory altitude and actual altitude, the triangle will lie exactly adjacent to the altitude rectangle. The GSI is termed a "fly to" indicator. When actual altitude is above the reference trajectory altitude, the triangle will be below the rectangle and vice versa. To correct the altitude error, the pilot must fly the Orbiter towards the triangle. The sense is comparable to the glide slope on the horizontal situation indicator (HSI). The GSI is automatically removed at weight on nose gear (WONG).

The scaling of the altitude reference monitor varies with altitude. Figure 9-11 depicts scaling above 10,000 ft.

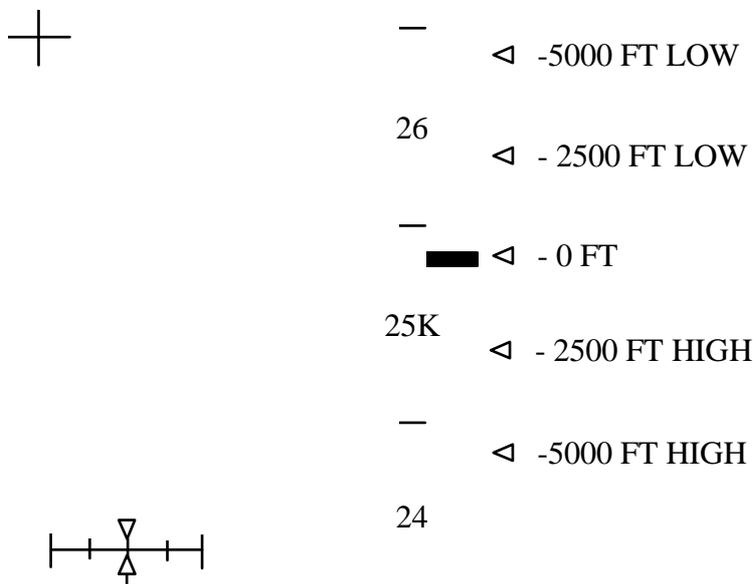


Figure 9-11. Scaling above 10,000 ft.

Below 10,000 feet altitude scaling changes linearly with altitude. Figure 9-12 depicts scaling at 5000 ft.

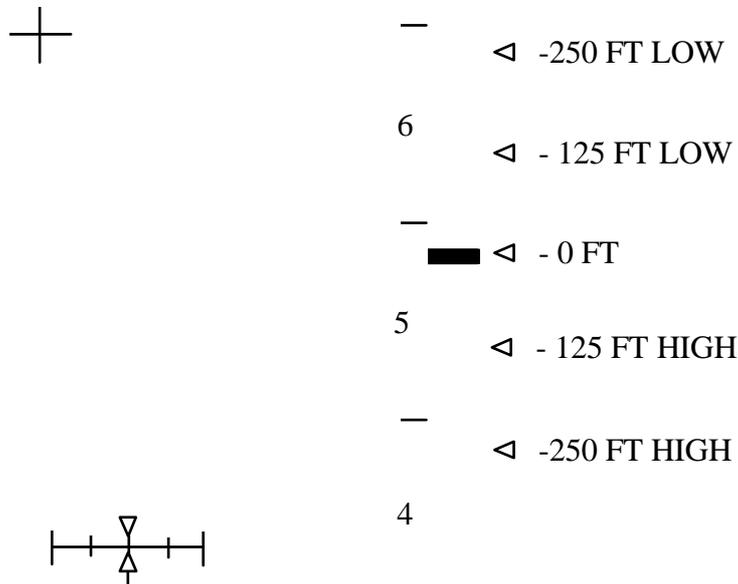


Figure 9-12. Scaling below 10,000 ft.

Below 5000 ft altitude, the scaling remains as shown in figure 9-12 until 5000 ft cross range to the runway threshold. At 5000 ft cross range, scaling changes slightly as a function of cross range.

9.3.2.7 Flight Director/Velocity Vector

Prior to TAEM prefinal (I_PHASE = 3), the HUD has a flight director symbol fixed in the center of the PDU. The flight director symbol consists of a small square with wings and a tail. This symbol can be used with the guidance diamond to null guidance-commanded errors.

At TAEM prefinal, the flight director symbol automatically releases in the pitch axis. The symbol can then be uncaged/caged in the horizontal axis through a GPC parameter controlled by the crew using the ATT REF switch. When uncaged in both axes, the symbol becomes a velocity vector. A 5-sec fader has been incorporated to smooth the transition from flight director to velocity vector.

The velocity vector (V) symbol consists of a small circle with wings and a tail to pictorially represent the Orbiter. The wings of the symbol always remain parallel to the wings of the Orbiter during maneuvering flight. A line-of-sight projection from the pilot's eye through the symbol depicts the instantaneous flight path of the Orbiter. Precise adjustments of flight path can be accomplished as required by overlaying the V symbol on the desired aim point. Wind effects are readily apparent as manifested by the V symbol drifting left/right or up/down on the PDU. Note that the guidance diamond, OGS reference triangles, and digital airspeed (DAS)/altitude all move relative to V position on the HUD. This results in a consistent cross-check on final approach for any wind condition.

A velocity vector filter has been incorporated to prevent velocity vector jitter during banked maneuvers. The filter is removed when flare guidance (P_MODE = 3) is initiated. Velocity data is truncated at 0.1 fps in the HUD to provide a smooth velocity vector for pitch control on final approach.

A pitch quickening term has been incorporated in the V control algorithm to improve the initial response of the symbol to flight control pitch commands. The pitch quickening term effectively negates the translational velocity component of the pitching motion. To the observer, the V symbol responds in the direction of the commanded pitch rate even though the initial response at the Orbiter c.g. to the pitch command might be in the opposite direction.

The V symbol is constrained to remain in the HUD total field of view (TFOV). When a large sideslip angle or angle-of-attack exists which forces the symbol to the limits of the TFOV, further travel of the symbol in the limited direction will be precluded, and an X will be superimposed in the center of the symbol. The X will remain superimposed in the symbol until the sideslip angle or velocity which drove the symbol to the TFOV limits is reduced.

The CDR or PLT can laterally cage the velocity vector in the center of both PDU's by pushing either ATT REF pushbutton. Pushing either ATT REF pushbutton again will provide true velocity vector (V). This feature insures uncluttered guidance, DAS, and digital altitude during crosswind approaches if declutter 2 is not selected.

9.3.2.8 Flight Path/Pitch Attitude Reference Scales

The flight path/pitch attitude scales display 5° increments above and below the horizon. Lines above the horizon are solid with end ticks pointing down toward the horizon. Lines below the horizon are dashed with end ticks pointing up toward the horizon. The numerical value of the scale line is displayed at the right end of each line in the elbow formed by the line and its horizon direction mark. The lines are positioned in the display by reference to the horizon line. This positioning technique insures that the incremental displacement is always with respect to the local horizontal. Scales are presented for +180° to -180° range from the horizon. The scales are occluded from view at the horizontal member of the speedbrake position pointer.

The horizon line and flight path/pitch attitude scales are roll and pitch stabilized so that Orbiter attitude changes cause them to rotate and translate in the FOV. The center of rotation of the entire symbol is about the center of the HUD FOV. The entire flight path/pitch attitude symbol is translated along its vertical centerline as a function of the bank angle.

The translation process insures that the boresight symbol (pitch) and velocity vector symbol (flight path angle after TAEM prefinal) are always accurately displayed with respect to the flight path/pitch attitude scales.

The flight path/pitch attitude scales can also be used to determine angle-of-attack, which can be observed as the absolute value of the scale difference between the boresight and the velocity vector.

9.3.2.9 Speedbrake

The speedbrake symbol consists of a horizontal scale and two opposed pointers. The upper pointer indicates speedbrake position and the lower pointer indicates speedbrake command. The scale is marked in 24.5 percent increments from 0 (on the left side) to 98 percent (on the right side). Zero percent is fully closed. The open/close sense is comparable to the head down speedbrake indicator. The speedbrake command pointer always represents the speedbrake auto command schedule. From Mach

9.8 (I-LOAD) to Mach 0.95 (I-LOAD), a speedbrake position difference greater than 20 percent (I-LOAD) from the auto command schedule will cause the speedbrake position pointer to flash. This occurs with the same logic that displays the CRT SBD BK class III warning.

9.3.2.10 OGS Reference

The OGS reference symbol consists of two opposed triangle indices. The OGS reference triangle automatically appears on the HUD at TAEM prefinal (I_PHASE = 3). The indices are roll stabilized so that a line drawn between them will always remain parallel to the horizon. Their center of rotation is collocated with the flight path/pitch attitude scales center of rotation at the center of the HUD FOV.

The symbol provides flight path angle reference information for OGS acquisition and tracking and for the preflare-to-touchdown flare maneuver. On the OGS, the vertical position of the symbol in the HUD FOV is controlled by the selected OGS flight path angle (currently 17° or 19° (I-LOAD)). The OGS indices are blanked when the flare indices cross them.

9.3.2.11 Normal Acceleration

Normal acceleration is displayed below and outboard of the left wing of the velocity vector. Symbol range is both positive and negative and is to the nearest tenth unit of gravitational force g. A decimal point separates the whole and fractional parts. Leading zeroes are displayed when N_z is less than unity. A minus sign (-) is displayed preceding the digits when N_z is negative. The symbol digits will flash when N_z exceeds the I-LOAD load limit (currently 2.0g). This is the same I-LOAD limit flag that flashes the digital normal acceleration on the HSD. Normal acceleration automatically blanks when guidance modes to prefinal (I_PHASE = 3).

9.3.2.12 Airspeed Scale/Tape

The airspeed scale consists of a fixed rectangle against which a moving scale of equivalent airspeed is read. The scale moves in the same direction as the alpha/Mach indicator (AMI) airspeed. Scaling is constant through the symbol range of 0 to 500 knots. Numerics are provided every 10 knots. A tick mark is used to define 5-knot increments between the numerics. Lower airspeeds are displayed at the top of the scale. A minimum of three airspeed numerics will be displayed at all times. The numerics roll on/off the scale as airspeed changes. A smoother has been incorporated to increase airspeed data rate from 1.04 Hz (on the Orbiter data bus) to 50 Hz (on the HUD).

9.3.2.13 Digital Airspeed

Declutter level 2 forces the DAS to be displayed on the HUD when the altitude is less than 100 ft. DAS is displayed above and slightly outboard the left wing of the velocity vector. The DAS moves with the velocity vector throughout the FOV. At touchdown, (WOWLON flag set), the DAS is repositioned to the upper left corner of the boresight. When WONG has occurred, a G is displayed in front of the DAS to indicate groundspeed.

If the declutter level 1 or 0 display is up, DAS is forced up when the altitude is equal to or less than 100 ft.

9.3.2.14 Guidance

The guidance symbol is a diamond. Its position in the display depicts the direction to which the Orbiter must be flown to satisfy the GPC-derived guidance solution. The diamond represents the intersection of the guidance needles on the attitude direction indicator (ADI). When the flight director symbol overlays the guidance diamond (), the guidance error is zeroed. The symbol is constrained to remain in the TFOV and when limited will flash.

ADI error sensitivity selection provided by the ADI ERROR select switch panels F6/F8 does not change the HUD guidance symbol sensitivity. Error sensitivity in the HUD is slightly less sensitive than ADI error sensitivity with medium scale (MED) selected.

The guidance symbol automatically blanks at final flare (P_MODE = 4) if control stick steering (CSS), all axes, and declutter ³ 1.

9.3.2.15 Guidance Mode Descriptor

The guidance mode descriptor displays guidance modes. The guidance I_PHASE descriptors which appear in the HUD are:

S-TRN for I_PHASE = 0
ACQ for I_PHASE = 1
HDG for I_PHASE = 2
PRFNL for I_PHASE = 3

A thorough discussion of I_PHASE transition criteria and guidance logic is contained in Functional Subsystem Software Requirements (FSSR) document GNC, part A: Entry through Landing Guidance.

The guidance P_MODE descriptors which appear in the HUD are:

CAPT for P_MODE = 1 (Trajectory Capture; starts at the A/L interface and continues until guidance lock-on to the OGS)
OGS for P_MODE = 2
FLARE for P_MODE = 3
FNLFL for P_MODE = 4

Guidance P_MODE logic is in the longitudinal guidance module, pitch trajectory subfunction of A/L GUID (4.10)/GRTL S A/L GUID (4.234).

9.3.2.16 Gear Annunciation

The gear annunciator is set whenever the gear is up and the altitude is less than 300 ft (I-LOAD) and airspeed is less than 300 knots (I-LOAD). When set, the mnemonic GEAR flashes. The //GR// mnemonic is set when the gear is in transit.

The GR-DN mnemonic is set when all three main landing gears are down and locked as indicated by 28-V discretes from the landing gears proximity sensor packages no. 1 and no. 2. If any one or more of the gear fail to complete the down and lock cycle, the GR-DN mnemonic will not be set.

The GR-DN mnemonic will only remain for 5 sec.

9.3.2.17 Flare Reference

At approximately 3800 ft, a set of flare reference indices (fig. 9-13) identical in shape and size to the OGS reference indices will appear in the bottom of the HUD and move up to merge with the OGS indices. When the two sets of indices (OGS and flare) merge, they become a single set that continues to display the flight path angle requirements of the preflare-to-landing maneuver. By maintaining the velocity vector wings precisely in line with the flare indices throughout the preflare to final flare, the desired flight path angle profile will be tracked. At final flare (P_MODE = 4), the flare indices automatically blank.

NAV altitude is the source of data for the flare reference indices; therefore, the merging of the OGS and flare indices will occur at the incorrect flare altitude if the approach is made to a non-microwave scanning beam landing system (non-MSBLS) vice station runway.

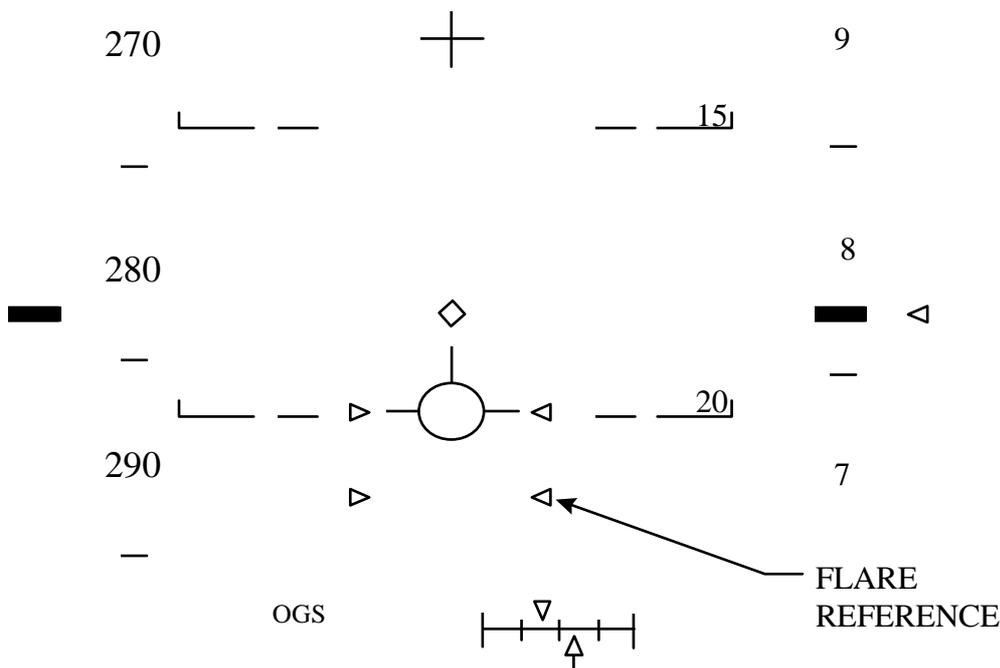


Figure 9-13. Flare reference.

9.3.2.18 Horizon

The horizon line (fig. 9-14) consists of a solid line with a 4° gap in the center. It is positioned with respect to the local horizontal.

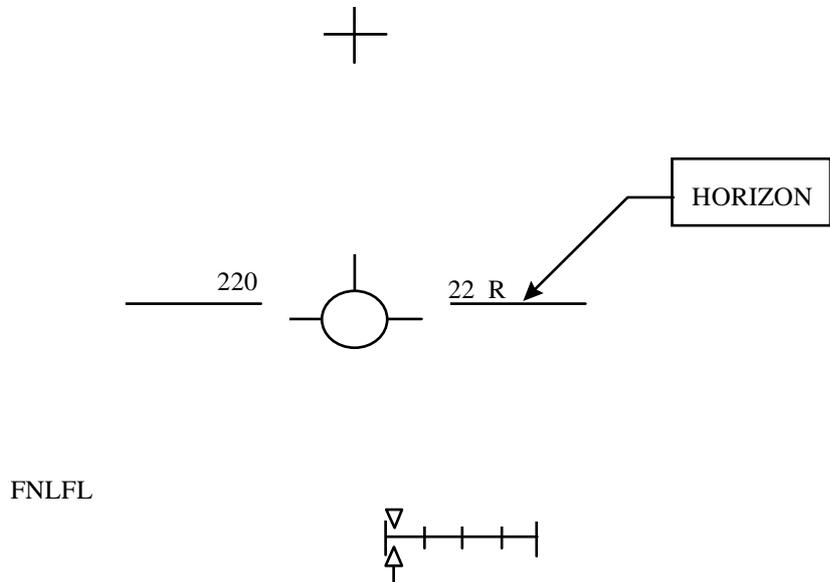


Figure 9-14. Horizon.

9.3.2.19 Deceleration Rate

The deceleration rate symbol (fig. 9-15) consists of a vertical scale and two opposed pointers and is displayed only when WOW occurs. The left pointer indicates actual deceleration rate and the right pointer indicates commanded deceleration rate. The scale is marked in four equal increments with zero deceleration at the top and 16 ft/sec² (I-load) deceleration at the bottom. A smoother has been incorporated to prevent the deceleration rate pointer from jittering.

The commanded deceleration rate displays required deceleration to stop with 1000 ft (I-LOAD) of runway remaining (for the selected runway). The commanded deceleration rate assumes the Orbiter NAV state is accurate.

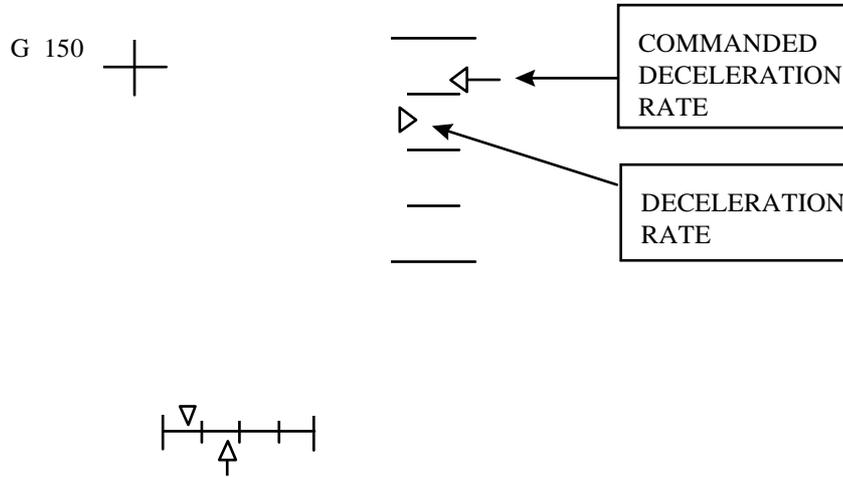


Figure 9-15. Deceleration rate.

9.3.2.20 Cues/Alerts

The cues/alerts are displayed in three different windows with a fourth window serving as a summary line. Each window is a reserved field for the cue/alert; however, there are no visible vectors defining the window and if no mnemonic is being displayed, the field will be blank. The windows are numbered 1 through 4. Their positions in the display are shown in figure 9-16.

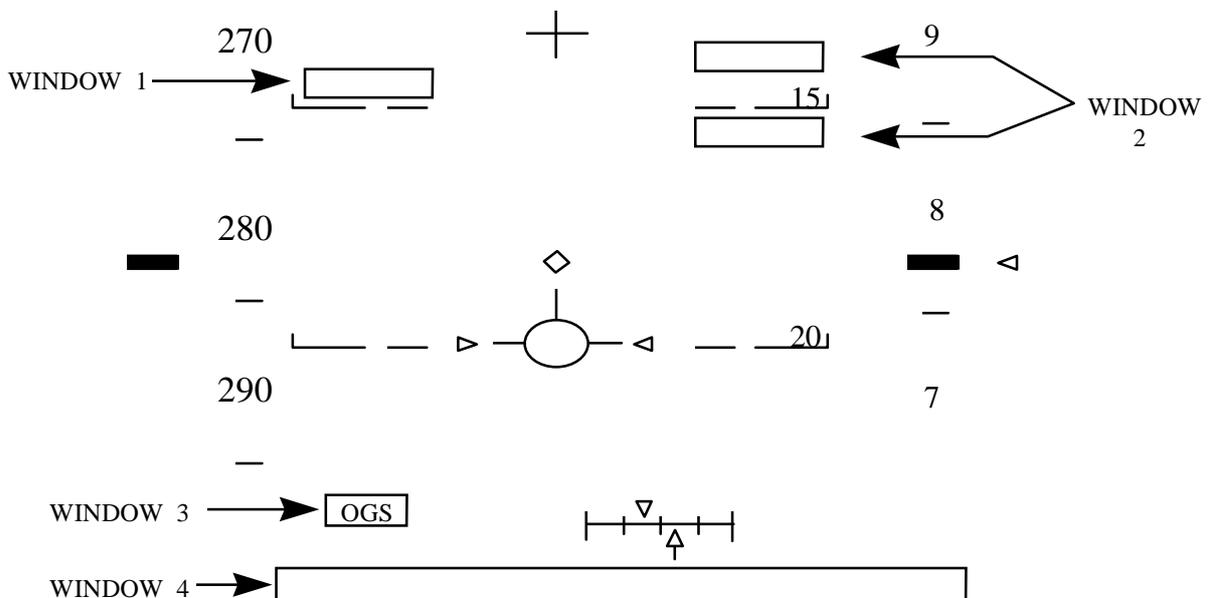


Figure 9-16. Cues/alerts.

Window 1 is reserved for landing gear cueing.

Window 3 displays guidance mode until main gear touchdown, at which time the field is used for the load balance display.

Windows 2 and 4 display three alerts to the crew:

| <u>Cue/alert</u> | <u>Comment</u> |
|------------------|--------------------------------|
| CSS | CSS control in all axes |
| MLSNV | NAV is not processing MLS data |
| B/F | Body flap (B/F) not at trail |

Window 2 consists of an upper and lower half which provide space for the simultaneous display of two mnemonics, if required. Upper and lower field use for window 2 mnemonics function as follows: The first cue/alert mnemonic set appears in the upper half of window 2. The mnemonic will be displayed until the condition setting the flap ceases or 5 sec elapse, whichever occurs first. After 5 sec, if the condition persists, the mnemonic will be blanked at window 2 and positioned first in line in window 4. The mnemonic will remain in window 4 until the condition setting the flag ceases. If a second mnemonic is set while the first mnemonic occupies the upper half of window 2, the second mnemonic will occupy the lower half until the condition setting the flag ceases or 5 sec elapse, whichever comes first. After 5 sec, if the condition persists, the mnemonic will be blanked at window 2, and positioned in sequence in window 4. A maximum of two mnemonics can be displayed in window 2 at any time. CSS and microwave landing system navigation (MLSNV) have priority. If B/F is occupying the second field in window 2 and the CSS or MLSNV cue/alert occur and the last field is full, then B/F will be put on hold to make room for the priority cue/alert.

A cue/alert that persists beyond the 5 sec time out will cause the mnemonic to be transferred to window 4 where the most recent mnemonic transfer will always occupy the last position in line. Five sec occupancy timing for window 2 does not begin until the mnemonic appears in the window. Mnemonics positioned in window 4 are left justified so that any leading mnemonic blanked will cause all following mnemonics to be repositioned left. The mnemonic will be displayed in window 4 until the condition ceases. Window 4 contains field for all three mnemonics.

All mnemonics appearing in window 2 flash. All mnemonics appearing in window 4 are steady.

9.3.2.20.1 CSS Cue

When both pitch and roll/yaw CSS axis are engaged, the CSS cue will be in window 2.

9.3.2.20.2 MLSNV Alert

At an altitude of 12,000 feet (I-LOAD) and below, NAV is continuously checked for use of MLS data. If MLS data is not available for greater than 6 sec (N_SEQ_MAX design dependent) after having once been acquired, the MLSNV alert will be displayed in window 2. The alert is not displayed head down.

If this alert is seen while monitoring an autoland, the CSS should be engaged. One may elect not to engage CSS if at a low altitude (100 ft) and the observed vehicle performance continues to be acceptable.

9.3.2.20.3 B/F Alert

The B/F mnemonic appears in window 2 whenever altitude is less than 5000 ft (I-LOAD) but greater than touchdown and the B/F is not within $\pm 0.5^\circ$ (I-LOAD) of the trail position. The alert is not displayed head down.

If this alert is seen, B/F should be set at trail.

9.4 BFS HUD

9.4.1 HUD Operations in the BFS

There are some minor differences between the PASS and BFS HUD processing software that the crew may notice. The BFS HUD:

- does not use an I-load runway length; rather, it is fixed at 10,000 ft. (So if, for example, the crew is landing on a 15,000 ft runway, the decel command will be more aggressive than it needs to be, because it doesn't know the full length of the runway.)
- has no CSS, MLSNV, or B/F messages
- does not display the "CAPT" or "OGS" P_MODE descriptors (because of no A/L in BFS)
- only displays "PRFNL", guidance diamond, and H-error until 2000 ft (BFS Guidance is not supported below 2000 feet)
- does not flash the "g" value when the high-g limit is exceeded

Note: B/F and high-g messages may be added with OI-34.

9.5 SYSTEM ANOMALIES, CHECKOUT, AND TEST

9.5.1 Data Validity/Data Bad

The HUDE examines the incoming GPC messages for several criteria to determine validity. Should any incoming word in a message fail to satisfy the validation criteria, no subsequent words will be received by the HUDE until a new message is transmitted and the control word of that message is validated. If a control word invalid condition exists for a period of 500 ± 100 ms, a full screen X superimposed over the display will appear on the HUD. Should any data word invalid condition exist for a period of 500 ± 100 ms, all symbology derived from that data will be caused to flash for 2 plus 1, minus 0 sec and then will be removed from the display. The flash rate is set to approximately two flashes/sec with a 60 percent ON duty cycle. If no data are received at all for 500 ± 100 ms in the NORM mode, a full screen X only (no background format) will be displayed. In all cases when valid data are received subsequent to a fault indication display, the fault indication will be removed within 100 ms.

Data bad flags sent by the GPC to the HUDE will either cause the selected data to flash and then be removed as in the validity test or will cause the preceding plus an alert message to appear on the HUD.

9.5.2 System Checkout

An OPS 8 on orbit and an OPS 9 ground checkout capability have been provided. This checkout is accomplished in the same manner as the dedicated displays test and requires no special keyboard unit (KBU) keystrokes to support it. When initiated, selected symbology on the display will be forced up and driven to low ranges to confirm system operation. The OPS 8/9 checkout sequence differs in rigor and display characteristics from the TEST feature available through the MODE select TEST switch. OPS 8 and OPS 9 test displays are presented in figures 9-17 and 9-18.

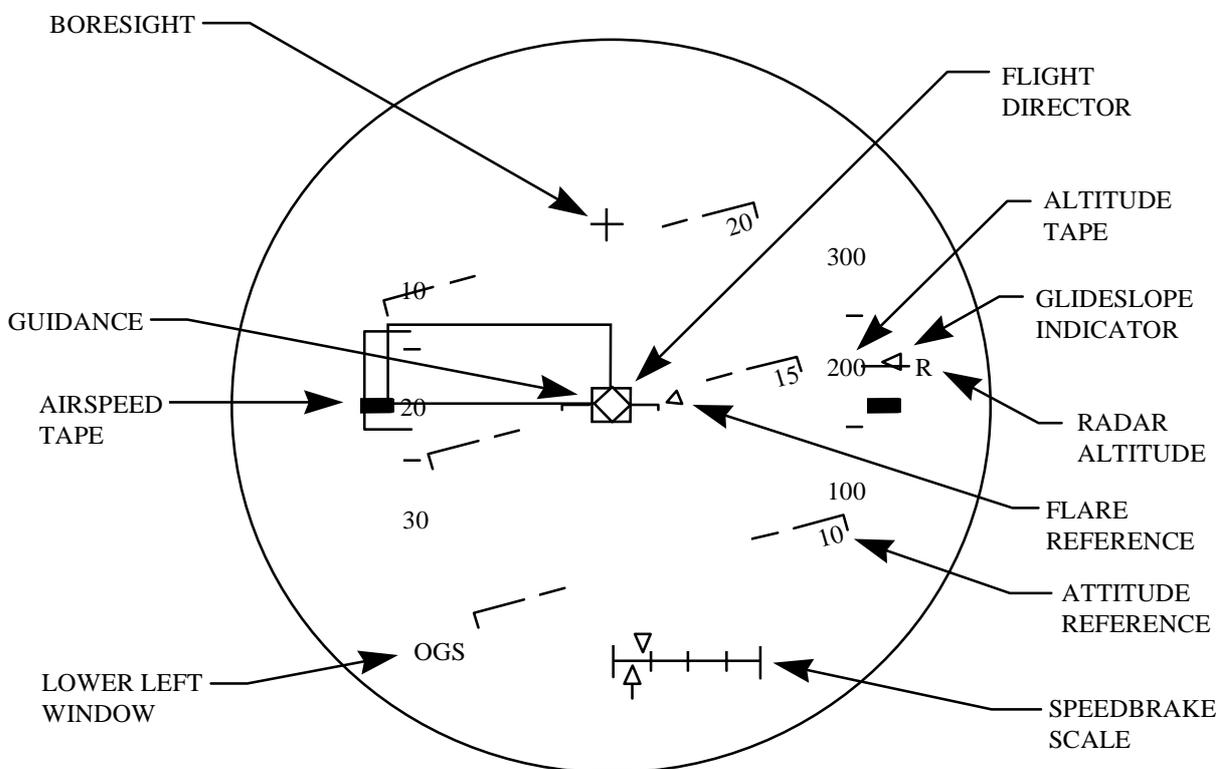


Figure 9-17. OPS 8 test format.

Control of the format to be displayed is provided by the GPC. When powered up, the HUD initializes in format 0. Automatic sequencing to the appropriate display is then accomplished in accordance with event flags toggled in the HUD principal function module. The automatic mode of sequencing is the normal mode of operation. A manual format selection override capability is provided through an L or R HUD item entry (item 37 and item 38) on the HSD (SPEC 50).

In the automatic sequence mode of operation for PASS 19, format 1 will be forced up at MM 304 to 305 or MM 602 to 603 transition. If desired, the HSD HUD item entry can be used to force up format 1 in MM 304 or 602; however, since not all the data required for format support are processed in MM 304/602, the display will be incomplete. As the approach progresses from TAEM interface to touchdown, various

symbols will be introduced and blanked as their data content is required. At touchdown (WOWLON flag set), the display is automatically changed to provide derotation/rollout symbology.

In the automatic sequence mode of operation for PASS 20, format 2 will be forced up at MM 304 to 305 or 602 to 603 transition. Format 2 which incorporates the autoland monitor displays is an overlay of format 1. Crew selection of format 1 or reselection of format 2 after rejection is provided through the HSD HUD item entry. As in PASS 19, the display can be forced up in MM 304/602 if desired.

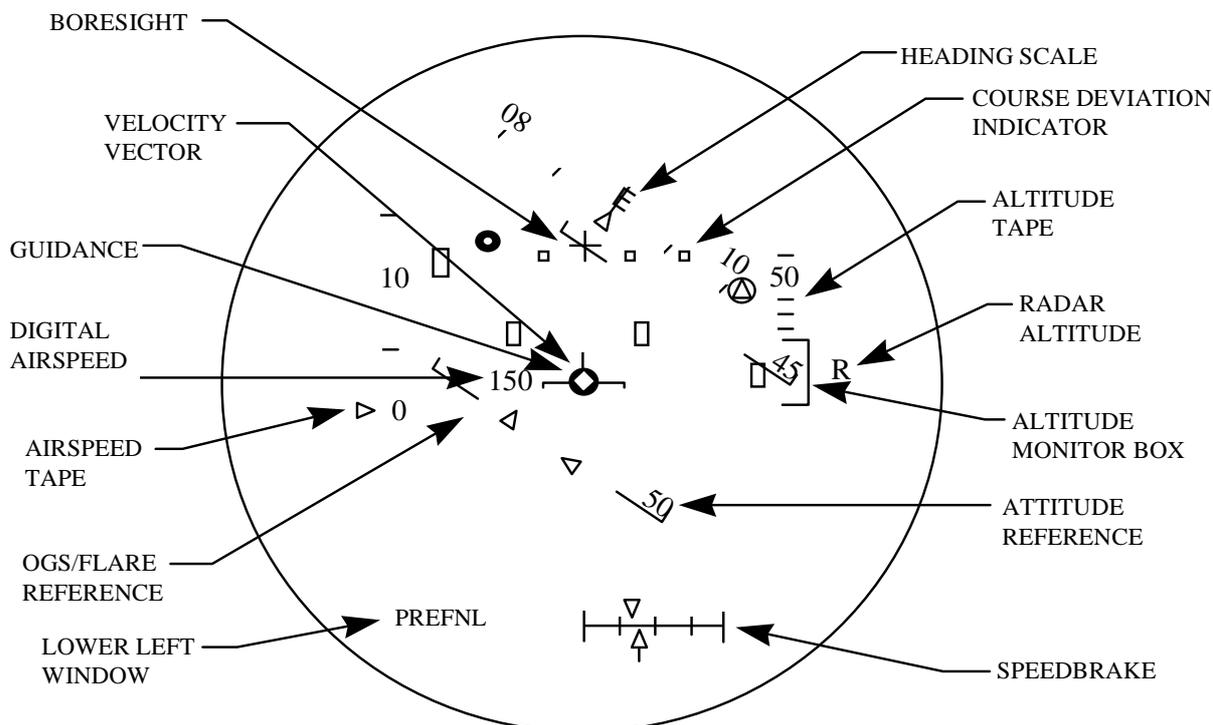


Figure 9-18. OPS 9 test format.

The test format (format 6) is available at any time through selection of the test position of the MODE switch.

9.5.3 System Test

The purpose of the test mode is to provide a means for an overall check of the HUD system integrity. This is accomplished by the display of simple, dynamic symbology, moving in a continuous fashion, and in parallel by continued processing of input data and bit. The entire test takes approximately 15.5 sec. This mode will also be utilized for calibrations of the HUD.

The test symbology has been designed to exercise all PDU electronics required for the display of current normal mode symbology. To the extent that an error affects the symbology, a pilot can detect erratic display motion throughout the dynamic display cycle. At the end of the test, the termination indicator

appears and the display becomes static. Until the test mode is exited, the display will remain static, giving the pilot an opportunity to inspect the positional relationships of the symbology for inaccuracies. Should a parity or other bit error occur, the display will become and remain static at the moment of error detection until the test mode is exited. The termination indicator, COMPLETE, will not appear in this case.

The test mode may be entered by two methods. The GPC can command the HUD to enter the test mode by setting the three bits, 14 through 16, to a binary 6, 110, in the HUD control word of the GPC's HUD message 1. Since the control word is recomputed at a 6.25-Hz rate, the GPC must continuously command the test mode for at least 96 recomputed transmissions.

The other method of entering the test mode is by the activation of a three-position MODE SELECT switch. The lower, momentary, position selects the declutter mode; the middle, on, position selects the normal mode; and the upper, on, position selects the test mode. In order to perform the entire test, the MODE SELECT switch must remain in the upper position until the termination indicator, COMPLETE appears.

Figure 9-19 shows the beginning of the test mode display. Symbology labels 1A, 1B, 2A, 2B, and 3 do not appear in the test display. The symbology contained in the display consists of four circles and four boxes with overlapping sides. The symbology is drawn at 60°, 150°, 240°, and 330° about the center of rotation COR, which is offset from the PDU's CFOV along the X- and Y-axes by -5 mr. The circles and boxes at 60° and 240° are centered 100 mr from the COR. The circle at 150° and the box at 330° are centered at 80 mr from the COR. The circle at 330° and the box at 150° are centered at 85 mr from the COR.

Figure 9-20 represents the symbology after about 2.5 sec. The boxes immediately begin to break up with sides 1A and 1B of each symbol moving together and maintaining their orthogonal relationship in a circular, counterclockwise direction. Sides 2A and 2B move similarly in a clockwise direction. Circle 3 and all circles remain stationary.

Figure 9-21 represents the display after about 5 sec. The sides from adjacent boxes have crossed over each other in their bidirectional course, while the circles remain stationary.

The sides of the boxes continue moving until they have traveled 180° from their initial position. Figure 9-22 represents the final symbology. Boxes have reformed, enclosing the circles opposite their initial position circles. A brightup window surrounds each symbol and is represented by the dashed square. Its effect is to cut off the overlapping portion of each side of the boxes, thus forming clean corners. The word "COMPLETE" appears and is positioned so that the lower left hand corner of the C is at coordinates (0,0), the CFOV. The dash/gap rectangle around it is drawn so that its lower left corner is at coordinates (-5, -5), the COR. The characters forming the word "COMPLETE" will be 10 mr x 5 mr and centered in the dash/ gap rectangle 5 mr from it.

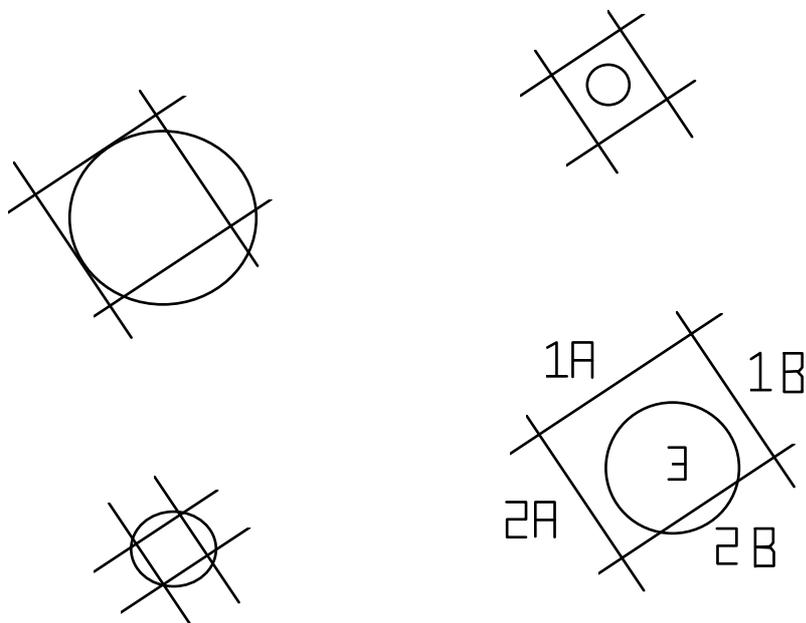


Figure 9-19. Beginning of test mode display.

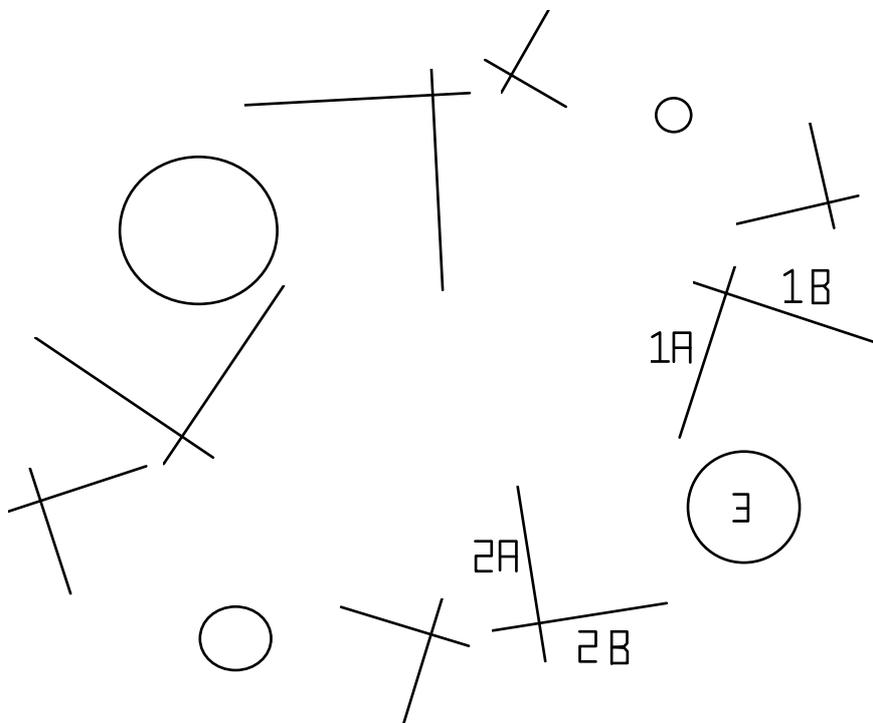


Figure 9-20. Test mode symbology after 2.5 sec.

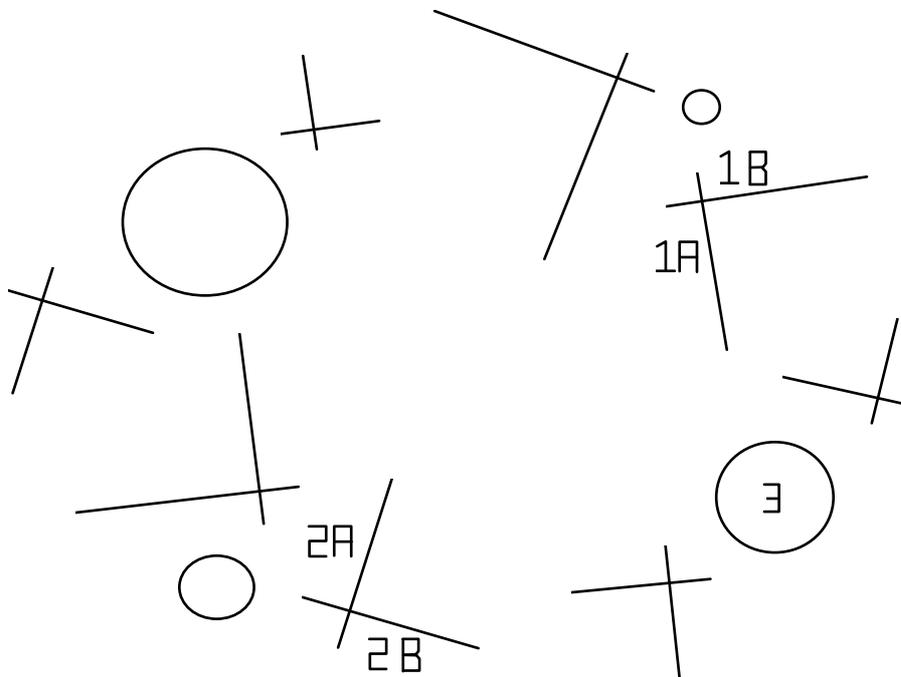


Figure 9-21. Test mode display after 5 sec.

9.6 REFERENCES

- FSSR 20 V3 (OI-32), Displays and Controls (STS 83-0020V3-32, F.4.252)
- OI-32 SCR 92938B BFS HUD Processor
- Boeing / Rafael de la Torre, FS&SE, per conversations 5/2007
- Boeing / Carrie Larcher, D&C HUD/C&W/Timers SSM, per emails 5/2007
- Head Up Display Computer-Based Trainer (CBT), Version 2.0, Modified May 2005
- "Landing and Rollout" NASA AMES VMS presentation, April 2007, by Astronaut Office

APPENDIX A: Operational Notes

A.1 Deceleration Cues

SYSTEMS DIVISION NOTE OF INTEREST

May 05, 1989

To: Distribution

From: DF62/LeRoy E. Cain

Subject: Deceleration Errors During Braking Due to Navigation State Errors

The purpose of this note of interest is to explain the potential errors in the acceleration drag (deceleration), as displayed on the ALPHA/MACH Indicator (AMI) and on the Head Up Display (HUD), due to navigation (NAV) state vector errors.

Background/Problem

In a recent integrated simulation, when a Microwave Landing System (MLS) range error of +5,000 feet was introduced into the NAV state vector, the crew reported that during rollout and braking the AMI and the HUD were displaying a deceleration of 10 to 20 ft/sec/sec, which averaged approximately twice the normal value (8 ft/sec/sec). In addition, the crew reported that the commanded deceleration rate (also on the HUD) was much lower than normal, and at wheel stop the AMI tape was reading 20 ft/sec/sec. During a nominal rollout with normal brake pressure, the AMI reads approximately 8 ft/sec/sec with a commanded deceleration on the HUD of approximately the same value, and 0 ft/sec/sec at wheel stop. The problem was that the apparent deceleration was normal (based on the crew's brake feeling and visual cues) and the software was computing commanded and actual deceleration numbers that did not match this (different from the normal magnitude).

Analysis/Solution

Several SMS runs were done to determine the cause of these errors and to evaluate the performance of the SMS models. This problem was duplicated several times, and in all cases showed the same results. Some runs were also done using good MLS data, and some with no MLS data at all.

The MLS range bias introduces an error into the NAV state vector. Once the vehicle is on the ground (based on the LD_GR_HYDR flag being set ON) and the relative velocity is less than a prescribed threshold value ($REL_VEL_MAG < LDGR_VEL_THRSH$), the acceleration drag (ACC_DRAG) is computed as follows (Ref. STS83-0004A FSSR).

$$ACC_DRAG = -(A) \cdot (V) \text{ (dot product)}$$

where A is the total sensed acceleration vector as determined by the IMU's, and V is a unit vector of the relative velocity vector (V_RHO) from the NAV state. The negative sign in the ACC_DRAG equation is used in order to display a positive number for vehicle deceleration. The velocity vector V_RHO is equal to V_REL, which is the vehicle velocity relative to the atmosphere and is a function of the M50 position and velocity vectors, r and v respectively. V_RHO is computed as follows:

$$V_RHO = V_REL(v,r) = v - EARTH_RATE(EARTH_POLE \times r)$$

where EARTH_RATE is the earth's angular velocity with respect to the M50 (inertial) axes and EARTH_POLE is a unit vector in the direction of the earth's axis of rotation, expressed in M50 coordinates. Both of these vectors are constants in the onboard software and could not have contributed to the anomaly. The latter part of this equation (parenthetic portion) is a cross product of the EARTH_POLE and r vectors. The MLS range error was introduced into the ACC_DRAG computation in two ways. The error provided an incorrect M50 position vector r , and also polluted the M50 velocity vector (v) due to the distribution of the residual in the covariance matrix to all six components of the NAV state. Therefore, since the MLS range error impacted v and r , the resultant value of V_RHO (and its unit vector (V)) was incorrect, which ultimately caused the error in the magnitude of the ACC_DRAG deceleration number. [Note that the ACC_DRAG number is computed differently prior to TD, and is set to zero during MM 305.]

The ACC_DRAG is displayed on the AMI and on the HUD during rollout. When the velocity vector is used from the NAV state vector in the ACC_DRAG computation (as shown above), it results in an error in the magnitude of the acceleration drag. Also, since the onboard software thinks the vehicle is 5,000 feet aft of its actual position, the commanded deceleration is much less than it should be. The commanded deceleration rate displays the required deceleration to stop with 1,000 feet of runway remaining (for the selected runway). The commanded deceleration rate computation assumes that the Orbiter NAV state is accurate.

An error of this magnitude is very unlikely for an MLS unit, but the covariance matrix weights the MLS data heavily once it is incorporated. This was a case in which the NAV state was polluted by bad MLS data; however, anything that introduces errors into the NAV state may have a similar effect on the ACC_DRAG computation.

Operational Notes

During rollout and braking, the crew uses the deceleration rate on the AMI (and the HUD) and the commanded deceleration rate on the HUD as braking cues. If the MCC Guidance Officer has determined that the NAV state vector is in error, and/or the GNC Officer suspects that errors are being introduced due to navaid sensor or IMU anomalies, then the crew should be alerted that their braking cues on the AMI and HUD are suspect. The crew will then rely on visual and brake pressure cues to accomplish braking on the runway.

LeRoy E. Cain

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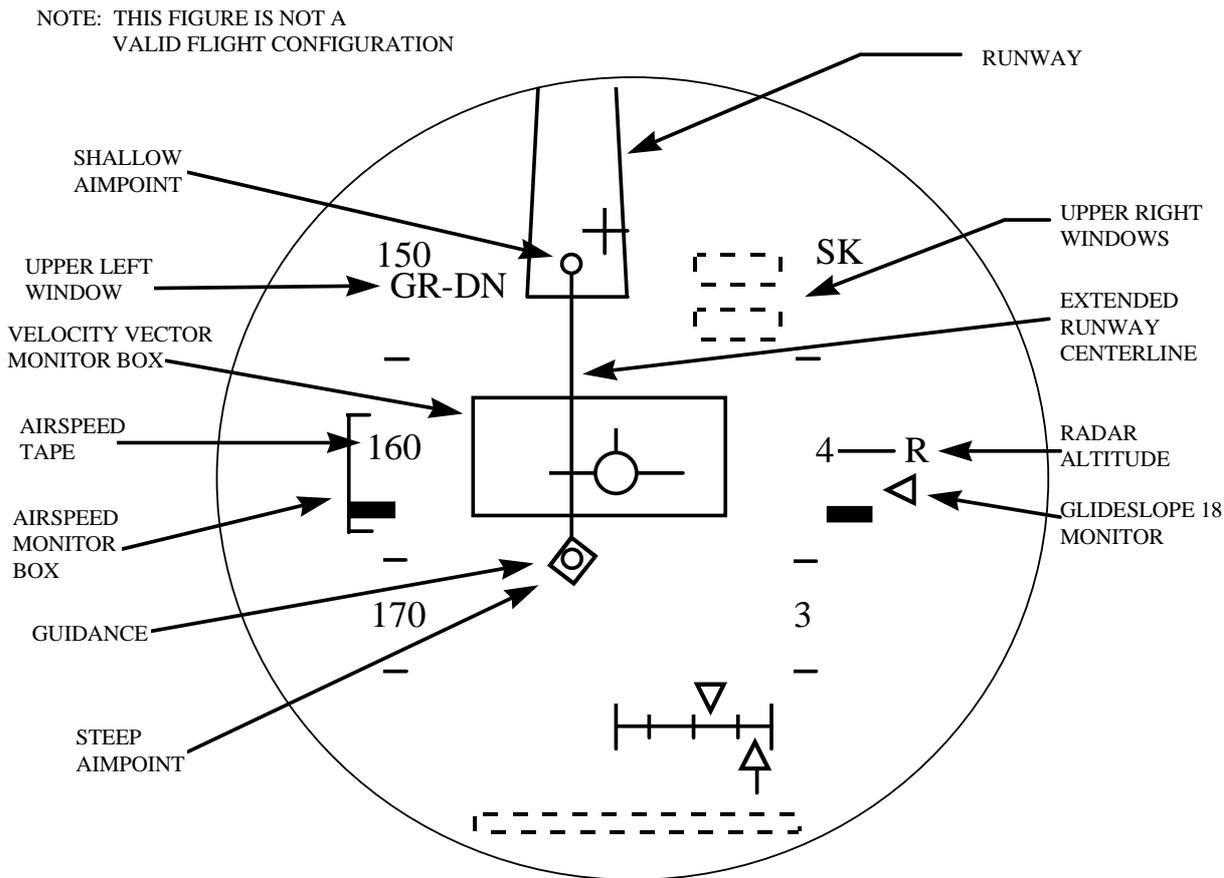


Figure A-1. Display format 2.

A.2 Velocity Vector Anomaly

A generic anomaly has been identified in the HUD firmware which can cause erroneous velocity vector indications under certain conditions. Upon the setting of the guidance parameter I_PHASE=3 at the beginning of the TAEM prefinal phase of entry (at an altitude of approximately 13 kft), the flight director symbol transitions to the velocity vector symbol. I_PHASE=3 allows the flight director symbol to uncage in the pitch axis and the setting of a GPC-generated flag allows the flight director symbol to uncage horizontally. The GPC flag is essentially a check for an inhibit to the horizontal uncage which is set if the ATT REF pushbutton is depressed prior to I_PHASE=3. Only when both axes uncage does the flight director symbol transition to the velocity vector symbol.

The anomalous behavior involves the HUDE Flight Control Computer (FCC) operational program velocity vector X and Y position smoothing routine during the uncage transition. The smoothing routine uses two counters, XCOUNT and YCOUNT, which are dedicated RAM addresses to provide a timing counter and intermediate velocity vector positions. The counters are incremented per-frame and compared to a value of 250 to create a 5-second fader timer. After 5 seconds the velocity vector is at its proper uncaged position.

The timers are preset to a known value of 1 during the FCC operational program prior to velocity vector transition. In extensive ground tests, it was discovered that cycling power with the velocity vector uncaged (i.e., post I_PHASE=3) bypasses the normal conditioning of XCOUNT and YCOUNT, which causes the counters to initialize at an unknown value. If the number is positive, the velocity vector will immediately move to the correct position, however if it is negative the delay depends on the value of the negative number. As an example, HUDE S/N 104 consistently initialized to a YCOUNT value of -6350 decimal which corresponded to a 2 minutes 7 seconds (50 Hz rate) delay before the velocity vector moved to its correct position.

Operationally, any HUD should be called suspect if the initial power-on or a power cycle occurs during the TAEM prefinal phase of entry.

APPENDIX B: Autoland Information Archive

B.1 AUTOLAND

As of OI-8A, the Autoland Monitor format has been deleted from the flight software. The following information is being retained for reference.

B.1.1 Purpose, Operation, and Organization

This format (fig. B-1) is the same as the A/L format (fig. 9-7 and fig. 9-8) but contains two additional symbols. Another symbol, the glide slope monitor, is displayed in both the A/L and autoland monitor formats but the data sources and control algorithms differ between the two formats.

The autoland monitor format has been designed to provide the pilot with the capability to monitor the autoland system performance with a minimum of workload and, in case of a takeover, provide the monitor with the information required to correct the flight profile. The autoland monitor format accomplishes two distinct functions during the approach. The first is to display the TAEM to autoland transition criteria for altitude, azimuth, and airspeed. The second function is to display the acceptable autoland system trajectory performance limits for the same parameters subsequent to transition to autoland. Additionally, the parameter altitude rate with respect to altitude is processed to derive flight path angle trajectory performance during final flare.

B.1.2 Autoland Glide Slope Monitor

Symbol displays the autoland derived altitude error. The symbol consists of a double width triangular pointer located on the right side of and facing the altitude tape. The pointer is referenced to and is free to move along the altitude tape. The pointer is driven by the A/L format algorithm until I_PHASE equals 3. At that time, it is driven by the autoland format algorithm. Symbol is displayed when autoland monitor is valid and I_PHASE is greater than 2. Symbol is blanked when WOW occurs or declutter level is 2.

B.1.3 Autoland Airspeed Monitor Box

Symbol displays the autoland derived airspeed error. The symbol consists of an open-ended box located on the left side of and facing the airspeed tape. The box is referenced to and is free to move along the airspeed tape. The box first appears when I_PHASE makes the transition from 2 to 3 and initially consists of dashed lines. The box becomes solid when P_MODE is greater than or equal to 2. Symbol is displayed during autoland format when equivalent airspeed (EAS) monitor is valid, I_PHASE is greater than 2, and speedbrake retract has not been commanded. Symbol is blanked when WOW occurs or declutter level is 2.

B.1.4 Autoland Velocity Vector Monitor Box

Symbol displays the autoland derived gamma and crosstrack errors. The symbol is a closed box that is free to move in the FOV. Above 600 ft altitude, the symbol is a fixed size. Between 600 and 100 ft altitude, the symbol increases in size. The symbol first appears when I_PHASE makes the transition from 2 to 3 and initially is composed of dashed lines. The symbol becomes solid when P_MODE is greater than or equal to 2 and is blanked when WOW occurs.

SECTION 10

AEROSURFACE ACTUATORS

10.1 GENERAL

The aerosurface actuator system is comprised of four aerosurface amplifiers (ASAs); four hydraulic actuators for the elevons; one hydraulic power drive unit (PDU) and four rotary actuators for the rudder/speedbrake; and one hydraulic PDU and four rotary actuators for the body flap (BF). Each elevon actuator and the rudder/speedbrake PDU contains two switching valves to control hydraulic switching from the three hydraulic systems.

In addition to these hardware systems, there is a considerable amount of software to support aerosurface commands, feedbacks (FDBKs), redundancy management (RM), displays, etc.

10.2 AEROSURFACE SERVOAMPLIFIER

The ASA is the avionics hardware used to control each actuator (Figure 10-1). Position, bypass, and override commands are sent from general purpose computers (GPCs) to their respective areas within the ASA. Normally, GPC 1 commands ASA 1, GPC 2 commands ASA 2, etc., although one GPC can command all or any ASA depending upon GPC/multiplexer/demultiplexer (MDM) stringing. Each ASA is assigned to a respective FCS channel, ASA 1 to FCS CH 1, etc. Position feedback (PFB) data (received from each actuator) and actuator channel fail signals are sent from each ASA to all GPCs. Position error commands (derived from the summation of position command inputs, primary delta pressure (ΔP) data, secondary ΔP data, and PFB from the actuators) are sent to each actuator servovalve (port) that the ASA controls. Based on secondary ΔP data FDBK from each servovalve channel and current from each servovalve driver, each ASA protects against failures through the activation of the isolation valve corresponding to the violating servovalve (also known as port popping). A discussion of how the ASAs perform these functions follows.

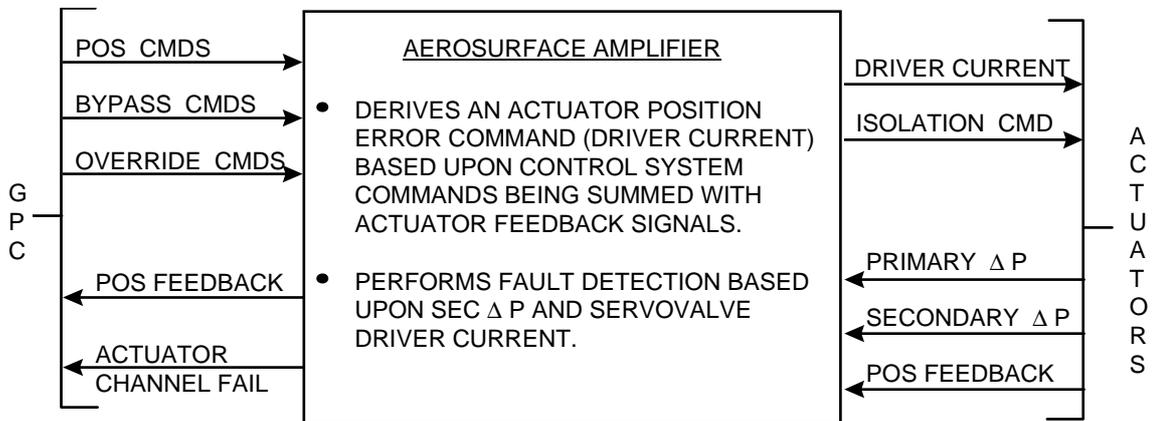


Figure 10-1. Aerosurface Amplifier

10.2.1 Position Command

Position commands generated by the flight control system (FCS) are sent to each ASA elevon driver via their corresponding flight aft (FA) MDM. The FCS position command is summed with a position feedback (PFB) from the primary actuator, producing a position error command (Figure 10-2). This position error command is sent into a servoactuator valve driver where it is amplified, producing a corresponding voltage which is used to command the servovalve torque motor. Note also that the PFB signal for the elevon actuator is also sent to the FCS caution and warning (C&W). This is needed for the crew displays and C&W.

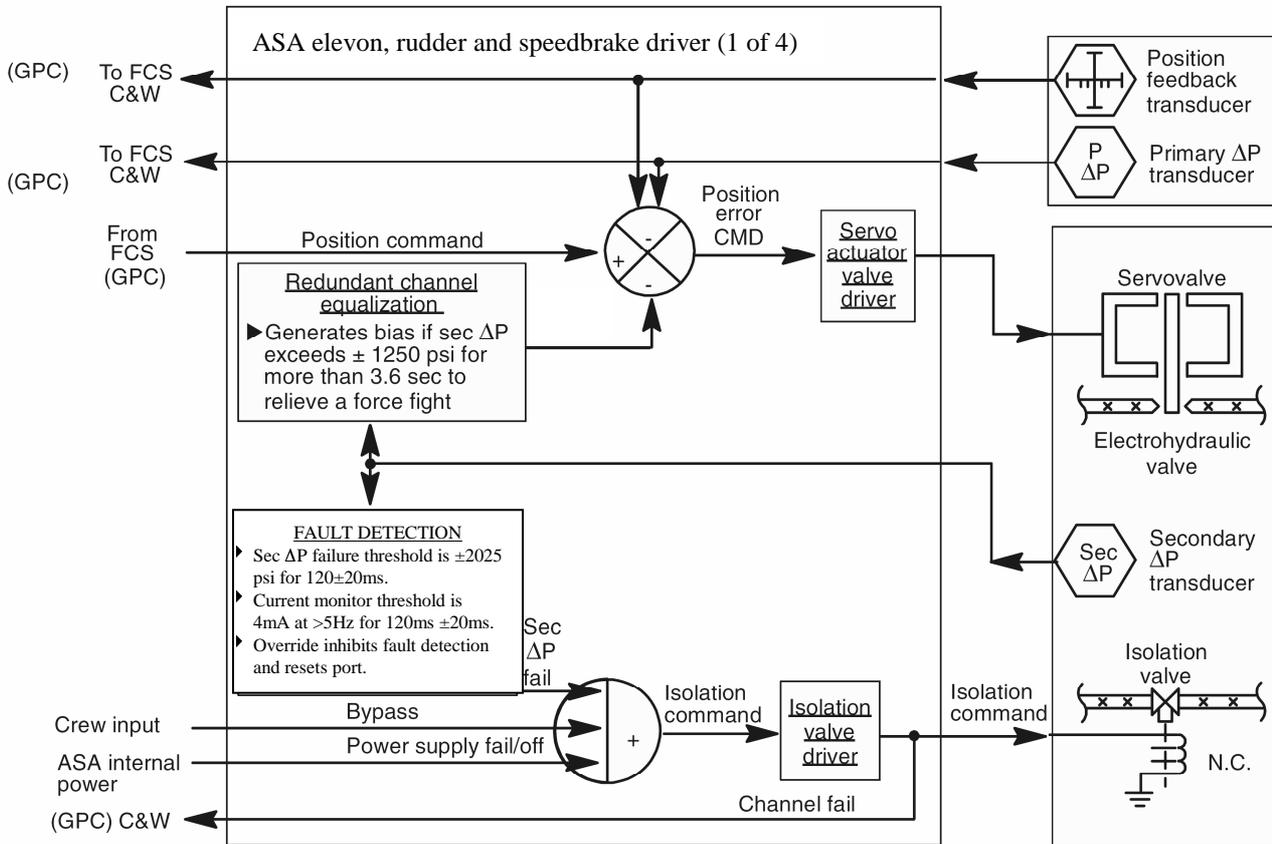


Figure 10-2. ASA Command, Fault Detection, Equalization

10.2.2 Primary ΔP Feedback

Primary (PRI) ΔP FDBK is used to measure hinge moments and to damp oscillatory motion. PRI ΔP FDBK is only used in the elevon drivers. The PRI ΔP signal is sent to the FCS C&W for the purpose of measuring hinge moments and into the position command summer where three values are used (position CMD, PRI ΔP FDBK, PFB) for the purpose of damping oscillatory motion. The summation of the three signals produces a position error command. Equalization is also included in the summer for fault protection. Reference Figure 10-2.

10.2.3 Fault Detection

Each servovalve has a secondary ΔP transducer used to measure the hydraulic resistance it encounters relative to the other servovalves. The hydraulic pressure signals are sent to the ASA controlling that particular servovalve where it is monitored in the fault detection logic (Figure 10-2). If the secondary ΔP signal exceeds a value of ±2025 psig for 120 ± 20ms, a secondary ΔP fail discrete is issued. This results in an isolation command being issued to an isolation valve driver where it is amplified, producing an isolation command voltage which is sent to an isolation valve for that servovalve. The isolation valve alleviates the hydraulic resistance generated by the servovalve and removes that servovalve's command pressure from the power spool. When the isolation command voltage is issued, a channel fail signal is also sent to C&W.

The ASA fault detection logic also contains a current monitor circuit that detects high gain oscillatory failures of the servovalve driver. Prior to the current increment III ASAs (prior to the delivery of OV-102), it was determined that a single resistor could fail in the servovalve driver which would result in the driver gain increasing by several orders of magnitude. This would result in the driver output oscillating hardover to hardover at a high frequency. If this failure occurs, the secondary ΔP fault detection will not cause a bypass in all cases, and, as a result, the current monitor circuit was added to the ASAs with increment III (STS-

6). The current monitor circuit filters the driver current with a 5 Hz high pass filter. If the filtered driver current exceeds 4 milliamps for 120 ± 20ms, an isolation command is sent to the isolation valve.

10.2.4 Effects of Feedback Biases

As stated earlier, all surfaces except the body flap use position feedback signals differenced with the command in the ASA to generate a position error command. If an individual position transducer develops a bias, this will adversely affect the command on that FCS channel. This will result in a force fight between this and the other good channels. If the position bias is large enough, the secondary delta pressure fault detection will cause the failed port to be bypassed. Rockwell performed a study to examine the level of position bias required to cause ports to bypass (ref RI 283-220-086-CG-002, 01/09/86). These data are contained in SODB Vol III, paragraph 4.5.1.1.1, as well as the Orbiter Flight Control Effector Systems Data Book (May 26, 1999, RSS99D0319) and are summarized below.

| Actuator | Position feedback bias required to bypass a channel (degrees) | |
|-----------------|---|--------------|
| | Nominal | Nominal + 3σ |
| Inboard Elevon | 4.7 | 5.8 |
| Outboard Elevon | 4.4 | 5.0 |
| Rudder | 5.6 | 6.8 |
| Speedbrake | 11.0 | 14.1 |

10.2.5 Displays

When an isolation command is issued to an isolation valve, a discrete is also sent to C&W which illuminates an FCS channel light and a backup C&W light (on panel F7) and activates a class 2 alarm tone. In conjunction with this, there is a GNC SYS SUMM display with an FCS channel cathode ray tube (CRT) message and a down arrow under the FCS channel that has isolated a servovalve (Figure 10-3). The four FCS channels shown in the CRT display correspond with the four ASAs used to control the elevon actuators.

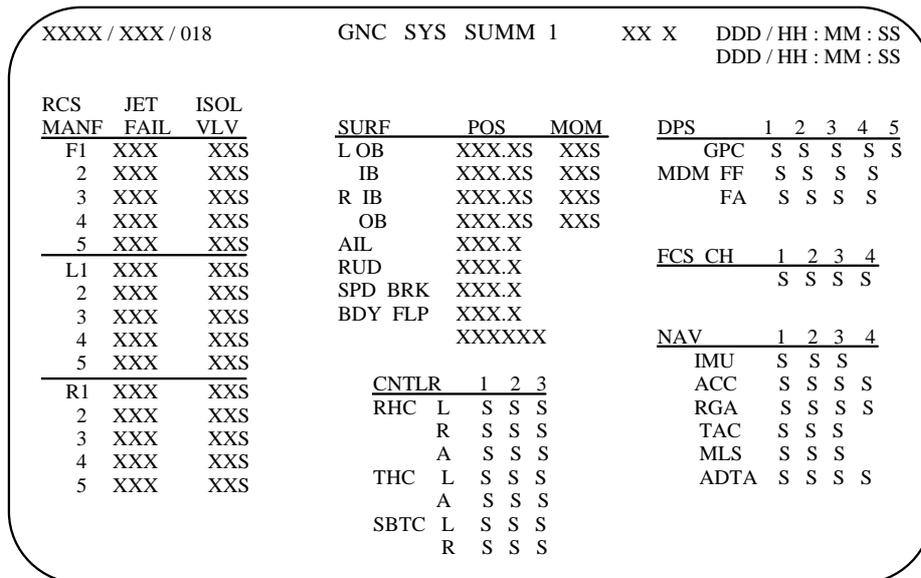


Figure 10-3. GNC SYS SUMM display

10.2.6 Fault Detection Integrator Curve

The fault detection logic consists of hardware which monitors the secondary ΔP and the driver current for a given servovalve. The hardware also includes a time counter or up/down integrator which counts up the time that either the secondary ΔP or

driver current exceeds their respective tolerances. When either of these inputs exceeds its tolerance for 120 ± 20 ms, the integrator will set and latch an isolation signal. If, however, one of the inputs exceeds its tolerance, but for less than 120 ms, and then subsequently falls below its tolerance, the time counter will count down at one-fourth the rate it counts up (the counter counts down at $\frac{1}{4}$ sec/sec). This is illustrated in Figure 10-4. When the integrated time exceeds 120 ± 20 ms, then the isolation command is issued. This isolation command can only be reset by the override command or by a power cycle of the ASA.

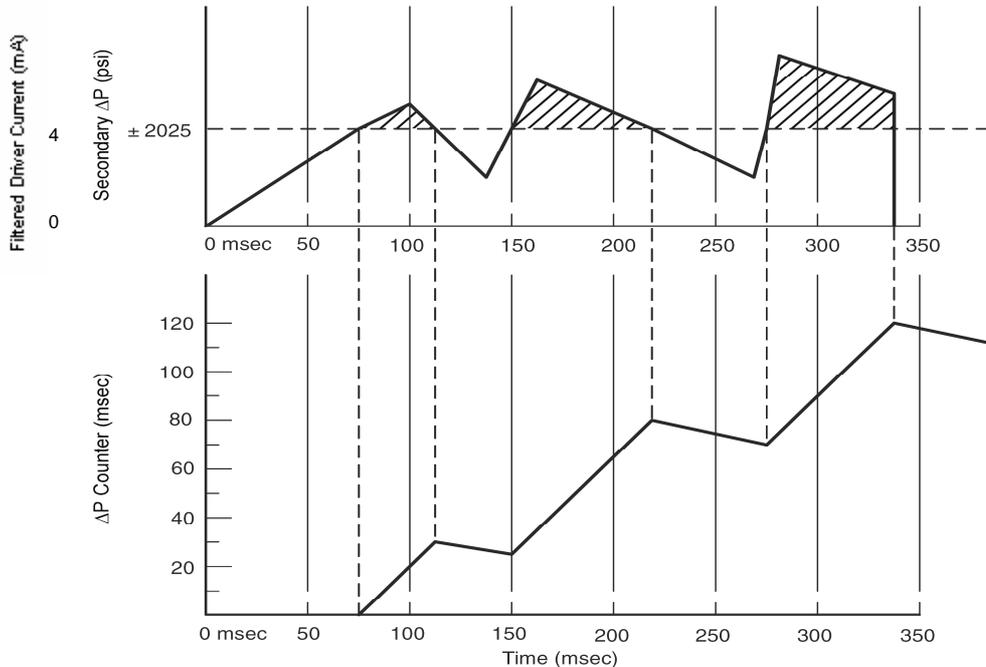


Figure 10-4. Fault Detection Time Counter Curve

10.2.7 Bypass Command

The purpose of a bypass command is to isolate any one aerosurface servovalve without affecting the other aerosurface servovalves controlled by the same ASA channel. Bypass commands to individual aerosurface channels can be issued in operational sequence (OPS) 8 from display 8011 FCS/DED DIS C/O and ENTRY CONTROLS display SPEC 53 in OPS 3. Once the bypass command is issued by the crew, it encounters a 120 ms delay in the bypass monitor to provide protection against transients and noise on the input line. The isolation valve driver then issues the isolation command to the isolation valve and sends an actuator channel fail signal back to the GPCs (Figure 10-2). This bypass will remain in effect until a reset command is issued from either of these displays. The bypass will also remain in effect for this aerosurface even if the FCS channel switch is placed in override.

10.2.8 ASA Power Supply Failure

Each ASA has a power supply which relays ac and dc power to all elevon driver electronics. There is also a 28 V dc power source which supplies electrical power to the isolation valve driver. If there is an ASA power supply low voltage condition, a power supply fail signal would be issued by the ASA power supply, isolating all the servovalves powered by the failing power supply. The power supply fail signal will be maintained until the power supply comes back up to an efficient voltage level. If there is an ASA power supply failure, isolation command voltage for all the elevon drivers on that channel will be supplied by the 28 V dc power source to the isolation valves via the isolation valve driver. If the ASA power supply comes back up to an efficient voltage level, the elevon drivers will drop the power supply fail signal and continue nominal operation (Figure 10-2).

10.2.9 Override Commands

Override commands are generated by the FCS channel switches whenever a crewmember places the switch to OVERRIDE (placing FCS channel 1, 2, 3, or 4 switch in an override position causes corresponding override status for that channel). They are used to prevent servovalve isolation (port popping). Override commands are used to override the fault detection logic. In order for the command to take effect, it must be present for a minimum of 120 ± 20 ms. The command inhibits the fault

detection logic from applying power to the isolation valve driver, thus preventing the servovalves from being isolated. If a servovalve has previously been isolated by the fault detection logic and an override command is issued, the fault detection logic will be reset removing the isolation command. The override request would also remove any CRT display and C&W indications related to the channel isolation command. Also, if the fault recurs after the override command has been removed, the fault detection can reissue an isolation command, isolating the faulty servovalve (Figure 10-2).

10.2.10 Redundant Channel Equalization

The redundant channel equalizer (Figure 10-2) is an integrator used to control hydraulic force fights between the servovalves when a servovalve will not be isolated. The redundant channel equalizer receives the same secondary ΔP signals as those being issued to the fault detection. If the secondary ΔP value exceeds ± 1250 psi, a bias signal is generated with a time constant of 1.7 sec. This bias signal is then sent to the position command summer where the offending position error command is modified so as to reduce the hydraulic force fight to ± 1250 psi or less. Each elevon driver has a redundant channel equalizer built into it. If the equalization logic cannot compensate for the excessive hydraulic pressure, the secondary hydraulic pressure will build up to the ± 2025 psig limit set by the secondary ΔP fault detection logic where an isolation command will be generated if the ±2025 psig limit is exceeded for 120 ± 20ms.

10.2.11 ASA Switch Configuration

Each ASA has two switches which control the electrical power: an FCS channel switch and an ASA power switch. The FCS channel switches are triple contact, three-position center lock switches, and the ASA power switches are triple contact, two-position (OFF and ON) switches. When the ASA power switch is in the OFF position, all power to the ASA channel is removed (Figure 10-5). In response to this, all the elevon servovalves in that channel are sent to a null driver, resulting in force fights if the other ASA channels are operating.

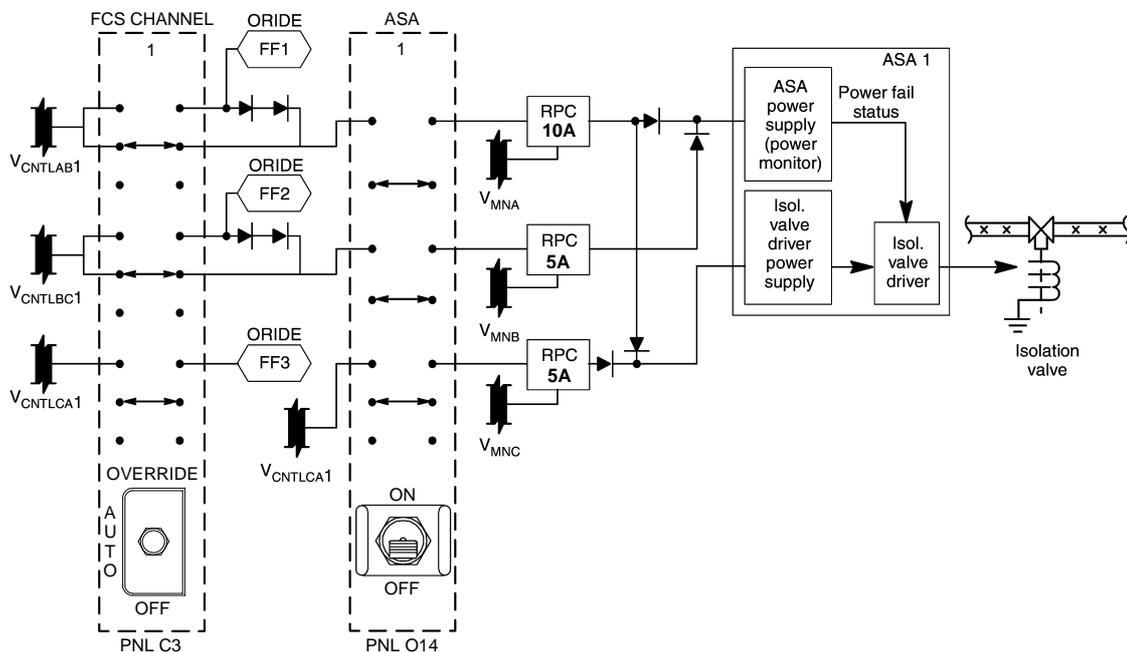


Figure 10-5. ASA/FCS Channel 1 Configuration

When the ASA power switch is in the ON position, there are three different configurations the ASA can be operated in, depending on the position of the FCS channel switch (OFF, AUTO, or OVERRIDE). When the FCS channel switch is in the OFF position, all control bus power to the remote power controllers (RPCs) is removed allowing only control bus power to activate the RPC to power the isolation valve driver power supply for that ASA channel. This results in the isolation of all the elevon servovalves on that ASA channel by the isolation valve power supply providing power to the isolation valve via the isolation valve driver. Note that the ASA power supply does not receive power in this switch configuration. When the FCS channel switch is in the AUTO position, main bus power is maintained to the RPCs which supply power to both the ASA power

supply and the isolation valve driver power supply. This allows the ASA to function nominally with the ability to isolate a failing elevon servovalve, if needed.

The final ASA configuration is that of the FCS channel switch being in the OVERRIDE position. This configuration allows main bus power to continue being sent to both ASA and isolation valve power supplies for the given channel. Also, override signals are sent to the aeroactuator subsystem operating program (SOP) via the guidance, navigation, and control (GN&C) switch RM. The aeroactuator SOP then sends the override commands to the ASA where all servovalves on that channel are overridden. This removes any isolation commands issued by the fault detection logic in that ASA channel and prevents it from issuing any further isolation commands while the FCS channel switch is in the OVERRIDE position. Figure 10-5 is for channel 1 only but is representative of all four FCS channels. Channel 4 does not have redundant power for isolation and each of the RPCs for the ASA 4 power switch are 5 Amps. See Space Shuttle Systems Handbook drawing 9.11 for details.

The FCS channel switch is a break-before-make switch. This means that when the switch is moved from AUTO to OVERRIDE, or vice versa, power will momentarily be interrupted to the ASAs and ATVCs. The resulting effect is a temporary bypass and subsequent reset of the channel associated with that switch for each of the TVC and aerosurface actuators. This effect is produced by a sequence of timing functions and delays within the ATVCs and ASAs. For the ASAs, the minimum time that the affected channel will remain bypassed is 0.51 seconds, and the maximum time is 1.37 seconds. The implication of this is that the channel switches should not all be switched in rapid succession, in order to ensure that adequate control of the actuators is not momentarily lost. This is discussed in the SODB paragraph 3.4.5.1.16 and in Rockwell International Internal Letter 283-220-86-009.

10.3 ELEVON ACTUATOR

Each of the four elevons moves by means of a separate actuator (Figure 10-6). Each elevon actuator contains four servovalves in which each generates a command hydraulic pressure in response to driver output commands from the ASA. The switching valves provide drive pressure to each servovalve. The servovalves provide hydraulic pressure which is summed together at the secondary force-sum actuator (power spool). These hydraulic pressures drive the power spool in order to generate primary command hydraulic pressure to drive the primary actuator (Figure 10-7).

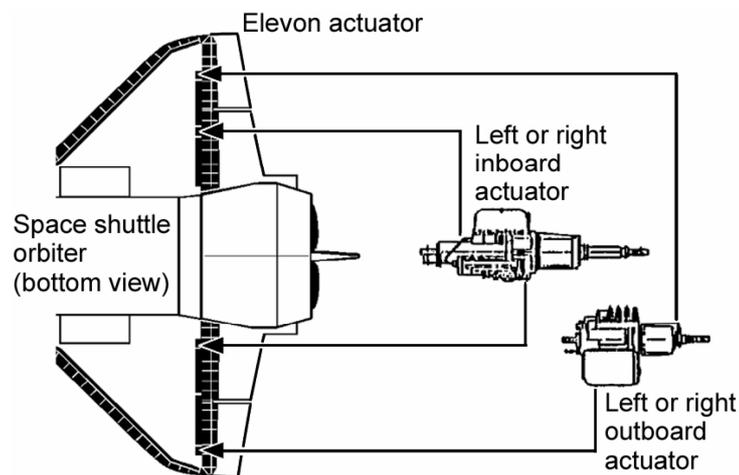


Figure 10-6. Elevon Actuator Location

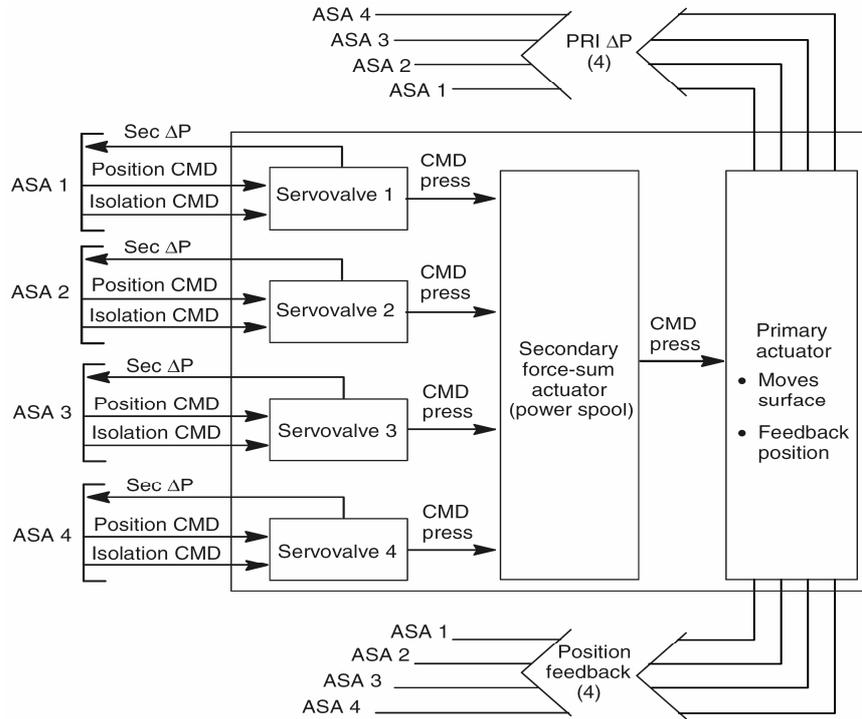


Figure 10-7. Typical Elevon Actuator

10.3.1 Switching Valve

Each elevon and rudder/speedbrake actuator has a switching valve assembly, consisting of two valves which connect to the three Orbiter hydraulic systems. The primary system and standby one system are connected to the first valve, and the standby two system and the output of the first valve are connected to the second valve (Figure 10-8). The output of the second valve is fastened to the actuator. Each valve's position is sent to downlink telemetry through a linear variable differential transformer (LVDT). Normal system operating pressure is 3000 psia and the spools will shift positions in the valve cylinders when the pressure of the controlling system pressure is less than 1200 to 1500 psia. For example, if the primary pressure was to fall below the controlling pressure (1200 to 1500 psia), the spool in the first valve assembly would be shuttled downward, removing the primary pressure from operation and allowing the standby one hydraulic system to take over. The output line from the first valve would show a slight drop in pressure, but not enough to cause the second valve spool to shuttle over. Note that each hydraulic system has a return system to the Orbiter hydraulics. The switching valve assembly is also used to receive hydraulic fluid returning from its elevon actuator and direct it to the hydraulic system in control at the time of operation.

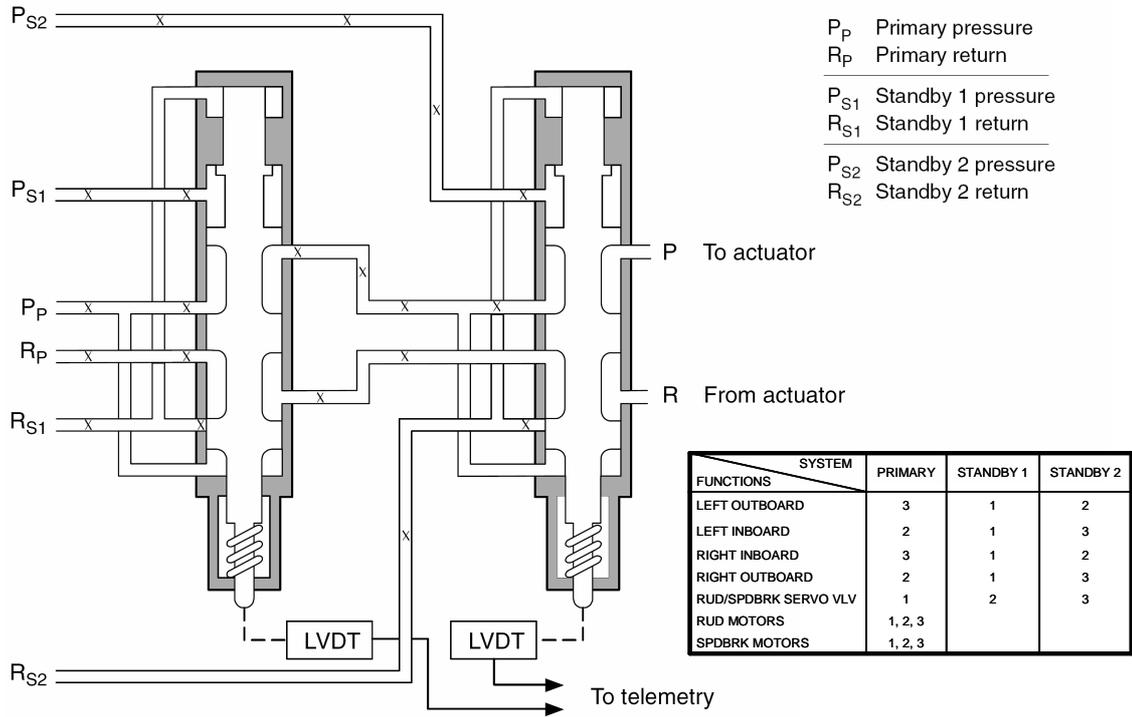


Figure 10-8. Switching Valve

10.3.2 Secondary Force-Sum Actuator

In order to generate the hydraulic pressure to move the primary actuator, a secondary force-sum actuator (power spool) is used, as seen in Figure 10-9 & 10-10. By shuttling the spool to the right, hydraulic source pressure of 3000 psig is allowed to flow to the back side of the ram in the primary cylinder, creating a positive pressure on the ram piston. In conjunction with this, hydraulic pressure in the front portion of the primary cylinder is relieved and routed back to the Orbiter hydraulic supply by way of the power spool. The pressure differential created by the power spool movement will cause the primary actuator to extend. By shifting the power spool to the left, differential pressure is generated on the main piston in the opposite way, causing the main ram to retract (shift to the left).

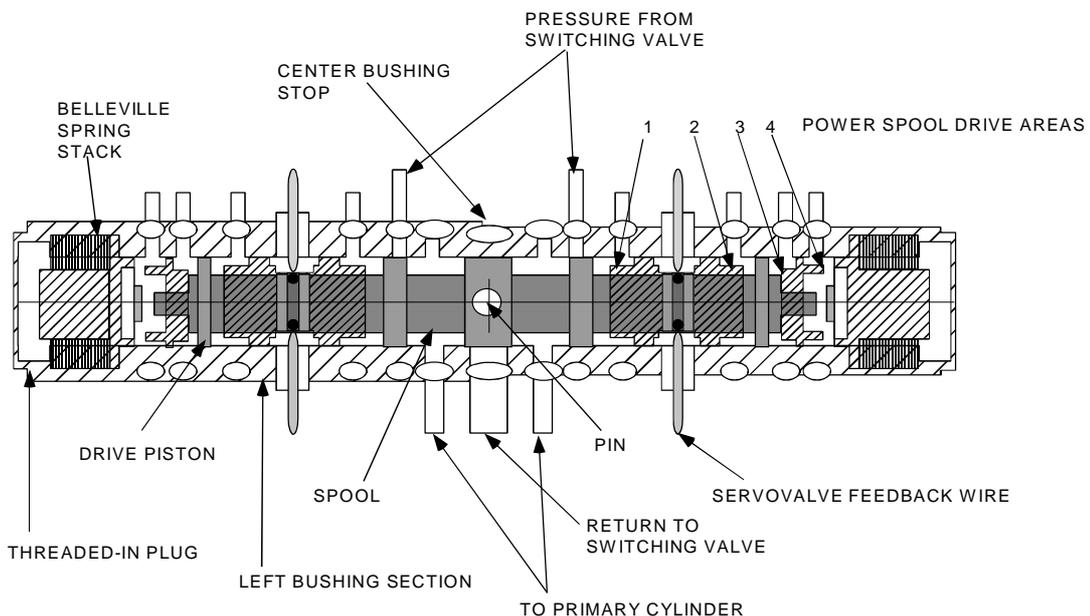


Figure 10-9. Power Spool

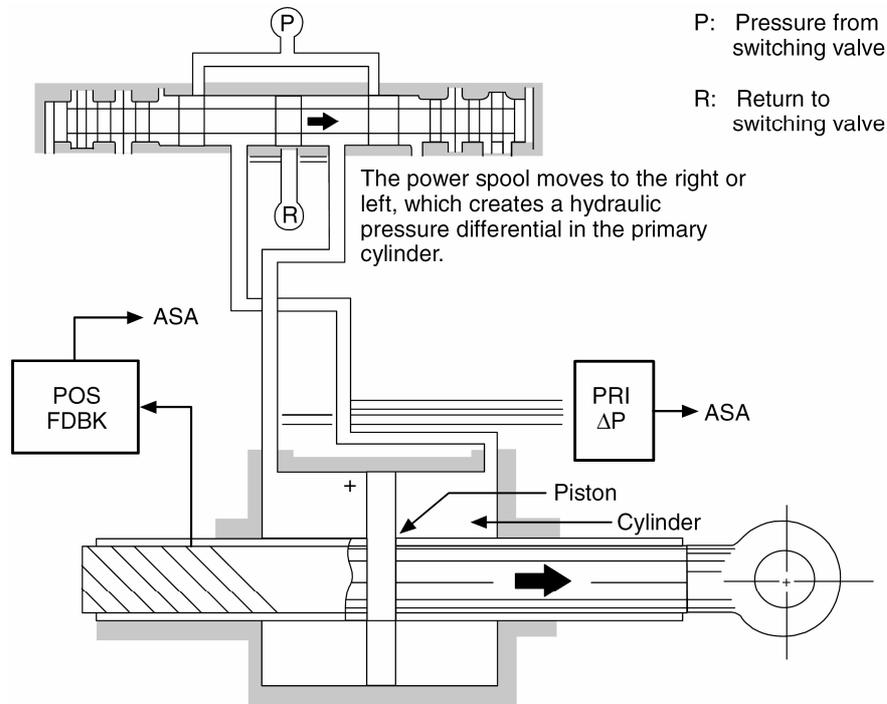


Figure 10-10. Secondary Force-sum Actuator

10.3.3 Servo Valve Assemblies

Each elevon actuator has four servovalves (Figure 10-11). These servovalves work together to shuttle the power spool. Each servovalve is made in two parts, a torque motor assembly and a hydraulic valve assembly (Figure 10-12). The torque motor assembly receives a position error command voltage which creates a magnetic force on the ends of the armature. This torquing motion rotates both the armature and the flapper about the flexure sleeve support. The flapper causes a hydraulic imbalance by closing off one nozzle and diverting flow to that end of the spool. This action moves the spool and opens fluid supply to one control port. The other control port is opened to the return flow (Figure 10-13). The fluid supply is the driving pressure which moves the secondary force-sum actuator.

Consider the example depicted in the Figures 10-13 and 10-14 below. The ASA has sent out a negative position error command voltage which causes the flapper valve to torque to the left. The end of the flapper then blocks the rightmost orifice into the central cavity. This raises the pressure on the right side, which translates to a push on the servovalve spool to the left. Its valve spool movement to the left blocks the rightmost fluid supply, opens the rightmost fluid return and the leftmost fluid supply. By opening the left fluid supply, high pressure fluid is allowed to flow to the secondary force-sum actuator, and pushes it to the left. In turn, this ports to the primary ram and causes it to retract, pushing the elevon up. Note that Figures 10-11 through 10-12 depict the opposite case of Figures 10-13 and 10-14. Note these figures depict a positive flow where the actuator extends.

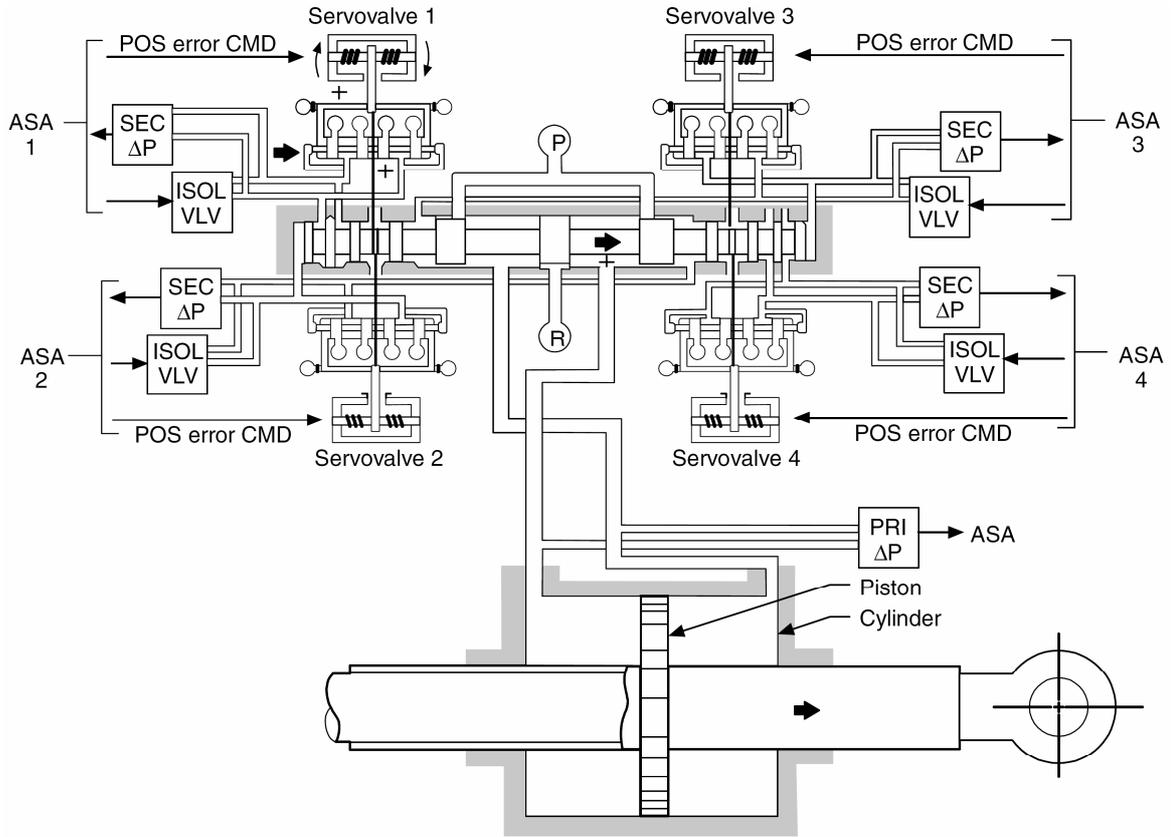


Figure 10-11. Typical Elevon Actuator

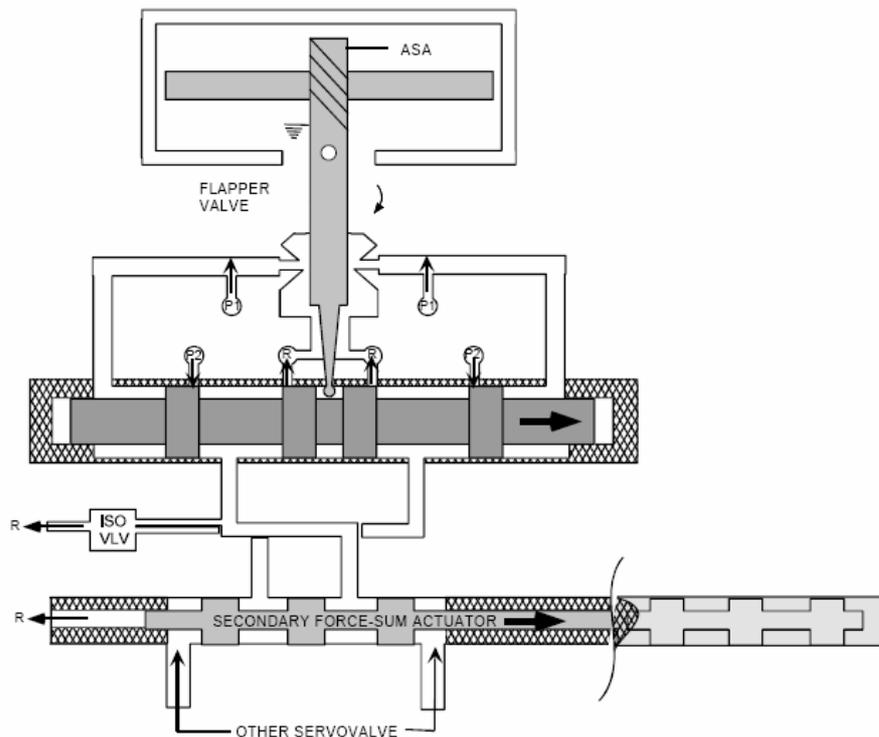


Figure 10-12. Servovalve Schematic including Secondary Force-Sum Actuator

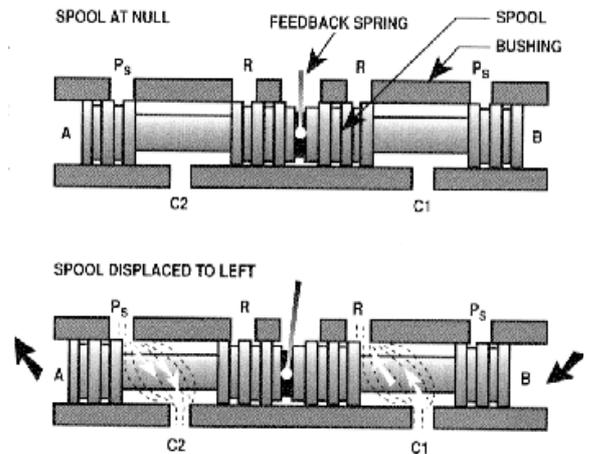
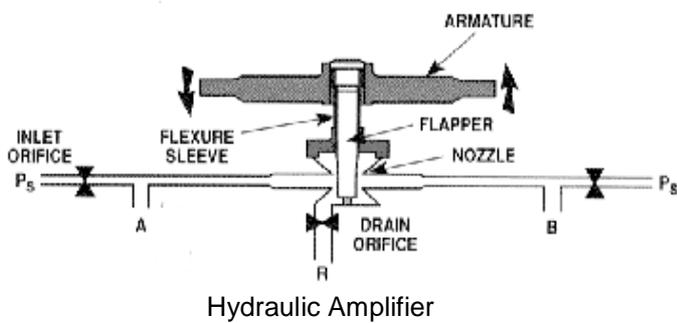


Figure 10-13. Flapper and Valve Spool assembly

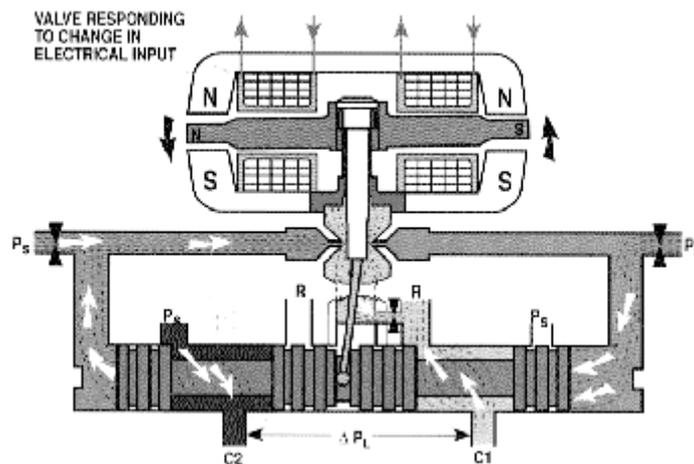


Figure 10-14. Servovalve Schematic (Secondary Force-Sum Actuator not shown)

Each servovalve receives driver output commands from its corresponding ASA. All ASAs for a particular actuator receive identical position commands and convert them to a proportional hydraulic pressure, which drives the secondary force-sum actuator. The four servovalves work together to push the secondary force-sum actuator (power spool) which in-turn generates the primary hydraulic pressure necessary to drive the main ram assembly. The main ram is connected to the elevon surface and an LVDT which sends back position signals to each ASA. As the primary actuator moves in response to the command pressure from the power spool, the PFB data is sent back to each ASA which sums the position signal of the primary actuator with the position command that is sent to the servovalve it controls on that actuator. As the primary actuator moves closer to the commanded position, the driver command to the servovalve is reduced. When the primary actuator reaches its commanded position, the servovalve driver command is canceled out and the command hydraulic pressure to the secondary force-sum actuator is removed.

10.3.4 Servovalve Feedback

The differential hydraulic pressure a servovalve generates on the secondary force-sum actuator (power spool) is known as secondary hydraulic pressure, while the differential hydraulic pressure the power spool generates on the primary actuator is the primary hydraulic pressure. The secondary hydraulic pressure generated by the servovalve is proportional to the position error command it receives from the ASA channel controlling it. All four servovalves receive the same position error commands, and thus issue similar differential hydraulic pressures to drive the power spool. If one of the four servovalves were to issue a

differential hydraulic pressure unequal to the other three servovalves, a resistive force is generated at the power spool where the four secondary hydraulic forces are summed to drive the power spool. An example: assume servovalves one, two, and three generate 1800 psia to drive the power spool to the right, as shown in Figure 10-11, but servovalve four only generates 1500 psia. Servovalve four is said to be dragging behind with a resistive pressure of 300 psia.

Each servovalve has its own secondary ΔP transducer designed to measure the resistance hydraulic force generated by its servovalve in relation to the other three counterparts as per the example given above. Also, each servovalve has an isolation valve. The isolation valve (when commanded) is used to alleviate the differential hydraulic pressure generated by its corresponding servovalve, thus relieving the resistance generated by the servovalve on the other three servovalves. Secondary ΔP data signals and isolation commands for a servovalve are received and sent respectively from the same ASA from which the servovalve receives its position error commands. In summary, each servovalve has its own secondary ΔP transducer and isolation valve. If that servovalve builds up a hydraulic resistance against its three counterparts, the pressure is measured by the secondary ΔP transducer and sent to the servovalve's controlling ASA.

If a problem develops within a servovalve, or it is commanded to a position different from that of the other servovalves within the actuator, secondary ΔP should begin to rise. Once the ASA detects the secondary ΔP at or above 2025 psia for more than 120 msec, it will send an isolation command to the servovalve. This will open the isolation valve, bypassing hydraulic pressure to the servovalve and causing its commanded pressure to the power spool to drop to zero. The bad servovalve is effectively removed from operation. Further discussion on isolation commands will be found in the text on the ASAs.

10.3.5 Position Feedback Circuit

The outer loop in the ASA is position feedback error. The position command from the GPC is differenced with the signal from the position feedback transducer in order to form an error signal or driver output. Thus, if a failure were to occur in the position feedback transducer on one channel, then driver output on that channel would be erroneous and would not agree with the driver output on the other channels. A resultant force fight would then occur between the channels, with the magnitude of the secondary pressure being dependent upon the size of the position feedback error. If the position feedback error is large enough, then the resultant force fight will cause the channel with the error to bypass.

The effects of aerosurface feedback bias levels on secondary ΔP were evaluated by Rockwell and previously described in section 10.2.4.

10.4 RUDDER/SPEEDBRAKE ACTUATION SUBSYSTEM OVERVIEW

The rudder/speedbrake actuation subsystem is comprised of a PDU and four geared rotary actuators (Figure 10-15). The geared rotary actuators move the rudder/speedbrake surfaces in response to command torque transmitted through the actuator drive shafts by the PDU. The PDU is made of five different subsystems: a mixer gearbox, a differential gearbox, six hydraulic motor and brake assemblies, two secondary force sum actuators (triplex power valve), and eight servovalves (Figure 10-16). The mixer gearbox drives the geared rotary actuators with rudder torque or speedbrake torque, in response to command torque from the rudder differential gearbox or the speedbrake differential gearbox, respectively. The differential gearbox drives the mixer gearbox in response to command torque from three hydraulic motors.

There are two differential gearboxes, one for rudder command torque and one for speedbrake command torque. Also, each differential gearbox sends position feedback data to each of the four ASAs. The hydraulic motor and brake assemblies receive hydraulic motor pressure from the secondary force-sum actuator and convert it into command torque to drive the differential gearbox. Three hydraulic motors work on each gearbox. When the hydraulic motors are not active, the brake assemblies engage, preventing the motors from free rotation. Hydraulic source pressure is routed to the hydraulic motors by the secondary force-sum actuator (triplex power valve) in response to the sum of the servovalve commands. The servovalve (secondary actuators) receives position commands and converts them to a proportional hydraulic pressure command to position the secondary force-sum actuator. In conjunction with this, each servovalve sends secondary hydraulic ΔP data signals back to its corresponding ASA channel to isolate failed channels or valves.

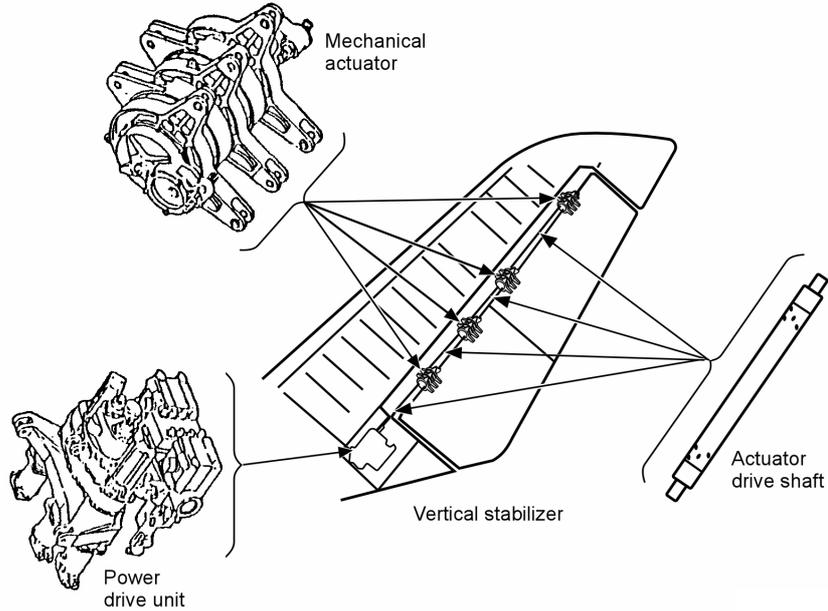


Figure 10-15. Rudder/Speedbrake Actuator Location

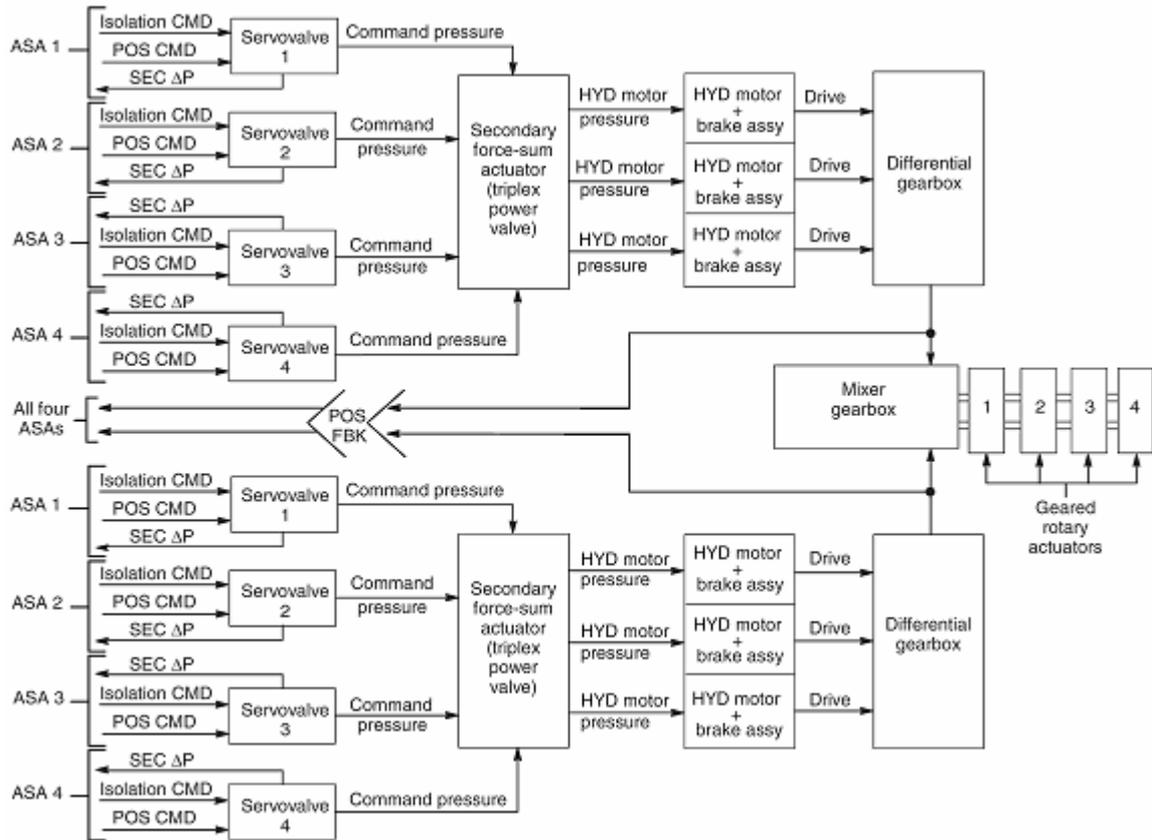


Figure 10-16. Rudder/Speedbrake Actuator System

In summary, there are two separate, identical command paths: one for rudder commands and one for speedbrake commands. Each path has four servovalves receiving position commands and converting them to hydraulic pressure to drive a triplex power valve (one for rudder and one for speedbrake). In response, the triplex power valve delivers hydraulic drive pressure to the

hydraulic motors. The motors convert this to mechanical torque energy which drives the differential gearbox. Depending on which gearbox it is, the mixer will be driven by either rudder command torque or speedbrake command torque. These torques are then transmitted to each of four geared rotary actuators connected in series by eight actuator drive shafts. Each subsystem making up the rudder/speedbrake actuation system will be discussed in detail in the following pages.

10.4.1 Rudder/Speedbrake Hydraulic Supply Overview

The rudder/speedbrake hydraulic system is supplied by a switching valve (see elevon switching valve Figure 10-8) that receives the three Orbiter hydraulic systems and produces one hydraulic output pressure source for the rudder/ speedbrake servovalves (Figure 10-17). Each rudder servovalve and speedbrake servovalve receives its hydraulic pressure from the same switching valve.

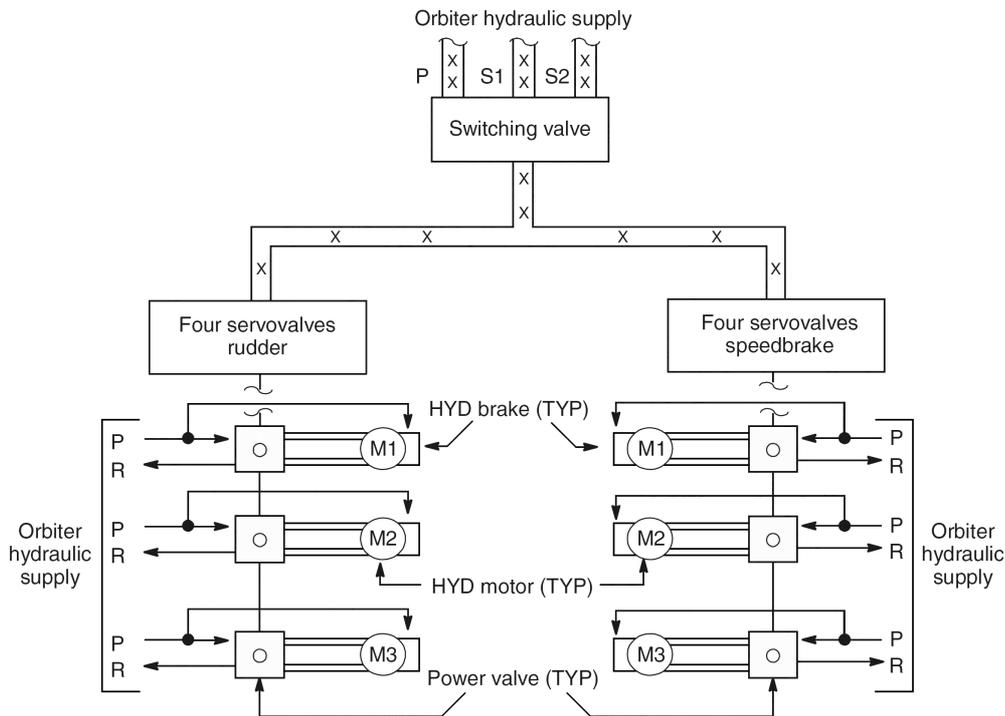


Figure 10-17. Rudder/Speedbrake Hydraulic Supply Overview

The servovalve command pressure actuates one power valve, which in turn mechanically actuates two more power valves by a mechanical link. Six power valves are linked together in two groups of three, each receiving hydraulic pressure directly from the Orbiter hydraulic supply. Six hydraulic motors receive hydraulic command pressure from the power valve (one power valve for one hydraulic motor). In conjunction with this, each motor has a hydraulic brake which receives hydraulic source pressure from the hydraulic feedline connected to its related power valve.

10.4.2 Hydraulic Motor/Brake Assembly Summary

There are three hydraulic motor/brake assemblies linked to each differential gearbox (Figure 10-18). Each hydraulic motor is activated by hydraulic drive pressure from its corresponding power valve. The hydraulic motors receive the same drive pressure which is transformed into identical clockwise or counterclockwise motion.. Per SODB vol-I 4.2.7.3.2, the rudder/speedbrake motor brakes are released whenever the hydraulic system pressure is up on a given system (e.g. if system 1 pressure is up, then motor 1 brake is released). The brakes release and relock (engage) at different pressure levels, as follows:

1. As pressure is increasing, the brakes initially release between 2100 and 2300 psi.
2. As pressure continues to increase, the brakes are fully released at 2700 psi, maximum.
3. As pressure is decreasing, the brakes relock between 2700 and 1000 psi, locking the drive shaft.

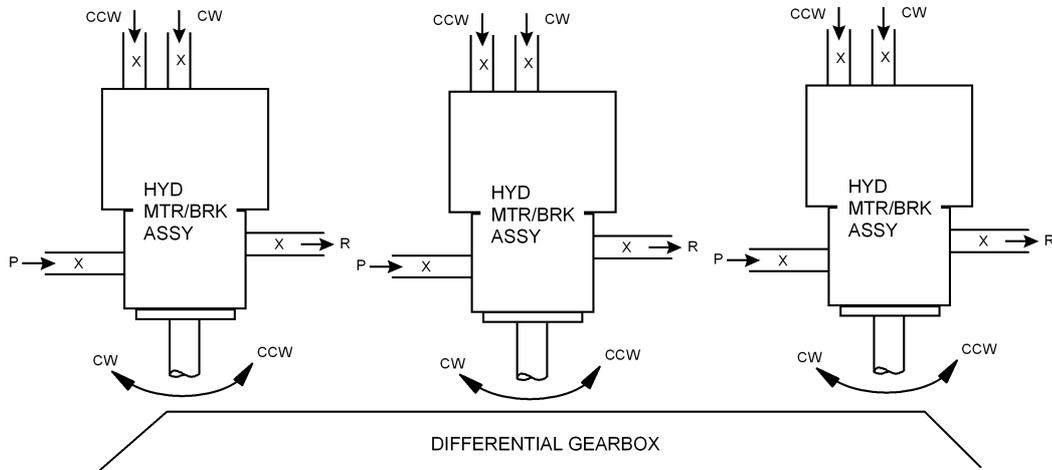


Figure 10-18. Hydraulic Motor/Brake Assembly

10.4.3 Hydraulic Motor/Brake Assembly

Figure 10-19 shows the internal mechanics of a hydraulic motor and brake assembly. Each hydraulic motor is comprised of a rotating barrel and drive shaft assembly, a fixed ramp (cam), and a hydraulic brake assembly. Hydraulic pressure from the power valve drives a series of pistons located in the rotating barrel against the fixed ramp, forcing the barrel to rotate clockwise (drive open) or counterclockwise (drive close). The rotation of the barrel causes the hydraulic motor drive shaft to rotate, which extends through the fixed ramp and through the hydraulic brake assembly. The hydraulic brake assembly consists of a disc brake assembly, which is fastened to the drive shaft, and brake shoes, which are fastened to the hydraulic motor casing. Orbiter hydraulic source pressure disengages the hydraulic brake by keeping the brake shoes from clamping down on the disc. Loss of hydraulic pressure allows the brake shoes to clamp down on the disc, preventing the drive shaft from rotating. In summary, the hydraulic motors convert hydraulic command pressure to mechanical torque energy.

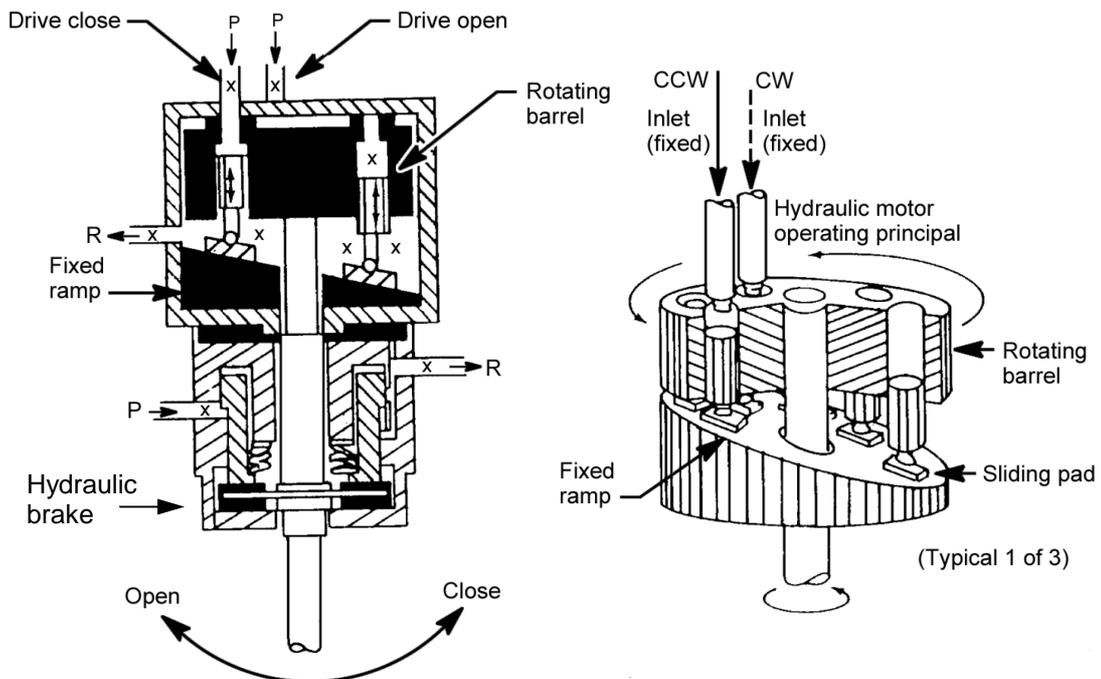


Figure 10-19. Hydraulic Motor/Brake Assembly (Internal Mechanics)

10.4.4 Triplex Power Valve (Secondary Force Sum Actuator)

Figure 10-20 shows the secondary force-sum actuator (triplex power valve) as it is fastened to three hydraulic motor/brake assemblies. The function of the triplex valve is to provide primary hydraulic pressure as needed for the rotation of the rotating barrel and drive shaft assemblies in the hydraulic motors. The triplex power valve is made up of two cylinder-spool assemblies connected by a mechanical linkage. The larger cylinder-spool assembly receives the secondary hydraulic command pressure from the four servovalves which causes the spool to shuttle to the right or left in a linear motion within the cylinder. The secondary hydraulic pressure inputs to the triplex valve are on either end of the larger cylinder with a valve assembly in the center of the cylinder which directs hydraulic pressure flow to one of two inputs on a hydraulic motor.

When the spool shuttles to the left, hydraulic pressure is supplied to the hydraulic motor, resulting in counterclockwise motion of the motor drive shaft and clockwise rotation when the triplex power valve spool shifts to the right. The large cylinder-spool assembly is fastened to a smaller cylinder-spool assembly by way of a mechanical linkage. This mechanical linkage shuttles the smaller spool to the left or right when the larger spool is shuttled to the left or right. The smaller cylinder-spool assembly is comprised of two valve assemblies which supply hydraulic pressure to two other hydraulic motors. These valves function in the same manner as the one located in the larger cylinder-spool assembly. Note that each of the three valves which supply primary hydraulic pressure to the motors utilizes separate Orbiter hydraulic systems. This is done so that if any one hydraulic system fails, the PDU will still be able to function nominally. The large cylinder-spool assembly also has four servovalve feedback wires which are used to relay the power valve position to the servovalve for the purpose of regulating the spool shuttle velocity.

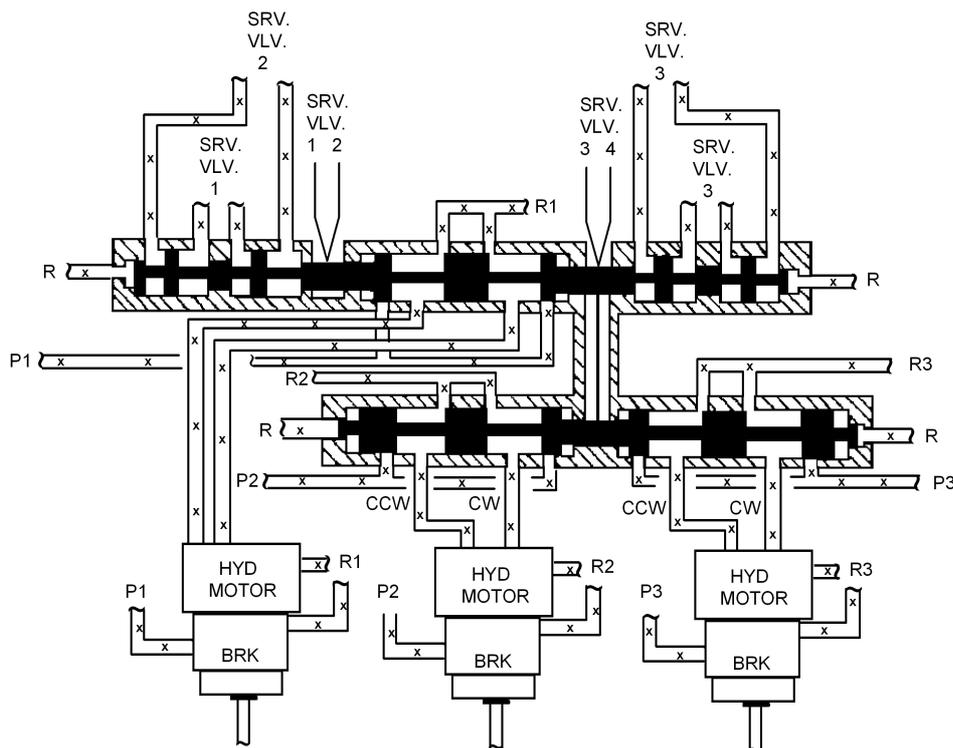


Figure 10-20. Triplex Power Valve

In summary, the triplex power valve receives secondary and primary hydraulic pressure as inputs. Its outputs are primary hydraulic pressure to the hydraulic motors and power spool position via the power valve FDBK wires.

10.4.5 Rudder/Speedbrake Power-Drive Unit (PDU)

Figure 10-21 shows a diagram of the rudder/speedbrake valve modules located in the PDU. Located in the lower center of the diagram is one of two triplex power valves with the PDU switching valve in the center of the structure.

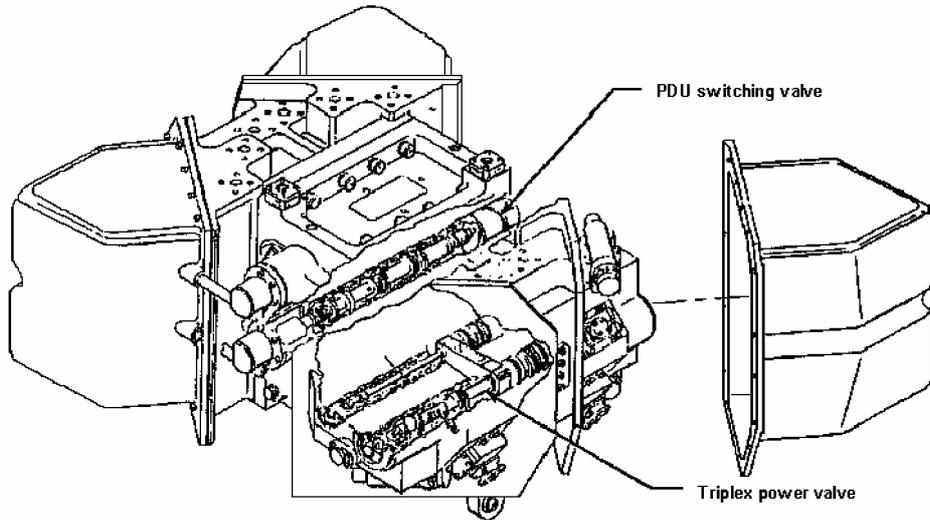


Figure 10-21. Rudder/Speedbrake Valve Module

10.4.6 Servovalve Assemblies

Four servovalve assemblies sum their hydraulic pressure to drive the triplex power valve. Figure 10-22 only shows two of the four servovalve assemblies for clarity. Each servovalve has its own secondary ΔP transducer and isolation valve, and functions identically to those servovalves used on the elevon actuators. Each servovalve's function is to generate differential hydraulic pressure (along with three other servovalves) to drive the triplex power valve to a commanded position. The inputs to the eight servovalves (four for rudder operations and four for speedbrake operations) are hydraulic pressure from the switching valve and position and isolation commands from the corresponding ASA. The servovalves output secondary hydraulic pressure to drive the triplex power valve and secondary ΔP FDBK data to their corresponding ASAs.

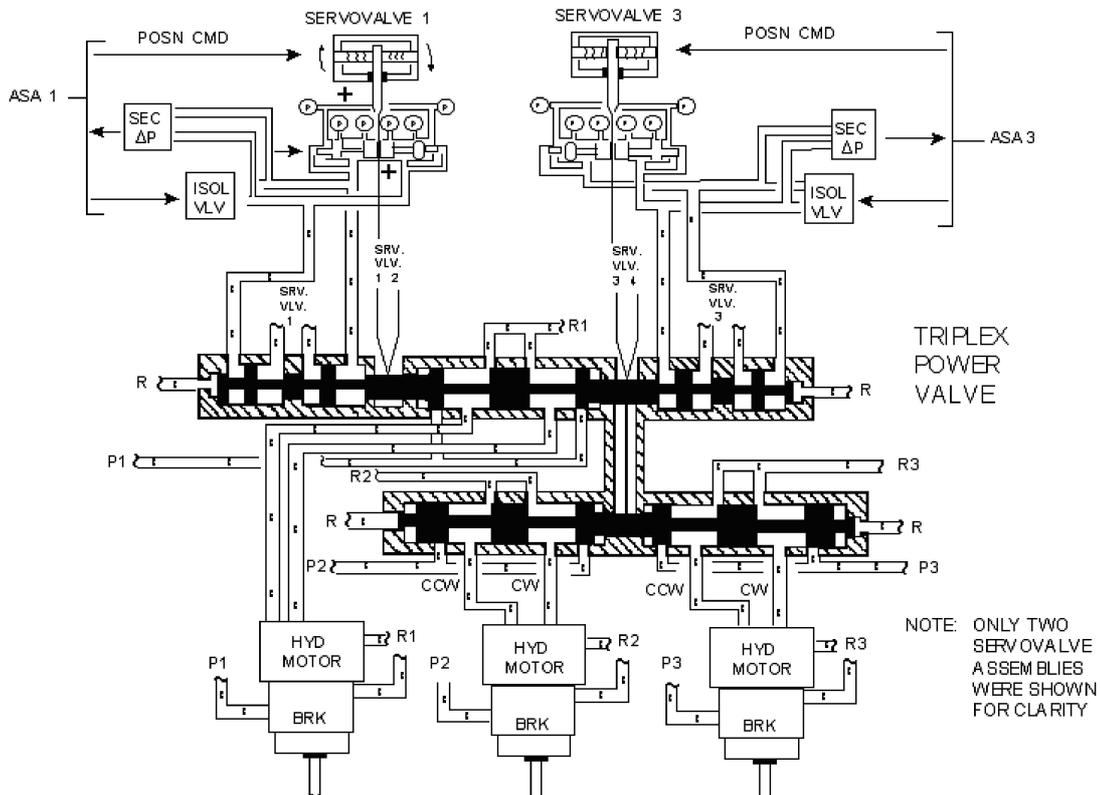


Figure 10-22. Servovalve Assemblies

10.4.7 Differential Gearbox

The differential gearbox is comprised of a gear assembly which combines three hydraulic motor drive shafts into one output shaft. There are two differential gearbox assemblies: one for rudder torque and one for speedbrake torque. Figure 10-23 shows the internal mechanism of a differential gearbox. Three hydraulic motor torque drives are transmitted to form a single torque output through orbital gear assemblies connected back-to-back. The center torque input (2) is geared up to the center axis of the first orbital gear assembly. The right torque input (3) is geared to the outer sprocket of the first orbital gear assembly and the left torque input (1) is geared to the outer sprocket of the second orbital gear assembly. The output of the first gear assembly is connected to the center axis input to the second orbital gear assembly. The second orbital gear assembly output is the combined torque of the hydraulic motors which is directed to the mixer gearbox. The output shaft to the mixer gearbox has a position transducer (not shown in Figure 10-23) which sends a position signal back to the ASAs. If one hydraulic motor were to fail, the differential gearbox would function at a normal rate. If two hydraulic motors were to fail, the differential gearbox would function at a reduced rate of 50 percent. In summary, the function of the differential gearbox is to combine the hydraulic motor output torque into one output torque to be used in driving the main gearbox.

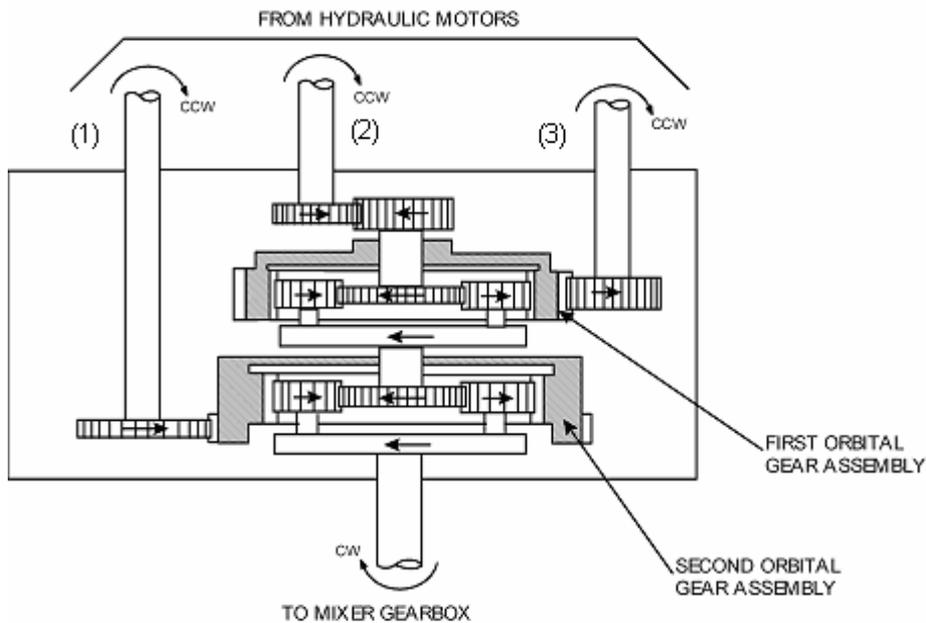


Figure 10-23. Differential Gearbox

10.4.8 Mixer Gearbox

The function of the mixer gearbox is to transmit speedbrake torque or rudder torque (both at the same or separate times) to two drive shafts leading to the geared rotary actuators. The mixer gearbox is comprised of two orbital gear assemblies being driven by the outputs of the rudder differential gearbox or the speedbrake differential gearbox. Figure 10-24 shows differential gearbox inputs and how they are geared to generate the output torque, which is transmitted to the geared rotary actuator. The speedbrake torque input is geared to the center axis of the orbital geared assemblies. This causes the output shafts to rotate in opposite directions. The rudder torque is geared to the outer sprocket of the orbital geared assemblies, which causes the output shafts to rotate in the same direction.

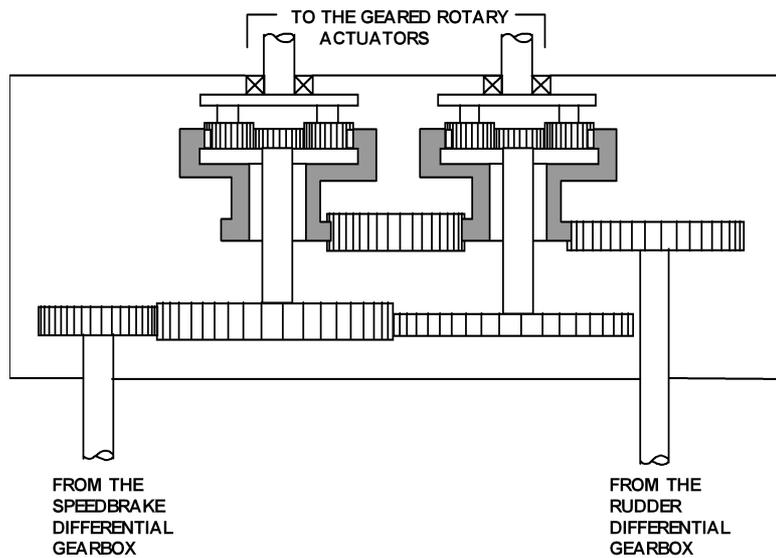


Figure 10-24. Mixer Gearbox

10.4.9 Rudder/Speedbrake Geared Rotary Actuator

The function of the geared rotary actuators is to move the rudder/speedbrake aerosurface panels in response to the drive torque output from the mixer gearbox. Figure 10-25 shows a rotary actuator. There are two actuator drive shaft inputs (one at either end of the actuator and two aerosurface panel fastening points - one per aerosurface panel). The aerosurface fastening links are the components which rotate about the actuator casing, causing the aerosurface panels to move.

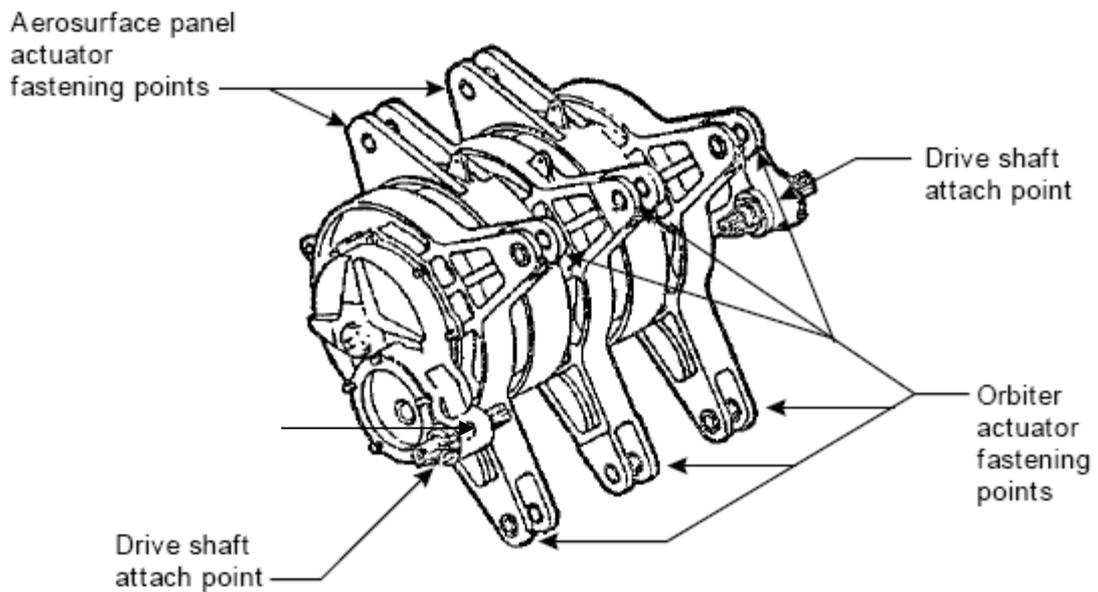
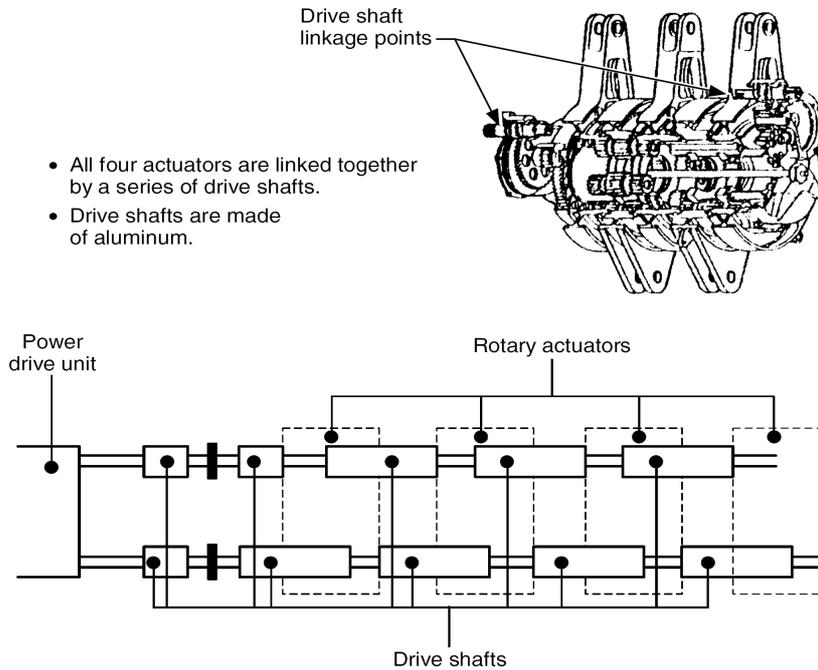


Figure 10-25. Rudder/Speedbrake Geared Rotary Actuator

10.4.10 Drive Shaft Connections

All four rotary actuators are linked together by eight aluminum drive shafts (Figure 10-26). They are configured in two rows of four actuators and are fastened to the rudder/speedbrake PDU. The PDU supplies the command torque to drive the geared rotary actuators.



- All four actuators are linked together by a series of drive shafts.
- Drive shafts are made of aluminum.

Figure 10-26. Drive Shaft Connections

10.4.11 Actuator Gear Assemblies

Each rotary actuator is made up of two driver gear assemblies, a series of satellite gear assemblies, and two center drum assemblies, as seen in Figure 10-27. The driver gear assemblies are driven by the rotary motion of the driver gear on the drive shaft. The rotary motion of the driver gear assembly turns a center drum assembly which rotates a series of satellite gears. The satellite gear assemblies are made up of three gears - two small gears which are driven by the center drum gears and one large gear which drives the drive bracket that moves the aerosurfaces. There are two drum assemblies, each having its own set of satellite gears and drive gear assemblies. The center drum assemblies are free to rotate in opposite directions (this causes speedbrake motion) or in the same direction (this causes rudder motion).

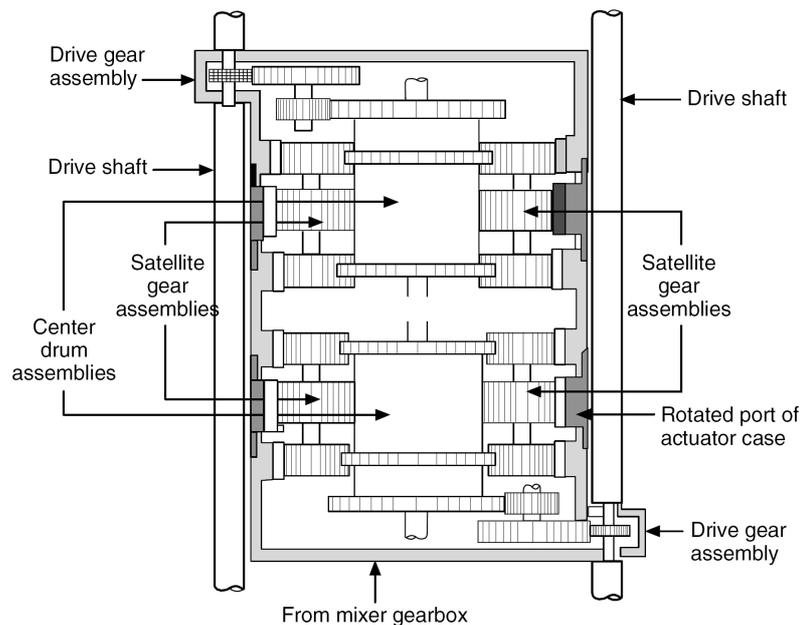


Figure 10-27. Typical Geared Actuator (Gear Assembly)

10.4.12 Rudder Movement

Figure 10-28 shows an axial view of a geared rotary actuator. Rudder movement is caused when the actuator brackets move in the same direction. To drive the actuator brackets in the same direction, the actuator drive shafts must rotate in the same direction.

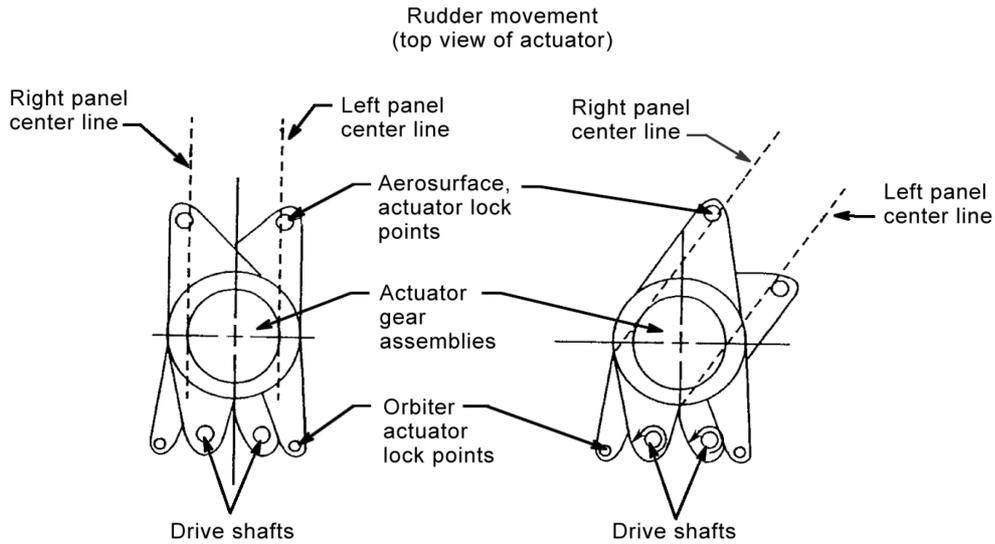


Figure 10-28. Rudder Movement (top view of actuator)

10.4.13 Speedbrake Movement

Figure 10-29 shows an axial view of a geared rotary actuator. Speedbrake movement is caused when the actuator brackets move apart from each other. Speedbrake movement occurs when both actuator drive shafts move in opposite directions.

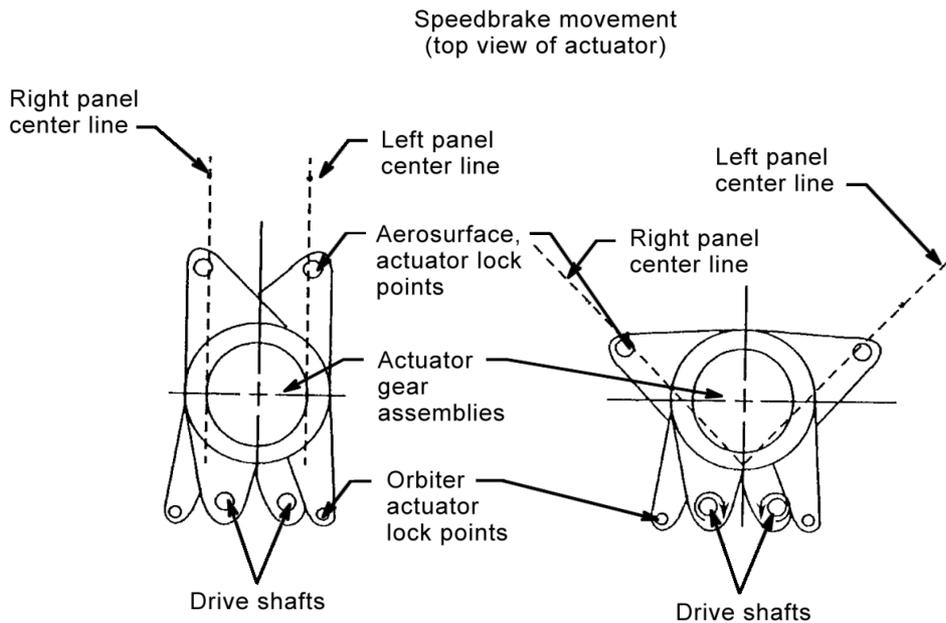


Figure 10-29. Speedbrake Movement (top view of actuator)

10.4.14 Rudder/Speedbrake ASA Driver

There are four ASA drivers controlling the rudder/speedbrake system (one rudder/speedbrake driver in each ASA channel). Each ASA driver subsystem functions identically to the elevon ASA drivers except where there is no PRI ΔP FDBK. Each rudder/speedbrake driver controls one of the four servovalves in the rudder/speedbrake PDU.

10.5 BODY FLAP ACTUATOR

The BF actuator system is comprised of four rotary actuators, a PDU, and four interconnecting drive shafts. All the BF actuator subsystems are located in the aft end of the Orbiter between the BF and the Orbiter structure (Figure 10-30).

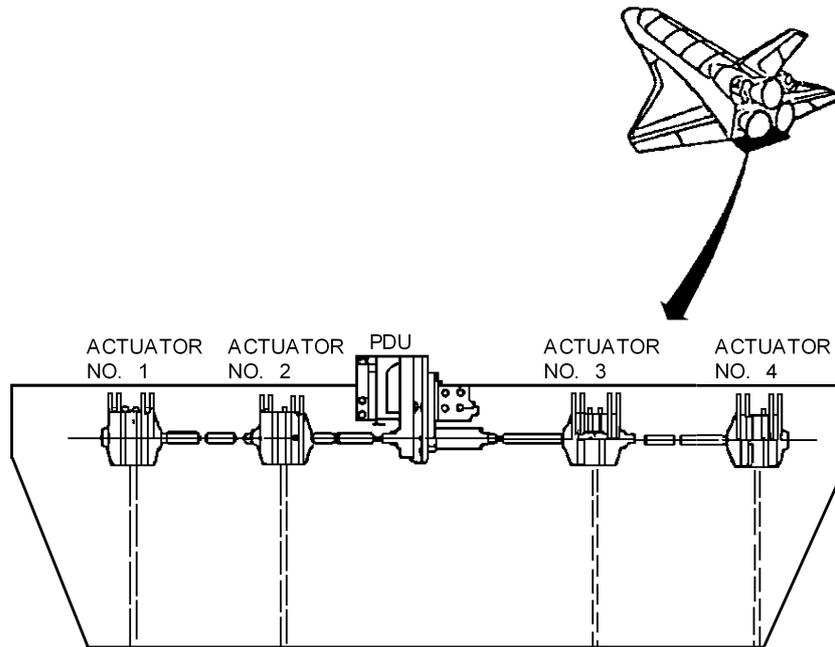


Figure 10-30. Body Flap Actuator Locations

10.5.1 BF PDU

The BF PDU is comprised of three solenoid actuator assemblies, three hydraulic motors, and a differential gearbox. The differential gearbox turns the geared rotary actuator drive shafts in response to command torque from three hydraulic motor/brake assemblies. Also, the differential gearbox issues PFB signals to each BF ASA channel. The hydraulic motors convert hydraulic command pressure into mechanical torque pressure to drive the differential gearbox. The brake assemblies disengage when the hydraulic brake pressure is applied, allowing the hydraulic motor drive shafts to rotate freely. When the hydraulic pressure is removed, the brakes engage, locking up the hydraulic motor drive shafts. The solenoid assemblies send hydraulic pressure to the hydraulic motors in response to up or down commands from the ASAs. Pilot solenoids are used to issue up or down commands to the actuators in response to enable commands from the system enable solenoid and up or down commands from the ASAs. The actuators are not only driven hydraulically but all three are mechanically driven by means of a mechanical linkage connecting the three actuators together. Each subsystem in Figure 10-31 is discussed in greater detail in the following pages.

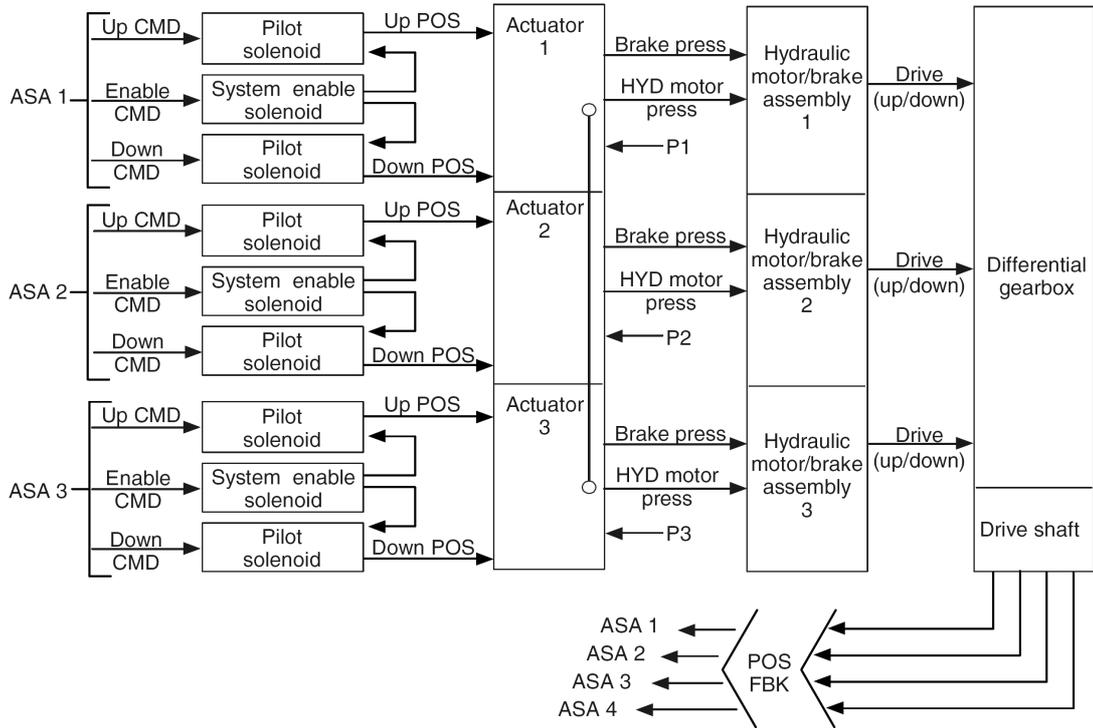


Figure 10-31. Body Flap PDU

10.5.2 BF Actuator Hydraulic Supply Overview

The body flap hydraulic supply (Figure 10-32) employs all three Orbiter hydraulic systems to drive the BF PDU (one hydraulic system per hydraulic motor). Three solenoid valves make up a control valve (two pilot solenoids and one system enable solenoid). The control valve activates the power spool by routing hydraulic pressure to it. The power spool (actuator) directs hydraulic pressure to a hydraulic motor in response to control valve hydraulic pressure. In conjunction with this, the brake assembly on each hydraulic motor is disengaged when it receives hydraulic pressure. The three power spools are linked together, providing identical outputs to their respective hydraulic motors by maintaining equal movement of each valve.

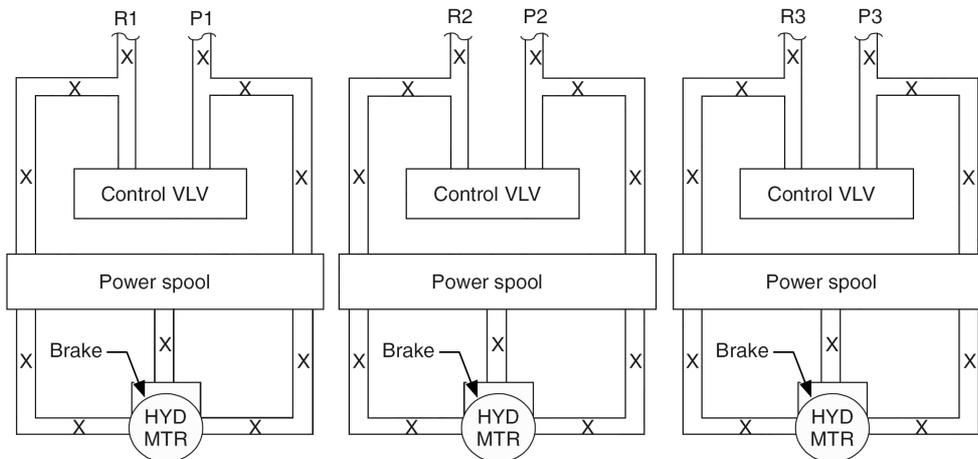


Figure 10-32. Body Flap PDU Hydraulic Supply

10.5.3 BF Hydraulic Motor/Brake Assemblies

There are three hydraulic motor/brake assemblies linked to a differential gearbox (Figure 10-33). Each hydraulic motor is activated by hydraulic drive pressure from its corresponding power valve. The hydraulic motors receive the same drive pressure which is transformed into identical clockwise or counterclockwise motion. Per SODB 4.2.7.3.2, the BF motor brakes are released only when the hydraulic pressure is up, the enable command solenoid is activated, and the up or down drive solenoid is activated. The brakes release and relock (engage) at different pressure levels, as follows:

1. As pressure is increasing, the brakes release between 1000 and 2400 psi.
2. As pressure is falling, the brakes relock between 2400 and 1000 psi, locking the drive shaft.

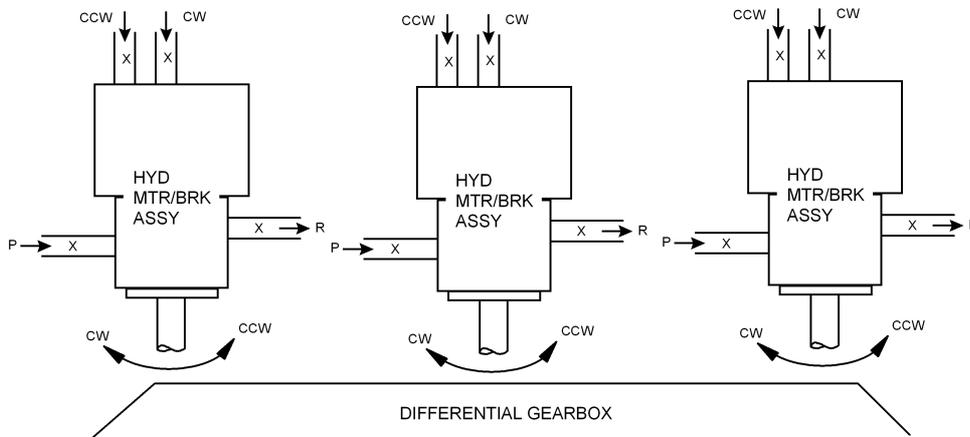


Figure 10-33. Body Flap Hydraulic Motor/Brake Assemblies

10.5.4 BF Actuator

The function of the BF actuator is to provide primary hydraulic pressure as needed for the rotation of the rotating barrel and drive shaft assembly in the hydraulic motor. Figure 10-34 shows the mechanism comprising a BF actuator. The actuator valve consists of a spring loaded valve which is held in a neutral position. Hydraulic pressure can be applied at either end of the spool, causing it to linearly shift to the left or right. When the valve is shuttled, it first allows primary hydraulic pressure to flow to the hydraulic motor brake assembly to disengage the brake. Then, depending on which way the valve spool has shifted, hydraulic pressure will flow to drive the rotary barrel in a clockwise or counterclockwise direction. Each power spool assembly has a recirculation valve which opens if the hydraulic pressure drops below 800 psig to allow hydraulic pressure to bypass the actuator (if the hydraulic pressure is this low, it is needed more elsewhere).

10.5.5 BF Solenoids

Each power spool assembly has its own control valve assembly (Figure 10-34). The control valve assembly is made up of a system enable solenoid, an UP pilot solenoid, and a DOWN pilot solenoid. The function of the system enable solenoid is to supply hydraulic pressure (3000 psig) to the two pilot solenoids in response to a command signal from the ASA BF circuit. The function of the pilot solenoids is to supply hydraulic pressure to the actuator power valve in response to a command signal from the ASA BF circuit. Each solenoid valve has an electrical solenoid which activates a spring closed cylinder-spool assembly opening the valve. In the case of the system enable solenoid, this allows hydraulic pressure to flow to the two pilot solenoids. When either pilot solenoid receives a discrete command, it opens, allowing the hydraulic source pressure to flow to the power spool. The power spool is then shuttled, allowing the hydraulic motor brake assembly to be disengaged and the hydraulic motor rotating barrel to begin rotation. To stop the process, the discrete command is removed from the pilot solenoid valve, allowing the power spool to shift to a neutral position.

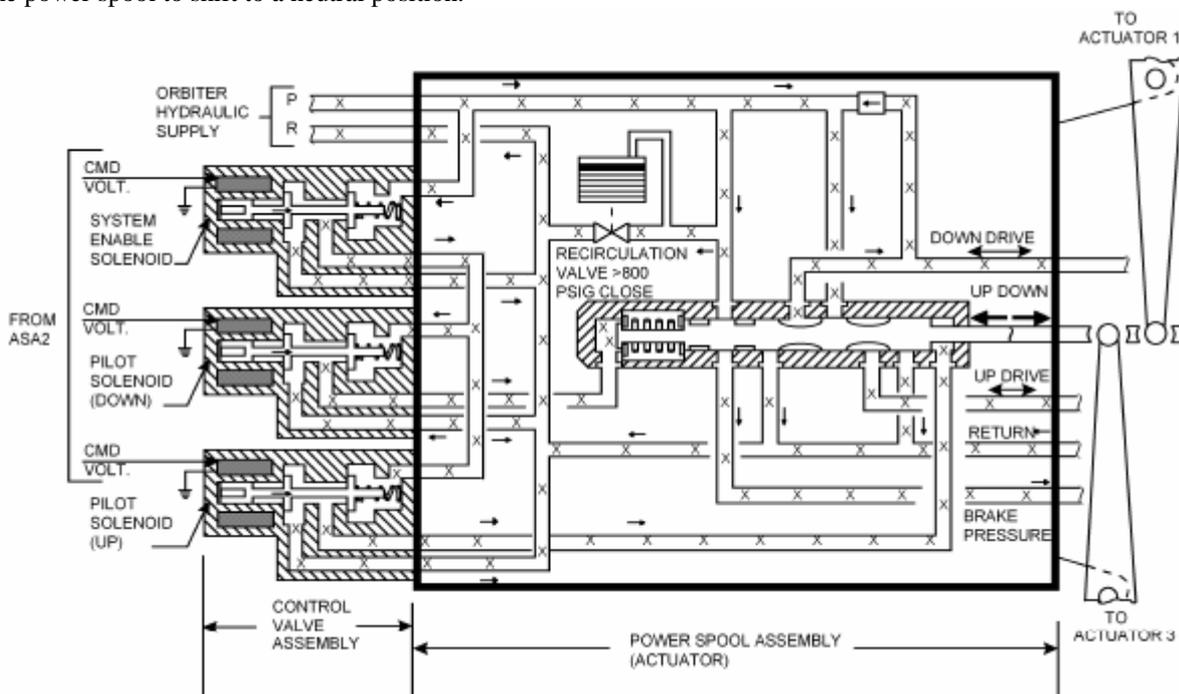


Figure 10-34. Body Flap Solenoids/Hydraulic Actuator

10.5.6 BF Actuator Linkage

Two pivot linkage assemblies are used to enable the BF actuators to mechanically work together (Figure 10-35). Any one BF actuator can mechanically drive the other two actuators through the actuator linkage assembly even if the non-driving ASA(s) has failed or in the presences of multiple channel failures. Note that actuators one and three have the same hydraulic output configuration, while actuator two's hydraulic output configuration is different, due to opposite linear motion in response to identical solenoid hydraulic command pressure and mechanical linkage actuation.

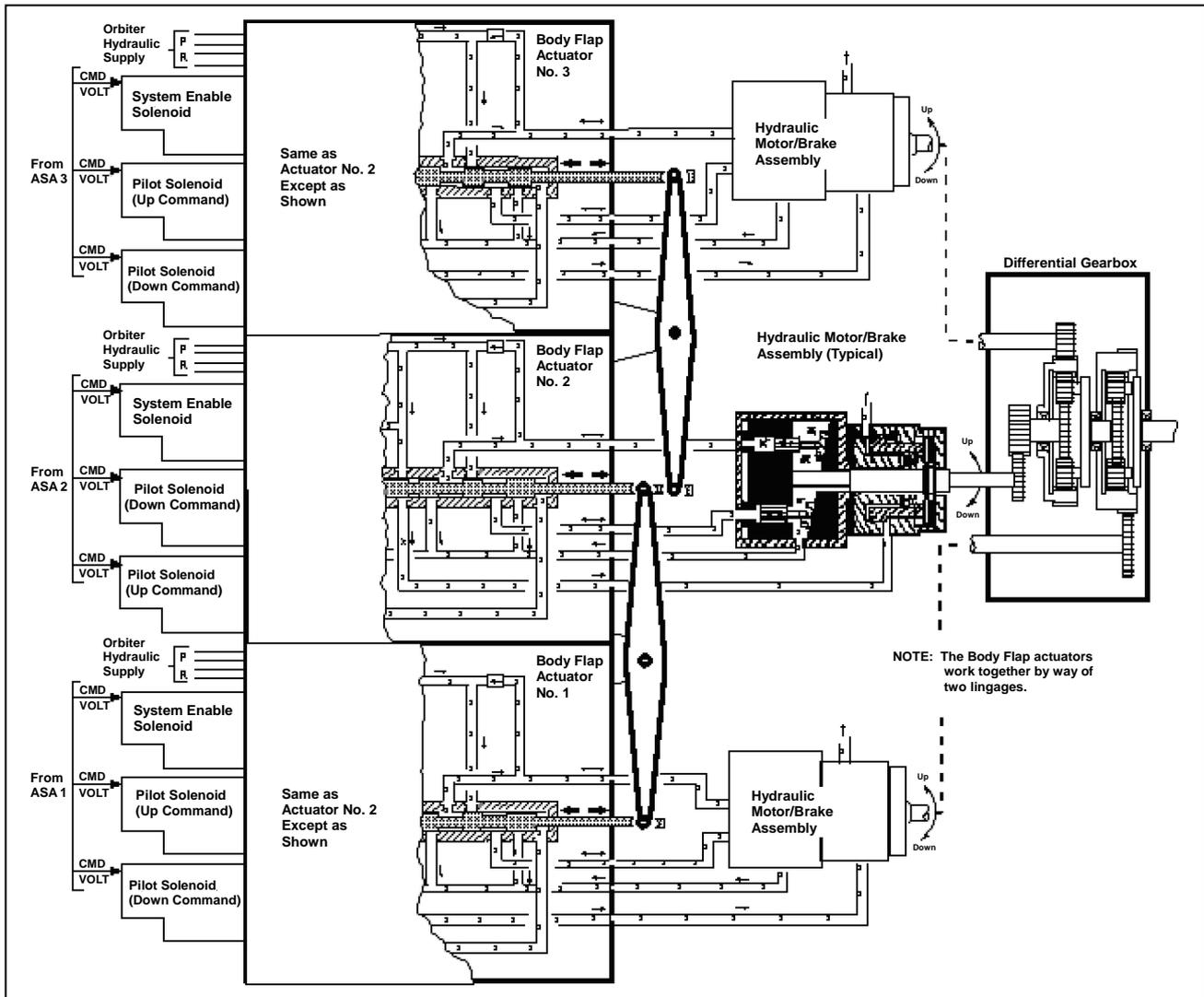


Figure 10-35. Body Flap Actuator Linkages

10.5.7 BF Differential Gearbox and Rotary Actuator

The differential gearbox is set up the same way as the differential gearboxes in the rudder/speedbrake PDU (Figure 10-36). The BF differential gearbox combines the three hydraulic motor torque outputs into one torque drive. This drive torque is transmitted to the four geared rotary actuator assemblies via a bevel gear assembly which is fastened to the actuator drive shaft. The output of the differential gearbox has four-position transducers which send PFB data to each BF ASA channel. Once the geared rotary actuators receive the drive shaft torque, they respond by generating a rotary motion which drives the BF.

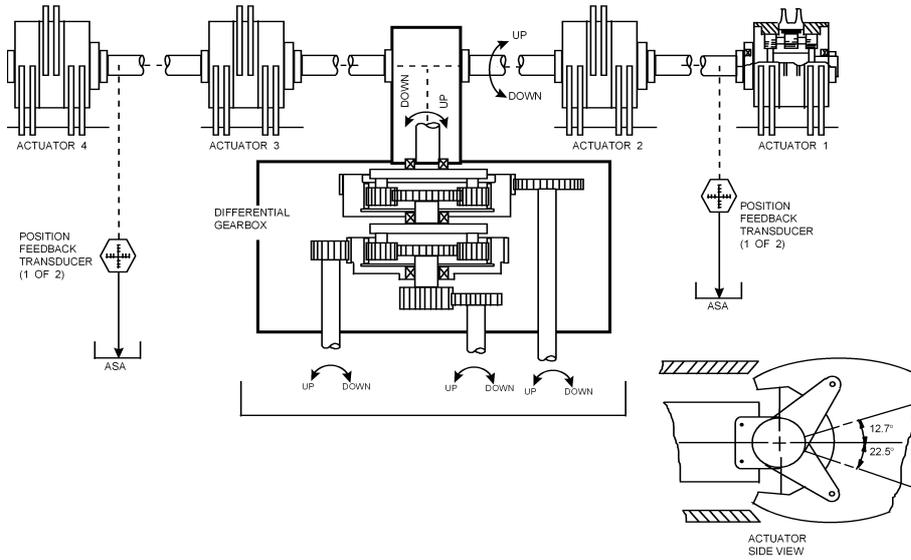


Figure 10-36. Body Flap Differential Gearbox and Rotary Actuators

10.5.8 BF Signal Flow

The BF enable commands, up commands, and down commands are generated in the GPCs and sent to BF channels 1, 2, and 3 via FA MDMs 1, 2, and 3. These discretives activate the solenoid valves in the BF PDU as needed (Figure 10-37). A four-position transducer on the differential gearbox output shaft issues position feedback signals to all four BF channels and on to the four GPCs via the FA MDMs.

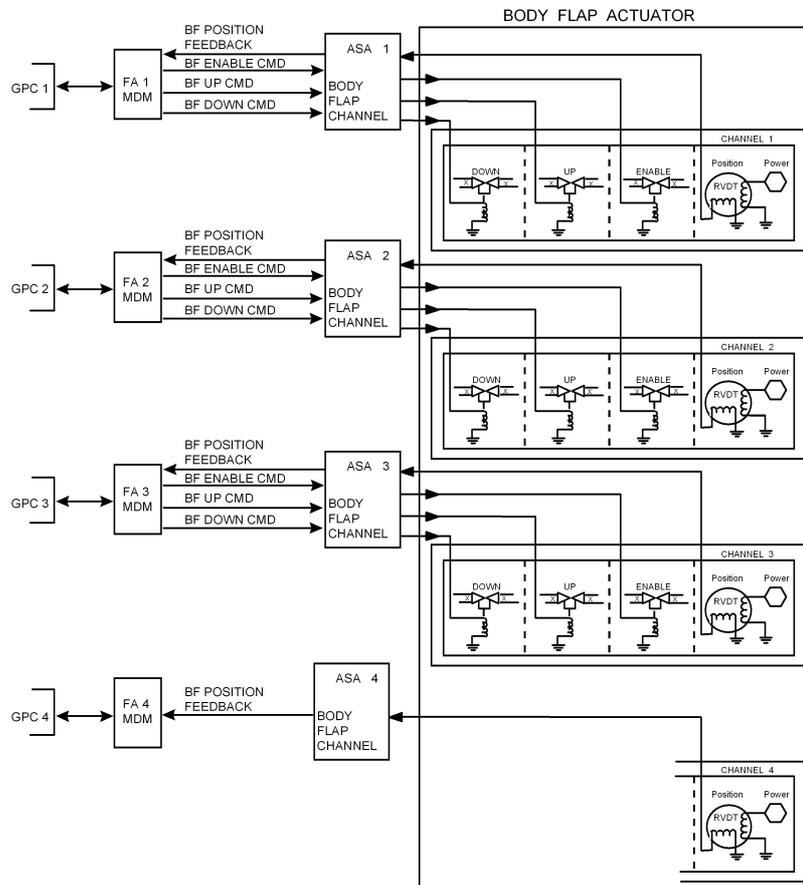


Figure 10-37. Body Flap Signal Flow

10.5.9 BF Channel

The function of the BF channel is to send command voltages to the system enable solenoid and pilot solenoids in response to enable, up, and down commands from its corresponding GPC (Figure 10-38). This is a function of BF channels 1, 2, and 3. A function of all four channels is to send PFB data to their corresponding GPC. The BF commands go through a gate, which monitors ac/dc power for the ASA channel, and then through an amplifier where discrete voltages are sent to the PDU solenoids. If there is a power failure, no discrete will be issued to the solenoids. Note that the loss of ASA isolation power will cause the loss of the corresponding ASA BF commands. This can be seen on drawing SSSH 9.11, area J-1.

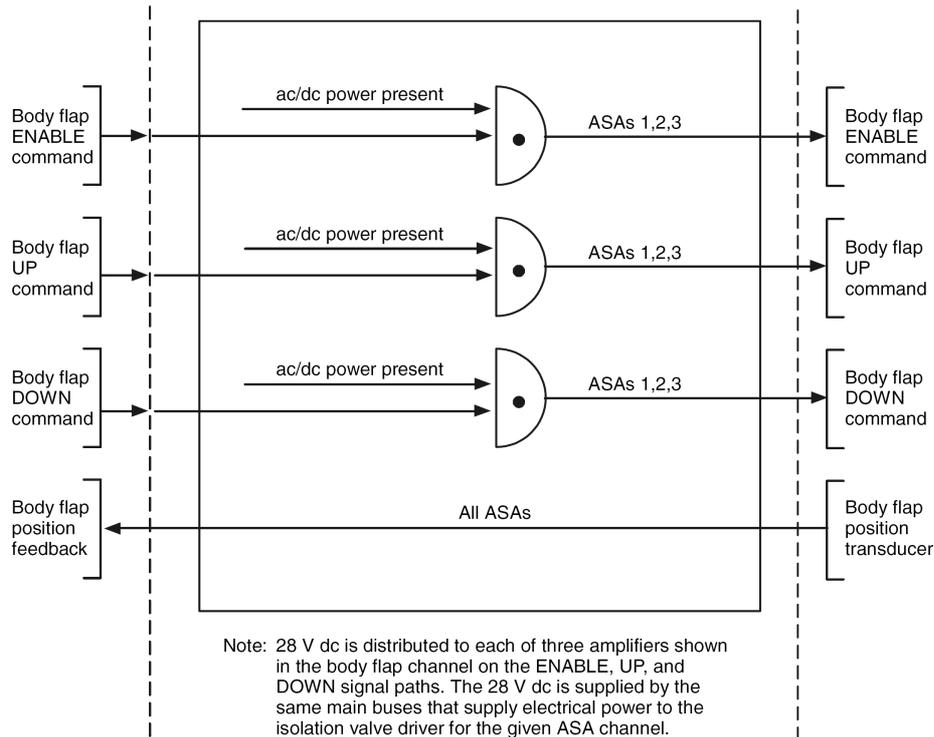


Figure 10-38. Body Flap Circuit

10.6 ELEVON, RUDDER AND SPEEDBRAKE COMMAND SOFTWARE DATA FLOW

Flight control software within the GPCs generates the aerosurface position commands that are forwarded to the aeroactuator Subsystem Operating Program (SOP). The SOP employs I-loaded biases and scale factors to convert the commands to volts dc that the ASA can use (referred to as command compensation). These compensated commands are sent to the ASA where they become position error commands that move the aerosurfaces. Refer to Figure 10-39.

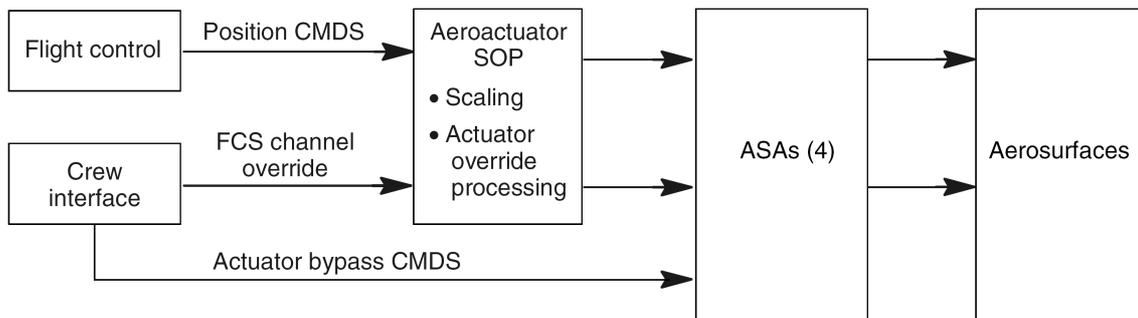


Figure 10-39. Aerosurface Command Software Data Flow

Channel override commands are generated when a crew member takes the FCS CHANNEL switches to OVERRIDE. These commands are also processed by the aeroactuator SOP and routed to the ASAs, where they reset any bypassed secondary actuators and override the fault detection logic. Reset commands are the equivalent of override commands and are set by crew keyboard entries via the FCS C/O DED display in OPS 8 or SPEC 53 in OPS 1, 3, or 6. Note that the override logic terminates with the Aerosurface Actuator SOP at the SSME Stow Complete Event (Event 43B) or transitioning to GN&C OPS 9. (Reference FSSR 16, 4.50.4)

Actuator bypass commands are generated by taking an FCS CHANNEL switch to OFF or by keyboard commands. SPEC 53 and the FCS C/O DED display both provide this latter capability. All bypass commands travel directly to the ASA. Taking an FCS CHANNEL switch to OFF will bypass all the secondary actuators on that channel. Keyboard entries will allow selection of individual actuators for bypassing.

10.6.1 Elevon Feedback Software Data Flow

Quad redundant elevon position feedbacks (PFBs) and primary delta pressures (ΔP s) are routed from the elevons through the ASAs to RM for processing. RM consists of two parts, Fault Detection, Isolation, and Reconfiguration (FDIR) and the Selection Filter (SF). FDIR compares the redundant data and checks it for out-of-tolerance conditions. The SF takes the available data inputs from FDIR and reduces them to a single selected parameter. The selected data is then forwarded to the SOPs, which convert them to engineering units for use in flight control software, crew displays, and C&W (Figure 10-40).

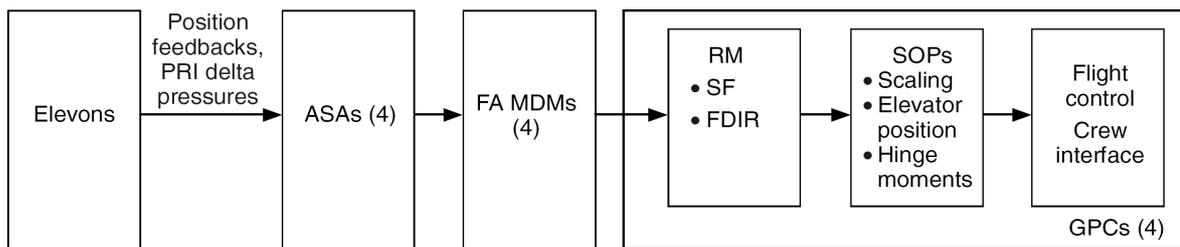


Figure 10-40. Elevon Feedback Software Data Flow

10.6.2 Elevon PFB Redundancy Management

The ELVN PFB RM function works on quad-redundant PFB's from each of the four elevon panels. As already stated, RM consists of FDIR and SF. Elevon PFB FDIR operates per elevon. That is, FDIR on the left inboard (LIB) and the left outboard (LOB) may be doing different things. For each elevon, FDIR compares the PFBs for channels 1, 2, and 3 against each other as follows:

$$\begin{aligned}
 |PFB1 - PFB2| &< \text{LIMIT} \\
 |PFB2 - PFB3| &< \text{LIMIT} \\
 |PFB1 - PFB3| &< \text{LIMIT}
 \end{aligned}$$

If any two results exceed the I-load limit (limit = 2.38°) three consecutive times, the common parameter is declared failed. Notice that the channel 4 feedback is not included in the scheme. This value is normally kept in standby. However, when a channel 1, 2, or 3 parameter is failed, channel 4 is used instead and FDIR then terminates. This means that if any of the remaining parameters go outside limits, no failure identification will be performed on them and they could influence the value the SF outputs. Should FDIR lose a channel's PFBs due to commfault or crew deselection, FDIR will terminate. Should the crew reselect the channel's PFBs or the commfault clears, FDIR will upmode to normal operation. See Figure 10-41.

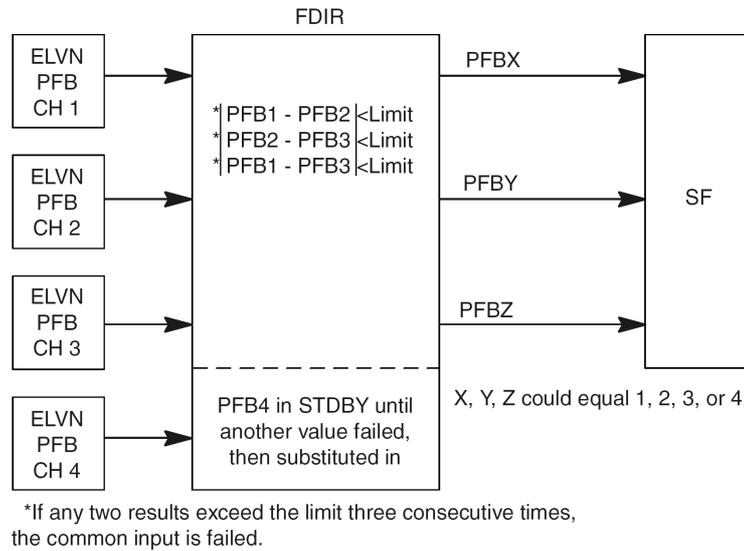


Figure 10-41. Elevon Position Feedback FDIR

The SF nominally MVSs feedbacks from channels 1, 2, or 3. As mentioned, if one of those parameters is declared failed, the channel 4 value is substituted for it. The SF MVSs the remaining three parameters. Because FDIR has quit, the SF will downmode only for commfaults or crew commands. If it downmodes to the two-level, it will average the two parameters. At the one-level it will pass the data through. For a zero-level case, it will remember the last data and pass it on. See Figure 10-42.

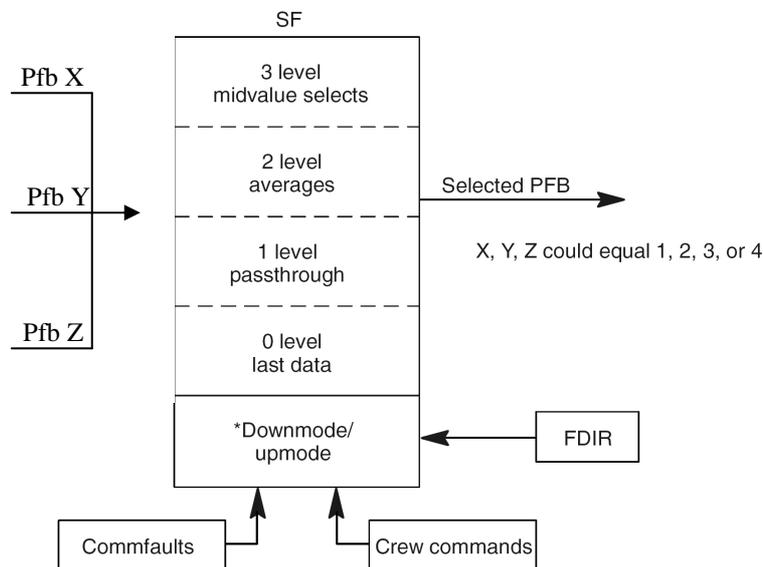


Figure 10-42. Elevon Position Feedback SF

The above description of PFB RM is how the system operates in OPS 3 and MM 602 and 603. In OPS 1 and MM 601, neither elevon PFB FDIR nor SF runs.

Unlike PASS, the BFS (all OPS) does not have FDIR. Moreover, the BFS SF initially Interchangeable Midvalue Selects (IMVSs) channels 1, 2, 3, and 4. However, the BFS SF does not recognize commfaults. Surface feedbacks can still be deselected by the crew with BFS SPEC 51 (OVERRIDE display). The first deselection causes the BFS to downmode to MVSing the remaining three PFBs. Further deselections result in averaging and then passthrough.

The elevon PFBs are also used to generate FCS saturation position (FCS SAT POS) messages, which occur when the elevon positions are outside of +12° down and -15° up. Saturation can also occur if the elevons are moving faster than 20°/s (faster than 13.9°/s if single APU) or for high PRI ΔP. This will cause a FCS SAT MON message. Saturation messages can be generated in OPS 3 and 6 (PASS and BFS).

It should be noted that, although elevon PFB RM functions in MM 602 and 603, elevon PFB fail flags are downlisted only in OPS 3.

10.6.3 Elevon PFB SOP

The purpose of the Elevon Position Feedback SOP (ELVN PFB SOP) is to perform compensation of the selected measured elevon position feedback signals and the computation of the elevator position. The ELVN PFB SOP is executed at 25 Hz during MM 304, 305, 602, and 603, and at 6.25 Hz in MM 301 through 303.

The RM Selection Filter sends selected elevon position feedback data for each of the four elevons to the ELVN PFB SOP (Figure 10-43). The SOP applies biases and scale factors to the feedback data to convert them from volts into engineering units (degrees) for use in flight control software, crew displays, and C&W (Figures 10-44 and 10-45).

The second function of the ELVN PFB SOP is to compute the elevator position by averaging the compensated elevon position feedback data from all four elevons, as follows:

$$\text{ELEVATOR POSITION} = \frac{\text{LIB} + \text{LOB} + \text{RIB} + \text{ROB}}{4}$$

The elevator position is then sent on to flight control.

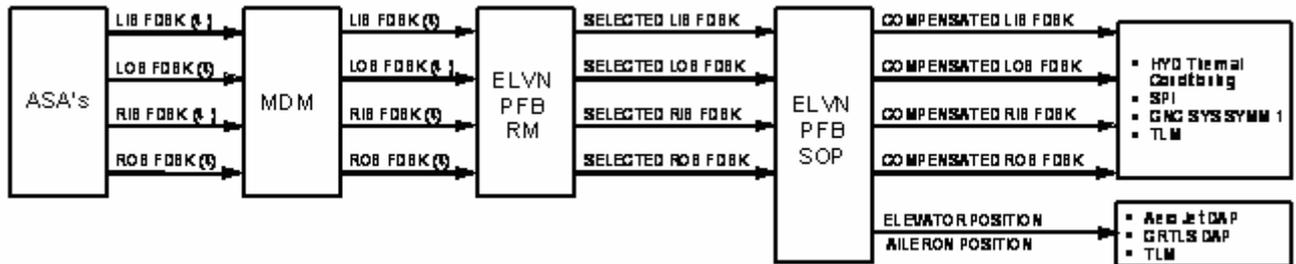


Figure 10-43. Elevon Position Feedback SOP

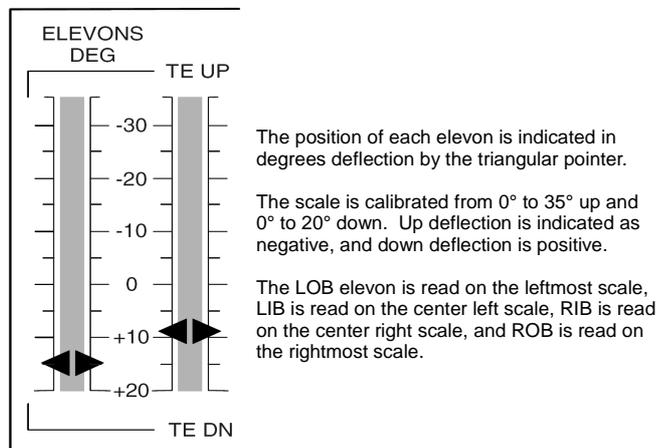


Figure 10-44. Elevon Position - SPI

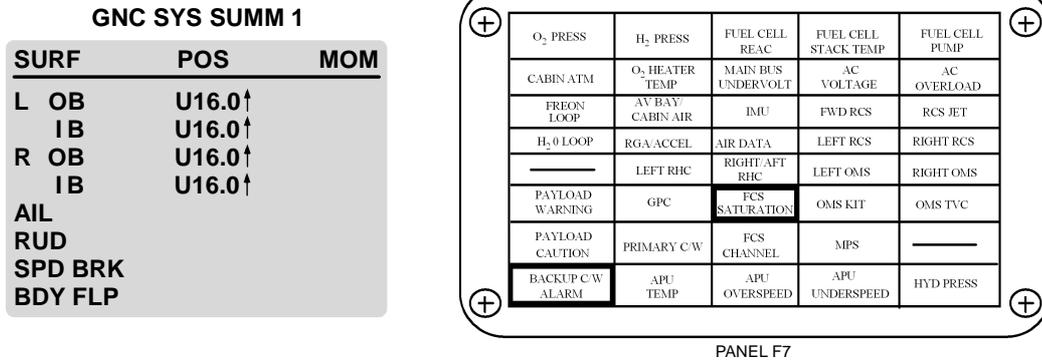


Figure 10-45. Elevon Position - Display and Crew Annunciation

During OPS 3 and 6, elevon surface position is displayed in degrees preceded by a D (down) or U (up) for LOB, LIB, RIB, and ROB on GNC SYS SUMM 1. Elevon position saturation status is displayed to the right of each surface as ↑ for limit exceeded and ↓ for below limit (limits are +12°, -15°).

Elevon position saturation generates a class two alarm:

- FCS saturation light and backup C&W light on C&W panel F7
- C&W tone
- Master alarm light on F2 and F4
- Fault message - FCS SAT POS - ↑ or ↓ next to position (degree) (limits are +12°, -15°)

During entry, the RCS activity lights (Figure 10-46) perform an important FCS function. In MM 304, 305, 602, and 603, when the pitch jets are deactivated (at a dynamic pressure of 40 psf) and the dynamic pressure is greater than 50 psf, both RCS pitch indicator lights (U and D) will illuminate when elevon rate saturation occurs. This is at a maximum rate of 20°/s, which is also the limit set by flight control in Priority Rate Limiting (PRL) for driving the elevons. Either left or right elevons moving at this rate will illuminate the lights. If only one auxiliary power unit (APU) is running, the elevon drive rate limit is 13.9°/s in PRL, which is when elevon rate saturation will occur and the RCS pitch indicator lights will illuminate.

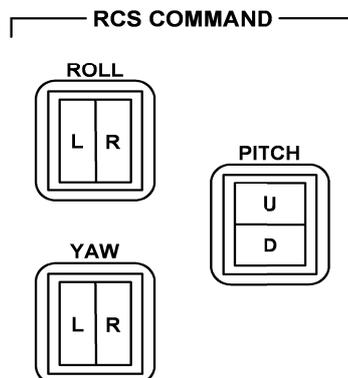


Figure 10-46. RCS Activity Lights

10.6.4 Aileron Position - SPI

Aileron position is shown in degrees (Figure 10-47). It is limited by the surface position indicator (SPI) to 5° left and 5° right of the 0° index. The triangular pointer is referenced to the deflection (0° to 5°) of the elevons that are deflected upward to produce left or right roll. Left roll (left elevons up and right elevons down) is read to the left of 0°, and right roll (right elevons up and left elevons down) is read to the right of 0°.

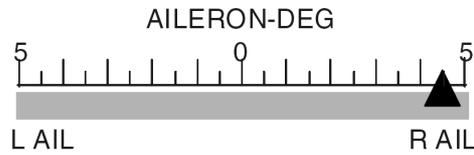


Figure 10-47. Aileron Position - SPI

The differential deflection is not necessarily with respect to the zero elevon point. The aileron position is calculated as follows (where up is negative and down is positive):

$$\text{AILERON} = \frac{\text{AVG LEFT ELVN} - \text{AVG RIGHT ELVN}}{2}$$

The maximum difference in differential deflection that will be indicated on the SPI is 10°. Aileron position is also displayed on GNC SYS SUMM 1 display (Figure 10-48).

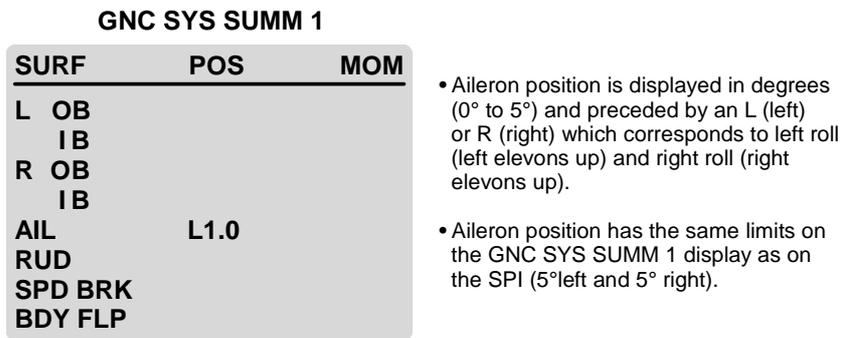


Figure 10-48. Aileron Position - GNC SYS SUMM 1

10.6.5 Elevon Primary Delta Pressures

Elevon primary ΔP is used to modify elevon position during first-stage ascent load relief and to generate FCS saturation moment (FCS SAT MOM) messages. During load relief (MM 102), the ascent digital autopilot (DAP) drives the elevons to correspond to an I-loaded schedule. The schedule is different for the outboard and inboard elevons. If the primary ΔPs exceed a set of predefined limits, the load on the elevons is too high and the elevons are moved to bring their hinge moments back within limits (unload them). In OPS 3 and MM’s 602 and 603, primary ΔP is used to calculate elevon hinge moments. If they exceed I-loaded limits, an FCS SAT MOM message is displayed. This tells the crew that elevon loading has become critical, and they may be reaching a point where flight control is marginal.

10.6.6 Elevon Primary Delta Pressure FDBK RM

In MM 102, elevon primary ΔP RM consists of both FDIR and SF. From each elevon actuator there are four measurements of primary ΔP, one for each FCS channel. FDIR uses data from channels 1, 2, and 3, holding channel 4 in standby until needed. For each elevon, FDIR compares the primary ΔPs for channels 1, 2, and 3 against each other. If any two limit checks exceeds a value of 600 psi three consecutive times, the common parameter is declared failed. When a channel 1, 2, or 3 parameter is failed, FDIR substitutes the channel 4 value in its place. For elevon primary ΔP RM, FDIR continues to run after the first failure. It will continue to run, performing its limit checks until a second failure occurs. After that, the SF downmodes only for commfaults.

SF in MM 102 uses a substitute MVS scheme on the available parameters. SF nominally MVSs primary ΔPs from channels 1, 2, or 3. As mentioned, if one of those parameters is declared failed, the channel 4 value is substituted for it. The SF MVSs the remaining three parameters. If it downmodes to the two level, it will average the two parameters. At the one level it will pass the data through. For a zero-level case, it will output the last valid data.

In OPS 3 and 6, the elevon primary ΔP FDIR does not operate. The SF alone MVSs channels 1, 2, and 3 primary ΔP s; the channel 4 primary ΔP is ignored. The SF downmodes only for commfaults. It will MVS, average, or pass through parameters, depending upon how many it has available.

10.6.7 Elevon Hinge Moment - Display and Crew Annunciation

During OPS 3 and OPS 6, the selected elevon primary delta pressures are converted to elevon hinge moments for left outboard and inboard (LOB, LIB) and right inboard and outboard (RIB, ROB) and are displayed as percent of maximum allowed, from -99% to +99% on GNC SYS SUMM 1. Hinge moment saturation status is displayed to the right of each hinge moment as \uparrow for high limit exceeded (the limit is 80 percent). See Figure 10-49.

| GNC SYS SUMM 1 | | |
|----------------|-----|---------------|
| SURF | POS | MOM |
| L | OB | 57 |
| | IB | 48 |
| R | OB | 55 |
| | IB | 85 \uparrow |
| AIL | | |
| RUD | | |
| SPD BRK | | |
| BDY FLP | | |

Figure 10-49. FCS Hinge Moment Saturation

Elevon hinge moment saturation generates a class two alarm:

- FCS saturation light and backup C&W light on C&W panel F7
- C&W tone
- Master alarm light on F2 and F4
- Fault message - FCS SAT MOM - \uparrow next to numerical value (percent) (limit is 80 percent)

10.6.8 Rudder and Speedbrake Feedback Software Data Flow

Rudder and speedbrake position feedback software data flow is identical to that of the elevon position feedbacks (see previous section). Quad-redundant rudder and speedbrake PFBs are routed from their respective control surfaces through the ASAs to RM for processing. The selected position signals are sent from RM to a SOP, where they are converted from volts to degrees. The speedbrake PFB SOP also calculates the percentage of speedbrake deflection. The compensated PFB data are then sent on to flight control and the crew interfaces (Figure 10-50).

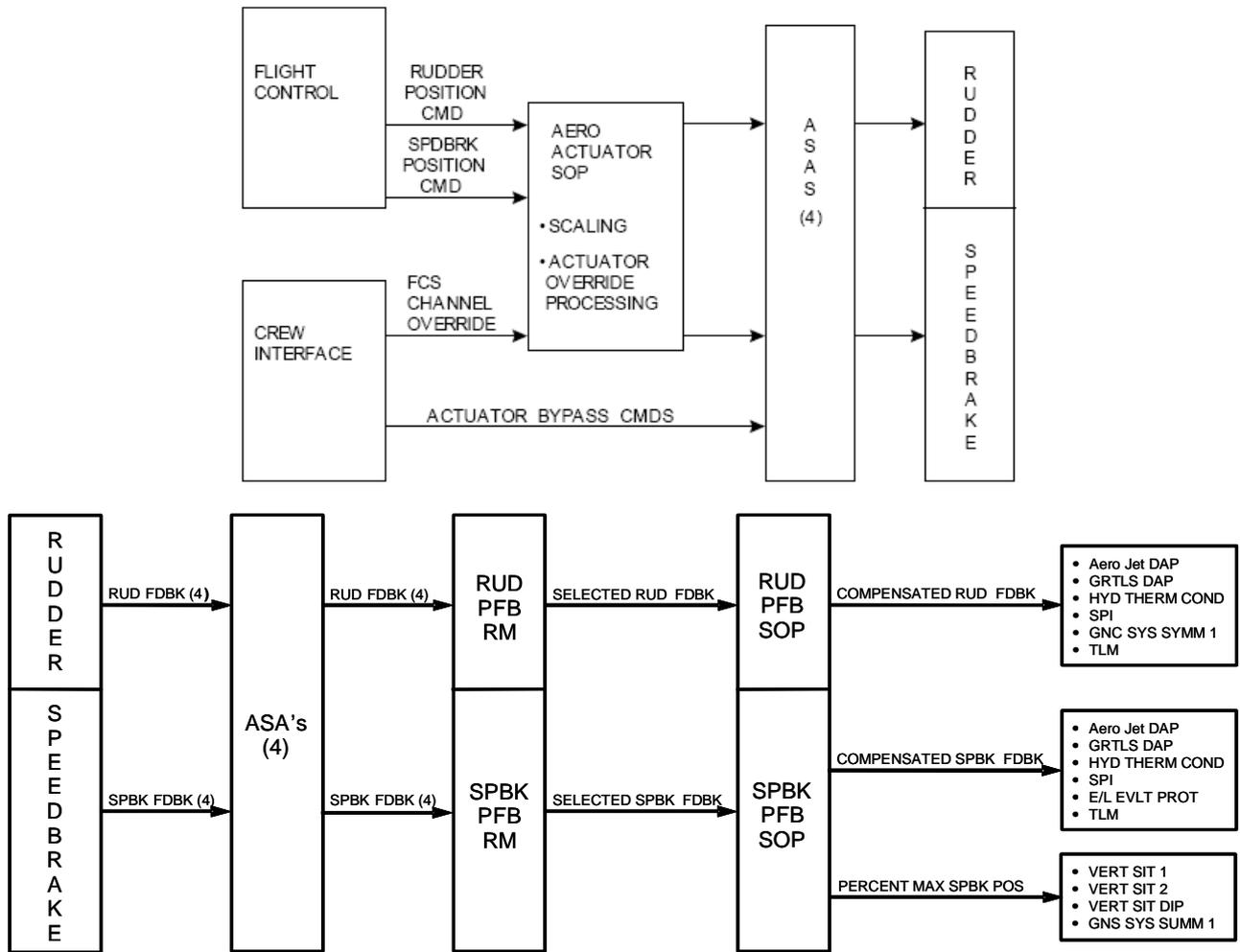
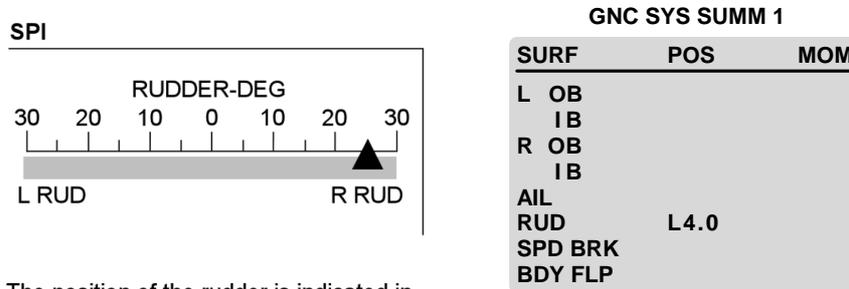


Figure 10-50. Rudder and Speedbrake Command and FDBK Software Data Flow

Crew displays for the rudder are shown in Figure 10-51.

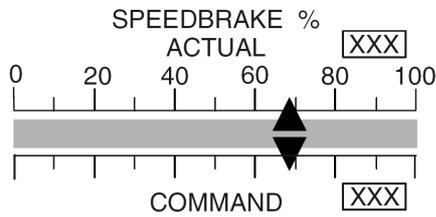


The position of the rudder is indicated in degrees of deflection by a triangular pointer. The scale is calibrated 30° to the left and right. Left rudder deflection is read to the left of 0° and right rudder deflection to the right of 0°. (The hardware limits for the rudder position are ± 27.1°.)

Rudder position is displayed in degrees preceded by an L for left rudder deflection or an R for right rudder deflection.

Figure 10-51. Rudder Position - Crew Interface OPS 3 and 6

Displays for the speedbrake are illustrated in Figures 10-52 and 10-53.



Speedbrake deflection in percent is indicated by the upper pointer. The auto guidance-speedbrake command via flight control in percent is indicated by the lower pointer. Both scales are the same and are calibrated from 0 to 100 percent. 0 indicates that the speedbrakes are fully closed. 100 percent corresponds to speedbrakes fully open (98.6° included angle).

Note: The 25% position is the speedbrake deflection angle such that the outer surfaces of each panel conform to the wedge angle of the vertical fin.

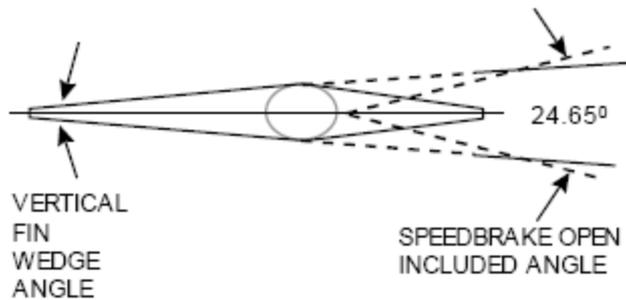


Figure 10-52. Speedbrake Position - SPI (OPS 3, 6)

| GNC SYS SUMM 1 | | |
|----------------|------|-----|
| SURF | POS | MOM |
| L OB | | |
| IB | | |
| R OB | | |
| IB | | |
| AIL | | |
| RUD | | |
| SPD BRK | 65.0 | |
| BDY FLP | | |

Speedbrake position is displayed in percent of maximum deflection

Figure 10-53. Speedbrake Position - GNC SYS SUMM 1 (OPS 3, 6)

10.6.9 Rudder Position Feedback and Speedbrake Position Feedback Redundancy Management (RUD PFB RM and SB PFB RM)

Rudder PFB RM and speedbrake PFB RM are identical to elevon PFB RM previously discussed in OPS 3 and MM 602 and 603 (see Section 10.6.2 Elevon PFB RM).

Fault detection for both RUD PFB RM and SB PFB RM is executed at 1.04 Hz. Selection filtering is executed at 6.25 Hz for speedbrake PFB data and at 1.04 Hz for rudder PFB data.

As with elevon position feedback, rudder and speedbrake position feedback fail flags are downlisted only in OPS 3.

VERT SIT 1 and VERT SIT 2 are GNC OPS displays in OPS 3 and 6 (Figure 10-54). Actual speedbrake (SPD BK) position and the guidance commanded speedbrake (CMD) position are displayed in percent. VERT SIT 1 is forced up automatically or when the crew performs and OPS 305 PRO or OPS 602 PRO in OPS sequences 3 and 6, respectively. VERT SIT 2 is forced up automatically in OPS sequences 3 and 6.

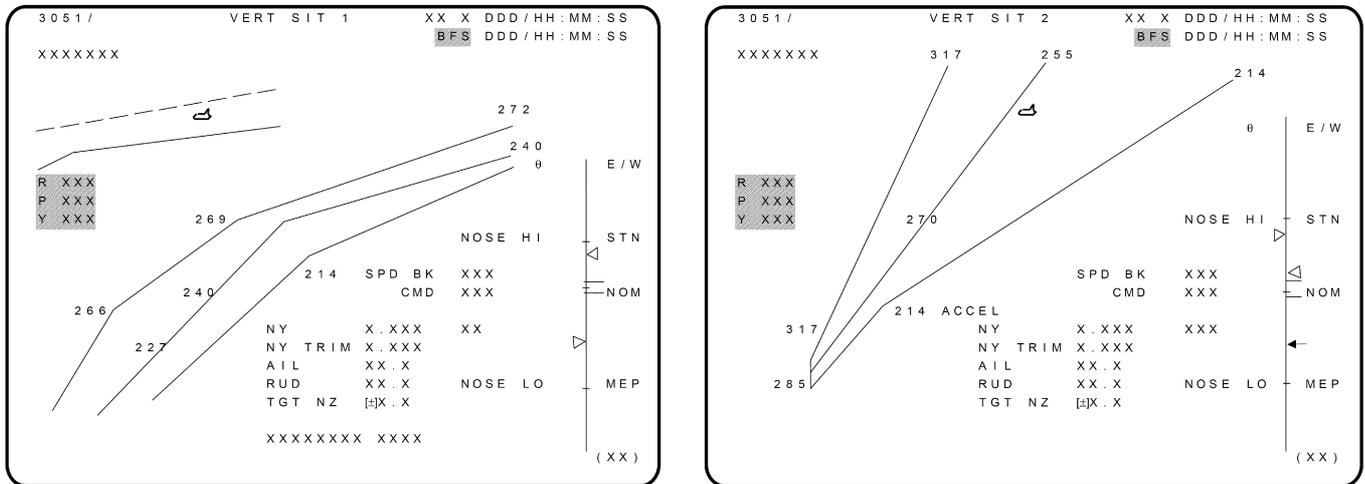


Figure 10-54. Vertical Situation Displays - Speedbrake Position

10.7 BF COMMAND SOFTWARE DATA FLOW

Flight control sends commands to move the BF to the aeroactuator command SOP and to BF command RM. BF command RM checks which pilot valves are selected and sends an ENABLE command to the aeroactuator SOP. The aeroactuator SOP checks that the BF is enabled (via RM) and then generates BF command discrettes UP, DOWN, and ENABLE (Figure 10-55). The aeroactuator SOP also tracks BF commands and position, issuing HOLD commands if the BF moves when it is not supposed to. The crew can manually select/deselect BF pilot valves via the FCS/DED DIS C/O in OPS 8.

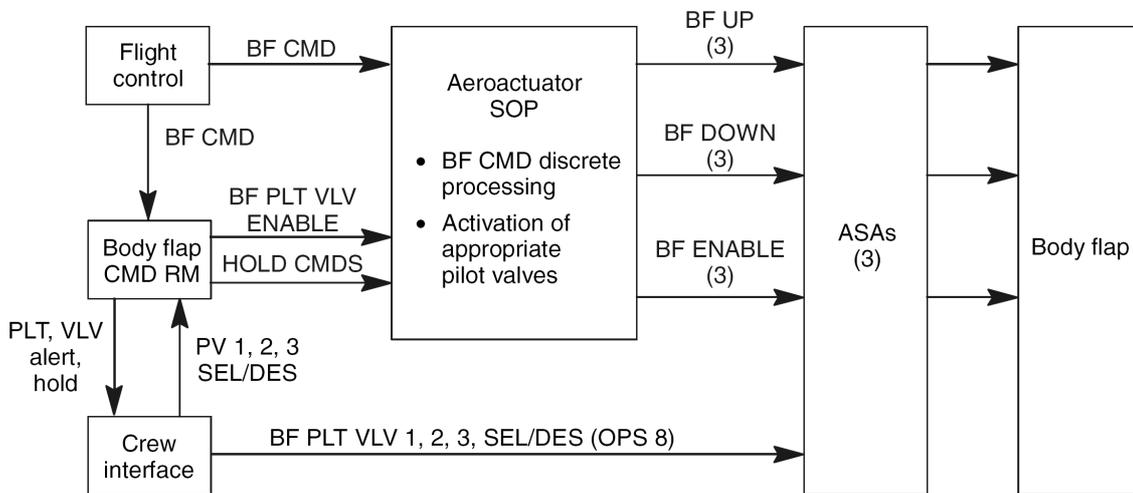


Figure 10-55. Body Flap Command Software Data Flow

10.7.1 BF FDBK Software Data Flow

Quad-redundant BF PFBs tell the story of BF position to the GPCs. Each one comes from a Rotational Voltage Differential Transformer (RVDT) on the differential gearbox output shaft and arrives at the GPCs through the ASAs via a different FCS Channel. Once there, they go to BF PFB RM for processing. The selected BF FDBK signal is sent from RM to the BF PFB SOP, where it is converted from volts to degrees. The compensated BF position feedback is then sent on to flight control, crew interfaces, and BF command RM (Figure 10-56). Quad midvalue select (QMVS) SF processing is used for the BF PFBs. In OPS 1, 3, and 6, BF PFB RM consists of FDIR and SF.

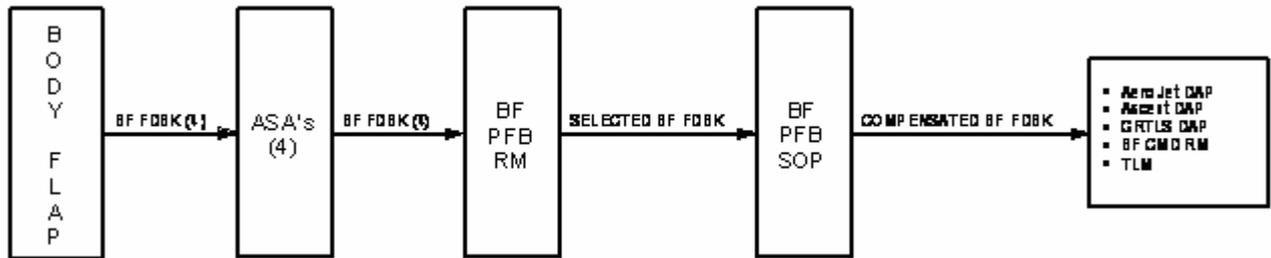


Figure 10-56. Body Flap FDBK Software Data Flow

10.7.2 Body Flap Position Feedback Redundancy Management

The BF PFB RM scheme works differently than for the other aerosurfaces. With no failures, all four PFBs are input into the SF, which uses a QMVS scheme to find a selected output. The selected output is fed to FDIR, as well as to other users. FDIR subtracts the selected output from the current position from each of the four FCS channels, takes the absolute value of the result, and compares it to a limit (1.76 deg). See Figure 10-57.

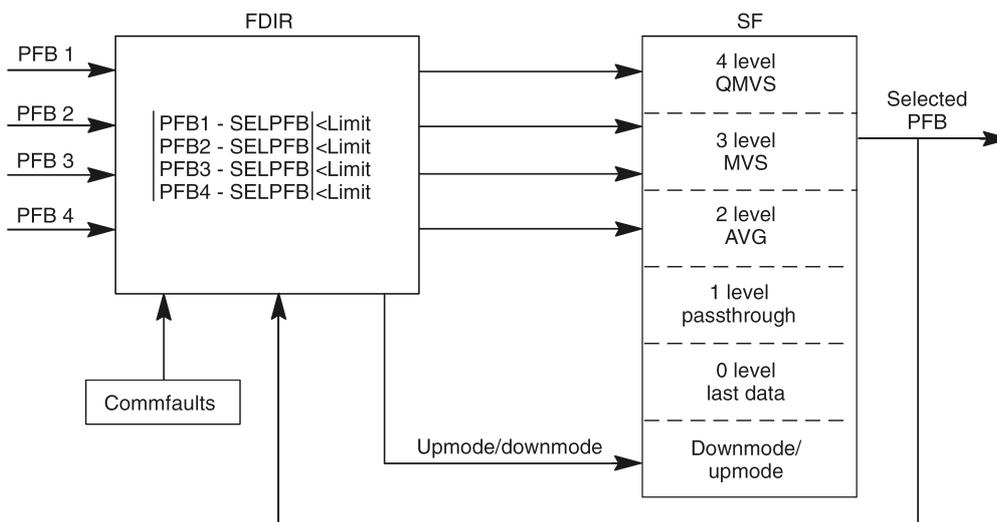


Figure 10-57. Body Flap PFB RM (No Failures)

For a first failure, FDIR will declare the failure, downmode the SF to MVS, and terminate processing. A commfault will be seen as a first failure. If the commfault is cleared, FDIR will upmode, as will the SF. After the first failure, since FDIR has terminated, the remaining feedbacks go to the SF, which downmodes only for commfaults or crew deselect.

BF position feedback fail flags are downlisted only in OPS 3, although BF PFB FDIR functions is OPS 1,3, and 6.

For BF PFB RM, FDIR executes at 1.04 Hz and SF executes at 6.25 Hz.

10.7.3 Body Flap Command Redundancy Management (BF CMD RM)

BF command RM enables and disables the BF pilot valves (PVs) based on crew selection/deselection of individual PVs. BF command RM also checks BF operation and issues HOLD commands and/or sets alerts if it is incorrect. Normally, all three BF PVs are selected (OPS 1, 3, and 6). Any PVs deselected by the crew in OPS 8 shall be carried over as deselected in OPS 3. BF crew displays are illustrated in Figures 10-58 and 10-59.

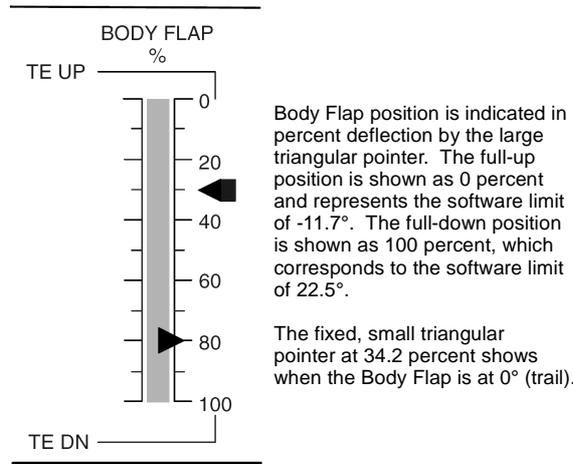


Figure 10-58. Body Flap Position - SPI

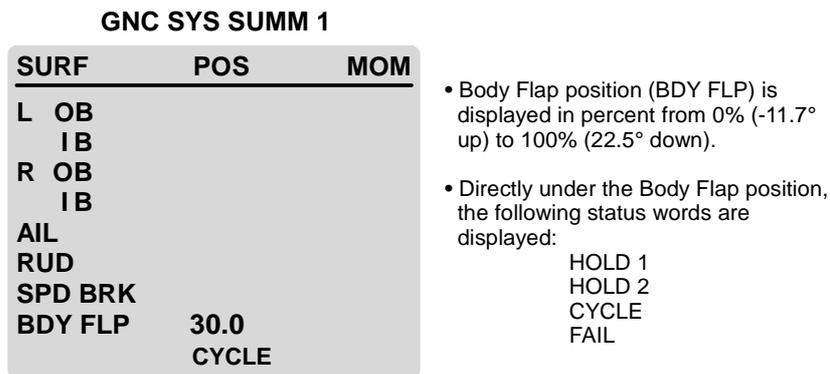


Figure 10-59. Body Flap Position - GNC SYS SUMM 1

BF CMD RM software computes the change in position of the BF and its direction of motion or lack thereof. It does this using BF position feedbacks.

If RM sees that a BF command from flight control is not present and the BF position change is greater than 1.5° for an error count of 6 (1.92 sec), a BF channel is failed ON. RM issues a HOLD command over channel 2; that is, channel 2 is commanded in the opposite direction of the ON failure. This will cause a HOLD 2 to appear on the GNC SYS SUMM 1 display underneath the BF position for crew information. If the HOLD on PV 2 was successful in halting BF motion, an SM alert and BF HOLD message is also issued for the crew.

If the HOLD 2 command does not work, the PV 2 HOLD is removed and a HOLD command is issued over channel 1. The PV 1 HOLD commands channel 1 in the opposite direction of the failed-on movement. A HOLD 1 will appear on the GNC SYS SUMM 1 display. If this command succeeds in stopping the BF movement, an SM alert and a BF HOLD message will be issued.

If the PV 1 HOLD does not fix the problem, the command is removed, CYCLE appears on GNC SYS SUMM 1, and an SM alert and BF CYCLE message are issued to the crew.

If RM detects that a BF command from flight control is present but the BF direction of motion is not following the flight control command for a rate count of 3 (0.96 sec), RM checks for a HOLD on PV 1 or 2. If a HOLD does not exist, RM sets the BF ALERT flag to state 4 which causes a FAIL to appear on the GNC SYS SUMM 1 display and issues a BODY FLAP FAIL fault message to the crew. If a HOLD does exist, the HOLD direction is compared to the direction of BF desired motion. If it is in the opposite direction, the HOLD is removed. If the direction of the HOLD is the same to the desired direction of BF motion, RM sets the BF ALERT flag to state 4, which issues the BODY FLAP FAIL fault message.

A class 3 alarm is issued whenever the body flap alert indication is output by RM. The annunciation is a Fault Message, SM alert light on F7, and C&W tone. The BF alert has 4 states which drive the following Fault Messages:

| BF Alert State | Fault Message |
|----------------|----------------|
| 1 | BODY FLAP HOLD |
| 2 | BODY FLAP HOLD |
| 3 | BODY FLAP CYCL |
| 4 | BODY FLAP FAIL |

BF CMD RM is processed at 3.125 Hz.

10.8 AEROSURFACE ACTUATORS RM SUMMARY

This section is provided to summarize the aerosurface actuator RM process.

The GN&C RM is designed to maximize the operational reliability of the GN&C system. This is accomplished by management of redundant data to minimize usage of incorrect or faulty data within the GN&C software application programs.

RM consists of two major functions:

- a. Fault Detection, Isolation, and Reconfiguration (FDIR)
- b. Selection Filter (SF)

Figure 10-60 is a general diagram of the RM used for aerosurfaces.

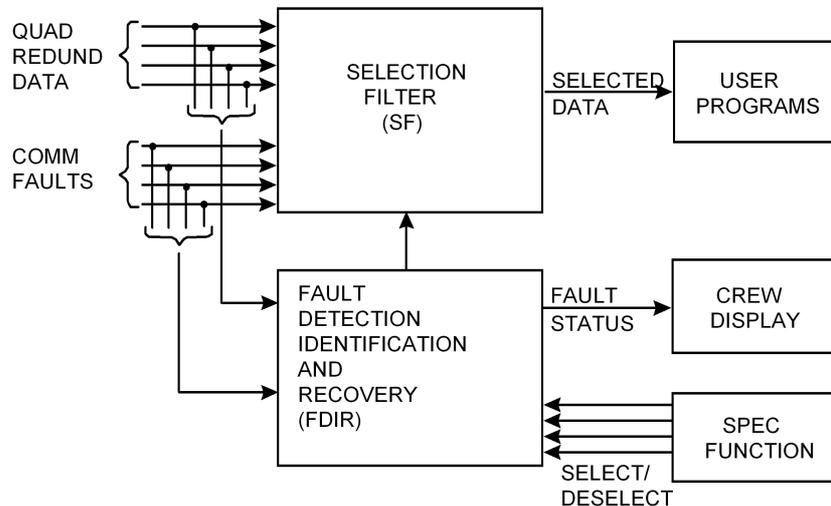


Figure 10-60. General Redundancy Management Black Diagram

RM processing for each aerosurface is not the same in all OPS or MMs.

Figures 10-61 and 10-62 show the specific major modes for each system for FDIR and SF processing. Interfaces exist between RM and the following major software functions:

- a. Flight Control
- b. Crew Interface
- c. SOPs
- d. Other application programs
- e. Flight computer operating systems (FCOSs).

FDIR (Figure 10-61) consists of three distinct submodules:

- a. Fault detection (FD) - compares redundant input data to determine whether an out-of-tolerance condition exists.
- b. Fault identification (FI) - determines whether a failure has occurred and attempts to identify the LRU that has failed.
- c. Fault reconfiguration (FR) - causes the SF to be moded in response to crew commands and identified failures.

The inputs to the FDIR and the SF are the same; i.e., redundant data and COMM commfaults plus SELECT/DESELECT commands from the crew. The redundant set of input data with which FDIR performs FD and FI is established by the crew select/deselect signals in combination with COMM faults and identified faults. The crew select/deselect capability is available for the elevon, rudder, and speedbrake PFBs in OPS 1, 3, and 6 via the SPEC 53 ENTRY CONTROLS specialist function and in OPS 8 via the SPEC 42 SWITCH/SURF.

The SF (Figure 10-62) provides a single data output to user programs which is derived from the redundant input data. The redundant data inputs to the SF are from the ASAs. The SF is controlled by an input from the FDIR, and by digital communication status (commfaults) from FCOS.

| LRU's | Major Modes | | | | | | | | | | | | Redundant Level | Proc Rate (Hz) | Crew Can Sel | Note | | |
|------------------------|-------------|---|---|---|---|---|---|---|---|---|---|---|-----------------|----------------|--------------|-----------|---|---|
| | 1 | 1 | 1 | 1 | 1 | 1 | 3 | 3 | 3 | 3 | 3 | 6 | | | | | 6 | 6 |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | | | | | 0 | 0 |
| Elevon FBK FDIR | | | | | | | X | X | X | X | X | | X | X | 4 | 1.04 | Y | 1 |
| Elevon PRI ΔP FBK FDIR | | X | | | | | | | | | | | | | 4 | 1.04 | N | 2 |
| Speedbrake FBK FDIR | | | | | | | X | X | X | X | X | | X | X | 4 | 1.04 | Y | 1 |
| Rudder FBK FDIR | | | | | | | X | X | X | X | X | | X | X | 4 | 1.04 | Y | 1 |
| Body Flap FBK FDIR | X | X | X | X | X | X | X | X | X | X | X | X | X | X | 4 | 1.04 | Y | 1 |
| Body Flap CMD FDIR | X | X | X | X | X | | | | | | X | X | X | X | - | 3.12 5 | Y | 3 |

- Notes:
- 1. FDIR terminates if any of the four LRUs is not available.
 - 2. FDIR terminates after second LRU becomes unavailable.
 - 3. Crew res/des in OPS 8 only.

Figure 10-61. FDIR Major Mode Applicability

| LRU's | Major Modes | | | | | | | | | | | | Proc Rate (Hz) | Note | | |
|----------------------|-------------|---|---|---|---|---|---|---|---|---|---|---|----------------|------|------|---|
| | 1 | 1 | 1 | 1 | 1 | 1 | 3 | 3 | 3 | 3 | 3 | 6 | | | 6 | 6 |
| | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | 0 | | | 0 | 0 |
| Elevon FBK SF | | | | | | | X | X | X | X | X | | X | X | 25 | 1 |
| Elevon PRI ΔP FBK SF | | X | | | | | X | X | X | X | X | | X | X | 12.5 | 1 |
| Speedbrake FBK SF | | | | | | | X | X | X | X | X | | X | X | 6.25 | 1 |
| Rudder FBK SF | | | | | | | X | X | X | X | X | | X | X | 1.04 | 1 |
| Body Flap FBK SF | X | X | X | X | | | X | X | X | X | X | X | X | X | 6.25 | 2 |

- Notes:
- 1. Substitute Midvalue Select (MVS) scheme used (CH 4 held in standby).
 - 2. Quad Midvalue Select (QMVS) scheme used.

Figure 10-62. SF Major Mode Applicability

10.8 ORBITER ACTUATOR SLEW CHECK - MM 101 - 25 Hz

During terminal count (MM 101), an actuator slew check is implemented to slew the elevons (four panels), rudder, speedbrake, and BF.

The slew check uses a trapezoidal function (Figure 10-63) such that for each set of elevon, rudder, and speedbrake actuators, the commands are incremented and decremented at the rates and deflections per the chart below. BF commands are incremented and decremented by 0.12°/cycle (3°/s). At each indicated position, the slew commands are held constant for 1 second.

| | Deflections (°) | Rate (°/sec) |
|------------|-----------------|--------------|
| Elevons | ±10 | 10 |
| Rudder | ±14 | 14 |
| Body Flap | ±3 | 3 |
| Speedbrake | ±2 | 2 |

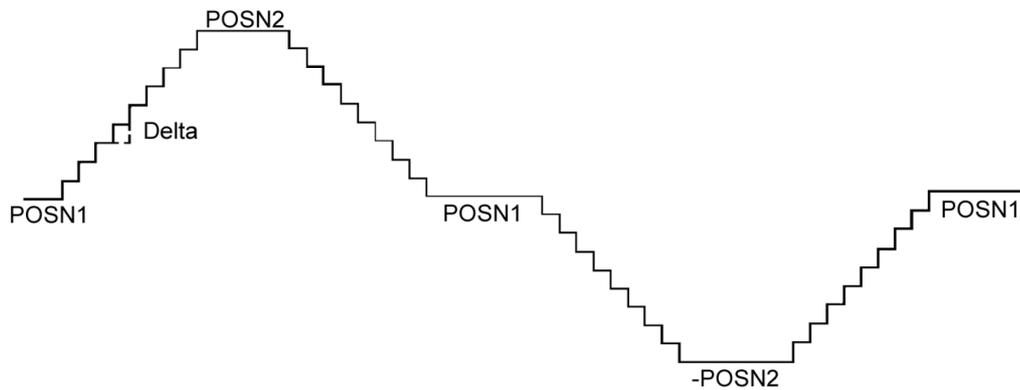


Figure 10-63. Trapezoidal Function

10.9 AEROSURFACE DRIVE TEST OPS 8

ITEM 10 EXEC on the FCS/DED DIS C/O display (Figure 10-64) initiates the aerosurface drive test (ADT). The primary purpose of the ADT is to warm up the hydraulic system in order to perform the secondary actuator test. Furthermore, it verifies that the FCS is responding properly to position commands.

The ADT incorporates a sawtooth ramp function to drive the aerosurfaces cyclically between two points. The surfaces are first commanded from their current position to point 1 or final value 1 (FV-1) at a specified rate for each actuator. They are then commanded to the second point, FV-2. The surfaces are driven between FV-1 and FV-2 until the crew performs ITEM 11 EXEC on the FCS/DED/DIS C/O. The surfaces are then commanded to FV-3 positions, which are the desired positions for performing the secondary actuator test.

The final values and drive rates for each actuator are as follows:

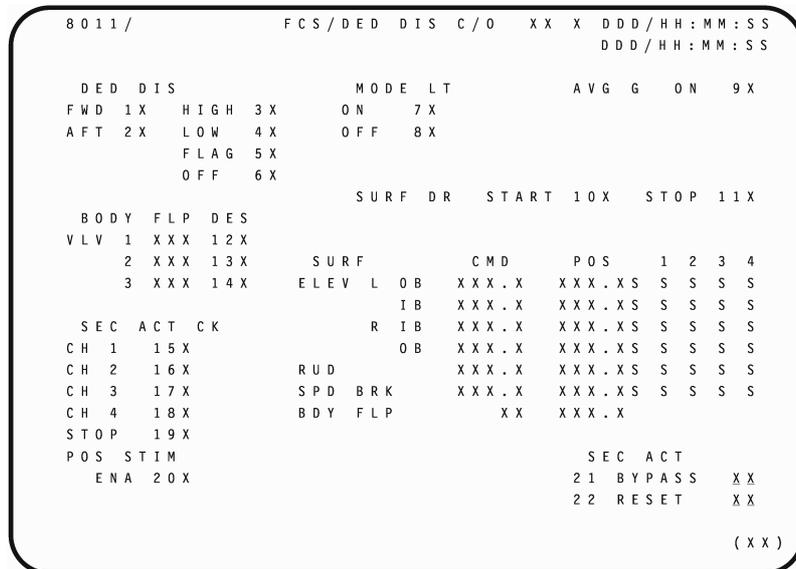
| Actuator | FV-1 (°) | FV-2 (°) | FV-3 (°) | Rate (°/s) |
|------------|------------------|-----------------|--------------|------------|
| Elevons | +18 | -33 | 0 | 5.8 |
| Rudder | +5 | -5 | 5 | 5.0 |
| Speedbrake | +10 | +2 | 10 | 2.7 |
| Body Flap | +20.3 (93.6%) | -10.5 (3.5%) | 0 (34.2%) | N/A |

If the actuator is not being driven within the allowed tolerance (the difference between actual and commanded position equal to distance driven in one second), a down arrow will appear on FCS/DED DIS C/O next to the affected aerosurface position - except for the Body Flap. Initially, all surfaces (except Body Flap) will have down arrows beside their display positions, but as the hydraulic fluid warms up, the actuators should drive at spec rates and the arrows should disappear.

The crew terminates the aerosurface drive test (ITEM 11 EXEC) 30 seconds after initiating it. If any down arrows remain in the surface position status column, that actuator is degraded (considered failed) and reported to the ground. The item 11 (stop) is mandatory regardless of whether an item 10 (start) is executed. The stop sequence ensures that the surfaces are commanded to the proper position for the secondary actuator check.

The capability to initiate and terminate the aerosurface drive test is also available via items 39 (ON) and 40 (OFF) on the DEORB MNVR display in MM 301 through 303. See Section 10.13 Hydraulic Thermal Conditioning below for more details.

An AERO DRIVE fault message can be generated only in OPS 8. The message will be annunciated if the commanded channel differs from the selected position feedback value by more than an I-loaded limit for any aerosurface except the body flap. The I-loads are 5.8°, 5.5°, and 10.5° for elevons, rudder, and speedbrake, respectively. The software inhibits the message from occurring when the secondary actuator check is active on channel 1 only. The AERO DRIVE message can occur if the FCS channel switches are configured so that the command/position delta for any of the six surfaces exceeds the aforementioned limits. The message will be annunciated every 4.8 seconds as long as the out-of-tolerance condition exists.



- Elevon and Rudder CMD and POS are displayed in degrees.
- Elevon POS is displayed as U (up) or D (down).
- Rudder POS is displayed as L (left) or R (right).
- Speedbrake CMD and POS are displayed in percent of full deflection.
- Body Flap CMD is displayed as remote test - either UP, DN (down), or blank (no drive).
- Body Flap POS is displayed in percent of full deflection.

Figure 10-64. FCS/DED DISP C/O Display

10.10 BODY FLAP VALVE CHECK - OPS 8

The capability exists to deselect the enable solenoid valves by item entry on the FCS DED DIS C/O display in OPS 8. At one time it was intended to perform a test in OPS 8 to verify the proper operation of the enable and pilot solenoid valves. This would have been done by selecting the enable valve for each channel one at a time with the aerosurface drive test in progress. This would have required all three APU’s to be running. It was decided prior to STS-1 that the FCS checkout would be done with only one APU, therefore this check was deleted. It should be noted though, that the capability does exist to deselect the enable valve for each body flap channel in the event of pilot valve failures.

Each of the three channel shutoff valves is enabled, one at a time, by item entries 12, 13, and 14 on the FCS/DED DIS C/O display (this inhibits the other two shutoff valves). If no movement of the BF is observed on the FCS/DED DIS C/O, the respective shutoff valve has failed. If BF movement is observed only in one direction (only up or only down), the respective pilot valve in that system has failed.

After each of the three systems has been checked, any failed pilot or shutoff valves are deselected by item entry and recorded as failed. If no failures are found, all three BF valve systems are left selected (all enabled by item entry with no asterisk in the DES column).

10.11 SECONDARY ACTUATOR CHECK

The secondary actuator check is a test designed to verify the functional integrity of the ASAs and aerosurface actuators. This test will verify the proper operation of several of the system components depending upon which option is used, but the primary reason for performing the test is to determine if any of the ASA servovalve drivers for the elevons have failed to null. A secondary purpose is to verify that the secondary ΔP fault detection (bypass monitor) circuitry is functioning properly. The test is performed on one channel at a time, first with positive stimulus and then with negative stimulus. The bipolar stimulus is required to fully check the bypass monitor circuitry; either positive or negative stimulus is sufficient to test for a null driver failure.

With a null failure, the driver output signal to the servovalve is failed to 0 mA (open circuit). This means that an ASA with this failure can no longer respond to position commands. This will not result in a force fight between the channels if the aerosurface is not being commanded to move. However, due to the nature of the dynamic feedback mechanisms in the actuator, a channel with this failure will resist the other channels if the actuator is moving and the amount of resistance is proportional to the drive rate. With maximum drive rates, enough secondary ΔP can be generated to bypass a channel.

The significance of null failures is that testing at the Rockwell Flight Control Hydraulics Laboratory has shown that, if an elevon has null failures on two channels, it is possible that the two good channels will bypass when the surface is commanded to move at a high rate. This is not the case for the rudder and speedbrake.

There are two options available for performing the secondary actuator check in OPS 8. One option uses an APU operating at normal pressure (3000 psi) and the other option uses a circulation (circ) pump which produces about 300 psi of hydraulic pressure. Both options are designed to detect null failures. Though as required by flight rules, the secondary actuator check will be done prior to entry for all situations except RTLS and TAL aborts. Crew procedures provide three procedural options: using an APU in OPS 8; using a circ pump in OPS 8; or using an APU post-deorbit burn. Nominally the check will be done in OPS 8 using an APU. If an APU has failed prior to OPS 8, it may be desirable not to start an APU until the deorbit burn. In this case the circ pump option may be used. If for some reason the OPS 8 checkout cannot be performed, then the checkout will be performed in OPS 3 post-deorbit burn using an APU. Flight Rule A8-104 may be referenced for more information.

10.11.1 OPS 8 APU Option

The nominal OPS 8 secondary actuator check uses a single APU operating at normal hydraulic pressure which allows for a complete functional checkout of the ASA and aeroactuator components. The FCS/DED DIS C/O display is used for this procedure. With this procedure a command bias is added to one FCS channel at a time while all FCS channels are in AUTO. The bias, which is 6° for the elevons, 12° for the rudder, and 16° for the speedbrake, causes a force fight between the channels that will generate enough secondary ΔP to cause the biased channel to bypass. This verifies the proper operation of the servovalve driver, the secondary ΔP transducer, and the fault detection circuitry.

10.11.2 OPS 8 Circulation Pump Option

This option is intended to be an abbreviated check of the secondary actuators in the event that an APU is not available. The procedure will verify the proper operation of the servovalve driver and the secondary ΔP transducer for the elevons only. A single circ pump is used for the test. The circ pumps produce only about 300 psi, which is enough to move the elevons, but will not move the rudder, speedbrake, or bodyflap (300 psi is insufficient pressure to release the brakes on the rudder/speedbrake and bodyflap).

If this option is to be used, then OPS 8 is entered with the ASAs powered off. This will initialize the commands to their null values (-7.5° for the elevons) and is done to prevent nuisance AERO DRIVE messages. The procedure is also managed so that two of the first three FCS channels are off at all times for the same reason. In OPS 8 the selected position feedback is always the middle value of the first three channels. Initializing the commands to the null value and ensuring that the selected feedback is also maintained at the null value eliminates the AERO DRIVE message since it is generated (continuously) if the channel 1 command and the selected feedback differ by 5.8° for the elevons, 5.5° for the rudder, and 10.9° for the speedbrake. Since the selected feedback is always null, the SWITCH/SURF display (Figure 10-65) is used for position information since it has all four channel positions.

The procedure is started with two FCS channels in AUTO and two channels off. A positive 6° bias is then added to the command on the channel that is to be checked (same as with APU option) and a small force fight should result between the two

channels. After a few seconds the channel that is not being checked is turned off and the elevon should move to the biased command value (-1.5°). The bias is then removed and the elevons should move back to -7.5°. The procedure is then repeated for the other channels. Moving the elevons with only one channel powered verifies that that channel does not have a null failure.

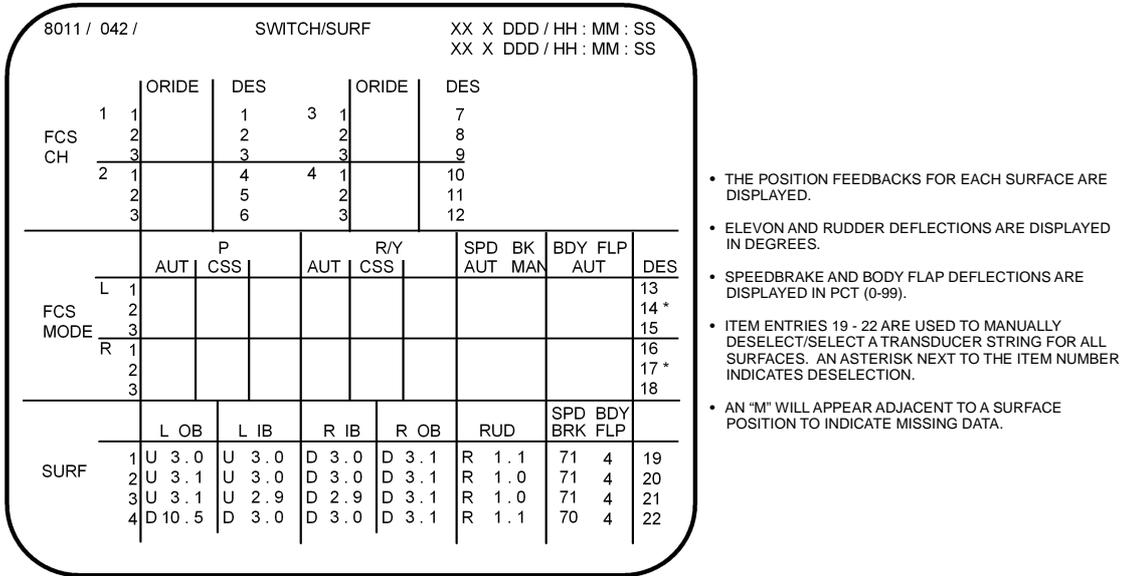


Figure 10-65. SWITCH/SURF Display - OPS 8 (SPEC 042)

10.11.3 OPS 3 APU Option

The OPS 3 option for performing the secondary actuator check is very similar to the OPS 8 APU option, except that the ENTRY CONTROLS display (Figure 10-67) is used instead of the FCS/DED DIS C/O display. The aerosurface drive test is not available on the ENTRY CONTROLS display; however, a similar slew capability, hydraulic thermal conditioning, is available on the DEORBIT MNVR display. Failed secondary actuators can be individually bypassed on the ENTRY CONTROLS display. Since this procedure is performed between the deorbit burn and entry interface, only the positive polarity is checked in order to save time.

10.12 HYDRAULIC THERMAL CONDITIONING

During MMs 301, 302, and 303, the crew can initiate (item 39) and terminate (item 40), on the deorbit MNVR display, a control surface drive function that simultaneously slews the four elevons, the rudder, the speedbrake, and the BF to thermally condition the hydraulic systems for entry.

The slew commands for elevons, rudder, and speedbrake incorporate a triangular function (Figure 10-66), and for the BF, an integer (±1) command. Each aerosurface is commanded from the initial position to an upper limit and then to a lower limit, incremented each cycle by a constant delta. The surfaces are repeatedly driven between limits until the surface drive function is terminated. The hydraulic thermal conditioning function runs at 25 Hz.

The surface positions will be displayed on the SPI and GNC SYS SUM 1 display while the slew function is operating.

The slew deflection limits are as follows:

| | Upper (°) | Lower (°) |
|------------|-----------|-----------|
| Elevons | +18 | -33 |
| Rudder | +5 | -5 |
| Speedbrake | +10 | +2 |

| | | |
|-----------|-------|-------|
| Body Flap | +20.3 | -10.5 |
|-----------|-------|-------|

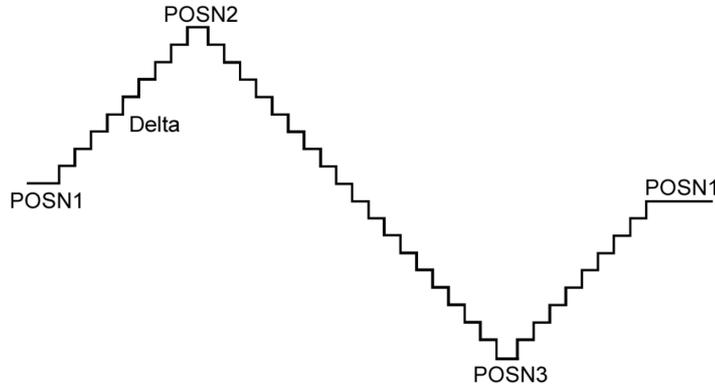


Figure 10-66. Triangular Function

The drive rates are as follows:

| | | |
|------------|----------------|-------------|
| Elevons | +0.232 °/cycle | (5.8 °/sec) |
| Rudder | +0.20 °/cycle | (5 °/sec) |
| Speedbrake | +0.108 °/cycle | (2.7 °/sec) |
| Body Flap | N/A | |

10.13 ENTRY CONTROLS DISPLAY (SPEC 53)

A crew display ENTRY CONTROLS (SPEC 53) is available in OPS 1, 3, or 6 and is described in the DPS Dictionary of the Flight Data File (FDF). A copy of the pertinent pages are provided here as Figure 10-67 with a description of each item on the display.

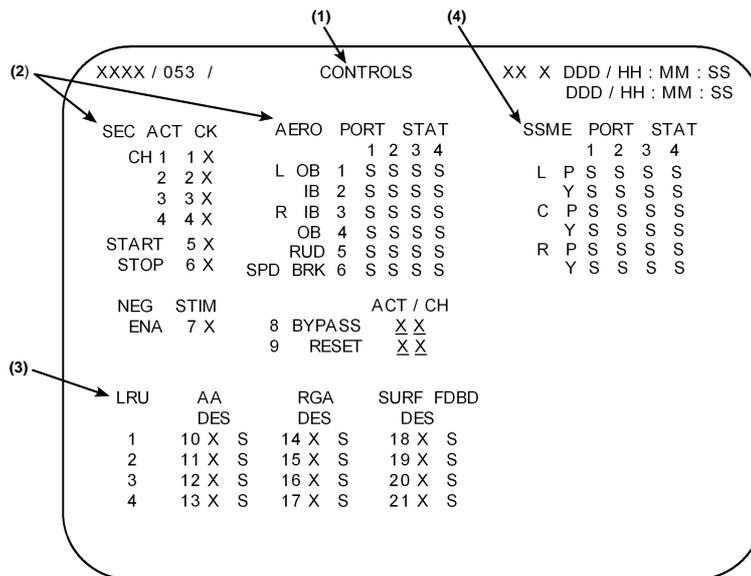


Figure 10-67. CONTROLS Display (SPEC 53)

- (1) ENTRY CONTROLS is a specialist function (SPEC 53) available in GNC OPS 1, 3, and 6. This display provides the capability to control secondary actuator checks and deselection/reselection of accelerometer assemblies (AAs), rate gyro assemblies (RGAs), and aerosurface position feedback (PFB) transducer strings. This display also allows the crew to monitor the status of the aerosurface and SSME actuator secondary ports.

- (2) SEC ACT CK. Items 1 through 4 select actuator channels 1 through 4 for testing and item 5 (START) initiates testing of the aerosurfaces on the selected channel. An * is displayed next to items 1 through 5 when selected. Execution of item 5 without a channel selected will result in an 'ILLEGAL ENTRY' message. Execution of item 6 terminates testing and blanks the *'s next to items 1 through 5. If a second channel is selected during testing, the *'s next to the previously selected channel and next to item 5 will be blanked and an * will appear next to item 6. Item 7 (NEG STIM ENA) changes the polarity of the secondary actuator check. An * next to item 7 indicates negative polarity; a blank indicates positive polarity. Items 1 thru 7 are legal in MM's 301, 302 and 303 only. Items 1 through 5 and 7 are initialized blank (no *'s); item 6 is initialized with an * next to the item.

The aerosurface port status (AERO PORT STAT) for each channel is shown as blank (no bypass), '↓' (bypassed), or 'M' (missing data). Aerosurfaces which do not show '↓' during testing can be checked by placing the appropriate FCS channel switch (on panel C3) in the OFF position. All ports in the off channel should then show '↓' on the display. These '↓'s will remain even if a new SEC ACT CK channel selection is made, until the switch is moved from OFF. Items 8 (BYPASS) and 9 (RESET) are used to bypass and reset, respectively, the aerosurface secondary actuator port specified by the ID entered below ACT/CH. Legal IDs are those shown in the table in paragraph 8, FCS/DED DIS C/O display (OPS 8011). These IDs are initialized at zero and entry of an ID for either item causes the other to be zeroed. When item 8 is used, the selected ACT/CH(s) will remain bypassed even if the FCS channel switch on PNL C3 for that channel is in 'OVERRIDE'. Item 9 will reset any selected port(s) that may have been bypassed with item 8 and will also momentarily reset a selected port that may have been bypassed due to a failure.

- (3) LRU AA/RGA/SURF FDBK. Items 10 thru 13, 14 thru 17 and 18 thru 21 provide for deselection/reselection of accelerometer assembly (AA), rate gyro assembly (RGA), and aerosurface feedback (SURF FDBK) transducer strings as candidates for the selection filter. An * next to an item number (10 through 21) indicates that the corresponding LRU/string has been deselected by the crew or, for AAs and RGAs (items 10 through 17), deselection can be triggered by RM fail detection. The deselect status of items 10 through 21 is carried over when transitioning from OPS 1 to OPS 3/6. In MM 101 through MM 106/MM 601, items 18 through 21 select/deselect body flap transducers only, however, these item entries also carry over to OPS 3/6 and will deselect all corresponding aerosurface position feedbacks once the aerosurface selection filter becomes active. Execution of any item, 10 through 21, with an * present, reselects the corresponding LRU and blanks the *. The status column for each item (10 through 21) displays an 'M' when a comm fault is declared, is blank (normal condition) or, for items 18 through 21 only, displays an '↓' when RM declares one or more transducers in the string to be failed.
- (4) SSME PORT STAT. The port status for each channel on the SSME TVC actuators is shown as blank (no bypass), '↓' (bypassed), and 'M' (invalid data).

SECTION 11 ASCENT TVC ACTUATORS

11.1 GENERAL

Ascent thrust vector control (ATVC) actuators are divided into two groups: main engine (ME) actuators and solid rocket booster (SRB) actuators. Each group has two actuators per device: a pitch and a yaw actuator for each (left, right, and center) ME and a tilt and rock actuator for each (left and right) SRB. Zone N7, Space Shuttle Systems Handbook (SSSHB), drawing 9.14, shows these actuators and their orientation to the Orbiter.

11.2 SYSTEM CAPABILITIES

ATVC actuators move in response to position commands processed through four ATVCs for slewing engine bells prior to lift-off, gimbaling engines to control trajectory, positioning engines during flight, driving ME's to dump position (same as stow positions starting with STS-26), and finally driving the ME's to the stow position.

11.3 MAJOR COMPONENTS

- a. Switching valves
- b. Servovalves
- c. Power valves
- d. Lock valves
- e. Bypass valves
- f. Piston/ram assembly
- g. ATVCs

11.4 HARDWARE

The hardware which comprises the ATVC actuator system includes the switching valves, the servovalves, the power valves, and the lock valves. Refer to SSSHB drawing 9.14 or figures 11-1 through 11-5 when reading the hardware description.

11.4.1 Switching Valves

Hydraulic pressure is supplied to the switching valve of each SRB actuator from hydraulic power units (HPU's) A and B of each SRB. Figure 11-1 shows the SRB switching valve arrangement. ME switching valves utilize two of the Orbiter's three hydraulic sources.

11.4.1.1 SRB Switching Valves

The SRB switching valve supplies approximately 3000 psi to the system from either HPU A or HPU B. Only one HPU at a time actually applies pressure to an actuator. The primary system drives the actuator until the primary system drops below 2050 + 150 psi at zero flow. If this should occur, the switching valve will automatically switch to the secondary system. Zone B7, SSSHB drawing 9.14, shows the SRB thrust vector control (TVC) hydraulic switching valve assignments. Indicators connected to each of the four switching valves provide a status report of which system (primary or secondary) is in use. Once the hydraulic fluid leaves the switching valve, it goes to the four servovalves on that actuator and to the lock valve.

11.4.1.2 ME Switching Valve

The ME switching valve also applies 3000 psi to the ME actuator but supply pressure comes from two of the Orbiter's three hydraulic systems. As with the SRB switching valve, only one hydraulic supply source actually powers an actuator. The primary system drives the actuator until the primary system drops below 1200 to 1500 psi. If this should occur, the switching valve will automatically switch to the secondary system. Zone O8, SSSHB drawing 9.14, shows the ME TVC hydraulic switching valve assignments. Once hydraulic fluid leaves the switching valve, it goes to the four servovalves on that actuator and to the lock valve.

11.4.2 Servovalves

There are four servovalves for each actuator. They are electrohydraulic valves whose output pressures are summed to drive the power valve. Each servovalve receives electrical signals from four ATVCs which are discussed in section 11.4.7. Servovalve 1 on each actuator receives a signal from ATVC 1, servovalve 2 on each actuator receives a signal from ATVC 2, and so on. A positive current to the electrohydraulic servovalve causes the actuator to extend, and a negative current causes the actuator to retract. Refer to SSSHB drawing 9.14. A positive current rotates the shaft in a clockwise direction forcing the servovalve to move toward the right. This movement toward the right sends a pressure signal to the power valve; that is, all four servovalves attempt to drive the power valve in response to their electrical signals. A power valve feedback wire connecting the power valve to each servovalve transmits the position of the power valve to each servovalve of that actuator. Each servovalve has a hydraulic pressure transducer used to measure the hydraulic resistance pressure the servovalves exert on each

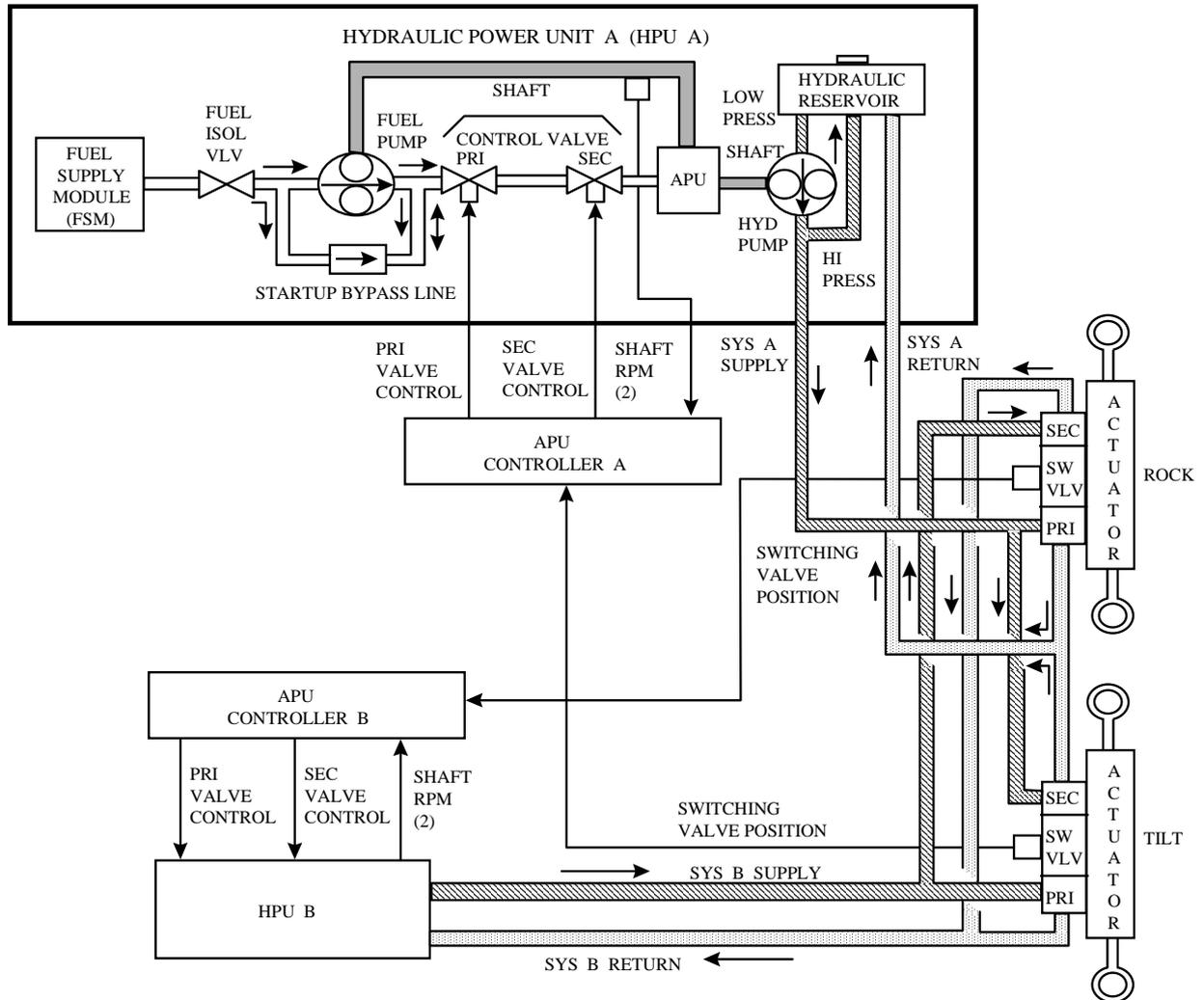


Figure 11-1.- SRB switching valve arrangement.

other. These resistance pressures are generated at the power valve where all four servovalve command pressures are summed to drive the actuator. They are called secondary delta pressures and are measured by the use of linear voltage differential transformers (LVDTs).

11.4.3 Bypass Valves

The bypass valve is a solenoid valve which contains a solenoid, two springs, two pistons, and a cylinder. When the isolation command is issued to the solenoid, piston I shuttles to the right against the spring on the right. This allows hydraulic source pressure to flow into inlet A and raise the hydraulic pressure such that piston II is shuttled to the left, inhibiting any further hydraulic flow from the right servovalve feed/return line to inlet B (fig. 11-2). The left feed/return line fluid flow from the servovalve is allowed to flow into inlet C, through the bypass reservoir, and out passage D. This allows the left return/feed line to equalize hydraulic pressure in either power valve reservoir connected to the servovalve. The secondary delta pressure piston will stay neutral, showing a zero secondary pressure value due to the equal fluid pressure on either side of the piston. When the bypass valve is activated, hydraulic pressure in both feed/return lines is equalized.

11.4.4 Lock Valve

Figure 11-3 shows a drawing of a lock valve. This is a two- or three-position valve used to control power valve hydraulic pressure to the piston/ram. The SRB lock valve is a three-position valve and the ME lock valve is a two-position valve.

11.4.4.1 SRB Lock Valve

The SRB lock valve is a three-position valve; the operate position, the lock position, and the bypass position. The operate position is achieved when system pressure is greater than 1000 psi. In the operate position, hydraulic fluid passes unrestricted through the lock valve from the power valve to the piston/ram because actuator (ACT) C2 is connected to valve C2 and ACT C1 is connected to valve C1. The locked position occurs when system pressure falls below 600 (plus 400, minus 0) psi (documented in United Technologies Spec. No. 10SPC-0055). Figure 11-3 shows the valve in the locked position. System pressure is isolated from the piston/ram with ACT C2 disconnected from valve C2 and ACT C1 disconnected from valve C1. The third position is the bypass position. This is a ground-use-only position. When in the bypass position, all four power valve ports and actuator ports are interconnected allowing piston/ram freedom of movement independent of power valve actuation.

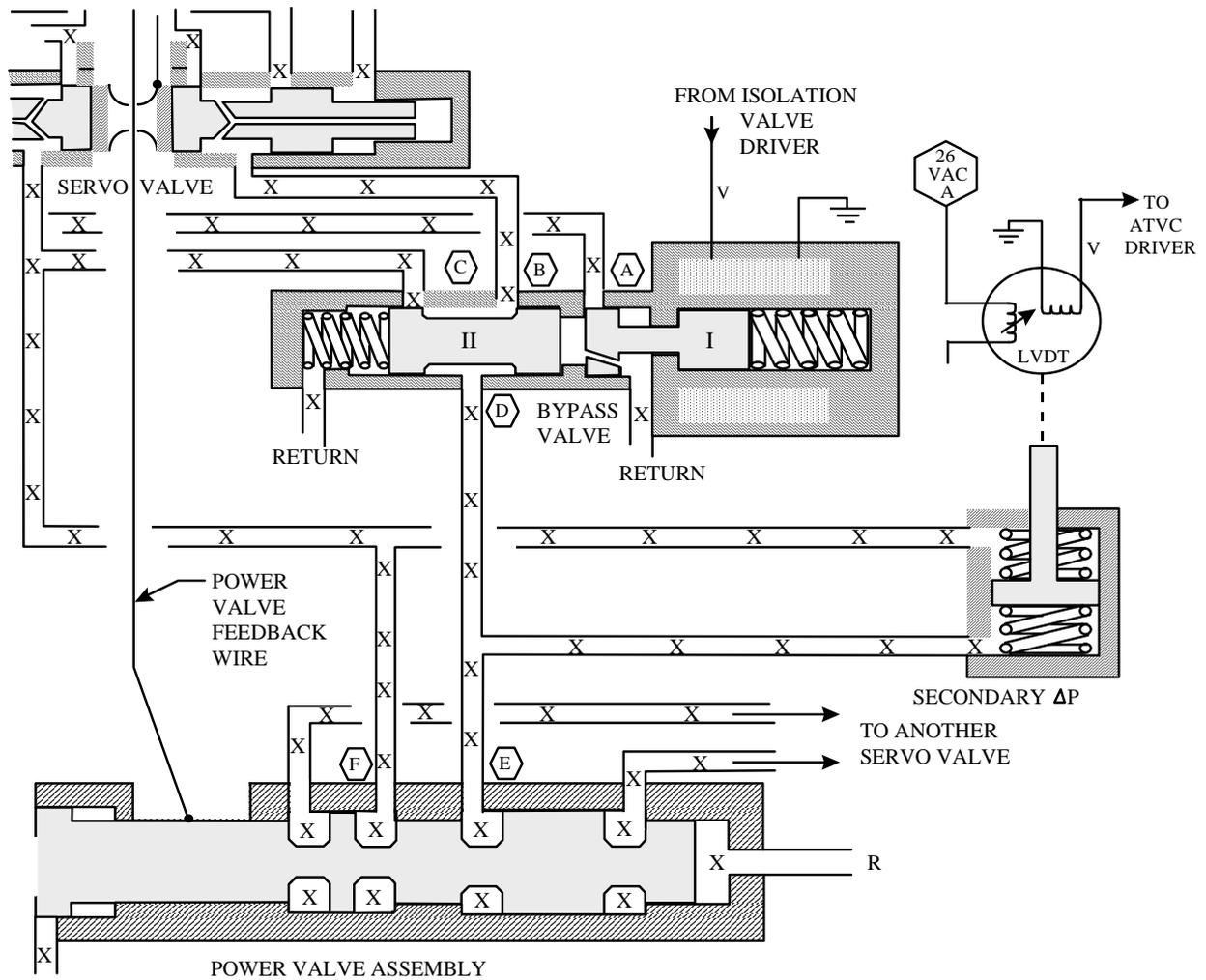
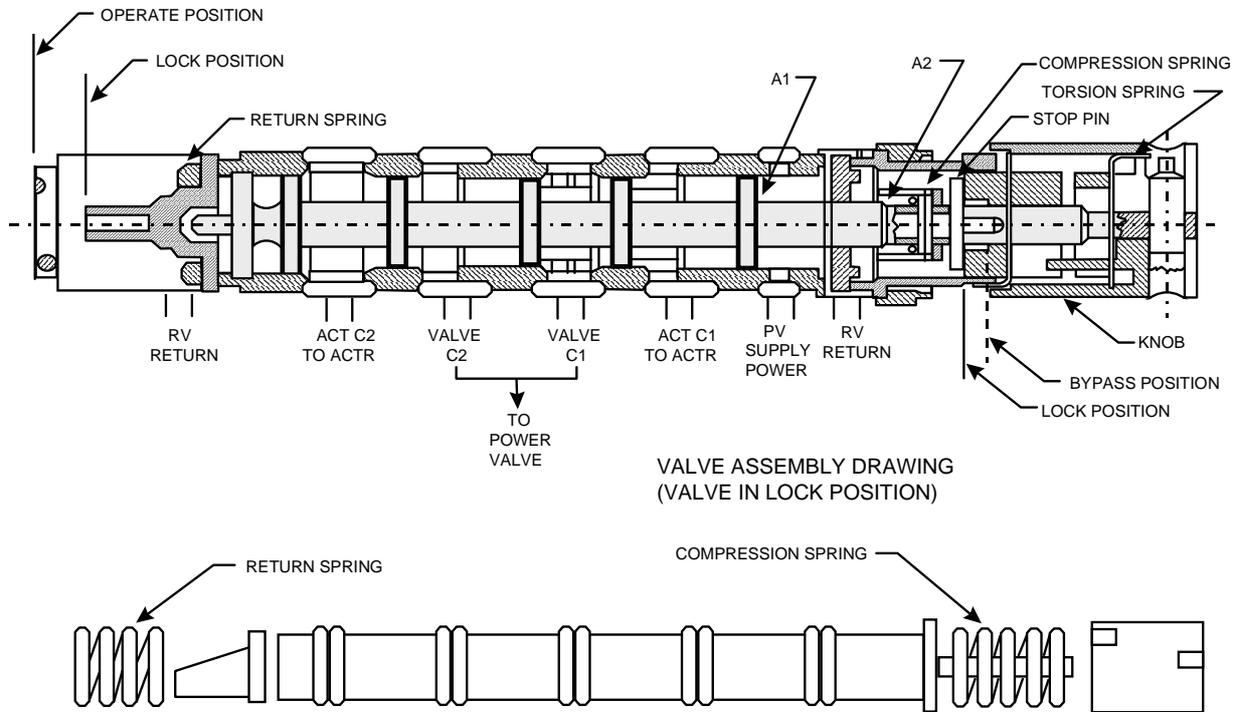


Figure 11-2.- Bypass valve.



- KNURLED KNOB WHEN PUSHED IN AND ROTATED CW 90 DEG AGAINST TORSION SPRING WILL CAUSE
- VALVE TO GO TO BYPASS POSITION (SPOOL MOVES TO RIGHT).
- SPOOL MOVES TO LEFT (OPERATE POSITION) WHEN SUPPLY PRESSURE TO PV IS >1000PSI (.5 IN.) AGAINST
- STOP INSIDE RETURN SPRING CONNECTING ACT C2 TO VALVE C2 AND ACT C1 TO VALVE C1.
- IN BYPASS POSITION, ACT C2, VALVE C2, VALVE C1, AND ACT C1 ARE ALL INTERCONNECTED.

Figure 11-3.- Hydraulic lock and manual cylinder bypass valve.

11.4.4.2 ME Lock Valve

The ME lock valve is a two-position valve, the operate position and the lock position. The lock position is achieved when system pressure falls to 600 psi plus 100, minus 0. If pressure subsequently rises above 2000 psig, the valve is returned to the operate position. (The lock valve pressure requirements are specified in the Rockwell International/Space Division document no. MC621-0015, Servoactuator, Main Engine, Electro-Command, Hydraulic (TVC), paragraph 3.2.1.16). In the operate position, hydraulic pressure from the power valve is allowed to flow uninhibited through the lock valve to the piston ram.

11.4.5 Power Valve

The power valve (fig. 11-4) consists of a cylinder and a linear power spool which can be shuttled to the left or right within a cylinder. At either end of the cylinder there is an outlet which allows hydraulic fluid to flow in or out of the reservoir at the ends of the power spool, without generating any fluid restriction, as the spool is shuttled in either direction. At either end of the cylinder are four inlets, which are linked to the four servovalves (two lines per servovalve). Note that each pair of lines flows on into a reservoir on either side of two pistons. For example, the hydraulic fluid lines for servovalve 2 enter on the left half of the cylinder. There are two reservoirs in this portion of the cylinder which house an inner and outer power spool piston. One fluid line is connected to the outboard reservoir A, on the left side of the outboard power spool piston, and the other fluid line is connected to the inboard reservoir B, on the right side of the inboard power spool piston. Servovalves 1 and 2 have their fluid inputs on the left half of the power valve cylinder. The center portion of the cylinder has two primary hydraulic pressure inlets at points C. The two inputs lead to the primary reservoir which contains three power spool bushings. The two outboard bushings are used to restrict primary hydraulic fluid flow into the primary reservoir from either source pressure input, when the power valve is in a neutral position. The third bushing is in the center of the power valve primary reservoir and is used to channel off hydraulic fluid to the Orbiter return lines.

Example:

If the four servovalves receive equal positive voltage commands, then a positive hydraulic pressure is applied to the left side of the four pistons. This would cause the power spool to shuttle to the right. In doing so, the three primary reservoir bushings would shift to the right. This would result in the hydraulic source pressure on the right inlet being opened, and the center bushing blocking off the return path on the right half of the primary return D. The left bushing will shift to the right and still restrict hydraulic inlet flow at inlet C to the left half of the primary reservoir. Once the preceding has occurred, primary hydraulic pressure will be directed to the cylinder and piston assembly by way of outlet E on the right, via the lock valve. Returning hydraulic fluid from the cylinder and

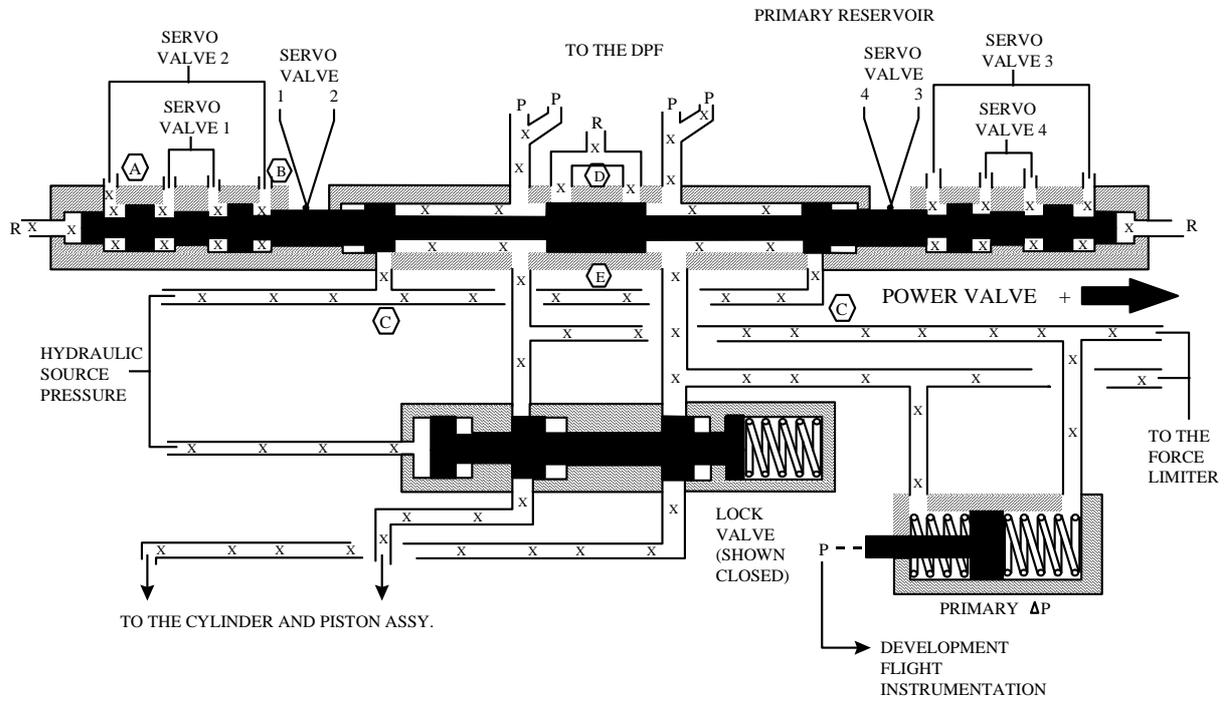


Figure 11-4.- Power valve.

piston assembly would enter passage E, on the left, into the left half of the primary reservoir and out through passage D to the Orbiter hydraulic supply. On either side of the return passages are canals which are directed to the dynamic pressure feedback assemblies.

11.4.6 Cylinder and Piston/Ram Assembly

The piston/ram assembly is used to produce movement of the Space Shuttle Main Engine (SSME) and SRB bells. Hydraulic fluid necessary to move the engine bell flows through the lock valve when system pressure exceeds 1000 psi for the SRB's and 2,000 psi for SSME's. This linear movement of the piston/ram produces rotational motion of the bell about a point. The linear motion of the piston/ram also causes opening and closing of a strain gauge position transducer. This transducer provides a mechanical position feedback linkage to the servovalves and also sends a position signal to downlink telemetry. Figure 11-5 shows a cylinder and piston/ram assembly.

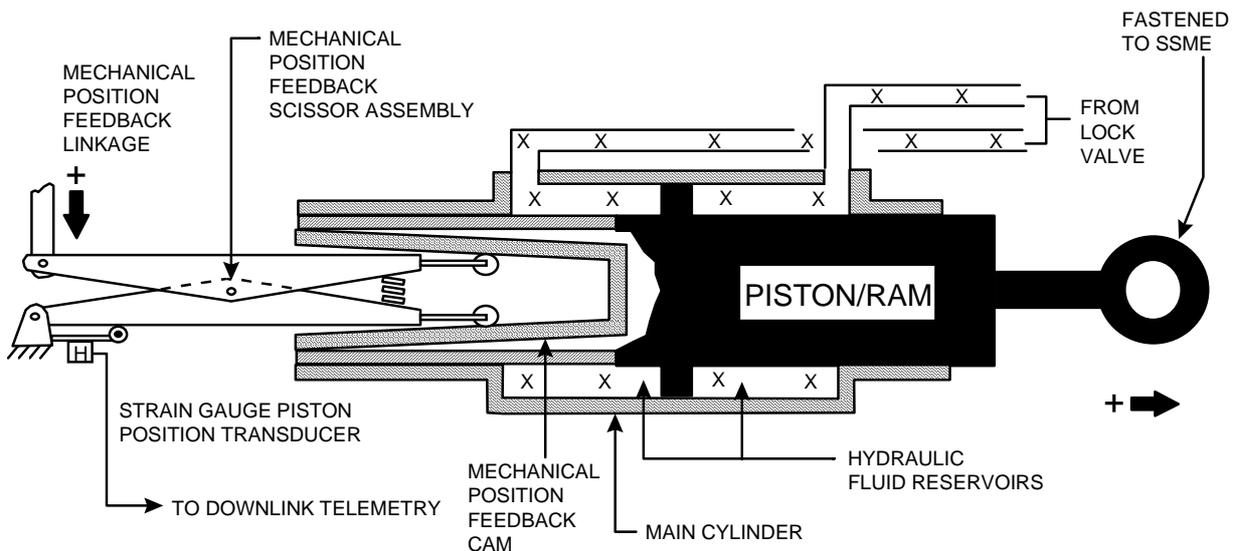


Figure 11-5.- Cylinder and piston/ram assembly.

11.4.7 ATVCs

As discussed in section 11.4.2, four ATVCs supply the signals which drive the electrohydraulic servovalves. Each ATVC may receive main propulsion system (MPS) and SRB actuator position commands, actuator channel bypass (capability does not exist with current Flight Software), and actuator channel override signals through flight aft multiplexer/demultiplexer (MDM's) from their respective general purpose computers (GPCs.) and secondary delta pressure feedback signals directly from the actuators. In addition, two master event controllers switch bus power to the actuators. The inputs are processed by each ATVC, and any necessary position or isolate commands are issued to the actuators. Reference SSSHB drawing 9.15 for SSME and SRB TVC signal flow.

With flight control system (FCS) channel switches in the off position, control bus power cannot flow through each ATVC switch to the ATVC power supply. Only isolation power is available, which is unaffected by FCS channel switches. In the auto (middle position), control bus power is available to operate each ATVC power supply. In the override position, control bus power is still available, and in addition a signal is present to prevent the delta pressure fault detection circuitry from isolating a channel. The ATVC switch, on the other hand, only has two positions: ON and OFF. In the OFF position, no power is supplied to the ATVC power supply. In the ON position, power is available to isolate a channel and if the appropriate FCS channel switch is on, then power is also available to operate the ATVC power supply.

With the ATVC power supply operational, bus power is also available for actuator driver power. Under nominal conditions, an actuator position command is issued to each of the four ATVCs. For example, a +2° center ME pitch actuator command would be issued to all four ATVCs. Each ATVC would process the command and send a signal to its respective electrohydraulic servovalve. The ATVC processing includes summing the command with the negative of the secondary delta pressure after it passes through a redundant channel equalization circuit. The redundant channel equalization circuit contains a delta pressure gain, a lag limiter gain, and a lag network. The delta pressure deadband gain outputs a zero if the secondary delta pressure supplied to it is between + 1175 psi. Once the delta pressure exceeds these limits, the output from this device begins to ramp upward, and after passing through a lag limiter, gain circuit is subtracted from the position command.

Therefore, redundant channel equalization provides feedback out of phase with the position command. This will null channel bias when delta pressure is greater than 1175 psi. The lag limiter inhibits this feedback when delta pressure fault detection responds to failures.

Secondary delta pressure feedback is also routed to the delta pressure fault detection circuit. This circuit monitors all incoming secondary delta pressure signals and should a signal greater than +2200 psi be detected, then the up/down integrator counts up. Should the signal fall below +2200 psi, then the integrator counts down at half the rate at which it counts up. It takes 120 ms for the integrator to count from zero to its maximum level at which time an isolation command is issued to the fault latch. Provided that there is no override command present, the isolation valve driver is activated. Subsequent to this time, even if the secondary delta pressure falls below +2200 psi, the fault latch holds the isolate command and that channel of the actuator will remain isolated. If, however, an override command is issued, then the up/down integrator is cleared, the fault latch is reset, and the isolate command is terminated. A command to isolate a channel of an actuator may also be issued in two other ways. First, the absence of either a +8.5 V dc signal, a +15 V dc signal, or a 1000 Hz signal will cause that channel of the actuator to isolate until the missing voltages are resupplied. The final method for isolation occurs when a bypass command is issued by a GPC through a flight aft MDM (capability does not exist with current Flight Software).

11.5 MISSION PERFORMANCE

The performance portion covers the period of time from lift-off minus 2 min to ME stow for STS-1 through STS-5.

11.5.1 SRB Actuators

SRB actuators performed nominally throughout all flights. SRB slew occurs approximately 23 sec prior to lift-off. Slewing normally produces the largest change in secondary delta pressures. Typical minimum and maximum secondary delta pressures are in the range of -700 psi to +700 psi.

11.5.2 ME Actuators

The SSME actuators functioned nominally from STS-1 through STS 51-F with three exceptions. These exceptions were (1) pitch oscillations as ME's were commanded to the dump position, (2) slightly high secondary delta pressure after 8 min of flight, and (3) a bypass of right yaw channel 4 at APU powerup prior to entry on STS-11. The last was attributed to silting. Silting has been defined as contamination of the hydraulic fluid due to long terms of ATVC inactivity.

During these missions, excessive oscillations ranging from 0.37° peak-to-peak to 0.6° peak-to-peak were encountered. Also during the period of these oscillations, jets fired to counter these excessive rates. Software change request CR39360A, implemented on STS-5, changed the actuator movement rate and increment from 1 Hz and 1°/step to 12.5 Hz and 0.08°/step. This actuation reduced the oscillations. It should be noted that the rate of engine movement is the same in both cases, 1 deg/s.

The other item worth mentioning is a high secondary delta pressure on the center ME channel 4 pitch axis. Between main engine cutoff (MECO) and ME stow, the secondary delta pressure on the channel 4 pitch axis was observed to climb at a faster rate and achieve a higher value than any other secondary delta pressure. The largest value noted was -1126 psi. If this value had reached -1175 psi, the redundant channel equalization in ATVC 4 would have prevented the pressure from exceeding -1175 psi.

11.5.3 Lessons Learned

The SRB actuators have been proven through all flights so far. Actuator positions followed commands and secondary delta pressures were relatively small. Although no circumstances have arisen to cause redundancy management (RM) to bypass a channel, these actuators are reliable at least under the circumstances noted.

SSME actuators have also performed quite well. The oscillations experienced as the engines were commanded to the dump position should not be present for future flights. Expect higher than average secondary delta pressure on channel 4 pitch axis of the center ME actuator.

DF NOI #171

March 6, 1984

REFER TO: DF6/84-8 (STS-9)

February 28, 1984

To: Distribution

From: DF62/F. DiGenova

Subject: FCS Channel/ATVC Anomaly During STS-9 Landing - OV102

BACKGROUND:

The Orbiter FCS Channel Switches (4) are three-contact, three-position, lever-locking (locking at center position) toggle switches on Panel C3. The switches control the power to both the ASA and ATVC boxes and, in addition, provide three-contacts to GNC SW RM via FF MDM's when the FCS CH SW is in the override position. The three-switch positions of the switch are override, auto (lock position), and off.

When the FCS switch is in the auto or override positions, power is applied to ASA/ATVC boxes, provided the respective ASA and ATVC switches are on.

- When the FCS switch is moved from the auto position to the override position and vice versa, there is always a power interruption to the ASA/ATVC boxes (in the order of milliseconds because of break-before-make switch contacts).
- This power interruption causes all the actuators on that FCS channel to temporarily "Bypass" until the internal power supplies come back up, which could take up to 900 msec in the ASA and up to 400 msec in the ATVC.

During the STS-9 (OV-102) rollout phase of the landing, an anomaly with the ATVC 3 box occurred (S/N 9) when the crew placed the FCS channel three-switch to the override position. Postmission testing and analysis to date has revealed the following:

- When the FCS SW went from auto to override, the ATVC three-power supply (internal) went down due to the temporary input power interrupt but did not come back up when the switch was in the override position, as it should have. When the FCS SW was subsequently moved to auto position, the ATVC power supply did come up, as it should.

POSTMISSION ANALYSIS:

Investigation and analysis of this anomaly revealed

- A jumper wire was missing between a resistor and the emitter lead of transistor Q16 in the internal power supply. A sneak circuit allowed the transistor to be on if power was only interrupted for a very short interval (2 to 5 msec). If power interruption was longer, transistor turned off due to the missing wire preventing ATVC from coming back on. During acceptance testing, power interruptions were very short and sneak circuit prevented the missing jumper from being detected. The delay when the STS-9 crew was switching the FCS channel switch was long enough to cause transistor Q16 to be off, preventing power up of the unit.
- This circuit does not normally get tested for this problem during normal vehicle turnaround.

CORRECTIVE ACTION:

OV-099 was tested on January 20, 1984 by cycling the FCS channel switches and does not have any problem with the power-on-reset.

- Consideration is being given to include a ATVC/override test in the normal prelaunch tests.

CONCLUSIONS:

The bottom line of the above discussion is as follows:

- The acceptance testing procedure of the transistors was reviewed and there are no issues of concern.
- There are a mix of new and old cards in use in ASA's and ATVC's. The new cards use printed circuit (Pc) instead of a "jumper" wire as in the old cards. For STS 41-C, ATVC 4 has an old card, all other have new cards. The FCS channel switch controlling ATVC 4 was cycled 20 times verifying the unit was good (no missing jumper).

CREW PROCEDURAL RECOMMENDATIONS:

- When the crew needs to place two FCS channel switches to override (for the two-channel fail case), delay at least 1 second before throwing the second FCS channel switch.
- In the event the crew gets an "FCS channel fail" message for the channel that was placed in override, they should move that switch back to auto. If that does not clear the fail, move the switch to off, then to auto.
- One last item of interest regarding the break-before-make characteristics of the FCS channel switch is that the reset function can be accomplished by moving the FCS channel switch from auto to override and then back to auto even if the override contacts are "failed open." This is because the reset occurs as a result of the input power cycle rather than the issuance of the GPC override discrete.

Frank DiGenova

DF NOI #190
REFER TO: DF6/84-52 (GENERIC)

June 28, 1984

To: Distribution
From: DF62/Frank DiGenova
Subject: SSME Second Stage Pitch Paralleling

The purpose of this NOI is to describe SSME (Shuttle System Main Engine) second stage pitch paralleling and to point out the conditions for inhibiting or disabling pitch paralleling.

BACKGROUND:

Pitch paralleling allows a 500-lb performance gain to be attained when permitted under no-failure conditions and is included in the flight software for a particular flight. Flight STS 41-F is planned to have pitch paralleling software.

When the pitch axis of the SSME's are at 0° position, the thrust vector pitch of each engine with respect to the spacecraft X-axis is as follows:

- Center engine thrust vector is -16°.
- Left and right engine thrust vectors are at -10°.

The center engine thrust vector is apparently not parallel to the two lower engine thrust vectors but converges forming an angle of 6° in pitch with respect to the lower engines.

When pitch paralleling is included in the flight software it occurs automatically at 12 seconds after SRB separation assuming no HYD failures have occurred. The center engine moves its thrust vector up (bell down) 4° while the two lower engines move their thrust vectors down (bells up) 2° which makes the thrust vectors parallel to each other in pitch. The SSME's are then nominally returned to their unparallel position before MECO (when the vehicle mass gets to less than 15,580 slugs (»500,000 lb).

When pitch paralleling was first flown on STS-8, the pitch biases were +3° for the center engine and -3° for lower engines. While this configuration still attained the parallel thrust vectors, it caused a noticeable vehicle transient since the new resultant thrust vector had to be repositioned by the flight control system. The new numbers of +4° and -2° eliminate this vehicle transient.

The RM threshold values are as follows:

- A Hydraulic system is declared failed if the selected HYD pressure is less than 1740 PSI.
- A hydraulic transducer will be declared failed if the delta pressure is greater than 250 PSI when compared with the two good transducers.

The basic CR's for pitch paralleling are as follows:

- CR 59279A "SSME pitch parallel second stage."
- CR 69441A "Second stage pitch parallel modification."

This MOD is applicable to both PASS and BFS.

Frank DiGenova

DF6/FDiGenova:dkj:7/3/84:5595

DF NOI #194
REFER TO: DF6/84-64 (STS 41-C)

8/14/84

To: Distribution

From: DF62/Jeff Bantle

Subject: Linear Accelerations Due To MPS Dump, STS 41-C

Various numbers and characteristics are typically mentioned regarding the MPS dump and its effect on the vehicle. The purpose of this note is to explain and disseminate linear acceleration data and total DV's due to the MPS dump seen on STS 41-C.

To get a better understanding of the linear acceleration and total DV due to the MPS dump, an analysis of STS 41-C IMU 1 DV (in M50) was completed. STS 41-C was of particular interest for such an analysis since a direct insertion was performed that meant the MPS dump was not done during an OMS burn.

The results of this analysis are shown in table 1 and figure 1. Table 1 shows IMU DV (in M50) derived accelerations for one 5-second interval and eight 10-second intervals. Any DV due to jet firings was removed. These accelerations were transformed to Orbiter body axes and then plotted in figure 1. Also, the total vehicle linear acceleration is shown in figure 1. The dump appeared to last for about 1 minute and 20 seconds; however, the acceleration for this time period was not constant and seemed to drop off after about 40 seconds. The linear acceleration due to the MPS dump is almost totally in the Orbiter body x axis. The total DV due to the dump was about 6.0 ft/sec (see table 1).

TABLE 1

IMU 1V (M50) Velocities and Accelerations (M50)
(Including correction for jets)

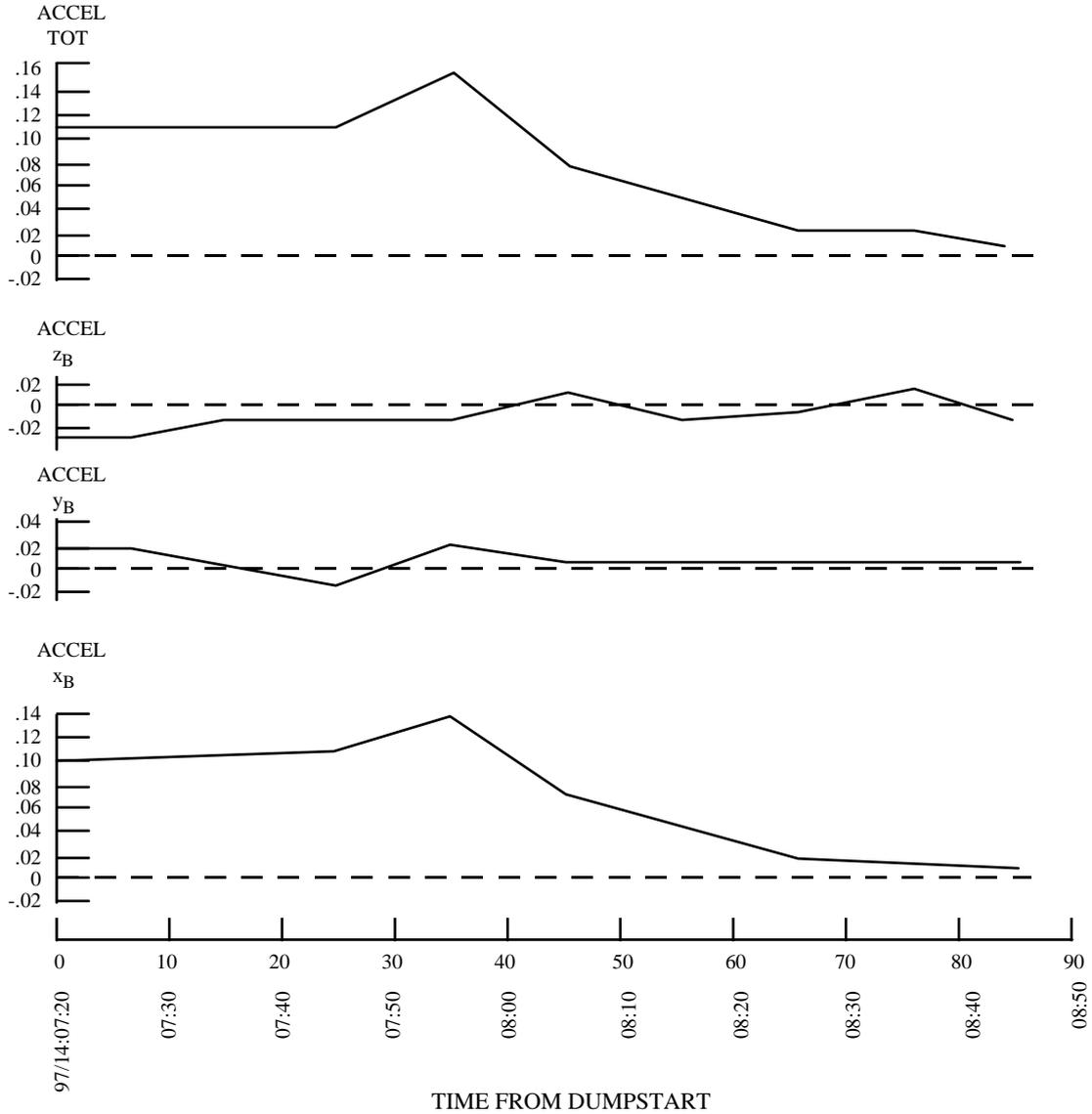
| TIME | lx | ly | lz | $\Delta V_x/A_x$ | $\Delta V_y/A_y$ | $\Delta V_z/A_z$ |
|-------------|-----------|------------|-----------|------------------|------------------|------------------|
| 97/14:07:20 | 59957.240 | -16884.973 | 32420.501 | | | |
| 07:25 | 59957.359 | -16884.438 | 32420.439 | .119/.024 | .535/.106 | -.062/-.013 |
| 07:35 | 59957.772 | -16883.398 | 32420.277 | .413/.041 | 1.040/.104 | -.162/-.016 |
| 07:45 | 59958.258 | -16882.385 | 32420.171 | .486/.049 | 1.013/.101 | -.268/-.027 |
| 07:55 | 59958.786 | -16880.931 | 32420.023 | .528/.053 | 1.454/.145 | -.054/.055 |
| 08:05 | 59959.188 | -16880.274 | 32419.995 | .402/.040 | .657/.066 | -.028/-.003 |
| 08:15 | 59959.296 | -16879.872 | 32419.967 | .108/.011 | .403/.040 | -.028/-.003 |
| 08:25 | 59959.331 | -16879.685 | 32429.962 | .034/.003 | .187/.019 | -.005/0 |
| 08:35 | 59959.474 | -16879.563 | 32420.080 | .143/.014 | .122/.012 | .118/.010 |
| 08:45 | 59959.495 | -16879.473 | 32420.117 | .021/.002 | .080/.009 | .037/.004 |

TABLE 1
(Continued)

Velocities and Accelerations (Body Axes)

| TIME | ΔV_{xB} | A xB | ΔV_{yB} | A yB | ΔV_{zB} | A zB | ΔV_{TOTB} | A $_{TOTB}$ |
|-------------------------|-----------------|------|-----------------|-------|-----------------|-------|-------------------|-------------|
| 97/14:07:20 | | | | | | | | |
| 07:25 | .52 | .104 | .11 | .0 | -.13 | -.025 | .55 | .109 |
| 07:35 | 1.12 | .112 | .12 | .012 | -.13 | -.013 | 1.13 | .113 |
| 07:45 | 1.15 | .115 | -.02 | -.002 | -.13 | -.013 | 1.16 | .116 |
| 07:55 | 1.51 | .151 | .32 | .032 | -.11 | -.011 | 1.54 | .154 |
| 08:05 | .77 | .077 | .07 | .007 | .06 | .006 | .78 | .078 |
| 08:15 | .40 | .040 | .09 | .009 | -.07 | -.007 | .42 | .042 |
| 08:25 | .18 | .018 | .06 | .006 | -.04 | -.004 | .19 | .019 |
| 08:35 | .16 | .016 | .07 | .007 | .12 | .012 | .21 | .021 |
| 08:45 | .08 | .008 | .06 | .006 | .01 | .001 | .10 | .010 |
| TOTALS | 5.89 | | .88 | | -.42 | | | |
| TOTAL $\Delta V = 5.97$ | | | | | | | | |

LINEAR ACCELERATION VS. TIME



SECTION 12 OMS TVC

12.1 GENERAL

The Orbital maneuvering subsystem (OMS) is the Shuttle system designed to provide thrust for orbit insertion, orbit transfer, rendezvous, deorbit, and launch aborts. The OMS is housed in two independent pods located on each side of the aft fuselage. The pods, which also house the aft reaction control system (RCS), are referred to as OMS/RCS pods. Each OMS/RCS pod contains one OMS engine and the hardware needed to pressurize, store, and distribute propellant to perform the burns. The two pods provide complete redundancy for the OMS.

12.2 GIMBAL ACTUATION

Each OMS engine is equipped with a gimbal and a control assembly that allows gimbaling of the engine in two axes to control the engine thrust direction in pitch and yaw. It consists of a gimbal ring assembly and a gimbal actuator assembly.

The gimbal ring assembly contains the mounting pads to attach the engine to the vehicle. It provides multiaxis engine gimbaling capability and transmits the engine thrust through two vehicle interface thrust pads.

The gimbal actuator assembly provides force to gimbal the engines for thrust vector control (TVC). Each engine has a pitch actuator, a yaw actuator, and primary and secondary servocontrol units (SCU's). The primary SCU controls the primary motor of the pitch and yaw actuators, and the secondary SCU controls the secondary motor for the same.

The pitch and yaw gimbal actuators provide linear travel output by driving a screw actuator arm. Each actuator consists of redundant brushless dc motors, gear train, jackscrew, and nut-tube assembly, and yields linear position feedback (PFB).

A position command (CMD) signal from the primary SCU energizes the primary dc motor which is coupled with a reduction gear and a no-back device. The output from the primary power train drives the jackscrew of the drive assembly. This causes the nut-tube to translate which results in an angular engine movement when the secondary power train is idle. If the primary power train is inoperative, actuation of the secondary motor drives the engine by applying torque to the nut-tube through the spline that extends along the nut-tube for the stroke length of the unit. Rotation of the nut-tube about the stationary jackscrew causes the nut-tube to translate along the screw resulting in an angular engine movement (dwg. 12-1); refer to Drawings section 12.7.4.

12.2.1 SCU

Each OMS engine contains a primary and secondary SCU. Each unit controls a pitch and a yaw actuator motor drive mechanism. The primary and secondary control units are electrically isolated and are not operated concurrently. The unit operates as follows: an engine gimbal position CMD is received from the computer through a multiplexer/demultiplexer (MDM) as an analog of engine position. This CMD is summed with a corresponding engine PFB to form an error signal. This error signal is scaled and sent to the motor control logic which drives the brushless dc motor to reposition the engine until the error is eliminated.

The engine position is sensed with a linear voltage differential transformer (LVDT) and sent back to the SCU where it is demodulated and used for forming the error signal. The demodulated position signal is sent to the computers via MDM's where it is used for the actuator fault detection process and to display engine gimbal position on the CRT maneuver (MNVR) displays. (xxxxx MNVR YYYYYY) (fig. 12-1)

| | | | | | |
|----------------|----------------------|------|---------------------------|------|------------------------|
| XOX1 | XXXXX | MNVR | YYYYY | XX X | DDD/HH:MM:SS |
| OMS BOTH | 1X | | | | DDD/HH:MM:SS |
| | L | 2XS | | | |
| | R | 3XS | | | |
| RCS SEL | 4X | BURN | ATT | | Δ VTOT XXXX . X |
| 5 TV ROLL | XXX | 24 | R XXX | | TGO XX . XX |
| TRIM LOAD | | 25 | P XXX | | |
| 6 P | [\pm] X . X | 26 | Y XXX | | VGO X \pm XXXX.XX |
| 7 LY | [\pm] X . X | MNVR | 27X | | Y \pm XXX . XX |
| 8 RY | [\pm] X . X | REI | XXXX | | Z \pm XXX . XX |
| 9 WT | XXXXXX | TXX | XX : XX | | |
| 10 TIG | | | GMBL | | |
| | XXX/ XX : X : XX . X | | L R | | HA HP |
| TGT | PEG 4 | P | \pm X . XS \pm X . XS | | TGT XXX \pm XXX |
| 14 C1 | XXXXX | Y | \pm X . XS \pm X . XS | | CUR XXX \pm XXX |
| 15 C2 | [\pm] X . XXXX | | | | |
| 16 HT | XXX . XXX | PRI | 28X 29X | | 35 ABORT TGT XX |
| 17 \ominus T | XXX . XXX | SEC | 30X 31X | | |
| 18 PRPLT | [\pm] XXXXX | OFF | 32X 33X | | FWD RCS |
| TGT | PEG 7 | | GMBL CK 34X | | ARM 36X |
| 19 Δ VX | [\pm] XXXX . X | | | | DUMP 37X |
| 20 Δ VY | [\pm] XXX . X | | XXXXXXX | | OFF 38X |
| 21 Δ VZ | [\pm] XXX . X | | | | SURF DRIVE |
| XXXX | 22 / TIMER | 23 | | | ON 39X |
| | | | | | OFF 40X |
| | | | | | (XX) |

Figure 12-1 MNVR Display

12.2.2 Actuator Assembly Power

Logic and instrumentation power as well as motor drive power to the actuators and to the SCU's is received from the main buses. Instrumentation and logic power comes from a main bus and goes to a power supply which provides excitation voltage to the pitch and yaw actuator electronics. There is a power supply for the primary control unit and a separate power supply for the secondary control unit of each OMS engine.

Motor drive power is applied to the pitch and yaw actuator motor controllers when general purpose computer (GPC) signals turn on two remote power controllers (RPC's), allowing main bus power to the

units. These two RPC's are both required (i.e., are not redundant since one RPC alone cannot carry the total current when both pitch and yaw axes are slewing simultaneously). If only one RPC is working, then the TVC can slew one axis at a time. Power signals from the GPC's go to the primary or the secondary units depending on the actuator selection item entries on the MNVR displays.

Selection between the left and right engine is independent, but only one item can be selected for a particular engine. The TVC CMD SOP in the primary avionics software system (PASS) or backup flight system (BFS) powers the selected units when the subsystem operating program (SOP) is operating.

12.3 OMS TVC

The main function of the OMS TVC is to convert automatic and manual MNVR CMD's to OMS engine deflections in the form of actuator extend and retract CMD's. This function is mainly performed in the digital autopilot (DAP) but TVC also includes processes and conditions for the engines to be trimmed, stowed, and checked, and for the software and crew interfaces needed for them to take place. The OMS TVC CMD SOP is the central point of any output from the GPC to the engine actuators. Outputs from the actuators, in the form of PFB, go through the OMS TVC FDBK SOP before being used by the other software modules in the GPC.

12.3.1 TVC Command SOP

The OMS TVC CMD SOP processes and outputs pitch and yaw OMS engine actuator CMD's and the primary/secondary/off (PRI/SEC/OFF) power selection discretetes. The CMD SOP is operational during major modes (MM's) 104, 105, 106, 202, 301 through 304, and 602. Actuator CMD outputs are dependent on different events and are selected as follows.

a. OMS control law position CMD's (two yaw and two pitch) come from the TRANS DAP during operational sequence (OPS) 1 and 3 and from the on-orbit DAP during OPS 2. These CMD's are processed whenever the OMS gimbal drive flag, the MM 303 flag, the transatlantic abort landing (TAL) declared flag, and the MM 602 flag are indicating false.

b. The OMS gimbal drive check flag is monitored in MM's 104, 105, 106, 301, 302, when the TAL declared flag is false, and from the ORB MNVR EXEC display during OPS 2. The gimbal drive check is performed via an item entry on the MNVR display. Pitch and yaw position CMD's for both engines are generated simultaneously. Only the selected PRI or SEC system will be driven.

The drive check is done in three steps, each 6 sec in length. Starting from where the engine is positioned (previous trim values), the pitch actuators drive in response to step CMD's of first +5.89°, then -5.89°, then back to +5.89° and then back to the previous trim values. Similarly, the yaw actuators are driven to +6.44°, then -6.44°, back to +6.44°, and then to the previous trim values. During the gimbal drive check, the drive check CMD's take precedence over the control law CMD's.

c. Upon transition to MM 303, the engines are commanded to the stow position. In the case of a TAL, the engines will remain in the stow position in MM 104.

d. In MM's 304 and 602, the actuators are commanded to the stow position unless a contingency dump is declared (for loss of two main engines (ME's) or crew selection), in which case they are commanded through the c.g. until dump completion when they are commanded to stow. Upon transition to MM's 305 and 603, the channels are deselected.

e. The zap CMD is issued anytime the crew selects channels via the MNVR display and lasts for 160 ms. Its purpose is to drive the engine into the hardstop in the event the TVC is jammed. The intent

is to free the tension on a no-back device which could be locking the TVC in a position against the hardstop.. It is a -8.37° CMD for 80 ms and then a $+8.37^\circ$ CMD for 80 ms. The movement in one direction is 0.4° with a nominal drive rate of 5 deg/s. The zap CMD is sent to all actuators on both engines and is reset to zero immediately after execution.

12.3.2 PRI/SEC Select Processing

Power to the actuator motors is output by the OMS TVC CMD SOP depending on crew selection of the actuator to be powered up. Crew selection of OFF for the left or right engine on the MNVR display inhibits motor power to the pitch and yaw actuators on that engine (fig. 12-1).

If the primary or secondary is selected, power is applied to the primary or the secondary motors of the pitch and yaw actuators in that engine anytime the TVC CMD SOP is running.

Channel selection is determined by monitoring the PRI/SEC/OFF indications from MNVR display. Due to a known software timing condition when using the ZAP command, it may be necessary to select a TVC a second time on the MNVR display when changing to the alternate system (DR 55241).

12.3.3 OMS TVC Feedback SOP

The purpose of the feedback (FDBK) SOP is to monitor the PRI/SEC/OFF channel select indicators and channel COMM-fault (CF) discretes to perform data selection and compensation of the pitch and yaw actuator FDBK data. It monitors the PRI/SEC/OFF CMD's on an engine basis and the CF discretes on an actuator channel basis.

For a CF on a primary/secondary TVC when it is selected, the FDBK will be set to the previous valid value, the data good (DG) flag will be set bad, and M's will appear on the MNVR display next to the gimbal positions for the respective actuator.

When both channels are commanded off, the FDBK SOP does not run and the previous selected channel data are provided on the MNVR display but not updated. If the previous data are COMM faulted, an M will appear until a valid input is selected. If the previous data are not COMM faulted, the data are provided independent of subsequent CFs.

When a transition to MM 301 is made with both channels commanded off, the FDBK values for both pitch and yaw are initialized to $+1.0^\circ$. Since the CMD's do not change during the transition, it is very likely that redundancy management (RM) will fail one or more actuators.

12.3.4 Ascent RM

On transition to operations sequence (OPS) 1, both channels are commanded off. In MM's 101 to 103 and 601, RM compares the primary and secondary FDBK's to the commanded stow position. If the absolute value of the difference between both the primary and secondary feedbacks (2 of 2) exceeds the OMS threshold of 0.7° , the fail counter will be incremented and after a count of two, the primary will be powered up to drive the engine back to the stow position and the counter will be reset and power will remain on (CR 92188). However, if the primary channel fails to drive the engine back to within 0.7° of the CMD, one more count (total of three) will cause the primary channel to be powered down and the secondary channel to be selected. If the primary and secondary feedbacks disagree and only one feedback (1 of 2) indicates motion of greater than 0.7 degrees, then no actuator is powered. Not powering and driving an actuator for a single transducer failure prevents a bad position feedback from faking out RM into driving the actuator out of the stow position. If one of two position feedbacks are commfaulted, then the remaining position feedback (1 of 1) is used to determine if the engine should be

powered and driven to the stow position. With one position feedback commfaulted, then the Ascent RM protects the OMS engine by assuming that any additional position feedback indications greater than the RM threshold are real and must be powered (on the remaining system) and driven to the stow position.

Individual cycle counters are used for each engine and are reset if an intervening cycle occurs in which the absolute value of the difference between the FDBK and the commanded position is less than the OMS threshold (0.7°).

Entering MM 104 or 602 selects the primary channel if no channel had previously been selected. Otherwise the previous channel remains selected. On all MM transitions, the selected channel is reflected by the last crew input via the MNVR display.

12.3.5 OMS Actuator Fault Detection and Identification

The objective of the OMS actuator fault detection and identification (FDI) is to detect and identify off-nominal performance of the pitch and yaw gimbal actuators of the two OMS engines, and generate OMS fail status indications. Since inputs for performance monitoring of the four actuators are made available and processed separately for each actuator, the failure detection and identification problem reduces to one of detection only. Failures are detected by testing the actuator PFB versus the actuator commanded position and comparing the result to an I-loaded threshold (CR 39620). A specified number of successive failure indications are necessary to set an OMS actuator fail flag. The four OMS actuator failure indications are generated for display to the crew.

The following apply to the OMS actuator FDI algorithm.

- a. In the case of a CF (DATA_GOOD = 0), OMS actuator FDI will not be processed on that actuator channel data and an M will be displayed.
- b. An unfailed OMS actuator achieves a steady-state nominal extension rate within two minor cycles (80 ms) in response to an applied voltage.
- c. The algorithm is not capable of detecting failures downstream of the actuator output; i.e., in the gimbal mounting structure.

The pitch and yaw actuators are identical in design, except for the stroke length, and contain redundant channels, which couple to a common drive assembly.

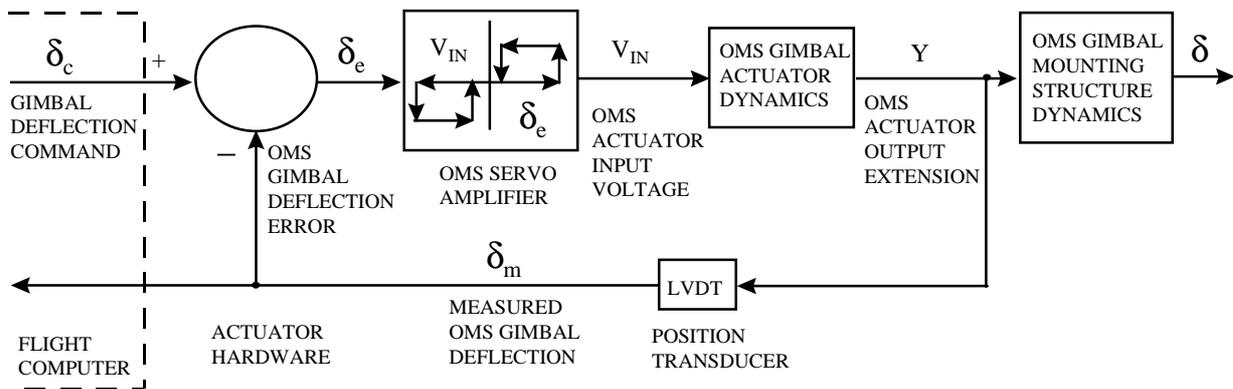


Figure 12-2 Simplified block diagram of OMS gimbal actuator Servo Loop

Figure 12-2 is a simplified block diagram of an OMS gimbal actuator servo loop. The gimbal deflection CMD, δ_c , is differenced with the measured gimbal deflection, δ_m , to obtain the gimbal deflection error δ_e . The error is then supplied to a servo amplifier with a dead zone and hysteresis. The output of the servo amplifier is the actuator input voltage, which drives the actuator motor. The output of the actuator motor is the actuator extension measured from null. An LVDT position transducer senses the actuator output extension and converts it to the measured gimbal deflection. The actual gimbal deflection is the output of the OMS gimbal mounting structure dynamics, while the measured gimbal deflection is inferred from the actuator output extension.

A failure is detected if the difference between the PFB and the commanded position exceeds 2.0° for a count of 24 (3.84 sec), at which time the fail flag will be set. The fail counter was changed from 4.80 sec to 3.84 sec by CR 39895B in order to detect actuators that are driving at less than the minimum spec rate of 2.9 deg/sec during gimbal checks.

Gimbal failures are annunciated by a C&W alert light - OMS TVC -, a down arrow by the failed actuator on the MNVR display, and an L or R OMS GIMBAL message. The associated fail counters for a given set of gimbals will be reset whenever the crew changes the selected channel via keyboard entry.

The associated fail flags for the active gimbal drive system shall be set to zero whenever the crew reselects the active drive system or selects a new active drive system.

12.4 BFS TVC

OMS burns in the BFS are done in manual TVC. Alignment and control for burns is the same as in the primary avionics software system (PASS).

12.4.1 OMS TVC Command Hardware Interface Program

The OMS TVC CMD hardware interface program HIP is the BFS counterpart to the PASS CMD SOP. Its purpose is to compensate position CMD's from selected software modules before they are sent to the two OMS engines' pitch and yaw actuators. It also monitors gimbal actuator selection from the MNVR display and outputs the redundant power-on CMD signals to the selected actuator unit.

Like the PASS, the BFS OMS TVC CMD output of power-on to the proper actuator unit is based on the TVC CMD HIP running and the actuator selection not in the off position. The CMD HIP is active in all MM's in which the MNVR display is operational.

12.4.2 OMS TVC FDBK HIP

The OMS TVC FDBK HIP monitors the actuator selection discrettes from the MNVR display and performs bias and scale factor compensation on the selected pitch and yaw actuators' FDBK data (ref. sec. 12.7.2). These data are then displayed to the crew via the MNVR display. The FDBK HIP, like the CMD HIP, is only active when the MNVR display function is active.

12.4.3 BFS OMS TVC FDI

BFS FDI is active concurrently with the CMD and FDBK HIP and monitors the actuator CMD's and PFB to determine if an actuator has failed. Like the PASS, the BFS declares an actuator failed whenever the difference between the CMD and PFB is greater than the threshold for an I-loaded successive failure count. Unlike the PASS, the BFS will clear a failure (down arrow disappears) whenever the difference between the CMD and PFB is less than or equal to the threshold.

The BFS outputs a class 2 fault message, R OMS GMBL or L OMS GMBL, and a down arrow by the failed actuator on the MNVR display. Like the PASS, the BFS will reset the associated fail counters and fail flags for a given set of gimbals whenever the crew changes status of the selected channel via keyboard entry.

12.4.4 BFS - PASS Interaction

Since the crew can engage the BFS to take over vehicle control during any part of an OMS burn, the BFS has to have the correct target data loaded. For this purpose, the BFS GPC listens on all display/keyboard (D/K) buses, including the PASS D/K buses.

In the PASS D/K buses, the BFS GPC will process only specific item entries. Those related to the OMS include engine select (BOTH, L, R), trim load (P, LY, RY), TGT (C1, C2, HT, qT, PRPLT, DVX, DVY, DVZ) LOAD and TIMER, BURN ATT (R, P, Y), and channel selected (PRI/SEC/OFF) (ref. sec. 12.7.3 and fig. 12-1). This capability eliminates the necessity to enter these items through both the PASS and the BFS keyboards.

The execute (EXEC) item is also listened to by the BFS which means that an EXEC made to the PASS is recognized by the BFS.

12.5 TVC COMMAND MODES

There are four operational TVC modes, auto or manual with two engines and auto or manual with one engine.

A two-engine auto burn can be performed in the PASS only. The engines are commanded parallel. Roll, pitch, and yaw are controlled by gimbaling the engines. The AUTO DAP push button indicator (pbi) must be engaged and the rotational hand controller (RHC) in detent. Taking the RHC out of detent will downmode the TVC to manual control as will selecting DAP: INRTL. The DAP: FREE and DAP: LVLH push buttons are ignored while the TVC DAP is active. RCS wraparound control is provided when TVC angular rate is insufficient. For an engine failure the TVC will reconfigure to one engine only by putting the ARM/PRESS switch to off.

A two-engine manual burn can be performed in both PASS and BFS and differs from an auto burn in that the actuator CMD's are controlled by the crew via RHC inputs.

Single-engine burns (PASS only) have the thrust vector aligned through the c.g. Pitch and yaw are controlled by gimbaling the engines and roll is controlled by the RCS. Everything else is the same as a two-engine burn for both AUTO/INRTL.

12.6 OMS ENGINE FDI

The OMS ENG FDI algorithm detects and identifies performance below a predetermined value. It uses independent measurements of engine chamber pressure and vehicle velocity to perform the FDI function. Chamber pressure is monitored continuously unless a CF is indicated. Velocity measurements are used after main engine cutoff (MECO) and external tank separation (ET SEP) to determine the number of engines actually firing by comparing the measured velocity increment over 0.96 sec to a threshold value.

An off-failure is defined as an engine providing thrust below a threshold value when it is CMDed on, and an on-failure is when the engine chamber pressure is above a threshold value when it is commanded off.

The output of the OMS engine FDI is a notification to the crew of an on/off failure condition. Reconfiguration is accomplished by crew operation of the OMS ARM, ARM/PRESS switch.

12.7 NOTES

12.7.1 CR's and DR's

CR 39895B OMS GIMBAL DRIVE CHECK. Extends gimbal drive check by a half cycle to detect low slew rate OMS actuator failures. Changes fail counter, 0_A_COUNT, to 24 for same reason.

CR 39620A OMS ACTUATOR FDI UPDATE. Approved for FSW, (20 this CR) modifies the FDI algorithm to function identically on each RM pass (this doubles the fail counter, 0_A_COUNT, to 30 since it was only incremented every other pass before this CR). This CR also deletes unused portions of the requirements previously made nonoperational by make-work patching the PASS software (S/W) only.

CR 59004 OMS ENGINE STOW IN TAL. OMS actuators remain in the stowed (STOW) position and the channels are deselected so that the engines are in STOW for entry.

CR 39897 LIMIT OMS ENGINE MOVEMENT. This CR minimizes the displacement that any single failure can cause on ascent by changing the OMS TVC RM threshold constant from 1° to 0.7° and the fail counter from 3 to 2 counts.

CR 59026 OMS TVC CONTROL. Adds a transport lag and initialization requirement for PFB's to provide single fault tolerance to protect OMS during ascent. The change allows the OMS TVC FDBK SOP to execute before the first execution of the CMD SOP.

DR 47780 (SMS B07456 BFS). Changes code to match requirement that the OMS primary and secondary actuator channels be deselected by FSW in MM 304 or 603 to remove power from the actuators.

CR 79135E OMS BURN IN MM 304. The OMS TVC stow commands are carried across the MM 304 transition. When an OMS burn is required in MM 304 the engines are commanded to the c.g. trim positions prior to initiating the burn and to the stow position upon completion of the burn. The primary and secondary actuator channels are deselected on transition to MM 305.

CR 89339 OMS TVC FAIL FLAG RESET. Allows OMS TVC fail flags to reset when either drive system is selected or reselected. Deletes the PASS requirement to reset the fail flags when both drive systems are powered off. Documentation change only to the FSSR (ref. DR 101412).

CR 92188 OMS TVC No-Back failure Logic Changes. Improves the Ascent OMS TVC RM fault tolerance. Under the old (pre-OI-28) flight software RM design, a single primary position feedback transducer failure could result in driving an OMS TVC into the airstream during the Hi-Q portion of the ascent. Starting with OI-28, 2 of 2 position feedback indications are required to be 0.7 degrees or greater from the commanded stow position before a TVC system will be powered and driven back to the stow position. This CR also monitors commfault indications on the gimbal drive systems to determine if a system is commfaulted. If a system is commfaulted, then only 1 of 1 transducer greater than 0.7 degrees from the commanded stow position is required to power a gimbal system and drive it back to the stow position.

DR 37561 OMS Gimbal Drive Check During OMS Maneuvers. Written against the PASS software to document the fact that an OMS Gimbal Check may be executed at **ANY** time via an ITEM 34 on the MNVR EXEC display. If an OMS Gimbal check were executed (via ITEM 34) during an OMS burn, the gimbal check would execute resulting in excessive propellant use and sloppy control. The current software design allows the ITEM 34 any time the MNVR EXEC SPEC is active, and does not make the ITEM 34 an "ILLEGAL ENTRY" under any circumstances.

DR 55241 Ineffective OMS Actuator ZAP Commands. Written against the PASS flight software to document the fact that whenever an ITEM 28-31 is executed, on the MNVR EXEC display, that the OMS TVC Command SOP is signaled to issue a new channel select command and a "ZAP" command to the selected actuator hardware. However, the channel selection is performed at 1.04 HZ, while the ZAP command is processed at 12.5 HZ. Therefore, there is an 11-out-of-12 chance that the ZAP command will be issued before the newly selected actuator channel is powered up. To successfully issue the ZAP command, simply re-execute the same item entry. This will force a second ZAP command to be reissued once the desired gimbal system is selected. Procedurally, this item entry is executed twice by the crew.

DR 63979 Written against the possible false failure annunciation of the OMS secondary actuator system after a hardover failure on the primary system. The user response is to ignore the annunciation and reselect the currently selected secondary system to clear the false failure.

DR 101412 Written against the PASS OMS RM software because an actuator fail flag can be reset when the currently selected gimbal drive system is reselected (ref. DR 63979). This disagrees with the flight system software requirements (FSSR) which state that the fail flag is reset only when the gimbal drive system status changes; i.e., alternate system selected or both systems powered off.

12.7.2 Bias and Scale Factor Compensation

Both the CMD SOP and the FDBK SOP have unique bias and scale factor terms by which they compensate their outputs.

CMD's are first provided to the OMS RM principal function and are then compensated by multiplying the selected CMD by a scale factor in volts dc/deg and then adding a bias in V dc. The MDM then outputs this value to OMS TVC and from there it goes to the actuator. The output position CMD's consist of two redundant compensated CMD's, one each for primary and secondary (two pitch and two yaw for each engine).

FDBK positions are compensated similarly, multiplying by a scale factor and adding a bias, except that now the positions are converted from volts dc back to degrees.

The *crew* reads *uncompensated* CMD's and positions on the MNVR display which come from the DAP and the *ground* reads the *compensated* values coming from the MDM's.

The scale factor converts the CMD's and positions into units usable by the hardware/software and the bias is a software provision to compensate for mechanical offsets.

12.7.3 Burn Target Data

Burn target data are located on the MNVR display, figure 12-1.

C1, C2 : Flight path angle coefficients, C1 in feet/second, C2 a dimensionless value
HT : Target radius in nautical miles
θT : Target central angle in degrees
PRPLT : Total propellant in pounds to be used by the deorbit maneuver

$\Delta VX, \Delta VY, \Delta VZ$: Body axis components of $\Delta VTOT$ (ft/s) in the local vertical,
local horizontal (LVLH) frame at ignition.

12.7.4 Drawings

Drawings included in this brief are two system software drawings OMS TVC RM and OMS TVC CMD/FDBK SOP (dwgs. 12-1 and 12-2) reflecting the information in the brief. Included also is OMS/TVC signal flow and drivers hardware drawing (dwg. 12-3).

12.8 NOTES OF INTEREST

One note of interest (NOI) was written on the OMS TVC. This NOI, OMS Burns With Fixed TVC, is included in this brief.

DF NOI #201

REFER TO: DF6/84-79 (STS-ALL) 11-16-84

To: Distribution

From: DF6/Frank DiGenova

Subject: OMS Burns With Fixed TVC

BACKGROUND

On August 10, 1984, a special SAIL test was run which consisted of ten OMS engine burns (TRANS DAP OPS 3) and a plus X RCS burn to obtain propellant usage data to compare nominal burns with off-nominal burns with the engine thrust vector fixed through the CG. Similar tests were run using the SSFS and the SMS back in the early part of 1983.

The purpose of these tests was to determine if, from a propellant stand point, it is feasible to consider burning an OMS engine if both its primary and secondary TVC system had failed, but the engine thrust vector has been "parked" through the CG and its TVC power turned off.

DISCUSSION

Figure 1 is a summary of the eleven burns and the propellant usage and cutoff errors. The first four burns (1, 1A, 2 and 2A) are nominal OMS engine burns with the thrust vectors through the CG as opposed to nominal parallel burns. This was done to obtain reference data to compare with the OMS burns 3, 3A, 4, 4A, 5 and 6 which had one or both engines burning with one or both thrust vectors fixed through the CG (TVC POWER-OFF). In all cases where an AUTO burn was made, when compared to its comparable MANUAL burn, the AUTO burn was better as would be expected so the remainder of this discussion will be limited only to the AUTO burns. There was no attempt to optimize pilot technique in the MANUAL burns and the pilots were instructed to attempt to "fly" the error needles keeping them near zero.

Case 2 (L-ENG CG NOM TVC AUTO) can properly be compared with case 3 (2-ENG CG with Left TVC-OFF, AUTO) since in a situation where an OMS engine is considered failed because of bad primary and secondary TVC systems but can be "parked" through the CG, the options would be to burn on only 1-ENG or 2-ENG with the bad TVC engine with fixed thrust vector. In this comparison, the case 3 uses 116 lbs more OMS propellant and 32 lbs of RCS and when case 3 is compared with a nominal 2-ENG BURN (case 1), the additional propellant is only 26 lbs more OMS and 32 lbs RCS. These differences are negligible when consideration is given to the additional redundancy such that either engine can fail and a burn can be completed with minimum impact. Note that case 5 (1-ENG CG with TVC OFF, AUTO) used 110 lbs more RCS but 111 lbs less OMS when compared with case 1 (nominal 2 ENG). It should

however, be pointed out that the assumption is made that the OMS TVC NO-BACK is parked close to the CG. See the Lockheed memorandum LEMSCO-19176 dated April 1, 1983, from M. W. Martin/B12 to E. T. KUBIAK/EH2 "Trans DAP OMS TVC study for off-nominal configurations," for additional information and a discussion of performance penalty as a function of pitch and yaw engine mistrims.

It should also be noted that the on-board MANEUVER display OMS ENG POSN data is not updated when TVC power is off but, the primary and secondary TLM is updated and valid at all times.

CONCLUSION

It is considered feasible to burn an OMS engine with a fixed thrust vector that is aligned through the CG from a propellant standpoint based upon analysis of the SAIL runs discussed in this NOI and the SSFS/SMS runs that were previously run.

RECOMMENDATIONS

Whenever in a situation where an otherwise good OMS engine is considered NO-GO for burns because of a failed or suspect TVC system but it is capable of being driven to its CG position, that procedure should be attempted as soon as possible. If the CG alignment operation is successful, the engine can then be considered usable for burns if there is confidence in the NO-BACK device that it will hold the engine in-place during the burn with TVC POWER-OFF. This confidence in the NO-BACK device is determined by an analysis of the OMS gimbal position data during the ascent phase to determine if there was any slippage. The past usage is also considered since an actuator with high previous usage has a higher probability of slippage than a relatively new actuator.

Frank DiGenova

DF6/FDiGenova:sls:11/13/84:5451
Reserved

| NO | CASE | PROPELLANT USES (LBS) | | | CUTOFF ERRORS (FT/SEC) | | |
|----|---------------------------|--------------------------|------|--------------|---------------------------|----------------|----------------|
| | | RCS | OMS | OMS + RCS | V _X | V _Y | V _Z |
| 1 | 2-ENG CG NOM TVC AUTO | 0 | 6026 | 6026 | +0.15 | +0.03 | +0.01 |
| 1A | 2-ENG CG NOM TVC MAN | 22 | 6026 | 6048 | +0.20 | +0.11 | -0.14 |
| 2 | L-ENG CG NOM TVC AUTO | 6 | 5904 | 5910 | -0.01 | +0.05 | +0.01 |
| 2A | L-ENG CG NOM TVC MAN | 79 | 5903 | 5982 | +0.04 | +0.23 | +0.04 |
| 3 | 2-ENG CG L-TVC OFF AUTO | 32 | 6020 | 6052 | +0.10 | -0.06 | +0.24 |
| 3A | 2-ENG CG L/R TVC OFF AUTO | 54 | 6058 | 6112 | +0.20 | +2.45 | +0.06 |
| 4 | 2-ENG CG L-TVC OFF MAN | 55 | 6016 | 6071 | +0.07 | -0.19 | +0.31 |
| 4A | 2-ENG CG L/R TVC OFF MAN | 304 | 6068 | 6372 | -1.19 | -0.95 | +2.60 |
| 5 | L-ENG CG TVC OFF AUTO | 110 | 5915 | 6025 | -1.16 | +0.10 | +2.38 |
| 6 | L-ENG CG TVC OFF MAN | 309 | 5941 | 6250 | -0.03 | +1.03 | -1.63 |
| 7 | 4-Jet + X RCS | 789 | 7745 | 8534 | -0.64 | -0.20 | -0.00 |

Figure 1- SAIL Test- OMS Burns with Fixed TVC

Figure 2- MASS PROPERTY AND BURN DATA

Vehicle WT (LBS) = 211025

CG X = 1099.1

CG Y = .3

CG Z = 377.0

IXX = 919770.8

IYY = 7064490.6

IZZ = 7349877.4

PXY = -7690.2

PXZ = 225017.9

PYZ = 389.4

PEG 7 Burn Targets

X = + 277

Y = - 1

Z = 66

ΔV Total = 284.7

TGO 2-ENG = 2:33

TGO 1-ENG = 5:07

TGO 4-JET = X RCS = 8:43

STS-41D (16) OI-4 FLT S/W Final

SECTION 13 CONTROLLERS

13.1 GENERAL SYSTEM DESCRIPTION

13.1.1 Introduction

Manual control of the vehicle is provided via the various controllers. The three rotational hand controllers (RHC's) are used to provide manual roll, pitch, and yaw inputs to the flight control system (FCS). The two translation hand controllers (THC's) provide manual, lateral, longitudinal, and vertical translation inputs to the FCS. The two rudder pedal transducer assemblies (RPTA's) generate an electric signal which is proportional to the displacement of the rudder pedal. This electric signal is used to control the position of the rudder during the final stages of the entry flight phase. The two speedbrake thrust controllers (SBTC's) are used to control the vertical stabilizer-mounted speedbrake during the entry flight phase. Additionally, the pilot's SBTC can be used for manual throttle control of the Space Shuttle main engines (SSME's) during the ascent phase. Data and information for this system brief were obtained from the Functional Subsystem Software Requirements (FSSR), Controller Workbook 2102, and software drawings.

13.1.2 Rotational Hand Controller

The two RHC's, located at the pilot's and the commander's stations, are usable during all mission phases (ascent, entry, and on-orbit). The RHC located at the aft station is usable only during the on-orbit flight phase. The RHC controls the Orbiter roll, pitch, and yaw movement during ascent by rate command/attitude hold through solid rocket boosters (SRB's) and main engine thrust vector control (TVC), and with the orbital maneuvering system (OMS) and the reaction control system (RCS). The RHC also controls movement on-orbit with OMS and RCS, and during entry with RCS and aerodynamic control surfaces.

13.1.3 Translation Hand Controller

The two THC's are located at the commander's station and the aft station. The THC is used for manual lateral, longitudinal, and vertical translation inputs to the FCS. The commander's THC can be used for the orbit insertion, on-orbit, and deorbit flight phases. The aft station THC can be used only during the on-orbit flight phase.

13.1.4 Rudder Pedal Transducer Assembly

The two RPTA's on the Orbiter are located at the commander's station and the pilot's station. The RPTA generates an electrical output signal which is proportional to the rudder pedal displacement. This signal controls the position of the rudder during the final stages of the entry flight phase.

13.1.5 Speedbrake Thrust Controller

The two SBTC's on the Orbiter, located at the commander's and pilot's stations, are used to control the position of the vertical stabilizer-mounted speedbrake during the entry flight phase. Additionally, the pilot's SBTC can be used for manual throttle control of the SSME's during the ascent phase.

13.1.6 Location of Controllers

The commander's station has an RHC, THC, SBTC, and a set of rudder pedals connected to an RPTA. The RHC is offset approximately 19° in yaw from the vehicle body axis and can be adjusted 5 in. in the vertical axis and 20° about the pitch axis. The rudder pedals can be adjusted 3.25 in. forward or aft from the neutral position. The pilot's station has no THC. Otherwise, the same equipment is available to the pilot. The aft station has only an RHC and a THC. Figures 13-1 and 13-2 provide views of both the forward and aft stations, highlighting controller location.

13.1.7 Power and Distribution

Each cockpit station has an associated device driver unit (DDU) which contains three power supplies - A through C. The A,B,C power supplies furnish power to the controllers at the commander's, pilot's, and aft stations as indicated in figures 13-3, 13-4, and 13-5. Each DDU is redundantly powered by two main power buses; therefore, the loss of one main power bus still leaves the DDU powered. Main power buses A and B power the commander's (left) DDU; main power buses B and C power the pilot's (right) DDU; and main power buses A and C power the aft station DDU. The DDU power supply then powers the associated controller with power supply A powering channel A of the controller, power supply B powering channel B, and power supply C powering channel C. The three channels in each controller are redundant, and the controller can operate with only one channel powered.

The power consumption for each DDU when Flight Controller power is ON is 44 watts and when Flight Controller power is OFF is 10 watts.

The flight controller switches control the ON/OFF state of the power supplies by providing an initialization of the excitation voltage, to turn the power supply on, and disables the excitation voltage, to turn the power supply off. There are failure modes associated with the power supply control switches such that with a "fail open" in the ON contact circuit, the power supply may still come on when the Flight Controller switch is turned on. In addition, if the OFF contact circuit is "failed open", the power supply will probably remain ON when the Flight Controller switch is turned off.

13.1.8 Flight controller Power Switch Transients

Before upgrading to the new DDUs, when the FLT CNTLR PWR switches were turned on/off, the THC/RHC, respectively, could generate spurious inputs to the flight control system. These transient inputs would appear to the flight control system as brief deflections of the various controllers. In the past, these transients have downmoded the DAP to manual, and fired jets unnecessarily. With the new DDUs there is a staggered power ON/OFF feature which prohibits transient outputs on more than one channel at a time. This will prohibit inadvertent jet firings in all DAP modes. There are some important points to consider:

1. The Staggered Power ON/OFF sequence only provides improved protection from transient if at least 2 channels/contacts are available. (See 13.1.8.3 Workarounds for Controller Transients for further information).
2. The Staggered Power ON logic is expected to work regardless of whether flight control power is turned ON before or after the DDU receives power.

The Staggered Power OFF logic will ONLY work if flight control power is turned OFF before removing power from the DDU. (See 13.1.8.3 Workarounds for Controller Transients for further information).

13.1.8.1 RHC Transients

The spurious RHC inputs were measured (in SAIL, February 1981) as approximately 450 millivolts, lasting approximately 50 milliseconds. The flight software interprets these voltages as RHC deflections of 1.2 degrees in yaw and 2.1 degrees in pitch and roll. These were outside the original detent limits for the Orbit and Trans DAPs. These transient inputs were interpreted as valid manual commands. The DAP downmoded to manual, and jets fired.

The Orbit DAP detent limits were changed (OFTP # XXX) to 2.5 degrees in yaw and 4.3 degrees in pitch and roll, putting the transient well inside the detent limits. This has eliminated the downmode problem in the Orbit DAP. Unfortunately, the detent limits cannot be similarly increase in the Trans DAP. The RHC must be sensitive enough to fly the deorbit burn in manual thrust vector control mode, in case this mode is required for deorbit. Increasing the detent limits reduces the sensitivity of the RHC to a level unacceptable to the crew.

BFS engage resulting from RHC transients: the same DDU power supplies for the RHC power the three "Engage" sense/driver circuits housed in the RHC. If the FLT CNTLR PWR is cycled several times in 3.5 seconds, then the instability of the operational amplifiers increases and may generate transient pulse outputs greater than +2.83 Vdc. The output from the RHC engage operational amplifier is sent to a differential amplifier in the Hand Controller Engage Driver (HCED), (see Figure 13.1a for a diagram of the HCED/DDU/RHC interface). This differential amplifier is used to reduce the noise effects by maintaining a low output until the input voltage is greater than +2.83 Vdc (which is required to latch the three engage circuits in the BFC. When this voltage level is exceeded, the amplifier's output becomes the +15 Vdc from the DDU power supply. Once the HCED recognizes the "Engage" discrete, the BFC will send an "Engage" discrete to all of the other BFC's. Each of the five BFC's are connected to separate GPC's, therefore all five GPC's will detect that an "Engage" discrete has been set true and will engage the BFS.

13.1.8.2 THC Transients

The THC generates spurious inputs to the flight control system when the Flight Controller power is turned on. These inputs do not downmode the DAP, but they do fire jets unnecessarily. Before OI-8D, the workaround for the Orbit DAP was to set the DAP to VERNIER mode before switching the Flight Controller power on. In VERNIER mode, the pre OI-8D the DAP ignored translation commands. With the Hotstick downmoding (OI-8D CR # XXXX), when in VERNIER mode, the orbit DAP will upmode to primary for the duration of THC out of detent. This can result in unnecessary primary jet firings.

13.1.8.3 Workarounds for Controller Transients

The current solution implemented to avoid spurious controller outputs making it to flight control is the new DDUs with their staggered powered ON/OFF feature. For Orbit DAP situations where we do not have at least 2 contacts available and spurious jet firings could be detrimental to payload hardware or data, switch RM can be disabled by a SPEC 25 ITEM 16 EXEC immediately before powering on the Flight Controllers and reenabling switch RM with another SPEC 25 ITEM 16 EXEC (toggle) immediately afterward. During the switch RM disable period all cockpit switch changes are ignored by the flight software, therefore it is important that the reenable occur before any other switch changes are made. Before the new DDUs were incorporated switch RM was disabled with a SPEC 25 ITEM 16 anytime flight control power was turned ON/OFF and the vehicle was in configuration where jet firing could be dangerous (i.e. docked ops).

There are two cases that are a couple failures deep that should be considered.

1. Down to One Contact AND Inadvertent Jet Firings is a Concern:

As stated before, the staggered power ON/OFF feature only provides improved protection from transient if at least 2 channels/contacts are available. Therefore, if needed, SPEC 25 ITEM 16 EXEC can be used if down to the 1 level on contacts. This should only be needed if we are in a configuration where an inadvertent jet firing is a concern (i.e. EVA, Mated to ISS, Sensitive payload on the RMS, etc.) AND down to one contact.

2. Flight Control Power and DDU Circuit Breakers Turned OFF Out of Sequence:

As stated before, the staggered power OFF logic will ONLY work if flight control power is turned OFF BEFORE removing power from the DDU. If powering down occurs out of sequence then the staggered power down logic does not execute and power is removed from all three channels of the associated controllers at once. Since the detent limit has been changed in the Orbit DAP, this is considered a minor concern.

In OPS 1 & 3 this can cause inadvertent jet firings and the DAP to downmode from Auto to Inertial. But in order for this to be a concern in OPS 1 or 3 all three of the following have to occur: Shuttle must be in OPS 1 or 3, there must be a configuration where an inadvertent jet firing is a concern, and the crew must pull the DDU breakers with the Flight Control Power switch ON.

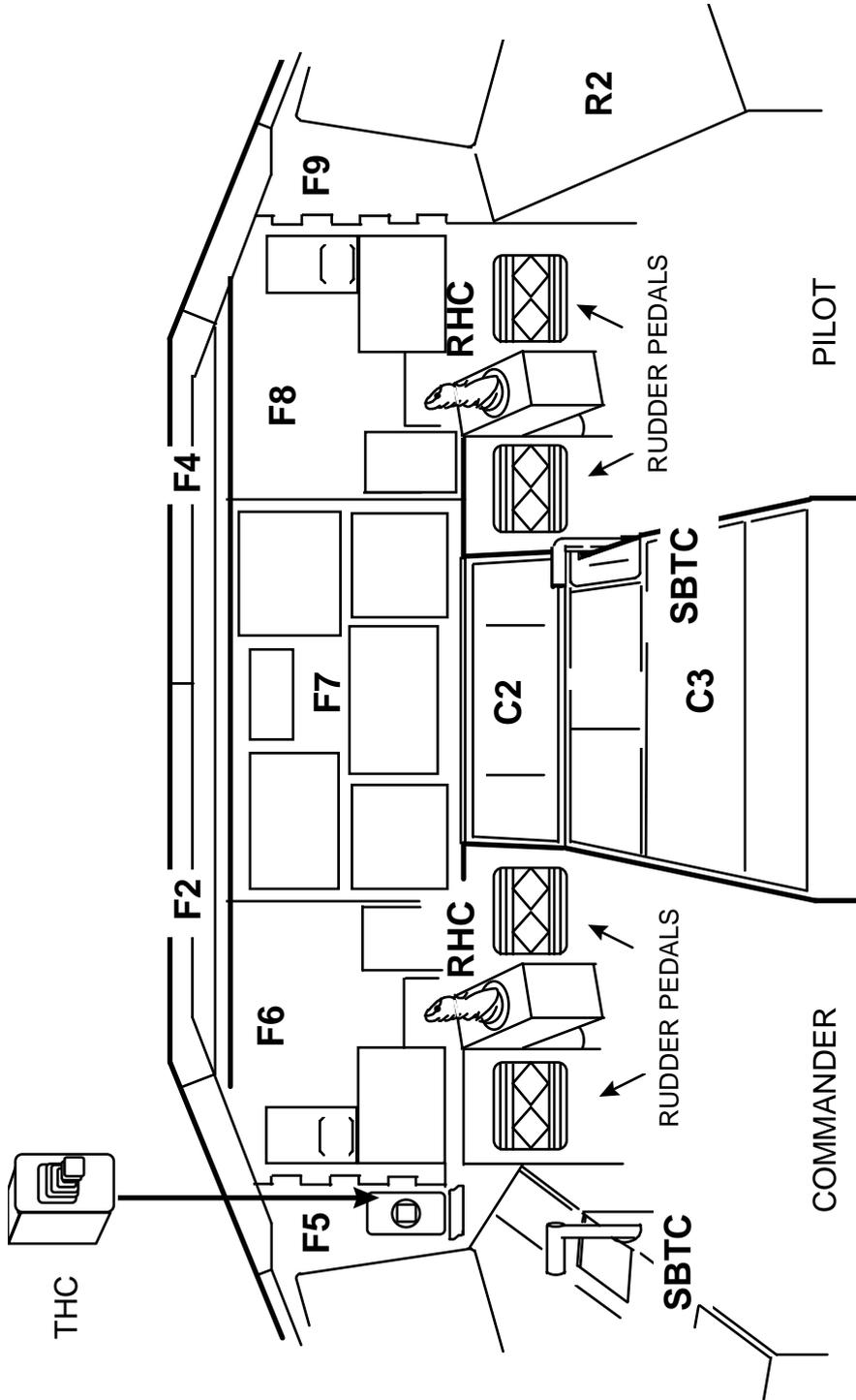


Figure 13 - 1. - Forward cockpit controllers.

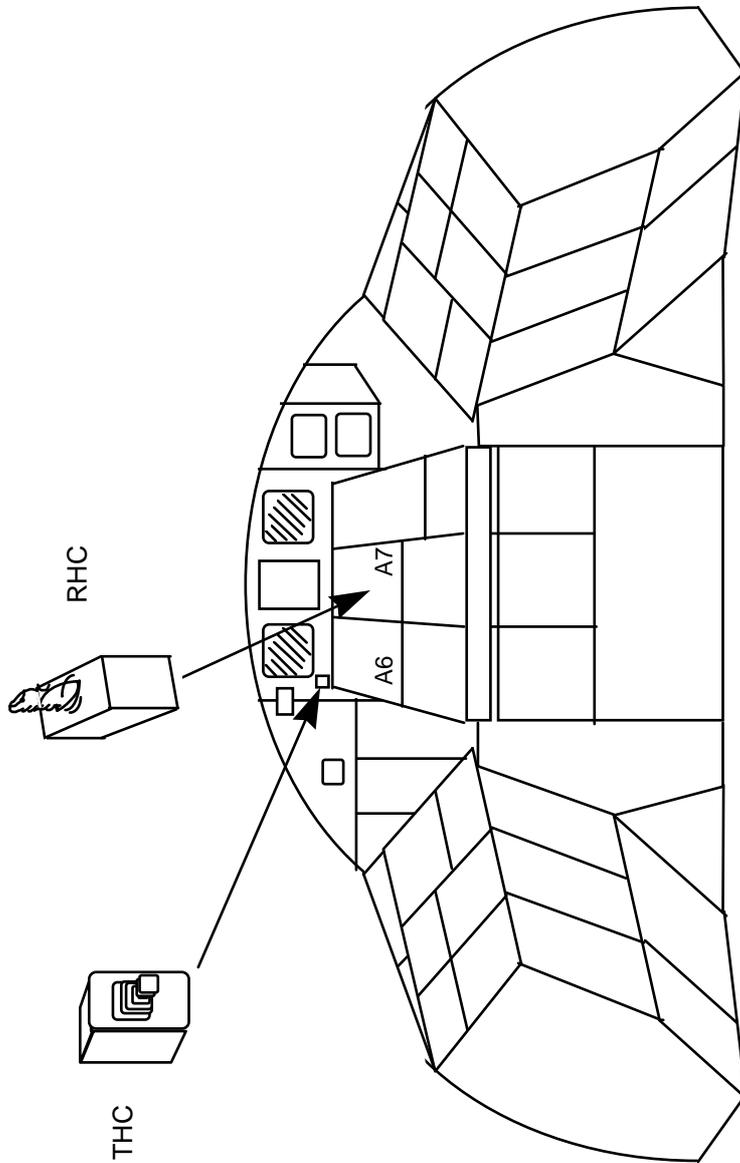


Figure 13 - 2 - Aft crew station controllers.

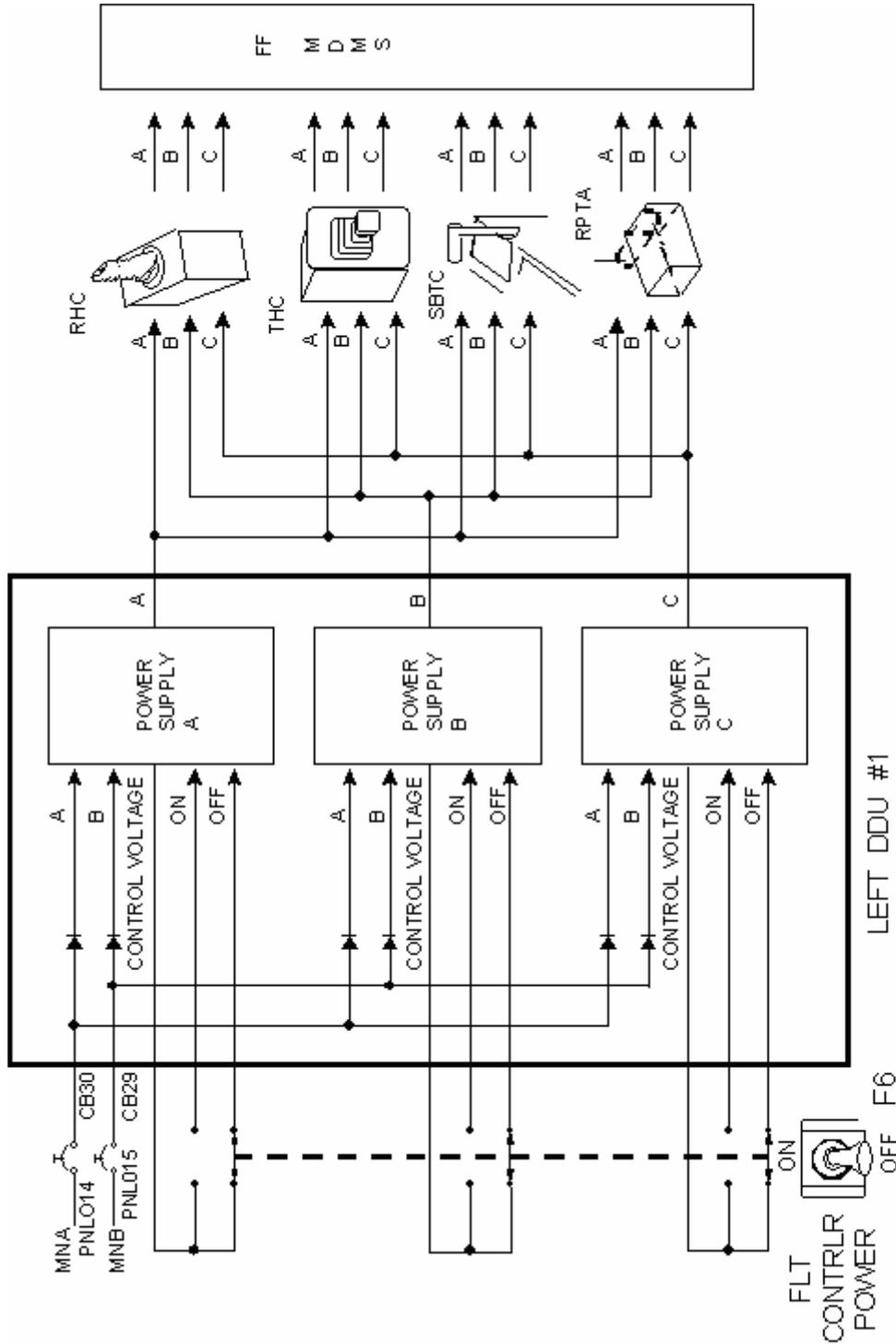


Figure 13 - 3. - Flight controller power distribution - commander.

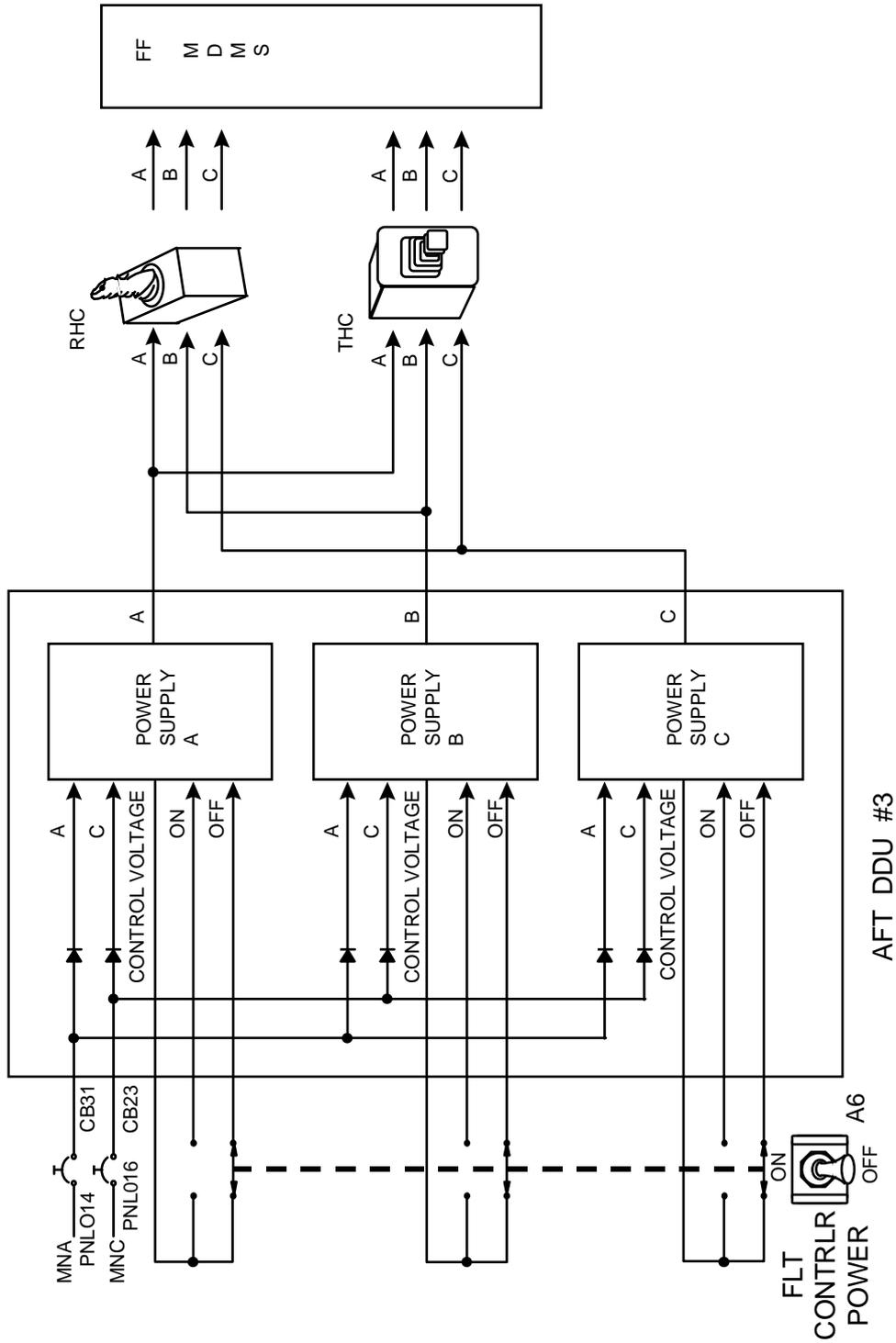


Figure 13 - 5. - Flight controller power distribution - aft station.

13.2 ROTATIONAL HAND CONTROLLER

13.2.1 Hardware Description and Operations

13.2.1.1 Physical Characteristics

The RHC has a backup flight system (BFS) ENGAGE switch, a TRIM switch, and a communications (COMM) switch (fig. 13-6).

The BFS ENGAGE switch is used to engage the BFS and is a momentary thumb-operated switch. The BFS engage requires the A, B, and C dc power to work. The BFS ENGAGE switch is inactive in the aft station.

The TRIM switch is used to introduce a constant rate of Orbiter movement about the body pitch or roll axis. (If an increased rate is desired, the switch is actuated again in the same direction. The Orbiter movement is proportional to the length of time the switch is actuated.) If a rate decrease is desired, or if it is desired to totally remove the rate, the TRIM switch must be actuated in the opposite direction. The TRIM switch at the aft station RHC is inactive.

The COMM switch is a push-to-talk switch. When the switch is depressed, the radio transmitters are enabled. The transmitters remain in operation as long as the switch is depressed.

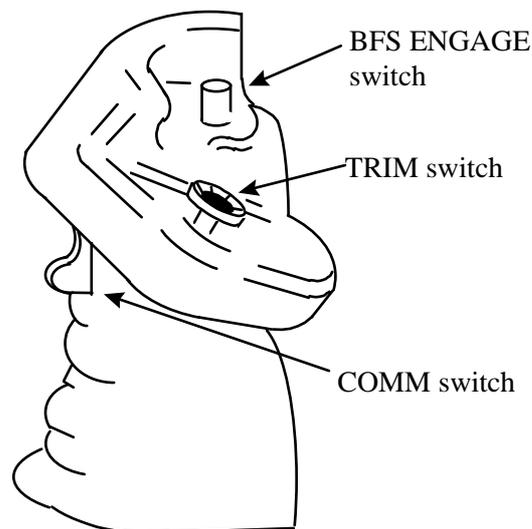


Figure 13 - 6. - The RHC switches.

Each RHC is redundantly powered by two main power buses which supply power to a DDU. Each main power bus has an associated circuit breaker. The output of the DDU is controlled by a flight controller power switch. Each DDU contains three power supplies, and each power supply powers one RHC transducer channel for each of the pitch, roll, and yaw axes. The pitch, roll, and yaw output signals for each transducer channel of one RHC are sent over the same flight forward MDM (except for Aft Roll B (FF4 vs. FF2) due to NWS channelization (see 13.2.2.6 RHC channelization)).

13.2.1.2 Availability

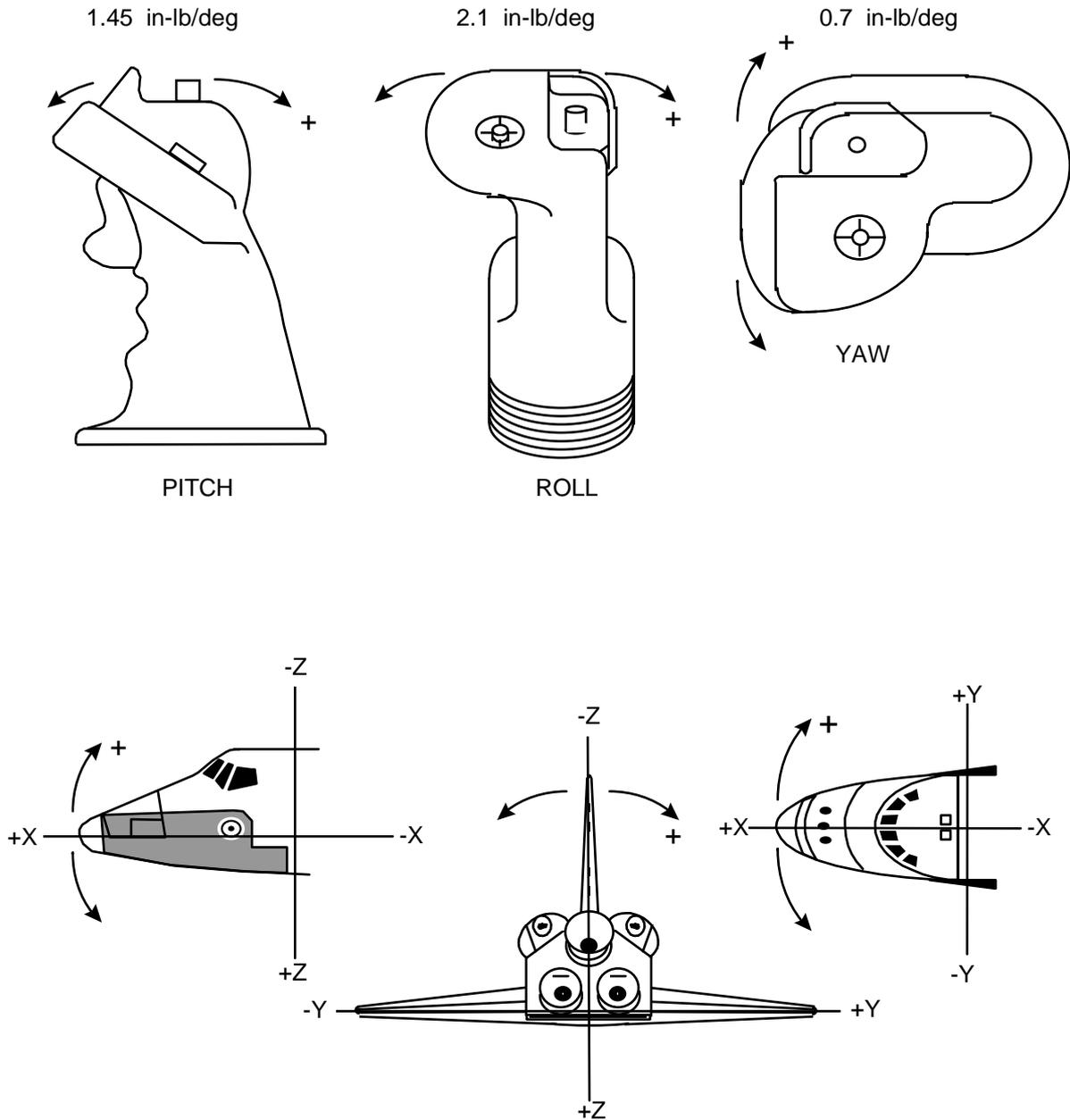
The commander's and pilot's RHC's are available to supply manual inputs or signals to the FCS throughout the mission except during major mode (MM) 101. During ascent operations sequence (OPS 1), both RHC's are available for MM's 102, 103, 104, 105, and 106. On orbit, the RHC is available for MM's 201 and 202 plus MM 801 for checkout. During entry (OPS 3 and 6), both RHC's are available for MM's 301, 302, 303, 304, 305, 601, 602, and 603. The aft RHC is only available for MM's 201, 202, and 801.

13.2.1.3 Mechanical Limits

The RHC mechanical hard stop occurs at $\pm 24.3^\circ$ in the pitch and roll axes and at $\pm 14.3^\circ$ in the yaw axis. The mechanical hard stop is the maximum deflection that can be obtained in an axis. The mechanical soft stop for the RHC occurs between $\pm 19^\circ$ and $\pm 20^\circ$ for the roll and pitch axes and between $\pm 9.5^\circ$ and $\pm 10.5^\circ$ for the yaw axis. The mechanical soft stop is a built-in resistance that occurs prior to the hard stop in an axis.

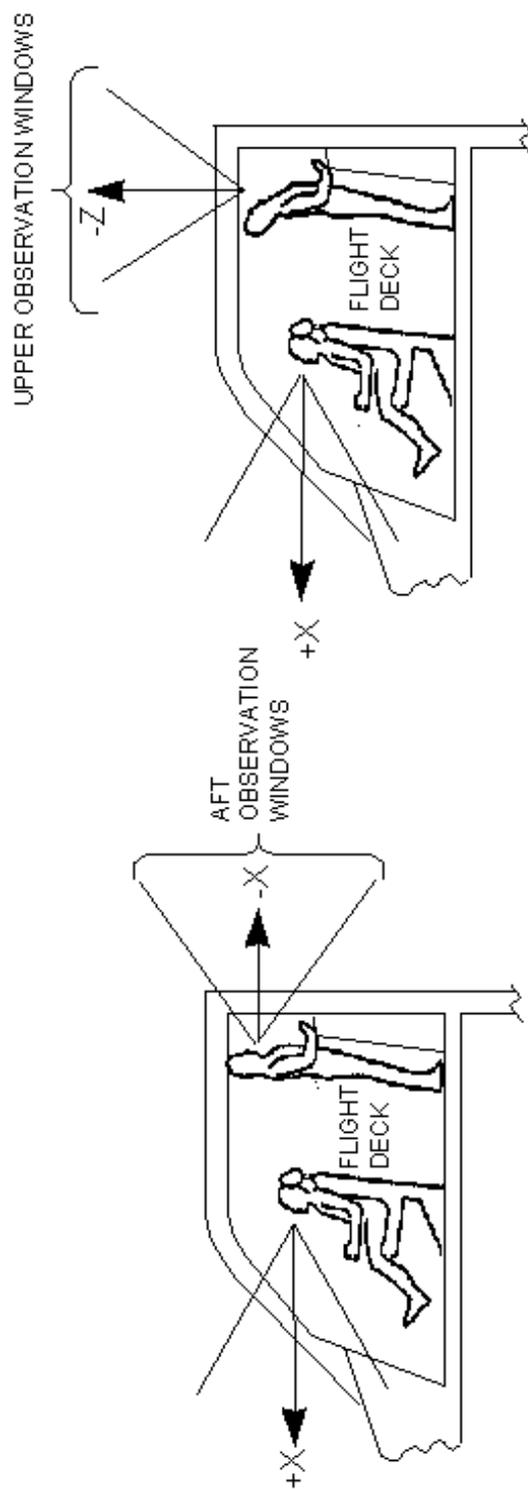
13.2.1.4 RHC References and Inputs

The commander's and pilot's RHC's provide inputs to the FCS based upon a +X reference system. Figure 13-7 provides a pictorial view of forward station RHC inputs and the relationship to the +X reference. The aft station RHC uses an entirely different reference system. The aft station has two sets of windows so that the crew can look along the -X or -Z axis of the Orbiter. Since there is only one RHC available in the aft station, the aft RHC reference is determined by the position of the aft SENSE switch located on panel A6. Figure 13-8 provides a view of the forward and aft station references. The -Z position of the switch selects the line of sight (LOS) along the -Z vehicle axis, while the -X position selects the line of sight LOS along the -X vehicle axis. Figures 13-9 and 13-10 provide a pictorial representation of vehicle movement through the aft RHC for both the -X and -Z SENSE switch positions.



ALL RHC COMMANDS TO THE FLIGHT CONTROL SYSTEM ARE SENT IN THIS REFERENCE.
TORQUE TO DEFLECT TO THE SOFTSTOP IS ALSO SHOWN.

Figure 13 - 7. - Forward RHC use - commander and pilot.



- TWO SETS OF WINDOWS - AFT (-X) AND OVERHEAD (-Z)
- ONE SET OF AFT STATION CONTROLS
- MUST ESTABLISH A VISUAL CONTROL REFERENCE BASED ON THE LINE OF SIGHT
- DONE WITH THE AFT SENSE SWITCH

Figure 13 - 8. - Aft station control.

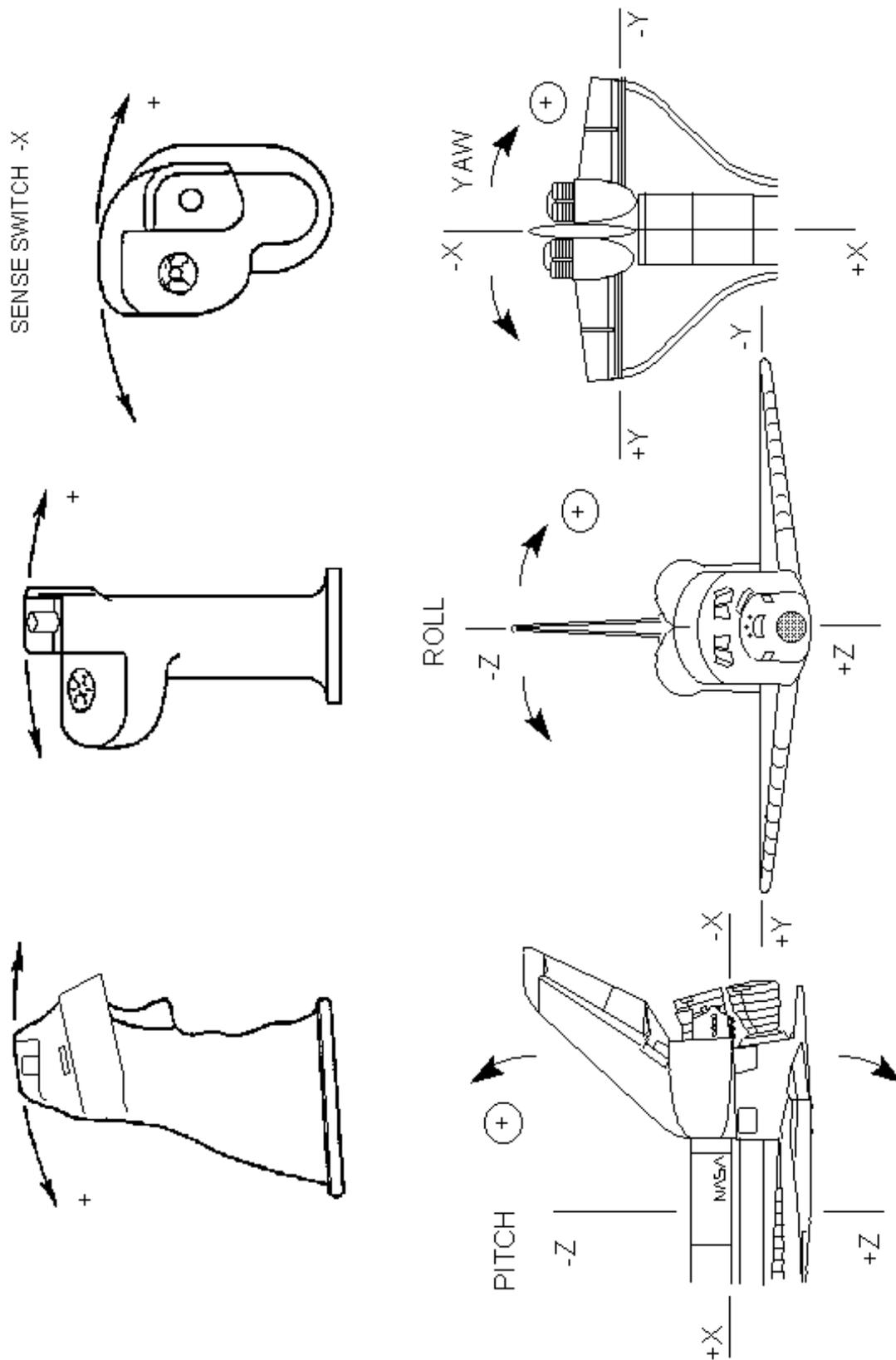


Figure 13 - 9. - Aft line-of-sight reference - aft RHC use.

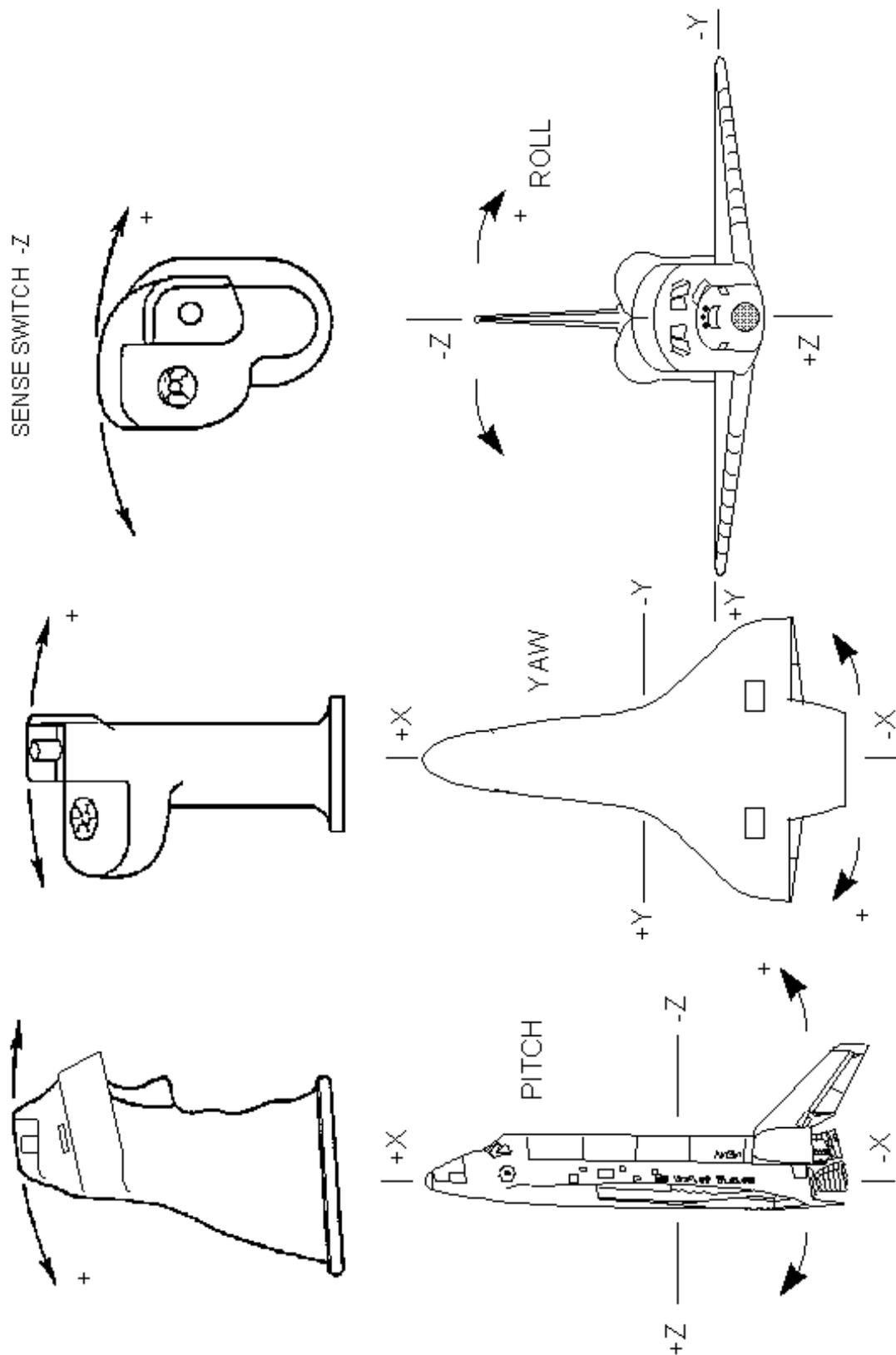


Figure 13 - 10. - Overhead line-of-site reference - aft RHC use.

13.2.2 Software Description and Operations

13.2.2.1 RHC Software Programs

The RHC software consists of three distinct programs. First, there is the three-axis RHC software available in OPS 102, 103, 104, 105, 106, 601, 301, 302, and 303. The second program is the three-axis orbital RHC software. The final program is the two-axis RHC software available in OPS 602, 603, 304, and 305. The first three-axis RHC software program uses inputs from the commander and pilot's RHC for roll, pitch, and yaw. The orbital three-axis RHC program uses the roll, pitch, and yaw inputs from all three RHC's. The two-axis RHC software utilizes only roll and pitch inputs from the commander and pilot's RHC plus trim inputs from the panel and RHC TRIM switches.

13.2.2.2 Three-Axis RHC

Triple redundant roll, pitch, and yaw signals are processed from both the commander's and pilot's RHC. Figure 13-11 shows the flow of the redundant signals going through redundancy management (RM) processing, through the three-axis subsystem operating program (SOP), to various outputs.

13.2.2.2.1 Three-axis RHC RM.- Three level RM during OPS 1 and 6 only runs selection filtering (SF) of the data. There is no fault detection at this level (CR89684C will make FDIR available in OPS 1 and 6 for OI 2O). The selection filtering is downmoded only by COMM faults (CF's) to the system. Also, in OPS 6 there are no yaw inputs into RM. The value selected by the SF uses the midvalue select (MVS) scheme.

In OPS modes 301, 302, and 303, the three-axis RHC RM is the same as the two-axis RM scheme in OPS 3; therefore, three-axis RHC RM for modes 301, 302, and 303 will be discussed in the section on two-axis RHC RM.

At the two level (assumes CF or fl/des on one RHC input), the SF will average the two remaining RHC inputs by axis. If the CF goes away, the SF will upmode to the three level. A data good (DG) test is implemented which consists of the two inputs for each axis being subtracted and compared to an I-load value. If the limits are exceeded, then the DG information is set "bad" and zeros are sent to the flight control for the affected RHC axis.

At the one level (combination of 2 CF/fl/des), there is no SF processing. The available data are sent directly to the SOP. The DG discrete is always set "good." Upmoding can occur if a CF is cleared.

At the zero level, the DG discrete is set "bad" and the SOP sends zeros to the FCS. Again, upmoding can occur, if a CF is cleared.

13.2.2.2.2 Three-axis RHC SOP.- Three-axis RHC SOP performs six major functions. Figure 13-12 is a block diagram of the SOP. The six major functions of the SOP for the three-axis RHC SOP are as follows. From the

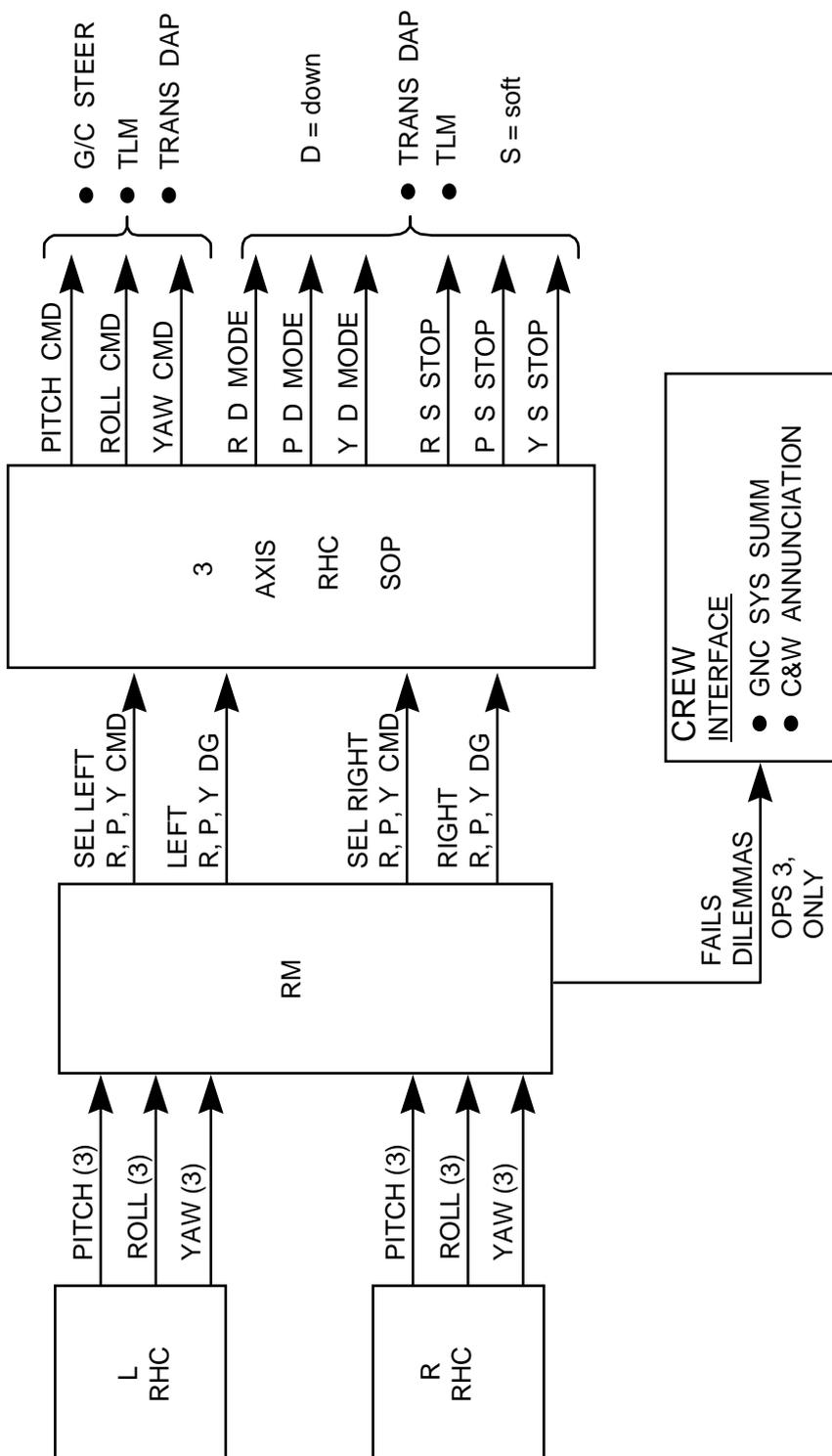


Figure 13 - 11. - RHC software data flow, three-axis program (OPS 1, 3, 6).

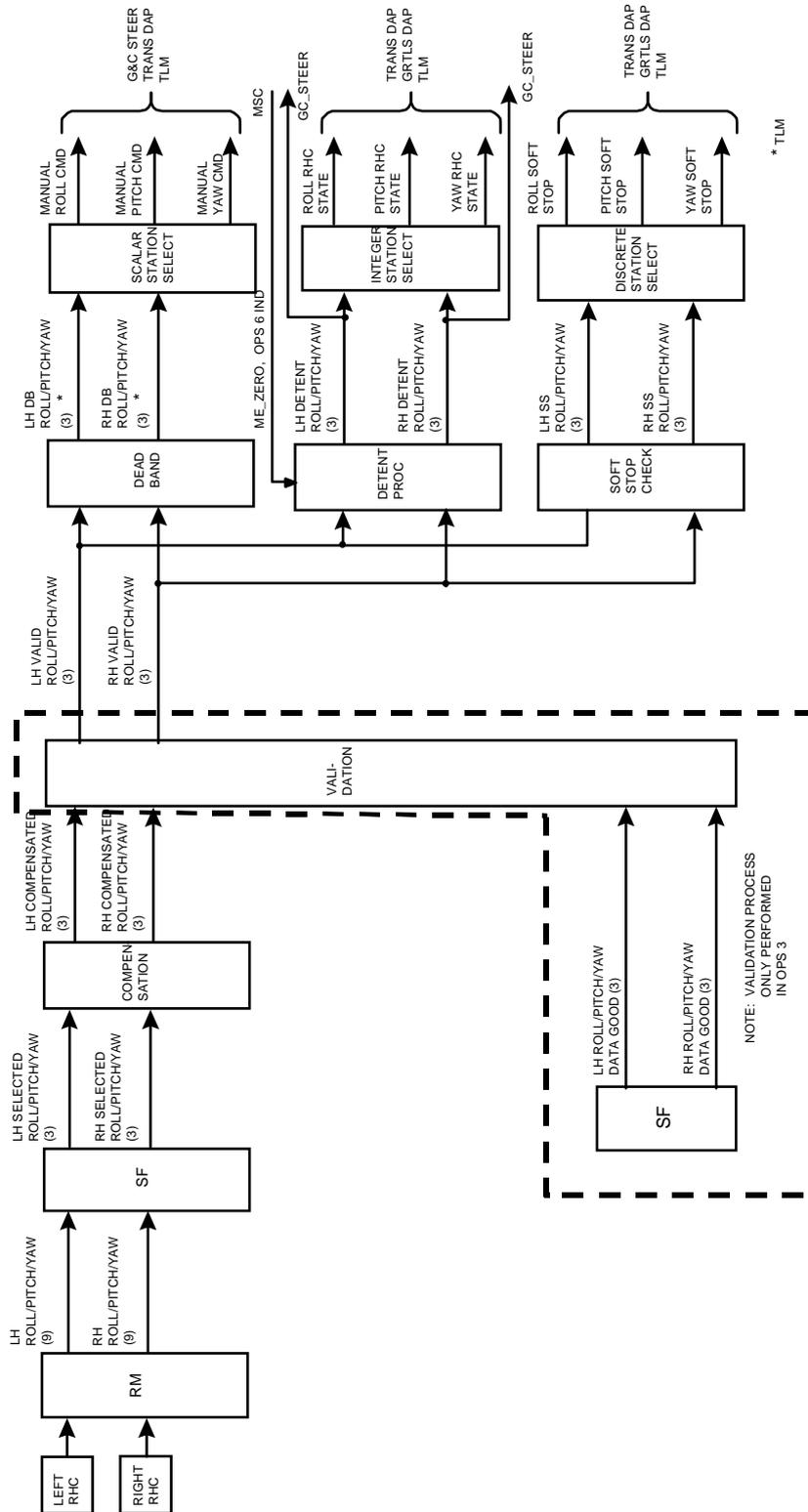


Figure 13 - 12. - Three-axis RHC SOP functional block diagram.

SF, the selected roll, pitch, and yaw transducer commands are compensated and scale factors and biases are applied. Voltages are changed to degrees. The RHC commands are then validated, resulting in DG flags. For each axis, the output command is the algebraic sum of the left and right RHC deadbanded scalar signals. Detent processing next uses hysteresis to generate a zero or plus or minus one output from the validated RHC signal. Soft-stop checks are then performed again using hysteresis on the validated RHC signals. Once again, soft-stop discretized (plus one or zero) are generated for processing to the DAP system. Breakout deadbanding is performed on the validated axis command.

This processing is performed on demand in MM's 102 and 103 until the main engines are at zero thrust, and it is also performed in 601. The three-axis RHC SOP will be processed whenever the FCS mode LH pitch is control stick steering (CSS) (manual mode; LCSS_PLAMP = 1); otherwise, the three-axis RHC SOP will not be performed in auto mode. Once the main engines are at zero thrust, the three-axis RHC SOP will be continuously processed at 12.5 HZ in MM's 103, 104, 105, 106, 301, 302, and 303. In other words, hot stick downmoding from auto to manual is not available in powered flight. Manual takeover must be requested via the FCS pushbutton indicators (PBI's) on the eyebrow panel.

13.2.2.3 Orbit Three-Axis RHC

Triple redundant pitch, yaw, and roll signals come from the forward RHC's (commander and pilot) and the aft station RHC when the orbital three-axis RHC software is processing. Redundant signals are processed through RM processing and then go to the SOP. The aft SENSE switch indication is used in both RM and SOP processing for all aft RHC signals. Failures and dilemma conditions are processed in RM and presented to the crew. Deselect/reselect capability of RHC signals can be performed by the crew through the keyboard during OPS 2 and 8. Figure 13-13 provides a pictorial view of the RHC software.

13.2.2.3.1 Orbit three-axis RHC RM.- Included in the RM scheme are the inputs for aft station RHC and RM for the SENSE switch. With these inclusions, the RM scheme is the same as the three-axis RHC RM as explained in 13.2.2.2.1. While in OPS 2, status indications of the RHC's can be viewed on the crew display entitled RM ORBIT; and, while in OPS 8, the same indications are available to the crew on CONTROLLERS.

13.2.2.3.2 Orbit three-axis RHC SOP.- The Orbit 3 Axis RHC SOP performs seven major functions. Six of these functions are the same as performed for the three-axis RHC SOP in section 13.2.2.2.2. The last function concerns the transformation of all aft station RHC signals as a function of the SENSE switch prior to the selection of the RHC station. The deadbanded signals from the left and right RHCs and the transformed signal from the aft RHC are algebraically summed by axis to generate the manual RHC output commands.. Figure 13-14 is a block diagram of this SOP's function.

It is generally necessary to exceed the upper softstop hysteresis limit in order to get a hardware commanded continuous jet firing, however, this SOP will fire the appropriate jets if the RHC is moved from the softstop state in one direction to within the softstop hysteresis in the opposite direction quickly enough so that the SOP does not detect having passed through detent (an apparent "jump" from the softstop state in one direction to within the other hysteresis). In this case it is not necessary to exceed the softstop hysteresis upper limit.

13.2.2.3.3 Ascent/Entry 3 Axis RHC SOP.- This SOP generally performs the same functions as the Orbit 3 Axis RHC SOP. There are, however, two differences: 1) the aft RHC is disabled and 2) a "jump" from the softstop state in one direction to within the hysteresis in the other direction results in no jets firing. This SOP requires the RHC to be deflected past the upper hysteresis limit in all cases to get the desired jet firing. The different handling of the "jump" between the two 3 Axis SOPs is of no practical significance to the piloting task because the pilot is unlikely to be able to detect the condition or to be able to

deliberately command it, and in any case, if the desired jet firing or rate change is not achieved the normal (and correct) response is to increase the RHC deflection breaking the upper hysteresis limit thereby causing the proper jet firing. The "jump" condition is mentioned here only to explain the associated code difference between the two 3 Axis RHC SOPs. As in the Orbit 3 Axis RHC SOP Left and Right RHC outputs (all available LRUs) are summed.

13.2.2.4 Two-Axis RHC

In MM's 304 and 305, triple redundant pitch and roll signals come from the forward RHC's (commander and pilot). These signals go through RM processing, then through SOP processing, eventually being passed to the DAP. Figure 13-15 provides a pictorial view of the two-axis RHC software.

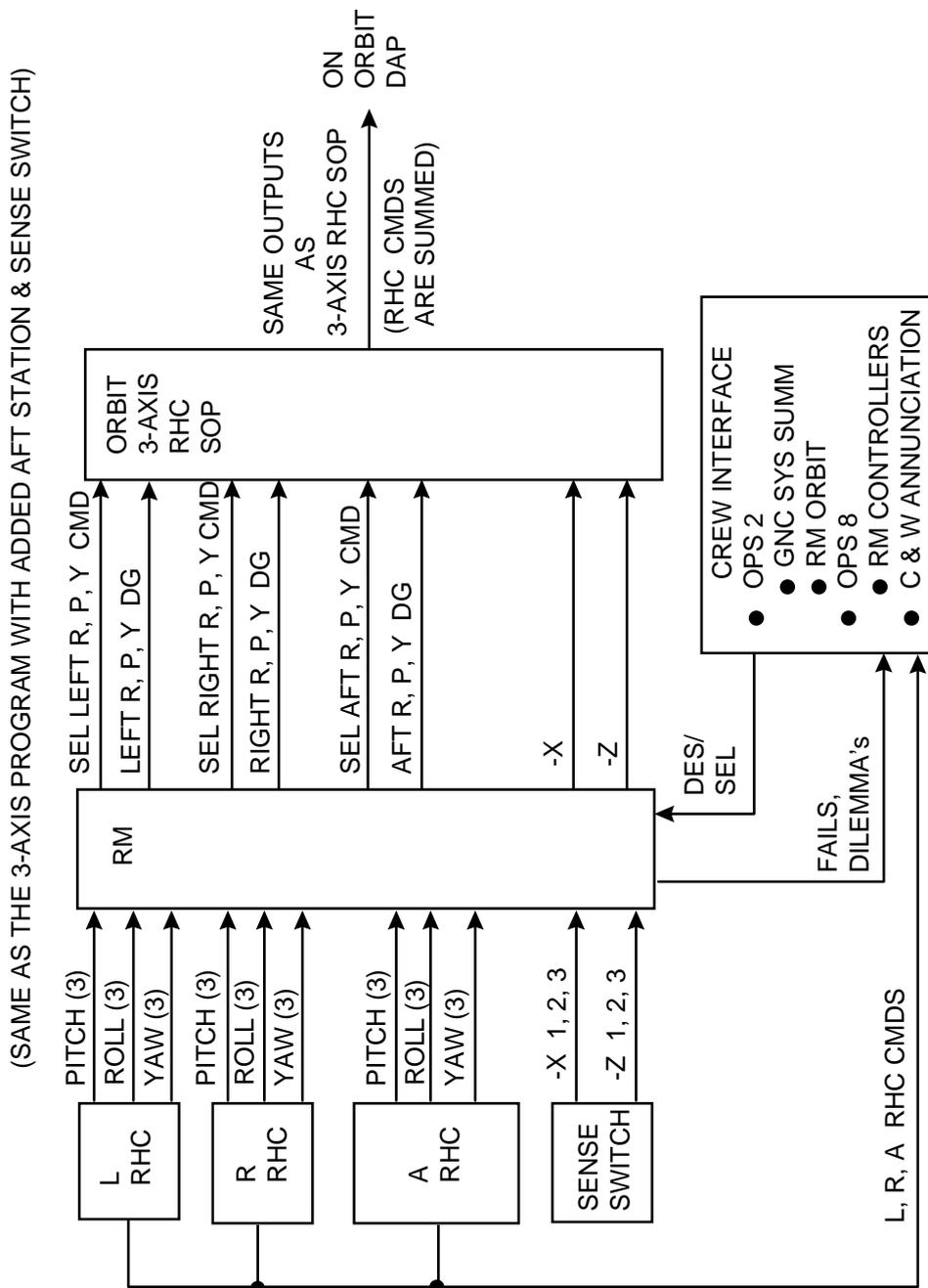


Figure 13 -13. - RHC software data flow, orbit three-axis program (OPS 2 and OPS 8).

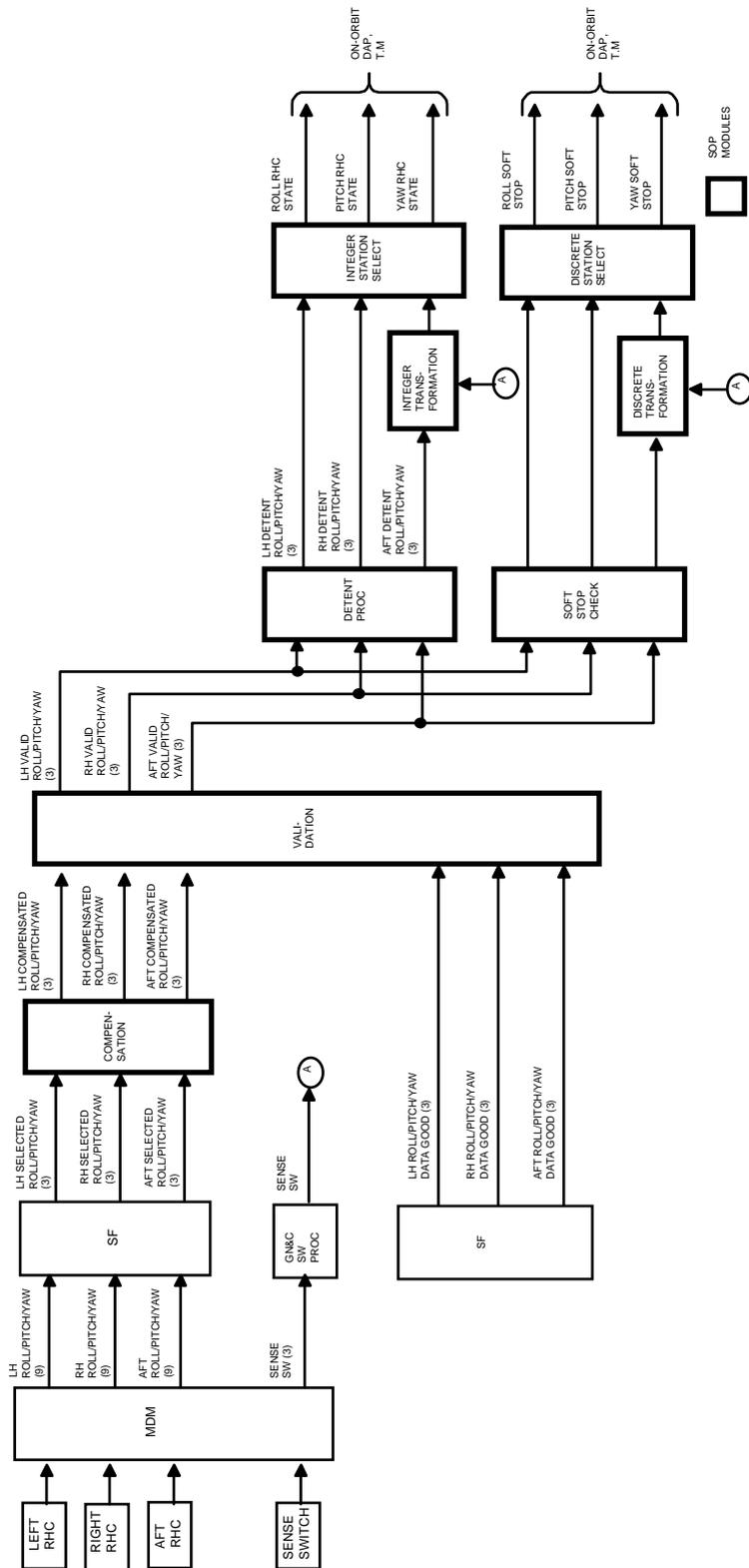
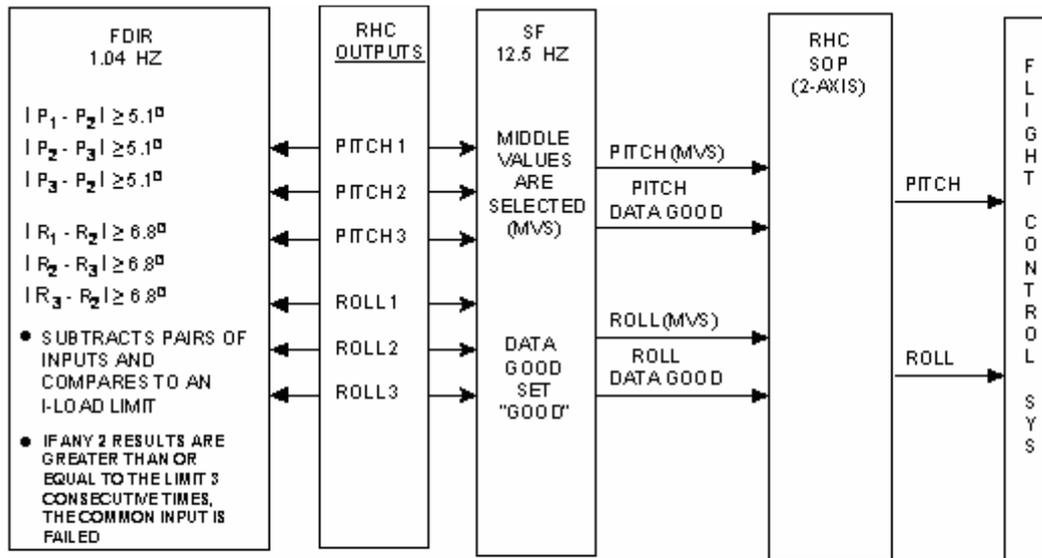


Figure 13 - 14. - Orbit three-axis RHC SOP functional block diagram.



P = PITCH R = ROLL MVS = MIDDLE VALUE SELECT

Figure 13 - 15. - RHC RM, two-axis control OPS 3 (MM 304, 305) nominal case - three-level RM.

13.2.2.4.1 Two-axis RHC RM.- Three level RM during OPS modes 304, 305, 602, and 603 uses MVS criteria for the SF function from the three pitch and roll signals for each RHC. FDIR is active by subtracting pairs of pitch and roll signals and then comparing the results to an I-loaded value. The I-loaded value for pitch is 5.1°; roll is 6.8°. If there is a miscompare three consecutive times, the common input is identified as failed.

If the first failure is a result of a data disagreement, the SF averages the two remaining axes individually. The DG test is implemented, which provides a subtracted comparison against an I-loaded value (pitch +5.1° (21.25% MCC), roll +6.8° (28.33% MCC)). Once deselected, a channel cannot be reselected in OPS 3 or 6. Fault detection, isolation, and reconfiguration (FDIR) deselects the failed command from both the SF and FDIR and also annunciates the failure to the crew. FDIR processing rate increases from 1.04 Hz to 3.125 Hz.

If the first failure is a result of a CF, the SF and FDIR are downmoded by the flight computer operating system (FCOS). The same action occurs in the SF, as mentioned in the previous paragraph. The SF and FDIR will upmode to the three level if the CF goes away. FDIR subtracts the two inputs for each axis and compares the result with the I-load limit. If a result exceeds the limit three consecutive times, a dilemma is declared.

The second failure resulting from a disagreement between the two remaining pitch or roll signals causes a dilemma condition to occur. The DG flag is set "bad" when the limit is exceeded, which causes the SOP to send zeros to the FCS for all RHC commands. Therefore, the RHC has lost its ability to control the vehicle. The FDIR sends annunciation to the crew, but this annunciation is after the DG test is set "bad."

If the second failure is caused by a CF, the SF is downmoded to the one level for both axes and passes the good data directly to the SOP. The FDIR is also downmoded to the one level, but there is no FDIR processing at the one level. If the CF goes away, both the SF and the FDIR upmode to the two level.

A third failure resulting from a bad command results in loss of control for that axis. No FDIR processing exists nor is the third failure annunciated to the crew. If the third failure results from a CF, then the SF and FDIR are downmoded to the zero level. Zeros are sent to the FCS.

13.2.2.4.2 Two-axis RHC SOP.- Two-axis RHC SOP performs six major functions. Figure 13-16 is a block diagram of the SOP. Scale factors and biases are applied to the selected roll and pitch command, changing voltages to degrees. Next the breakout deadbanding is processed for both the roll and pitch commands. Then there is processing of the discrete indications of RHC pitch deflections beyond soft stop. RHC deflections are then processed to determine AUTO or CSS operation. Processing of deadband signals for both crew stations produces a resultant output command (station select). The output command is the sum of the left and right RHC select. Finally, there is station select processing for the roll and pitch panel and RHC TRIM switches. Roll soft stop does not exist because roll and pitch transducer commands are compensated by the application of scale factors and biases. Furthermore, when a compensated RHC deflection in an axis exceeds a specified threshold, a discrete shall be set to true. Otherwise, the discrete shall be set to false. These discretetes are for use by the AERO_JET DAP in downmoding from AUTO to CSS. The RHC outputs must be processed at a rate of at least 12.5 Hz in MM's 304 and 305, except for the final phases of approach and landing (A/L) when the required rate is 25 Hz.

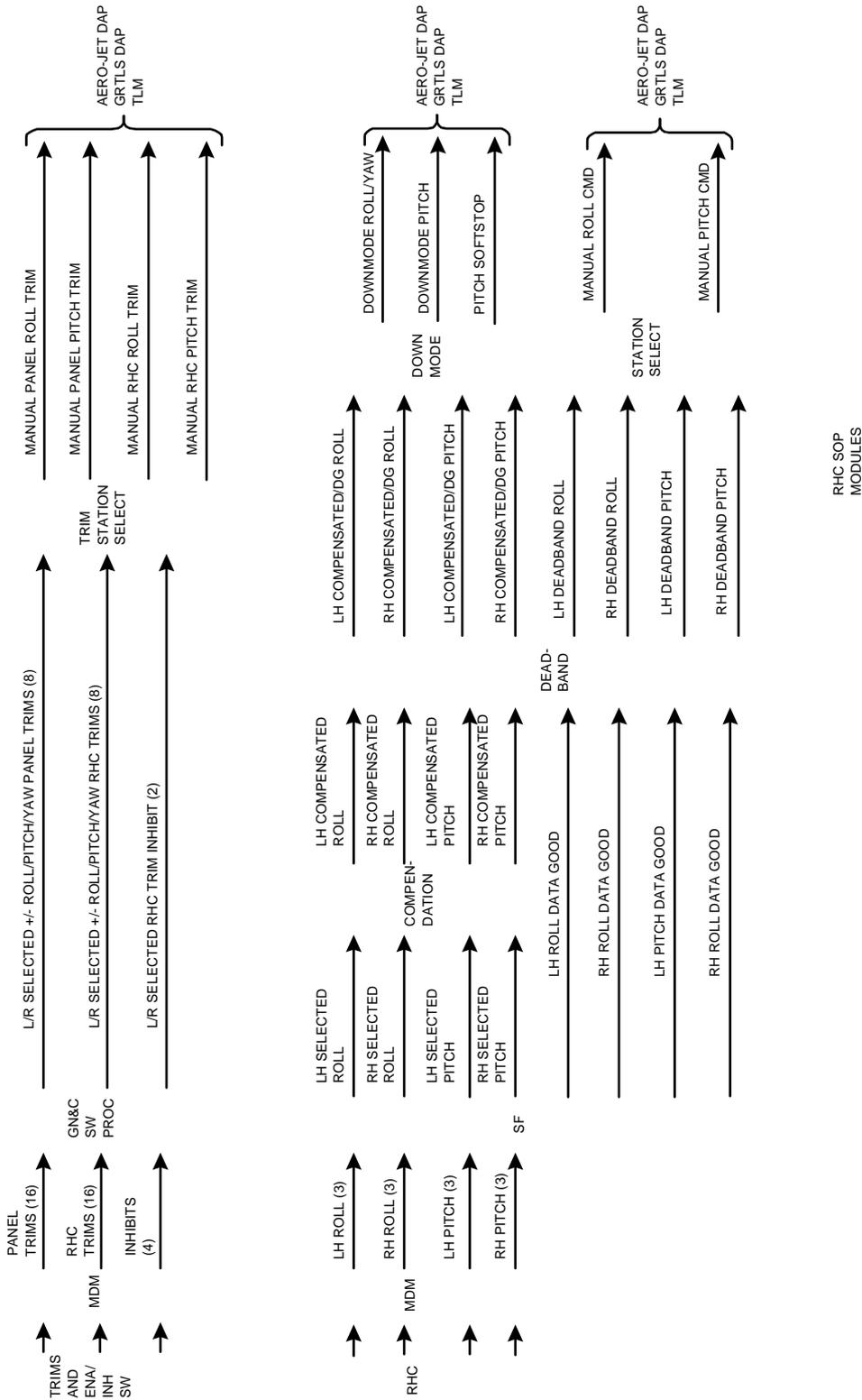


Figure 13 - 16. - Two-axis RHC SOP functional block diagram.

13.2.2.4.3 Beep Trim Derotation Logic

The RHC Beep Trim switch is used to derotate the orbiter after the main gear is on the ground and the orbiter is in a 2-point stance. The RHC Beep Trim switch provides a means of derotating the vehicle in a smooth and consistent manner. The improved derotation characteristics associated with using the RHC Beep Trim switch helps reduce the peak loads on the main gear and the slap down rate on the nose gear. The RHC Beep Trim switch's physical characteristics are described in section 13.2.1.1.

PASS (OI-28) software uses the Weight On Wheels (WOW) flag as the "Enable" for Beep Trim derotation. The WOW flag is triggered when the main landing gear touches down. Before WOW is set, the RHC/PNL-ENABLE/INHIBIT switch on F3 must be in the ENABLE position for a Beep Trim command to be honored. The RHC/PNL-ENABLE/INHIBIT switch acts as the "Enable" for both RHC Beep Trim and Panel Trim switches. There are two RHC/PNL-ENABLE/INHIBIT switches on F3, one for the commander's trim switches and one for the pilot's trim switches. Once WOW is set, PASS software ignores the position of the RHC/PNL-ENABLE/INHIBIT switch and RHC Beep Trim derotation commands are accepted and processed. This allows the RHC/PNL-ENABLE/INHIBIT switch to remain in the INHIBIT position throughout entry, while preserving the use of Beep Trim Derotation. Since the RHC Beep Trim is a two contact switch, leaving the RHC/PNL-ENABLE/INHIBIT switch in the INHIBIT position provides essentially two fault tolerance while the vehicle is in the air.

After the WOW flag is set, PASS software ignores the position of the RHC/PNL-ENABLE/INHIBIT switch and honors negative pitch commands from the left or right RHC Beep Trim. On the first cycle in which the WOW flag is set, the PASS software looks for negative pitch RHC Trim contacts which have failed "high" or "on" (See Figure 3-17). There are two contacts per RHC in each of the following directions: positive pitch, negative pitch, positive roll and negative roll. On the first cycle in which the WOW flag is set, negative pitch contacts which indicate "1" are set and latched to "0". If both negative pitch contacts on one RHC indicate "1", only one is set and latched to "0". All roll and positive pitch contacts are also set and latched to "0" at this time. Commfaulted contacts are also set to "0" by the software. This logic is processed for both the Left and Right RHC.

The negative pitch "Derotate" command is honored if both contacts on either RHC are set high after WOW is set. Also, the "Derotate" command is honored if one negative pitch contact on either RHC is set high after WOW and below an Equivalent Air Speed (EAS) of 195 kts.

The logic, as described above and in Figure 13-17, allows for protection from contact failures while the vehicle is in the air. After the WOW flag is set, there are several possible failure scenarios which can require a "Manual" derotation, "Auto" derotation at 195 kts, or "Uncommanded" derotation. Most of these cases are fairly obscure due to the short window of exposure and none of them result in Loss Of Control (LOC).

Manual Derotation Cases:

The logic will not recognize contacts that have failed "low". For this reason, if both contacts fail low or if one is commfaulted and the other fails low, a "Manual" derotation using the RHC is required. If one contact fails high before WOW is set and the other contact is commfaulted or failed low, a "Manual" derotation will also be required. This is due to the fact that the failed high contact will be set and latched to "0" at WOW, leaving essentially no negative pitch contacts on the affected RHC. If Beep Trim is enabled while the vehicle is in the air this case would degrade control and possibly result in an LOC. If only one of the two contacts fails low or is commfaulted and the other contact is NOT commfaulted, the software will not honor the derotation command until the vehicle is below 195 kts and WOW has been set.

Auto Derotation Cases:

For the case were two contacts have failed high before the WOW flag is set, the vehicle will derotate on its own or "Auto" derotate after WOW is set and the EAS is 195 kts or less. This occurs because only one

of the two failed high contacts is set and latched to "0". At or below 195 kts, with WOW set and one of the two contacts failed high and the other latched to "0", the software initiates the derotate command. If Beep Trim is enabled while the vehicle is in the air this case will result in degraded control and a possible LOC.

Uncommanded Derotation Cases:

More obscure cases which can cause "Uncommanded" derotations include the failure of two contacts high after WOW or the failure of one contact high after WOW and below 195 kts.

The BFS does not use the WOW flag as an "Enable" for Beep Trim derotation. In order for RHC Beep Trim derotation to be available in the BFS, the RHC/PNL- ENABLE/INHIBIT switch must be in the ENABLE position. The BFS only processes inputs from the commander's RHC Beep Trim switch. Inputs on the pilot's side are ignored.

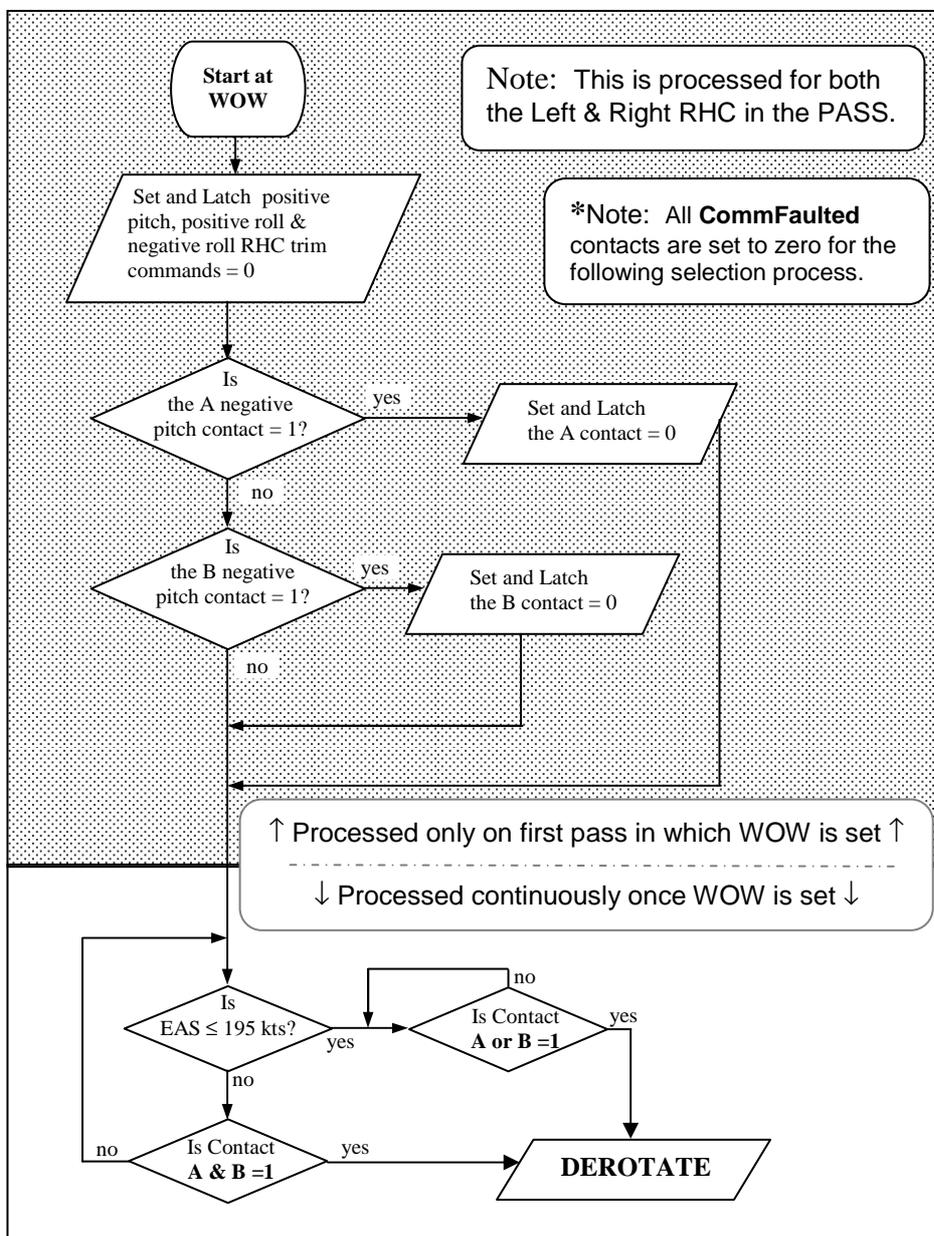


Figure 13-17 PASS (OI-28) Beep Trim Derotation logic.

13.2.2.5. RHC BFS

The BFS RHC processing differs from the primary avionics software system (PASS) as follows:

- a. BFS does not validate the RHC inputs by using a DG indication. No CMFLT, FDIR, or sel/des. Selected signals use an MVS scheme only.
- b. BFS does not support RHC processing in ascent. Only auto flight control is available until main engine cutoff (MECO), then only CSS is available (MM 103-6, 301-5, 602-3).
- c. FWD only.

13.2.2.6 RHC Channelization

The left, right, and aft RHC transducer outputs are channelized through the flight forward (FF) multiplexers/demultiplexers (MDM's) as follows:

- a. Left RHC A, B, and C commands (roll, pitch, yaw axes) go through FF1, FF2, and FF3, respectively. All commands pass through card 01. Roll outputs pass through channel 00, pitch through channel 01, and yaw through channel 02.
- b. Right RHC A, B, and C commands (roll, pitch, yaw axes) go through FF2, FF3, and FF4, respectively. All commands pass through card 14. Roll outputs pass through channel 00, pitch through channel 01, and yaw through channel 02.
- c. AFT RHC A, B, and C commands (roll, pitch, yaw axes) go through FF1, FF2, and FF3, respectively except for the channel B roll command. Roll channel B goes through FF4. All commands pass through card 01. Roll outputs pass through channel 05, pitch through channel 06, and yaw through channel 07.

13.3 TRANSLATIONAL HAND CONTROLLER

13.3.1 Hardware Description and Operation

13.3.1.1 Physical Characteristics

The THC is utilized to perform translation movements of the vehicle by utilizing propulsion systems associated with the movement of the controller. The THC has three-axis movement respective to itself: up-down (vertical), left-right (lateral), and in-out (longitudinal). See figure 13-18 for a pictorial view of the THC. The movement of the THC in any axis is a maximum of half an inch from the center position. As previously stated, there is a THC on the commander's side forward and a THC in the aft station. The THC is the only controller that uses dc power exclusively.

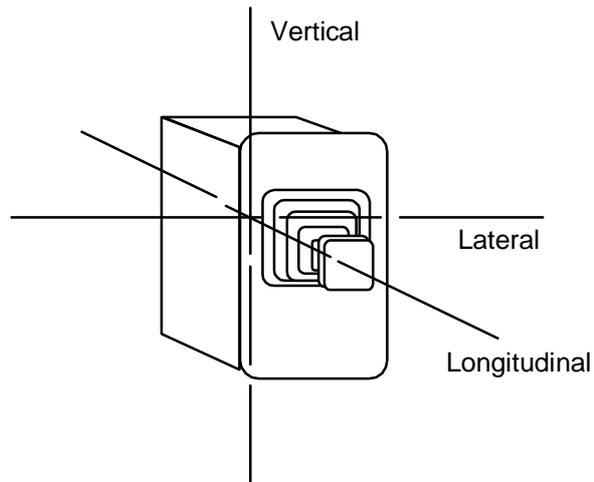


Figure 13 - 18. - Axes of THC control.

13.3.1.2 Availability

The left (commander) THC is available for use in OPS MM's 103, 104, 105, 106, 201, 202, 801, 301, 302, 303, and 601. The aft station THC is available only in 201, 202, and 801.

13.3.1.3 THC References and Inputs

The commander's THC provides inputs to the FCS based upon a +X (LOS) reference system. Figure 13-19 provides a view of the forward THC inputs and their relationship to the +X reference. As with the

ALL THC COMMANDS TO THE
FLIGHT CONTROL SYSTEM
ARE SENT IN THIS REFERENCE.

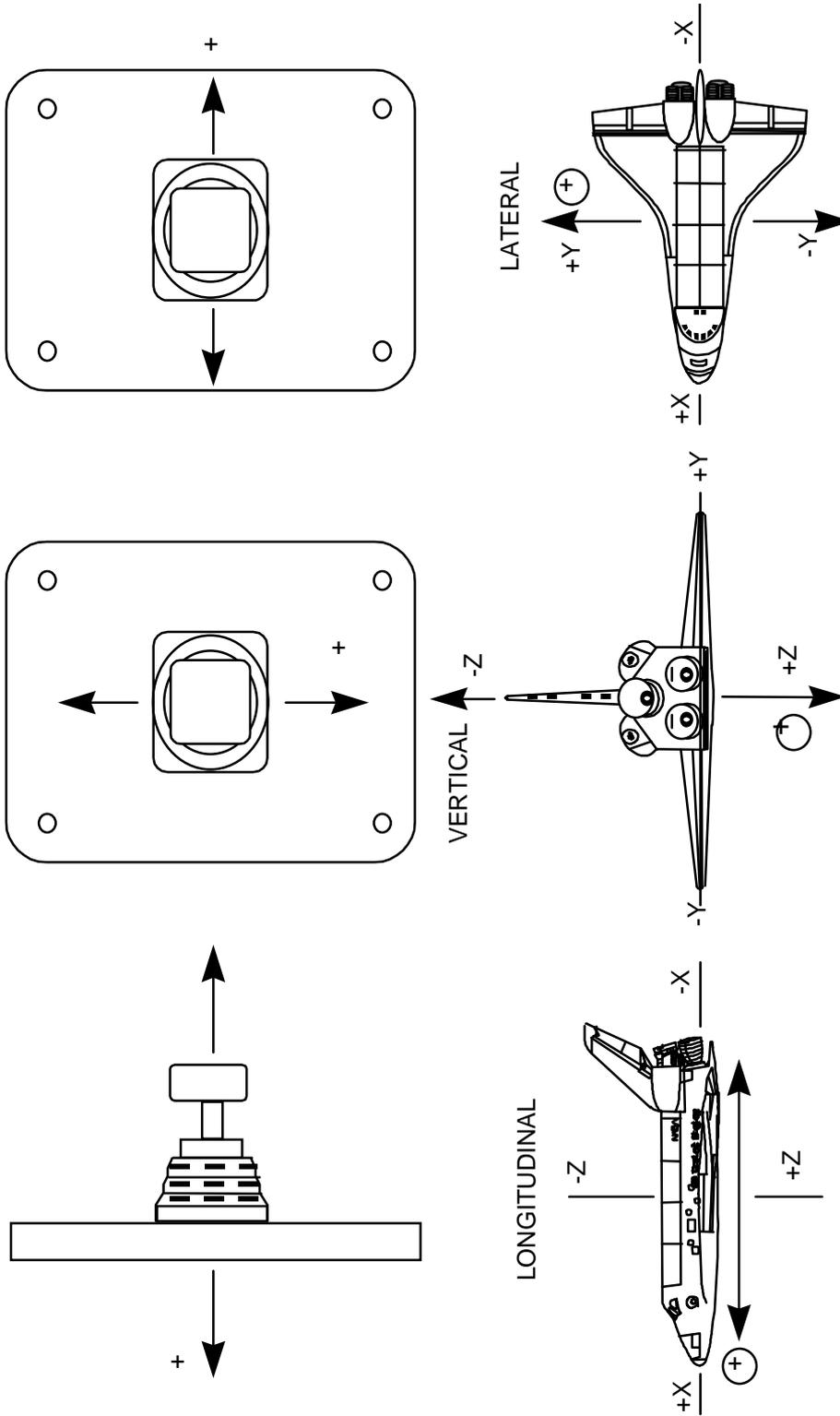


Figure 13 - 19. - Forward line-of-sight reference - THC use, commander.

RHC, the aft THC has two references, the -X and -Z reference. The -X reference uses the aft window to give proper and desired Orbiter references. Figure 13-20A provides -X SENSE switch reference and corresponding vehicle body axis movement. The -Z reference uses the overhead window to provide LOS reference. Figure 13-20B provides -Z SENSE switch reference and corresponding vehicle body axis movement.

It should be noted that a peculiar situation exists with the aft station THC. The THC is physically mounted 90° off nominal. This mounting is as pictured in figure 13-21 for the aft station. As a result, inputs to the aft THC, the resulting display indications on CONTROLLER display, and the actual movement of the vehicle are as follows.

| AFT THC physical movement (wrt panel) | Display outputs | Transformed aft THC outputs (body axis) | |
|---------------------------------------|-----------------|---|----------|
| | | -X sense | -Z sense |
| +X (in or push) | +X | -X | -Z |
| -X (out or pull) | -X | +X | +Z |
| +Y (right or vehicle port) | -Z | -Y | -Y |
| -Y (left, or vehicle starbrd) | +Z | +Y | +Y |
| +Z (down) | +Y | +Z | -X |
| -Z (up) | -Y | -Z | +X |

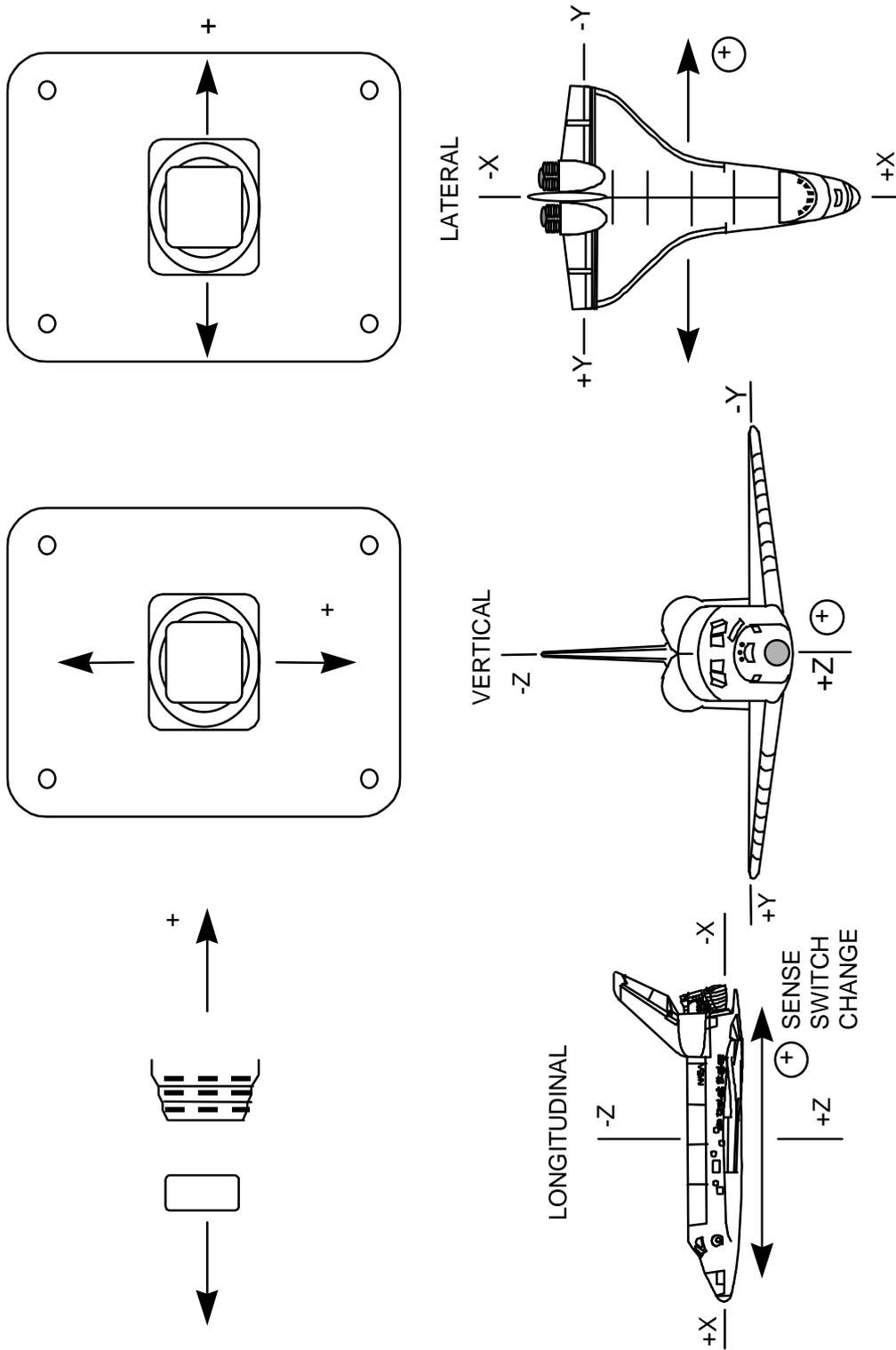


Figure 13 – 20A. - Aft line-of-sight reference - aft THC use SENSE switch -X.

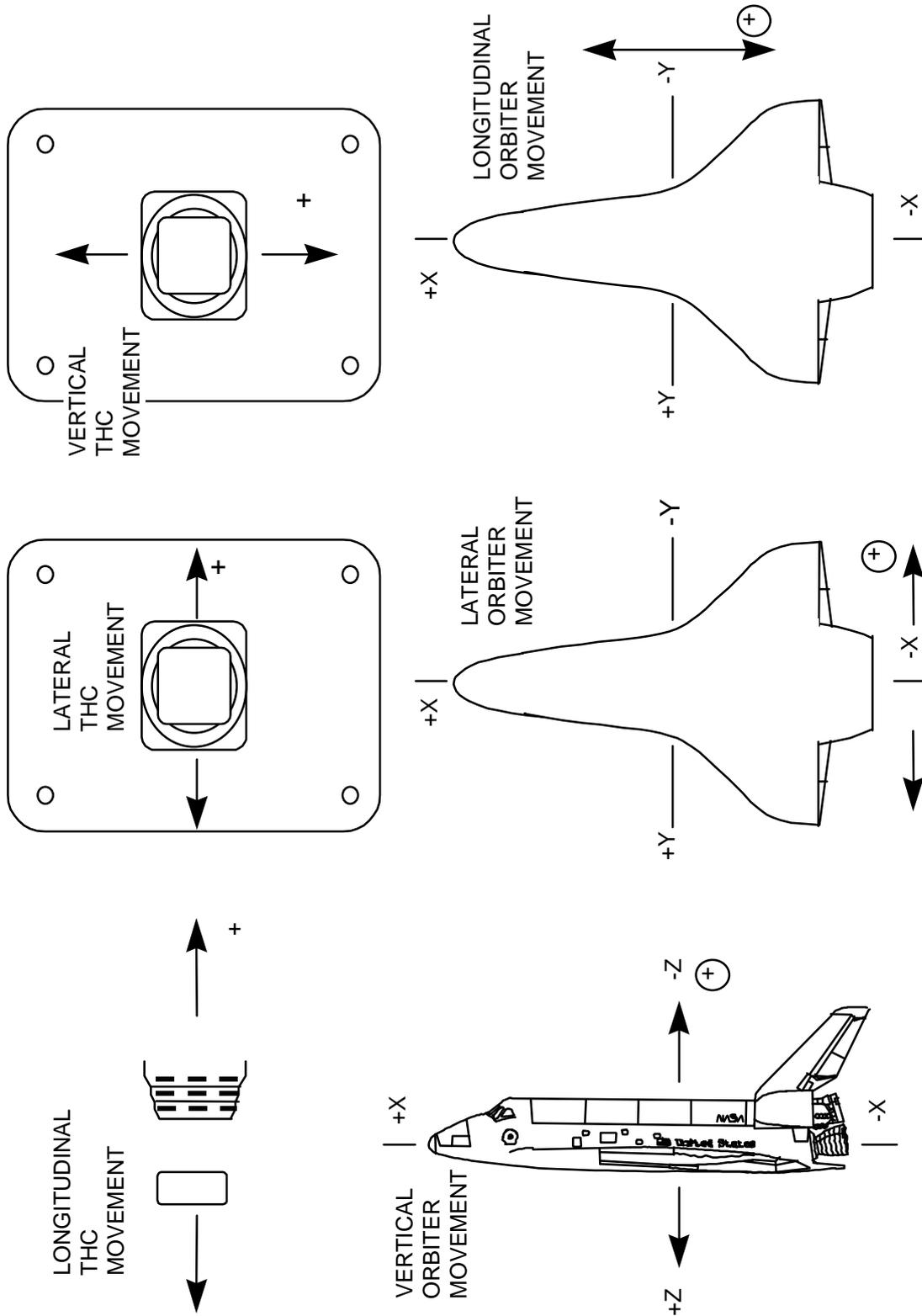
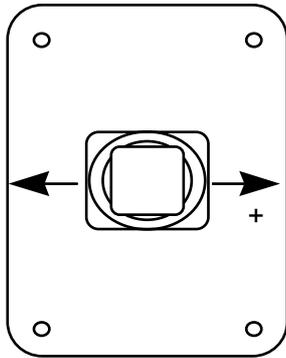
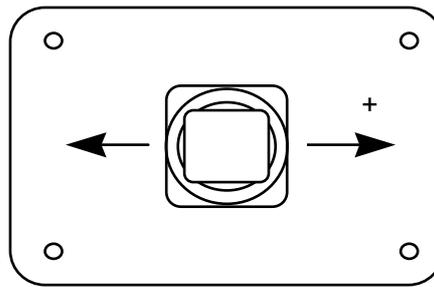


Figure 13 – 20B. - Overhead line-of-sight reference, aft THC use SENSE switch -Z.



Forward station THC



Aft station THC

Figure 13 - 21. - Aft THC mounting

13.3.2 Software Description and Operation

13.3.2.1 THC Software

The THC has six positions consisting of a positive and negative position for each axis. Each position has three contacts similar to a three-contact switch. The redundant signals from the forward and aft THC's go through RM processing and then to an SOP before being passed to the FCS in OPS 2 and 8. In OPS 1, 3, and 6, only the forward THC information is processed. The aft SENSE switch indication is necessary for proper transformation of the aft THC signals in OPS 2 and 8. Failures and dilemma conditions from RM go to the crew interfaces. The crew has deselection/reselection capability of the THC through keyboard item entry during OPS 2 and 8. Figure 13-22 provides a block diagram of the OPS 2 and 8 software flow for the THC. OPS 1, 3, and 6 software are similar except for the aft THC inputs and the SENSE switch inputs. Finally, it is important to note that the aft THC is not supported in BFS.

NOTE

Prior to OI-32, the 6.25 Hz sampling frequency of the THC contacts by the FF MDMs occasionally resulted in missed THC deflections. (If the deflection was less than 160 ms, the software may not see it, thus no jets would fire). This was observed on STS-63 and STS-112. Software changes in OI-32 will update the SF rate to 12.5 Hz in an effort to avoid this problem. Complications in the software prevented the FDIR rate from increasing as well, but this was determined to be not required.

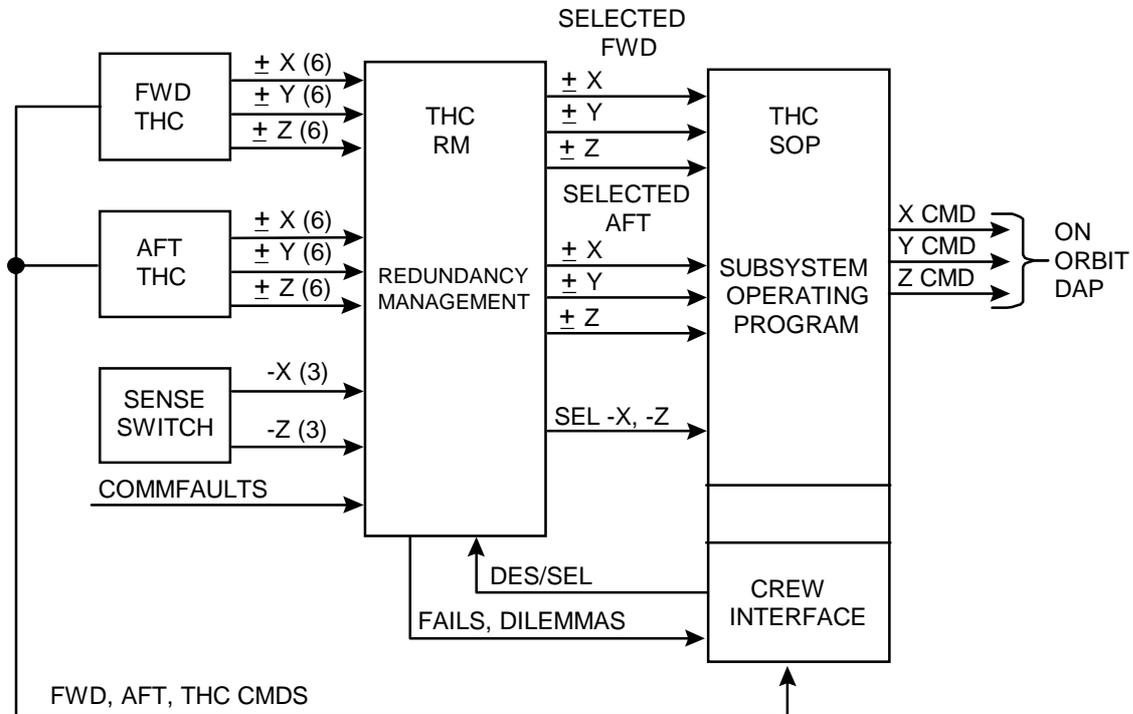


Figure 13 - 22. - THC software data flow (OPS 2, 8)

13.3.2.2 THC Software RM

Figure 13-23 provides a pictorial view of THC RM. RM for the THC is the same as three-contact switch RM since it is processed by switch RM software modules for both the SF and FDIR.

RM cases for the THC are as follows:

NOTE

FDIR and SF exist in all OPS/MM. The crew will perform MALF procedure and may deselect channels as procedures suggest. Channel deselection takes out all channel inputs from all axes. RM deselect only takes out failed input.

| <u>Case</u> | <u>SF</u> | <u>FDIR</u> |
|-------------------------------|---|--|
| Three level | <ul style="list-style-type: none"> The output is the majority vote of the three inputs. The processing rate is 12.5 Hz. | <ul style="list-style-type: none"> Compares all three contacts. A contact is declared failed if it disagrees with the other two switch inputs for three consecutive FDIR cycles. The processing rate is 6.25 Hz. |
| First fail | <ul style="list-style-type: none"> Failed contact deselected by FDIR. Remaining contacts are compared. Both must be "made" (closed) for switch to be in position. | <ul style="list-style-type: none"> Deselect failed contact from SF and FDIR. Compares remaining contacts; if miscompare for three cycles, declares dilemma. |
| First CF | <ul style="list-style-type: none"> CF channel data is declared missing by FCOS and SF downmodes to the two level. SF compares remaining contacts. SF upmodes to three level if CF goes away. | <ul style="list-style-type: none"> CF data is declared missing by FCOS and FDIR downmodes to the two level. FDIR compares remaining contact. If CF goes away, FDIR upmodes. |
| Second failure Fail open | <ul style="list-style-type: none"> SF does not show contact mode when switch closed. | <ul style="list-style-type: none"> Declares dilemma after three consecutive miscompares when switch closed. |
| Second failure Fail closed | <ul style="list-style-type: none"> Output from SF with one failed closed contact and one good contact. | <ul style="list-style-type: none"> Declares dilemma after three consecutive miscompares when switch opened. |

| | | |
|---|---|---|
| <p>Second fail- ure CF</p> | <ul style="list-style-type: none"> • SF to the one level. • Passes only single contact data. • Upmodes if CF goes away. | <ul style="list-style-type: none"> • FDIR downmodes to one level and stops processing. • Upmodes if CF goes away. |
| <p>Third fail- ure Contact fail</p> | <ul style="list-style-type: none"> • If previous failure was an open, and final contact is failed open, zero sent to FCS; if final contact was a failed close, dilemma results. • If previous failure was a close, output is determined by failure mode of final contact; i.e., if fail open, dilemma will result; if fail closed, contact is made and jets fire. | <ul style="list-style-type: none"> • No FDIR processing. |
| <p>Third fail- ure CF</p> | <ul style="list-style-type: none"> • SF outputs zero. • Will upmode if CF goes away. | <ul style="list-style-type: none"> • No FDIR processing. • Will upmode. |

13.3.2.3 THC SOP

The purpose of the THC SOP is to transform the aft station THC commands from the -X/-Z sense to the +X sense, and to process the forward or transformed aft THC signals to produce translation commands. Figure 13-24 provides a pictorial view of this function.

Note: BFS - Left only MM 103-6, 301-3, 601. COMMAND based on 2 of 3 majority vote. No COMMFAULT, FDIR or sel/des available.

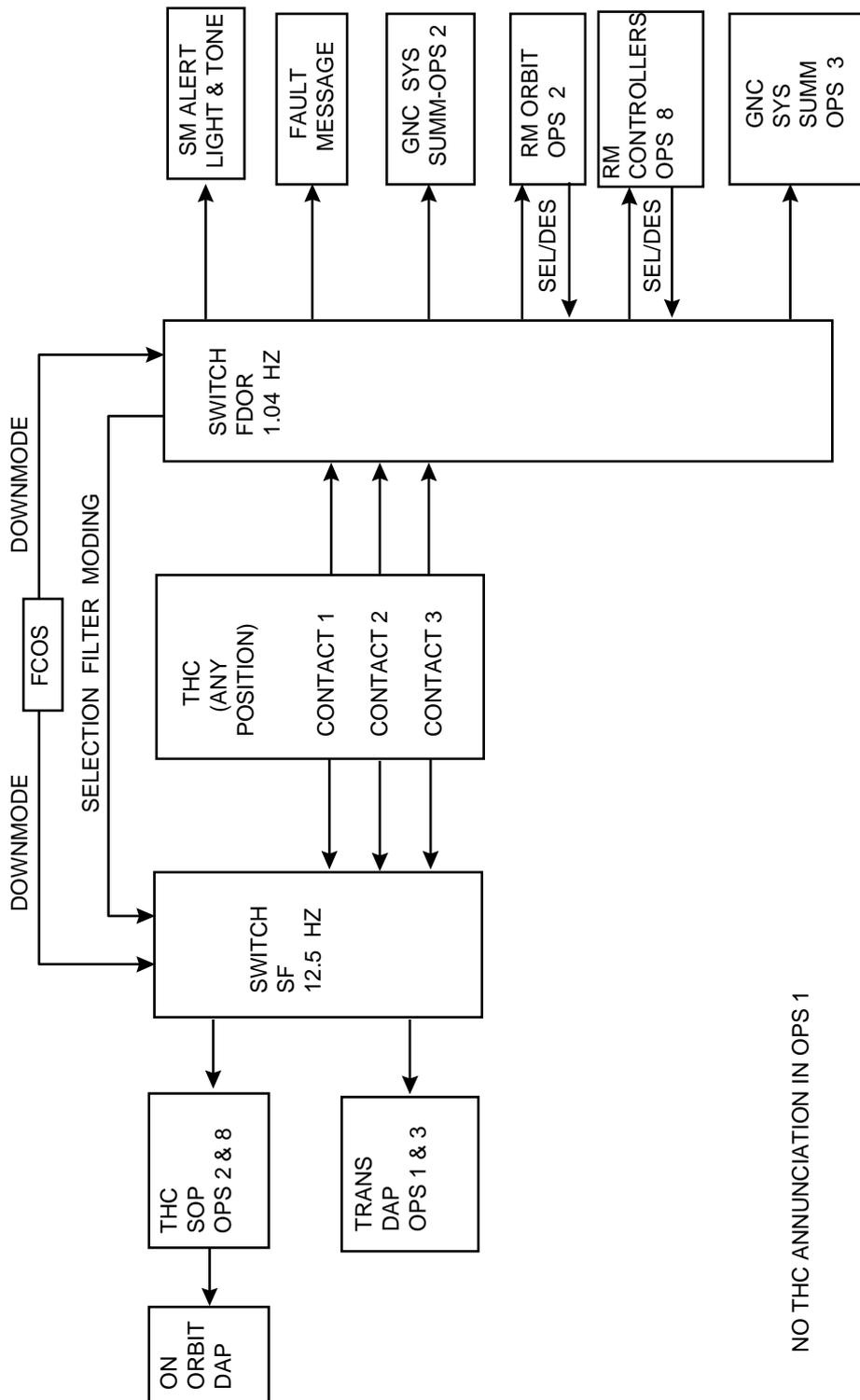
13.4 RUDDER PEDAL TRANSDUCER ASSEMBLY

13.4.1 Hardware Description and Operation

13.4.1.1 Physical Characteristics

The two RPTA's are located at the commander's and pilot's stations. Their purpose is to provide signals from the transducer to control the position of the rudder during the entry final stages. Figure 13-25 provides a pictorial view of the triple redundant transducers and its mechanical linkage. The left and right rudder pedals are mechanically linked together such that depression of the left rudder pedal will deflect the rudder to the left and depressing the right rudder pedal will deflect the rudder to the right. Each RPTA is redundantly powered by two main power buses which supply power to a DDU. Each main power bus has an associated circuit breaker.

- RM IS PERFORMED ON THE 6 POSITIONS OF THE AFT AND LEFT THC'S DURING OPS 2 AND OPS 8 AND ON THE LEFT THC DURING OPS 1 AND 3.
- EACH POSITION HAS 3 CONTACTS.



NO THC ANNUNCIATION IN OPS 1

Figure 13 - 23. - The THC RM.

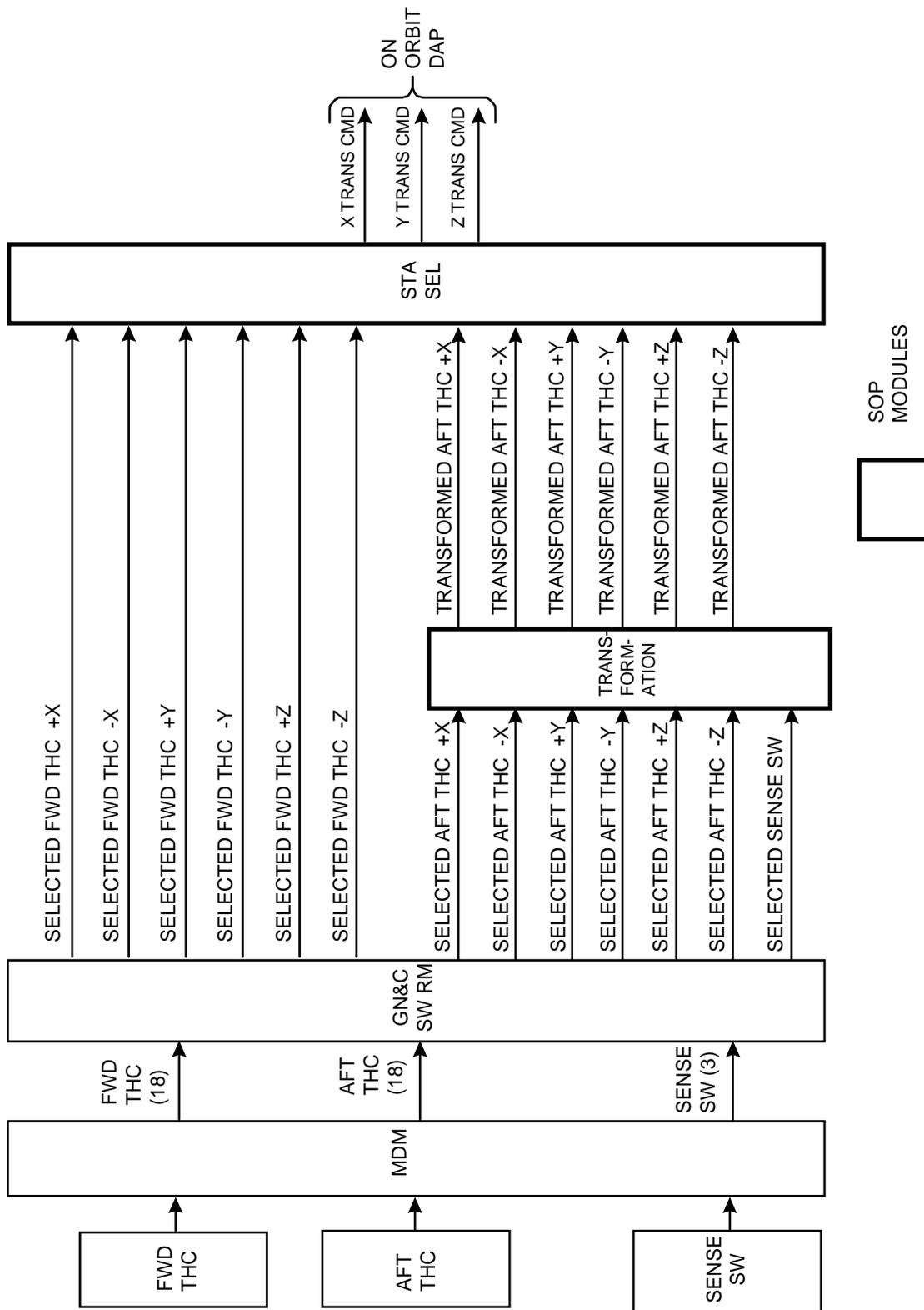


Figure 13 - 24. - THC SOP functional block diagram.

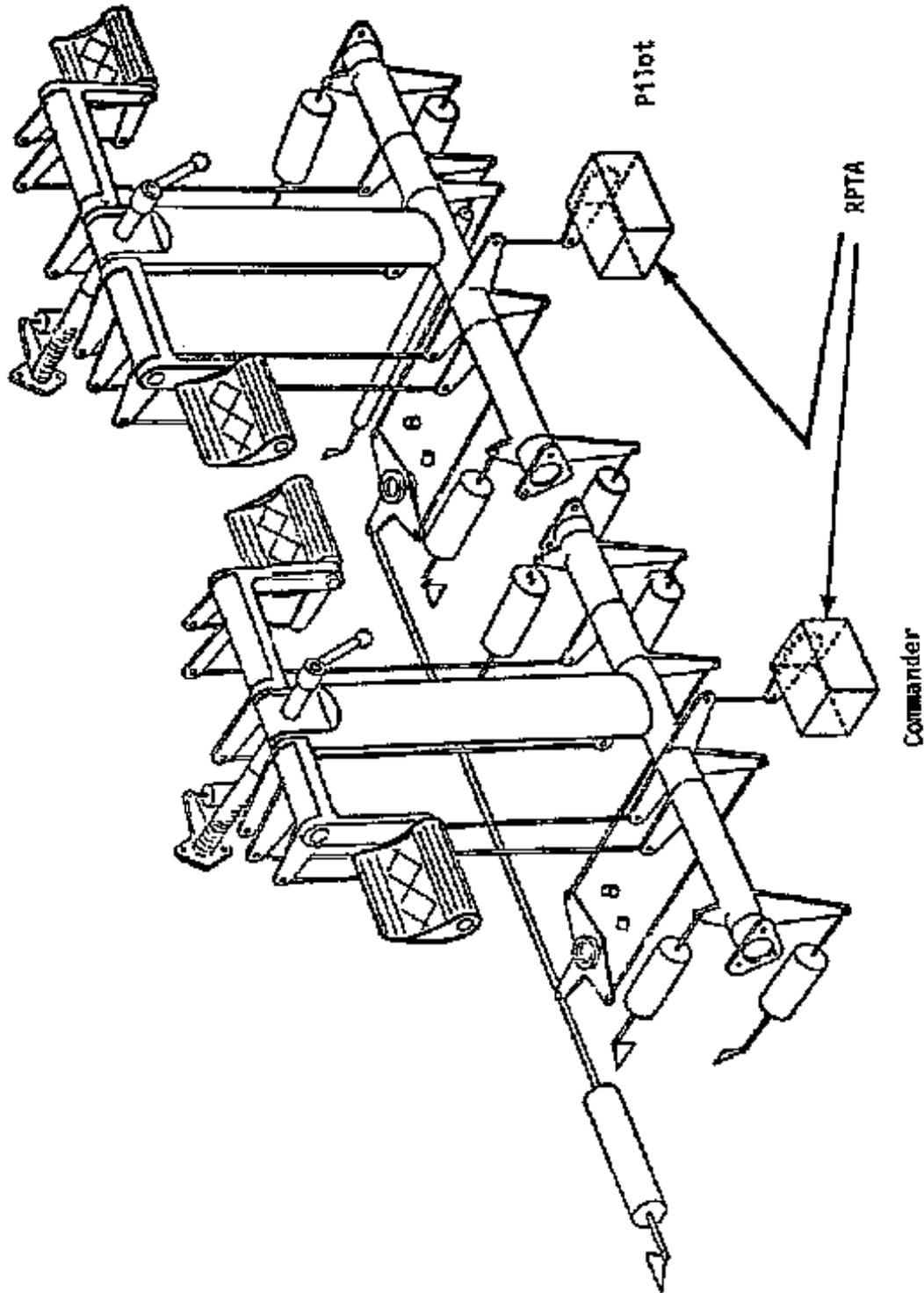


Figure 13-25.- Rudder pedal transducer assembly.

A flight controller power switch controls the power output of the DDU. See figure 13-5 for an overview of the DDU, flight controller power, and power switches related to the RPTA.

13.4.1.2 Availability

The commander and pilot RPTA's are available only in OPS modes 304, 305, 602, and 603. During OPS 8, the crew has the capability of viewing the rudder pedal commands and deselecting failures.

13.4.1.3 Mechanical limits

The RPTA has a 30° lever displacement from null going both up and down for a total spread of 60°. The breakout force is 10 lb with a total force of 70 lb required to move the pedal maximum forward or aft.

13.4.2 Software Description and Operations

13.4.2.1 RPTA Software Program

Software processes the triple redundant signals from the commander's and pilot's RPTA before the signal is passed to the FCS. The signal flows from the RPTA to the RM processing to the SOP to the FCS. Figure 13-26 is an example of data flow for the RPTA software.

13.4.2.2 RPTA RM

RM processing is performed on both the left and right side RPTA's at the same time but separately. Three level RM uses the MVS scheme for selection of the signal to be sent to SOP from the SF. Fault detection calculates the difference between two selected signals and compares the results to an I-loaded value (currently 3.6°). If an I-load is exceeded, then the fault detection will downmode the SF. From the SF, a DG bit is sent to the SOP. FDIR does not annunciate RPTA failures.

At the two level, the SF will average the two signals and send the results to the SOP. The DG test is implemented in the filter and results are sent to the SOP. The FDIR does its comparison of the two remaining commands to the I-load and will downmode the SF if necessary.

At the one level, the SF passes the single command to the SOP and sets the DG bit to "good." There is no FDIR.

At the zero level, the DG bit is set "bad" and the SOP sends zeros to the FCS.

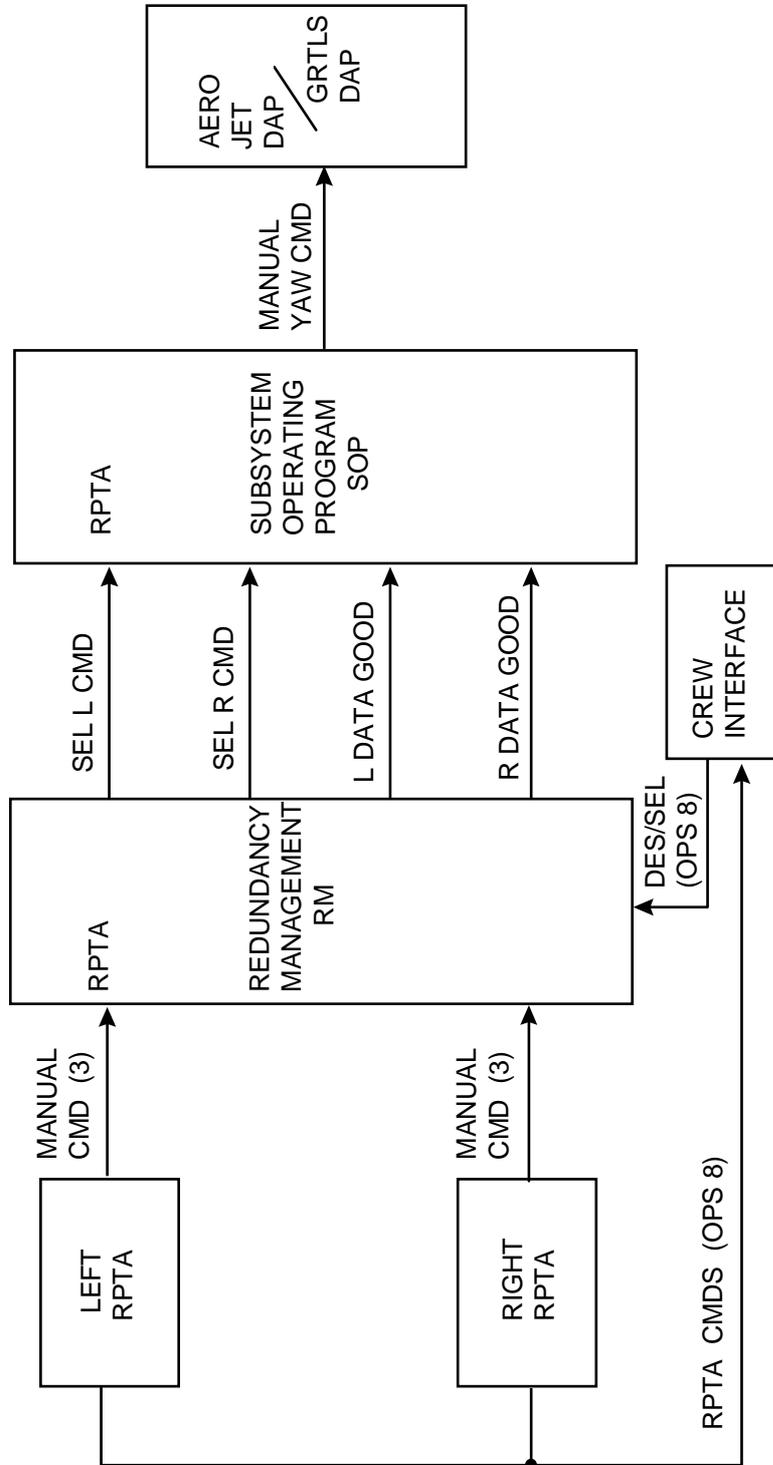


Figure 13 - 26. - RPTA software data flow (OPS 3, 6; MM 304, 305, 602, 603).

In OPS 6 there is no FDIR processing, and downmode of the SF is only by CF's from the flight computers.

13.4.2.3 RPTA SOP

The RPTA SOP provides the following functions. First, the command signals are converted from volts to degrees. Second, the SOP performs station selection by selecting the right or left RPTA signal, if DG flag is "good" and commands are present. Compensation of the selected transducer commands is also accomplished. Also the processing of breakout deadbanding of the selected station is performed. Figure 13-27 shows a functional diagram of the RPTA SOP.

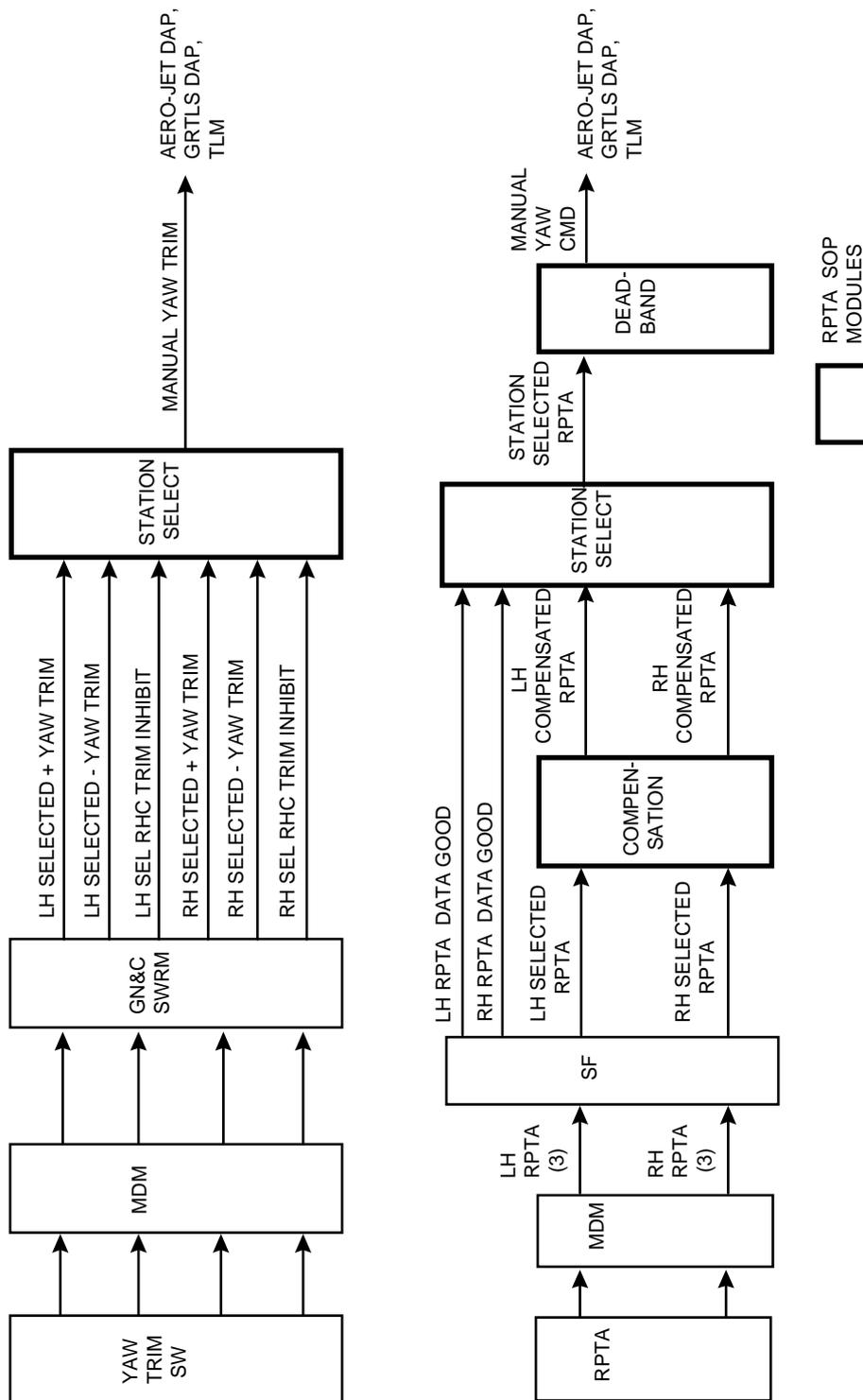


Figure 13 - 27. - RPTA SOP functional block diagram.

13.5 SPEEDBRAKE THRUST CONTROLLERS

13.5.1 Hardware Description and Operation

13.5.1.1 Physical Characteristics

The two SBTC's on the Shuttle, located at the commander's and pilot's stations, are used to control the position of the vertical stabilizer-mounted speedbrake during the entry flight phase. Additionally, the pilot's SBTC can be used for manual throttle control of the SSME's during the ascent phase.

The lever and cockpit mounting configuration of the SBTC makes the control suitable for left-hand use only. The takeover switch located at the thumb position is used to gain manual control of the speedbrake.

For manual control, when the lever is rotated forward to the stop, the speedbrake is commanded completely closed. The lever rotated completely backward to the stop yields a speedbrake full open command. Alternatively, during ascent these two settings correspond to a 100 percent throttle command (full forward) and to a 65 percent throttle command (full back). See figure 13-28 for view of the SBTC.

13.5.1.2 Availability

Pilot only SSME manual thrust control using the SBTC is available in OPS MM 102, 103, and 601. Manual speedbrake control is available in MM's 304, 305, 602, and 603 for either the commander or pilot.

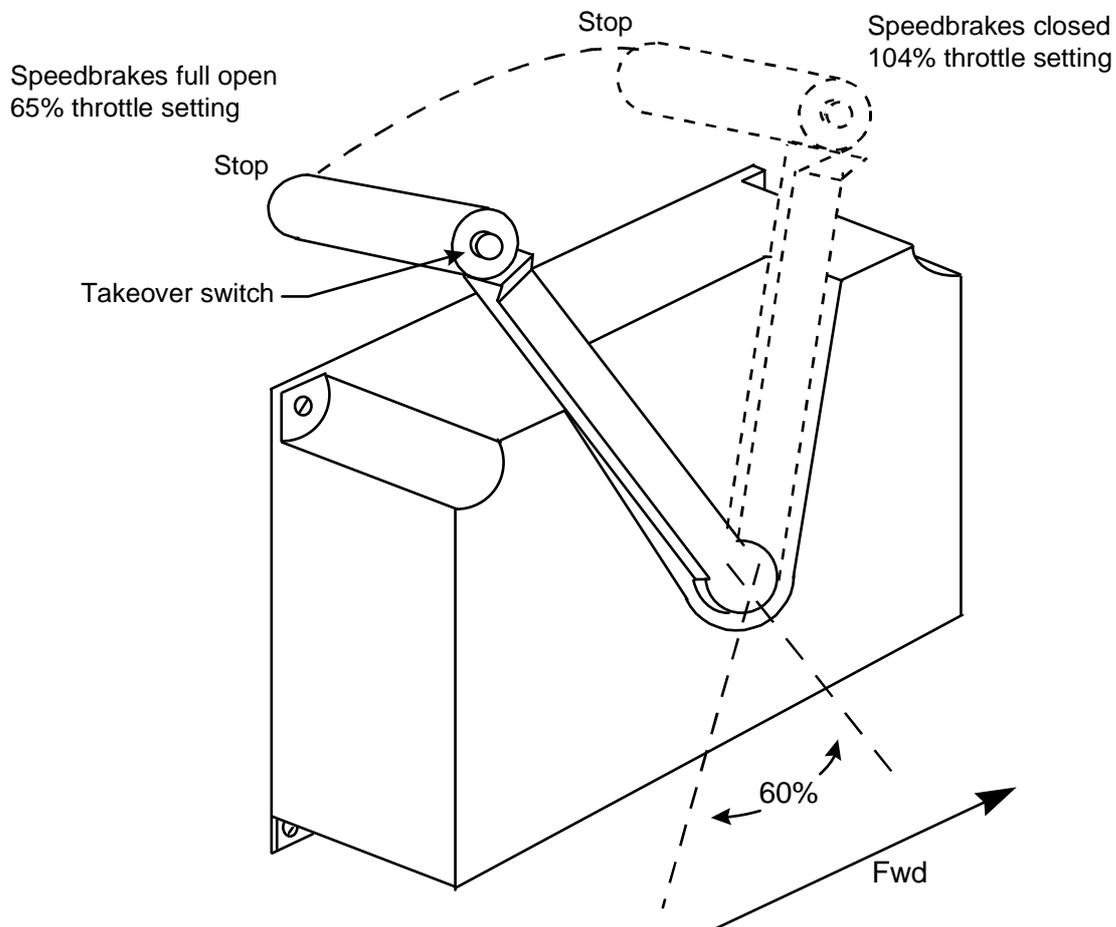


Figure 13 - 28. - Speedbrake thrust controller use.

Manual speedbrake control using the speedbrake takeover switch (fig. 13-29) is as follows:

If the SBTC mode is auto, momentarily depressing the pushbutton downmodes the control to manual and the SBTC of the crewmember who depressed the switch is granted control. The speedbrake is driven to the position currently commanded by the SBTC.

If the SBTC mode is under manual control, the crewmember who does not have control can gain control by momentarily depressing his SBTC takeover switch.

If both switches are activated simultaneously, the commander shall have control.

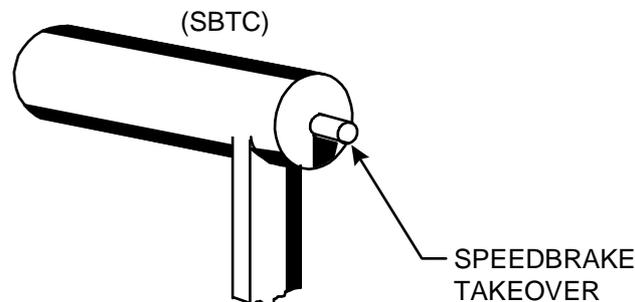


Figure 13 - 29. - Speedbrake control.

Manual throttling using the speedbrake takeover switch is as follows:

If the throttling mode is auto, manual control for the right SBTC is established as follows:

- a. Depress and hold the takeover button on the right SBTC. (This causes the computer throttle command to be frozen at its current value.)
- b. While holding the button depressed, move the SBTC to match the frozen computer command.
- c. Manual control is established when the match is achieved within 4 percent. (When the match is achieved, the SPDBK/THROTTLE PBI "MAN" light shall illuminate on the pilot's side of the cockpit.) The button may then be released.
- d. If the takeover button is released prior to achieving a match, the system shall revert to computer-generated (auto) commands.

13.5.2 Software Description and Operation

13.5.2.1 Manual Throttling Control Software

During MM's 102, 103, and 601, the pilot can perform manual throttling of the main engines.

The selected right-hand manual takeover, manual throttle command, and DG indicator are output by RM to the SBTC SOP as well as the station selected pbi auto command. When the SOP receives the manual takeover indicator, the manual throttle discrete is output to guidance which causes the guidance-generated auto throttle command to be frozen. Also, both the commander's and pilot's auto SPD BK/THROT lights are extinguished. A comparison of the auto throttle command from guidance and the pilot's manual throttle command is performed in the SOP, and when a match within 4 percent tolerance is made, manual control is established and the manual throttle command is output by the SOP. The pilot's man SPD BK/THROT light is illuminated after manual throttle command is established.

The data flow for manual throttling control is pictured in figure 13-30.

13.5.2.2 Manual Speedbrake Control Software

During MM's 304, 305, 602, and 603, the pilot or commander can manually control the speedbrake. RM processes the takeovers and the manual commands from the SBTC's as well as the AUTO pbi's. The SBTC SOP performs station selection of the manual SBTC commands for output to the autopilot. The autopilot handles the lighting of the AUTO/MAN SBTC lamps.

The crew has channel deselection capability for signals from both SBTC's in OPS 8.

The data flow for manual speedbrake control is pictured in figure 13-31.

13.5.2.3 SBTC Redundancy Management

RM for the SBTC can be easily visualized by review of figure 13-32 and table 13-I.

13.5.2.4 SBTC Crew Displays

The GNC SYSTEM SUMMARY 1 is available in OPS 3 and 6 for displaying speedbrake status. During OPS 6, only CF status is indicated. RM-declared failures and dilemmas, as well as CF's, are displayed in OPS 3.

The CONTROLLERS display is available in OPS 8 for displaying left and right SBTC commands. The triple redundant commands are displayed in percent (from 0 to 100) of speedbrake extension. The crew has channel deselection capability for each SBTC.

13.5.2.5 SBTC SOP

The SBTC SOP performs the following functions:

- a. Converts the selected manual speedbrake command from volts to degrees for flight control.
- b. Performs station selection for manual speedbrake commands - OPS 3 and 6.

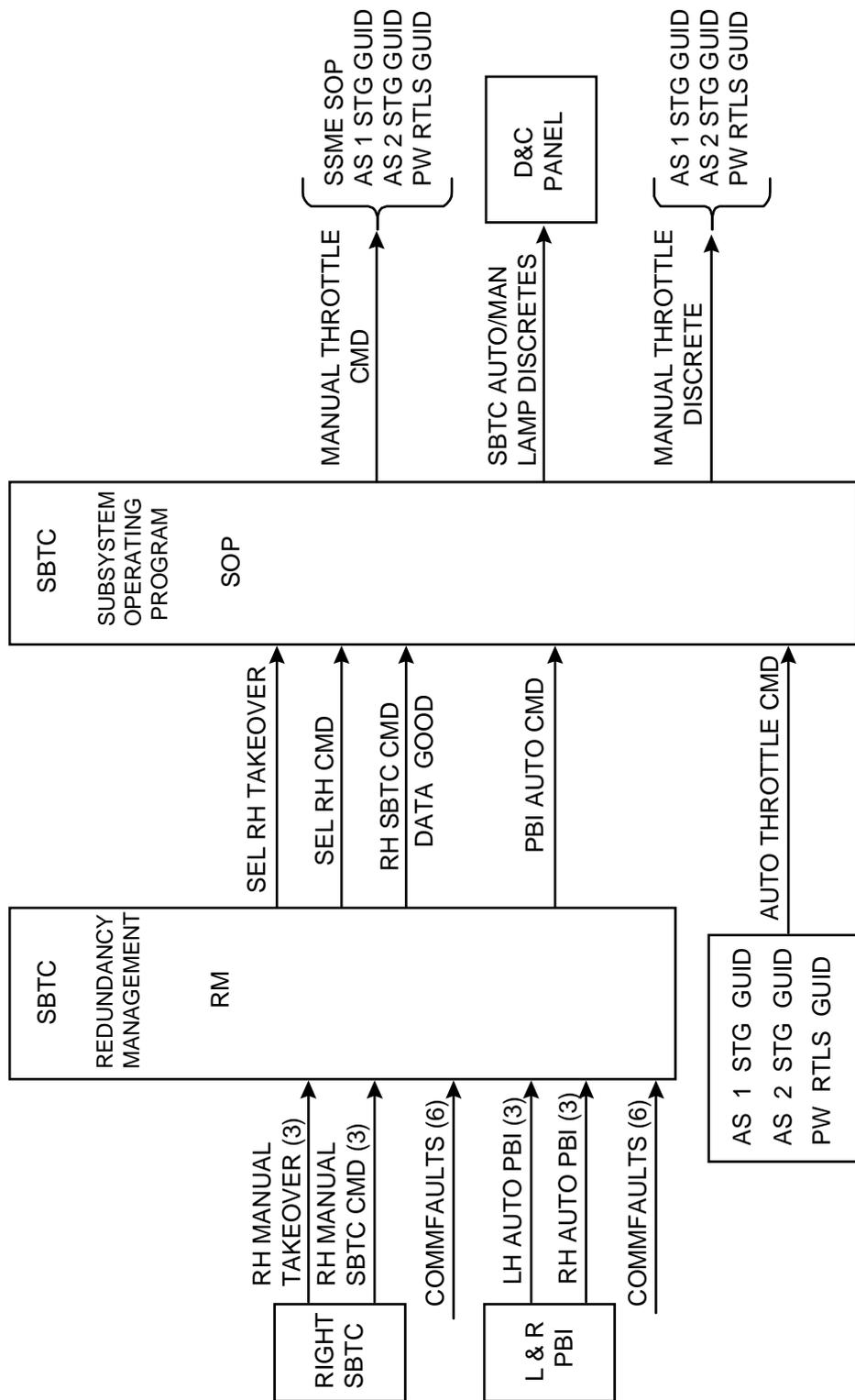


Figure 13 - 30. - SBTC software data flow (manual throttling control) (OPS 1, 6; MM 102, 103, 601).

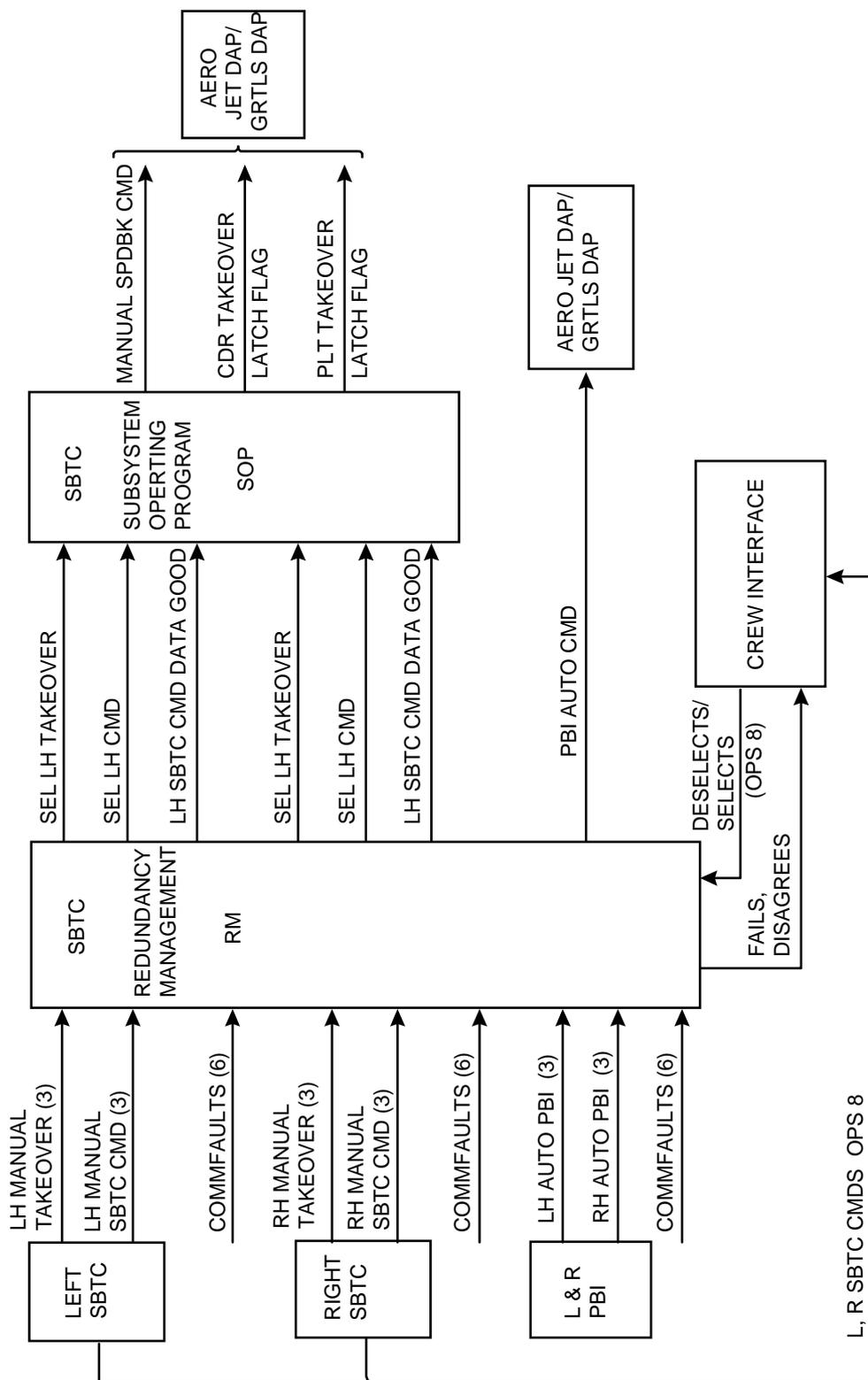
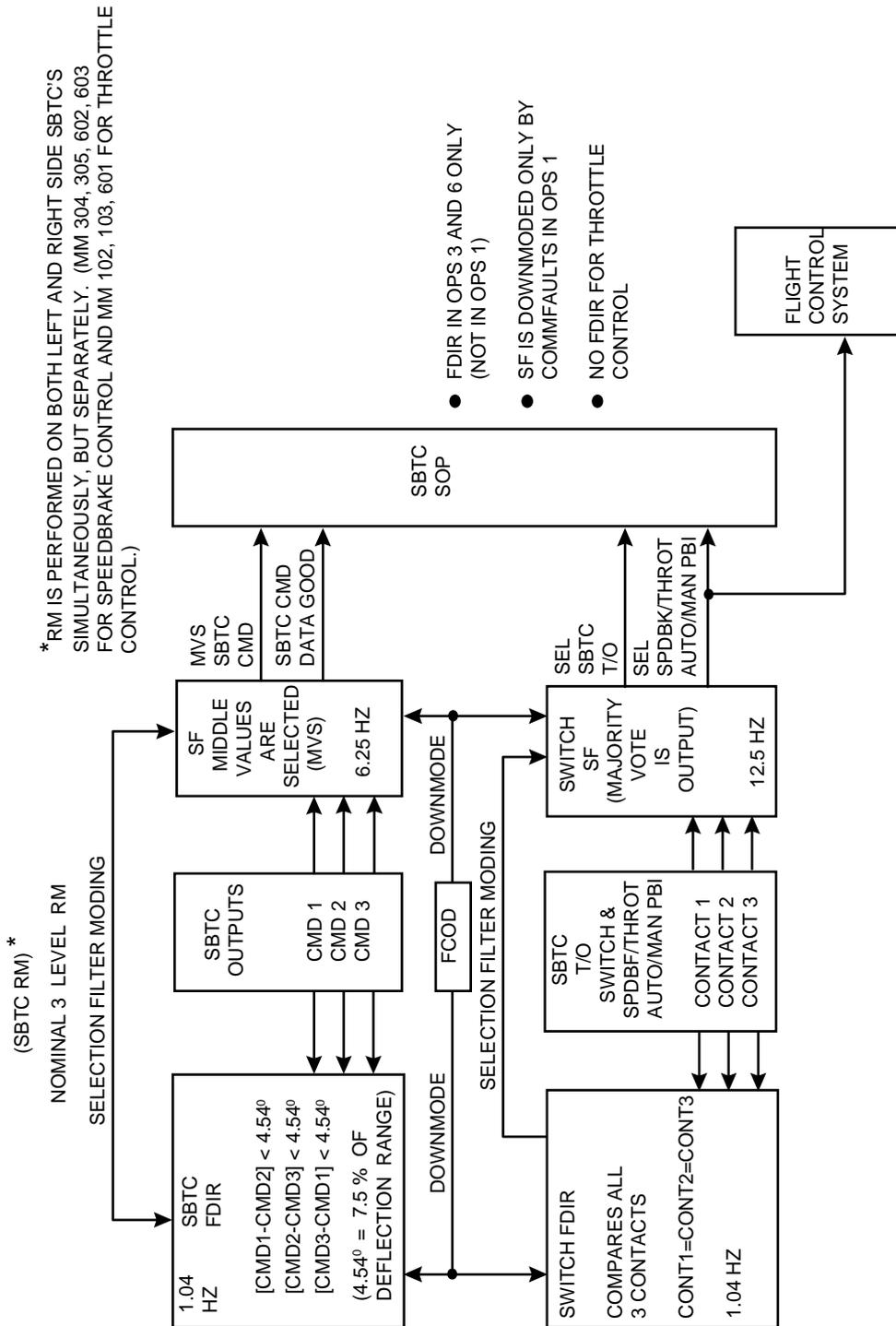


Figure 13 -31. - SBTC software data flow (manual speedbrake control) (OPS 3, 6; MM 304, 305, 602, 603).



*RM IS PERFORMED ON BOTH LEFT AND RIGHT SIDE SBTC'S SIMULTANEOUSLY, BUT SEPARATELY. (MM 304, 305, 602, 603 FOR SPEEDBRAKE CONTROL AND MM 102, 103, 601 FOR THROTTLE CONTROL.)

Figure 13 - 32. - Speedbrake thrust controller RM.

TABLE 13 - I. - SBTC RM

| Selection filter | | FDIR |
|---|---|--|
| <ul style="list-style-type: none"> • 3 level (nominal) • 2 level (1 failure) implemented. • 2 level (DLMA) • 1 level (2 failures) • 0 level (3 failures) | <ul style="list-style-type: none"> - Middle values are selected. Data good is set "good" - 2 values are averaged. Data good test is implemented. Tolerance is 5.35 - Data good set bad, output 0 - Data is passed through. Data good is set "good" - Data good is set "bad." SOP sends zeros to flight control | <p>Annunciation is (OPS 3):</p> <ul style="list-style-type: none"> • SM alert light and tone and fault messages "SBTC/THC L (R) 1234 HH:MM:SS" • GNC system summary 1 <ul style="list-style-type: none"> - for channel (CMD) failed (OPS 3) ? for dilemma (OPS 3) M for COMM fault (OPS 3&6) (from FCOS) |
| <ul style="list-style-type: none"> • RM for the SBTC takeover switch and auto pbi is the same as aft sense switch. RM - no annunciation • RM for SBTC commands is the same as for RHC commands. | | |

- The output command shall be chosen from the crewmember who last depressed the takeover switch.

- If the command selected for output has a "bad" DG flag, then the output command will be held to the value of the last output command.

- If both crew stations generate simultaneous takeover commands, the commander shall take precedence.

c. Provides takeover latch flags for flight control (OPS 3 and OPS 6) to identify the crewmember presently in control. This allows the correct pbi (bottom half) to be illuminated.

d. Converts the compensated manual throttle command to a manual throttle setting in percent for guidance.

e. Provides auto and manual flags to illuminate the SBTC pbi's in MM's 102, 103, and 601.

13.6 NOTES OF INTEREST

Two notes of interest have been written for controllers concerning an RPTA anomaly on STS-6 and flight controller power switch transients. These notes are included here in their entirety along with an associated internal letter from Rockwell and a memo from The Charles Stark Draper Laboratory, Inc.

| CONTROLLERS | | NOVEMBER 1988 | | | | | | | | | | | | | | | | |
|-------------|-------|---------------|--------|-------|-----------|---------|--------|---------|--------|---------|--------------|---------|----------|--------|---------|---------|----------|-------|
| | | CNTLR | AXIS | UNITS | DEAD BAND | DETENT | | | | RM (DG) | CSS DOWNMODE | | SOFTSTOP | | | | HARDSTOP | |
| | | | | | | OPS 1.3 | | OPS 2.8 | | | Flight | Post TD | Lower | Upper | Lower | Upper | Lower | Upper |
| RHC | Pitch | Pot (MCC) | 4.701 | 2.395 | 7.186 | 2.395 | 17.872 | 21.247 | 24.995 | 49.092 | 79.154 | 83.320 | 88.157 | 88.199 | 95.818 | 101.234 | | |
| | | Pot (OB) | 4.726 | 2.363 | 7.089 | 2.363 | 17.630 | 20.968 | 24.667 | 49.314 | 78.080 | 82.190 | 86.961 | 7.002 | 94.518 | 99.861 | | |
| | | Deg Volts | 1.150 | 0.575 | 1.725 | 0.575 | 4.290 | 5.100 | 6.800 | 12.000 | 19.000 | 20.000 | 21.161 | 21.171 | 23.000 | 24.300 | | |
| RHC | Roll | Pot (MCC) | 5.208 | 2.395 | 7.186 | 2.395 | 17.872 | 28.329 | 24.995 | 24.995 | 79.154 | 83.320 | 88.178 | 88.219 | 95.818 | 101.234 | | |
| | | Pot (OB) | 5.137 | 2.363 | 7.089 | 2.363 | 17.630 | 27.945 | 24.667 | 24.667 | 78.080 | 82.190 | 86.982 | 87.023 | 94.518 | 99.861 | | |
| | | Deg Volts | 1.250 | 0.575 | 1.725 | 0.575 | 4.290 | 6.800 | 6.800 | 6.000 | 19.000 | 20.000 | 21.166 | 21.176 | 23.000 | 24.300 | | |
| RPTA | Yaw | Pot (MCC) | 4.545 | 3.496 | 6.818 | 3.496 | 20.559 | 19.580 | | | 66.431 | 73.424 | 79.767 | 79.837 | 90.906 | 99.997 | | |
| | | Pot (OB) | 4.581 | 3.523 | 6.871 | 3.523 | 20.718 | 19.731 | | | 66.946 | 73.993 | 80.385 | 80.465 | 91.810 | 100.771 | | |
| | | Deg Volts | 0.650 | 0.500 | 0.875 | 0.500 | 2.515 | 2.800 | | | 9.500 | 10.500 | 11.407 | 11.417 | 13.000 | 14.300 | | |
| SBTC | X | Pot (MCC) | 5.000 | | | | 16.001 | | | | | | | | 102.231 | | | |
| | | Pot (OB) | 4.995 | | | | 15.984 | | | | | | | | 102.120 | | | |
| | | Deg Volts | 1.125 | | | | 3.600 | | | | | | | | 23.000 | | | |
| SBTC | X | Pot (OB) | | | | | 8.842 | | | | | | | | 102.556 | | | |
| | | Deg (MCC) | | | | | 5.348 | | | | | | | | 60.000 | | | |
| | | Volts | -0.250 | | | | 0.799 | | | | | | | | -5.106 | | | |

| CONTROLLER CONVERSION FACTORS AND NOTES | | NOTES |
|---|-------|--|
| RHC | Pitch | Two axis RHC (P, R) active in MM 602, 603, 304, 305. Three axis RHC (P, R, Y) active in MM 102 - 106, 601, 301-303, OPS 2.8. |
| | Roll | |
| | Yaw | |
| RPTA | Yaw | Right rudder pedal input equals positive percent, negative RPTA command (deg), negative RPTA volts, negative rudder surface deflection (towards starboard wing). |
| SBTC | X | 0 DEG = full forward, 80 DEG = full aft Takeover manual speedbrake/throttle with SBTC takeover PB |

Onboard (OB) display units, Mission Control Center (MCC) display units use conversion factors to convert from MCC units to other units

DF NOI #231

REFER TO: DF6-86-74

DATE: 10/8/86

To: Distribution

From: DF6/Stewart Cobb

Subject: Flight Controller Power Switch Transients

When the Flight Controller Power switches are turned on or off, the Rotational Hand Controller (RHC) and Translation Hand Controller (THC) can generate spurious inputs into the flight control system. In the past, these transients have downmoded the digital autopilot (DAP) to manual mode, and fired jets unnecessarily. This note will explain the cause of the transients, and the actions which have been taken to work around the problem.

Background

In the Orbit and TRANS DAP's the DAP will downmode automatically from AUTO to MANUAL when the RHC is deflected out of detent, even briefly. When it gets to MANUAL mode, it will interpret the RHC deflection as a manual rotation command and fire jets to accomplish the command. The Orbit DAP will interpret a deflection of the THC as a manual translation command, and will also fire jets.

The controllers (RHC, THC, RPTA, SBTC) contain transducers which required +/-15 volt and +5 volt power. This power is provided by dedicated power supplies inside the Display Driver Units (DDU). The Flight Controller Power switches at each of the crew stations (CDR, PLT, AFT) turn these power supplies on and off.

As these power supplies are cycled on or off, power transients pass through the transducers in the flight controllers, producing transients on the transducer outputs. These appear to the flight control system as brief deflections of the various controllers. These spurious inputs were measured (for the RHC, in SAIL testing in February 1981) as approximately 450 millivolts per axis, lasting approximately 50 milliseconds.

The flight software interprets these voltages as deflections of the RHC of 1.2 degrees in yaw and 2.1 degrees in pitch and roll. These are outside the original detent limits for the Orbit and TRANS DAPs. The original flight software interpreted these transient inputs as valid manual commands. The DAP downmoded to manual, and jets fired. This was, at the least, an annoyance to the crew, and potentially a major impact to the flight. A fix for this problem was sought.

During OFT, the Orbit DAP detent limits were changed to 2.5 degrees in yaw and 4.3 degrees in pitch and roll, putting the transients well inside the detent limits. This has eliminated the downmode problem in the Orbit DAP.

Unfortunately, the detent limits cannot be similarly increased in the TRANS DAP. The RHC must be sensitive enough to fly the deorbit burn in manual thrust vector control mode, in case this mode is required for deorbit. Increasing the limits reduces the sensitivity of the RHC to a level unacceptable to the crew.

Workarounds

For the TRANS DAP, no permanent solution has been found. The current workarounds are to minimize the number of times the flight controller power is switched on or off, and to have the crew check that the DAP remained in AUTO each time they switch flight controller power.

The THC also generates spurious inputs due to power cycles. These do not downmode the DAP, but they do fire jets unnecessarily. In the Orbit DAP, the workaround is to set the DAP to VERNIER mode before switching Flight Controller Power. In VERNIER mode, the DAP ignores translation commands. In the TRANS DAP, the only workaround is, again, to minimize the number of times the flight controller power is switched on or off.

No hardware fix for these transients is anticipated.

Reference DR 55357

Notes: Trans DAP downmodes observed on STS-28 (OV102) when both L & R fit cntlr power turned off.

Trans DAP downmode occurred on STS-36 (OV104) when L fit cntlr pwr turned off post-MECO.

H. Stewart Cobb

NOTE OF INTEREST #149

DF62 (83-12)

TO: Distribution

FROM: DF62/Harold Hardwick

SUBJECT: RPTA Anomaly During STS-6 FCS Checkout - OV99

Background

During the STS-6 Ops 8 FCS checkout of the rudder, the full deflection of the left rudder pedal was 91.5 percent and the right rudder pedal was 90.6 percent. Current criteria states that both rudder pedal full deflection values should be greater than 91 percent. The right rudder pedal was below passing criteria by .4 percent. No action was generated during STS-6 for this anomaly. Entry was unaffected.

Post Mission Analysis Conclusions

Analysis of the rudder pedal transducer assembly (RPTA) and rudder rigging on OV99 and OV-102 resulted in the following conclusion (see enclosed full Rockwell report dated May 19, 1983, same subject).

- a. The problem was determined to be mechanical in nature and not associated with the RPTA directly.
- b. Not taken into account in previous development of pass/fail criteria for Ops 8 checkout of the rudder system was lower than nominal pedal throw, and null and limit shifts resulting from cabin motion while in zero gravity for OV99.
- c. In a gravity environment; i.e., prior to pre-flare and landing, cabin motion does not present the structure problem as it does on-orbit.
- d. Data from previous flights indicate that the maximum entry and rollout rudder deflection was 5.5° and 12.5° respectively.

Recommendation

Change the current Ops 8 FCS checkout procedures for all vehicles as follows:

- a. Change the full deflection percent check from 91 percent left and right to full left 87 percent and full right 30 percent.
- b. At full deflection also check to see that all channel readings are within 9 percent.
- c. The null check reading should be between 14 percent left and 4 percent right.

Action

- a. Informed the STS-7 crew and GN&C personnel of recommended change through this note of interest.
- b. Develop Crew Procedures Change Request (482) for STS-8.

Harold Hardwick

Distribution:

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SHUTTLE OPS
GNC/JSC-18863

ORIG: M.INTERBARTOLO
7/26/2007:FINAL , PCN-1

CONTROLLERS
SB 13.0

Date: • May 19, 1983 No: 283-100-TLB-83-001

TO: *(Name, Organization, Internal Address)*

- D. A. Johnson
- Group Engineering
- D/283-200, 041/TB75

FROM: *(Name, Organization, Internal Address)*

- T. L. Brown
- Group Engineering
- D/283-100, 041-FB98

Subject: RPTA Anomaly During STS-6 Orbit Checkout

Rudder Pedal Transducer assembly Orbit Checkout on STS-6 had low output on right pedal extension, both stations.

Flight Anomaly tracking list number J-35

KSC 1PR 7V-0169CS

The conclusion is that the rudder pedal system operated normally during the STS-6 flight and that orbit checkout pass/fail criteria were too restrictive.

The null offset and travel limit changes show all electronics channels (6) were tracking and data handling procedures and scale factors were not abnormal.

The problem was therefore determined to be mechanical in nature. A shortened rigging check verified that the pedal rigging had not changed since installation.

The structures group had not performed a pedal attach point motion analysis, but data was available from which to obtain this stress/strain data for pedals.

Requests were submitted for OV-102 ground and orbit checkout data to look for similarities in pedal shift. These data showed similarity in that OV-102 ground checkout data was consistent before and after flight.

Successive flights showed the same pedal shift for orbit checkout which was a left RPTA output null shift and values to support the same output travel shifts for left pedal and for right pedal maximum deflection checks.

The null shifted slightly more on OV-102 than OV-099, but the travel limit changes for OV-099 were greater than for OV-102. All changes favor left pedal deflection.

The structures data were applied to the pedal installation data and structure shifts of less than 0.160 inch per axis caused RPTA linkage shifts of 0.060 inch (1.4°). Linkage play plus real vehicle installation and tolerances can easily account for differences from the model predictions for both OV-102 and OV-099.

D. A. Johnson
RPTA Anomaly During STS-6 Orbit Checkout (Cont.)
Page 2

The structures data was for maximum deflection at zero external cabin pressure which is compatible with orbit checkout environment. The structure is designed to flex/extend and return to the original slope and dimensions when pressure is re-established on the crew compartment exterior. RPTA data shows this to be the case for the pedal system.

This analysis was not previously accomplished because the RPTA's are not active in flight control on the orbit or entry flight regimes. They become active in the TAEM phase and are basically not used until pre-flare and landing.

By averaging the OV-099 and OV-102 data, there is a 5% left shift in null and right maximum travel output and a 2% left shift in left maximum travel output. The redundancy management software used in OPS 3 is set to declare any transducer channel different from the other 2 channels by more than 9% to be a failed transducer. The resulting RM limit plus shift says a reading of less than 14% left deflection and less than 4% right deflection at null pedal position would be appropriate orbit checkout limits.

The flight control system provides adequate gains to allow a transducer output less than 85% to command full rudder authority. Therefore, the full deflection travel greater than 87% left and greater than 80% right should be obtained to satisfy flight control input requirements. The redundancy management would require a further screen that the RPTA outputs be within 9% of each other at full deflection. These newly established limits are compatible with OMI tolerances limits.

The final conclusion is that rudder pedal system operation during the STS-6 mission was normal for the cabin configuration tested.

A request to the Flight Operations Directorate has been initiated that in-flight checkout tolerances be changed to open the full throw tolerance limits to 85%, but leaves the null readings at the RM limit of 9%. Additionally the left shifts of 5% for null and right travel extension and the 2% left shift for left travel extension should be applied to the orbit checkout. The tracking of channel to channel for each RPTA to be within the RM limit of 9% should also be imposed.

A subsystem by subsystem review should be conducted on ascent only or entry only equipment in the pressurized compartment for damage/deterioration of performance as a result of physical shifts of this magnitude. Questionable findings should be verified with the structure group to determine if the concern is even applicable to the specific location(s) involved.

D. A. Johnson
RPTA Anomaly During STS-6 Orbit Checkout (Cont.)
Page 3

T. L. Brown, Tech Manager
Systems
Flight Systems Design
and Performance

TLB:cbl

| | | | |
|----------------|--------------|------------------------|-----------------|
| CC: W.B. Fouts | | R.M. Holt 288-203 AC36 | B.E. Antletz |
| R.A. Weaver | 286-300 FA41 | J.A. Hansen | P.C. Richardson |
| W.B. Owens | 286-001 AA91 | M.J. Kronfly | J. Bowen |
| B.M. McAnally | 288-200 AC36 | L.A. Bartley | E. Tulridy |
| J.G. Ripley | 288-203 AC36 | M.A. Mavers | P.A. Gureid |

ANOMALY J-35 IDR 7V-0169C.S.

The STS-6 on-orbit checkout of the rudder pedal transducer assemblies (2) produced an out of tolerance low reading for right pedal deflection test.

Data Summary:

| | | OV 099 | | OV 102 | |
|------------------------------------|-------|-----------|------------|-----------|------------|
| | | left RPTA | right RPTA | left RPTA | right RPTA |
| Ground checkout null | (DEG) | .05 | .05 | .2 | .2 |
| Ground checkout left throw max | (DEG) | -22.3 | -21.9 | -23.0 | -22.3 |
| IN-Flight checkout left throw max | (DEG) | -22.6 | -22.7 | -23.02 | -23.3 |
| IN-Flight checkout right throw max | (DEG) | 20.5 | 20.3 | 21.4 | 21.3 |
| Ground checkout right throw max | (DEG) | 21.6 | 21.7 | 22.2 | 22.3 |
| In flight checkout null | (DEG) | - .8 | - 1.1 | - .6 | - .6 |
| <hr/> | | | | | |
| Right Pedal decrease at max | (DEG) | 1.1 | 1.4 | 0.8 | 0.8 |
| Left Pedal increase at max | (DEG) | .3 | .8 | >0.02 | >0.3 |
| Null Shift from Ground | (DEG) | -0.8 | -1.1 | -0.8 | -0.3 |

Analysis Summary:

Structural data for shift due to no external pressure on the crew compartment was extracted from the prediction model and confirmed the RPTA linkage shift required to achieve the in-flight checkout results. The shift is considered normal for pedal system location and mounting method.

Conclusion:

The rudder pedal system for OV-099 operated normally during in-flight checkout, but checkout tolerances did not account for: (1) lower than nominal pedal throw rigging and (2) the null shift and limit shifts due to cabin motion.

Corrective action to change the checkout to tolerances to be:

- R Pedal max >80%
 - L Pedal max >87%
 - null between 14% Left and 4% Right
- < 9% difference between RPTA channels

This change opens in-flight checkout to be compatible with OMI tolerances and protects the rudder pedal transducer redundancy management for RM tolerance exceedances of 9% channel to channel. The Flight Control gain structure provides maximum rudder deflection with inputs 85% in the near landing and rollout regime.

The Charles Stark Draper Laboratory, Inc.

SSV-90-010
EGC-90-070

To: Ken Lindsay/EH2
Rick Hieb/CB
Jennifer Newman/FR25
Jean Zophy/IBM

From: Doug Zimpfer

Date: March 6, 1990

Subject: Failure of THC Hot Stick CR To Respond to All THC Commands

Copies: Ray Barrington/EH
George Bull/D31
Duane Johnson/EH2
Kim Kirchway/MS #4C
Mark Nakano/Rockwell
Lamar Bolling/RSOC-R12A
Darryl Sargent/MS #4C
Brad Schoenbauer/DF621
Charlie Thomas/EH2
Bob Williams/Honeywell
Bob Wu/Rockwell

SSV Distribution

In December 1989, a series of tests were completed in SAIL to verify nominal onorbit FCS performance of the OI 8D software release. The orbit DAP CR's that had been implemented in OI 8D were tested at this time. The results of the test case to verify correct implementation of the "THC Hot Stick" capability (CR89685A/CR89795C) indicated that it did not work as desired for all THC commands. The intent of the CR was to cause the DAP to upmode to PRCS for all THC commands, then mode back to the previous jet mode (NORM/VERN) upon completion of the translation firings. From the SAIL testing it was found that the DAP did not upmode to PRCS for a second THC PULSE command, if the first pulse had completed and the THC had not been cycled back through detent. The purpose of the memorandum is to document this anomalous behavior to the onorbit community.

This anomalous "Hot Stick" behavior was discussed with various members of the onorbit software community including the Principal Function Manager (K. Lindsay), SSD (J. Newman), CR Author and Crew Office (R. Hieb). The results of these discussions were that an immediate code fix is not required, nor is a USER's NOTE required. Therefore, since the anomaly results from a requirements deficiency, no DR will be opened. Design of a CR fix is in work at CSDL and will be submitted into the system as soon as possible. The current plan is to submit the CR in time for the OI 22 baseline. The current crew workaround simply consists of returning the THC back to detent between all short translation firings. This will ensure that all translation commands are honored.

The scenario seen in SAIL was as follows:

DAP in VERN Trans:PULSE/PULSE/PULSE (Pulse size = 1.0 ft/sec/pulse)
THC out-of-detent in the +Y direction and held
DAP responded by moding to NORM and commanding the +Y translation jets
When the translation had completed the DAP correctly downmoded to VERN
Second THC command in +Z direction (THC still out-of-detent in +Y)
The DAP did NOT mode to NORM and the translation request was ignored

For this scenario the DAP would have been expected to re-upmode to NORM and command the +Z translation, then mode back to VERN. The "Hot Stick" logic worked as expected for all other moding tested, including the case where the second THC command was issued prior to completion of the first. The logic also downmoded correctly at the termination of the translation firings.

A review of the requirements for this CR indicates the problem. The OI 8D FSSR PCN gives the "Hot Stick" upmode logic requirements as:

THC_HOT_STICK = ON if RCS and [THC] = 0 -> ≠ 0 and PRIMARY_VERNIER_PBI = OFF

THC_HOT_STICK = OFF if TVC or ([TRANS JET CMD] ≠ 0 -> = 0 or PRIMARY_VERNIER_SW = ON -> OFF or PRIMARY_VERNIER_PBI = ON) or OFC_RESTART = ON

Currently, a SAIL IDR (DIT96451) is open to document this undesirable "Hot Stick" performance. Since this anomaly results from a requirements deficiency and the crew feels no USER's NOTE is required, the plan is to not open a DR. Therefore, the SAIL IDR is the only open paper work documenting this discrepancy. The crew position on this anomaly is that a CR fix is required, but not at the risk of a patch or impact to the current software schedules. It is the onorbit verification team's intent to close the SAIL IDR in reference to this memorandum and implement a software change as soon as it is feasible.

As stated, an anomaly was found in the performance of the THC "Hot Stick" CR. This anomaly will be corrected by a future CR to the orbit DAP FSSR. Between now and implementation of this future change the logic will not correctly perform for non-simultaneous THC pulse inputs unless the THC is returned to detent. This necessitates that the crew take the THC back to detent for a period of at least 160 msec between THC pulses. The upmode continues to work correctly for the TRANS_ACCEL mode. No additional documentation will be opened against this anomaly unless desired by other members of the onorbit flight software community.

SECTION 14 ASCENT DAP

14.1 GENERAL DESCRIPTION

The ascent digital autopilot (DAP) is the software module that controls the vehicle in response to automatic steering commands during the ascent flight phase by generating position commands for the Space Shuttle main engine (SSME) and solid rocket booster thrust vector control (SRB TVC) actuators. Other functions performed by the ascent DAP include hinge moment load relief for the elevons and reaction control system (RCS) roll control if two SSME's are failed.

14.2 DETAIL DESCRIPTION

The DAP is comprised of seven separate software modules. These are TVC, main propulsion system (MPS) command processing, SRB command processing, RCS command processing, aerosurface control logic, reconfiguration logic, and linear interpolations logic. The reconfiguration and linear interpolations modules provide support functions of bookkeeping and table lookups for the DAP, whereas the TVC and command processing modules contain the basic control logic. The aerosurface control logic is used to hold the aerosurfaces in place and to provide hinge-moment load relief.

14.2.1 Thrust Vector Control

The TVC module is a feedback (FDBK) control system that contains attitude and rate control loops. During the first stage there is also an acceleration loop based upon an I-loaded reference acceleration profile. Inputs to the TVC module are attitude errors and rate commands. A generic control loop with attitude FDBK, rate FDBK, and acceleration FDBK is shown in figure 14-1.

The TVC pitch loop, shown in figure 14-2, generates body axis pitch errors that are to be flown out by the SSME's and SRB's (first stage only). The pitch loop contains gain scheduling as a function of relative velocity (VREL) that gives emphasis to acceleration error during the high dynamic pressure region ($547 < VREL < 2600$ ft/s). At a VREL of 547 ft/s the load relief logic becomes effective and will actually fly the vehicle away from the planned attitude profile, if necessary, to relieve loads on the vehicle that could be caused by high winds and wind shears not accounted for in the Day-Of-Launch I-Load Uplink (DOILILU) wind profile design. Flying away from the desired trajectory is accomplished by increasing the gain in the acceleration error loop and decreasing the gain in the attitude error loop. Load relief events are considered rare, and have not occurred more than 5 times in the history of the Shuttle program.

In the attitude error loop, which can be considered to be the forward loop, the pitch attitude error (in degrees) is multiplied by a scheduled gain that converts it to an attitude rate error (in degrees per second). This error is then limited to ± 10 deg/s due to structural constraints on the vehicle.

The attitude rate error is then summed with the rate feedback loop which consists of the difference between the average pitch rate measured by either the SRB or Orbiter rate gyro assemblies (RGA's) and the pitch rate command generated by guidance control (GC) steering. This combined error signal is filtered and then summed with the acceleration FDBK loop.

The first stage pitch feedback has an acceleration feedback loop: the measured normal acceleration from the body-mounted accelerometer assemblies (AA's) differenced with an I-loaded reference acceleration profile. The AA's, however, are not mounted at the center of gravity (c. g.) of the vehicle and, therefore, a compensation term must be added in so that rotational acceleration will not be sensed along with the normal acceleration. This is done by bringing in the pitch RGA signal, gaining and filtering it, then adding it with the acceleration error. The resultant signal, a compensated acceleration error, is filtered and then gained before it is added to the rate error signal that was generated from the inertial measurement units (IMU's) and RGA's. The gain that the acceleration error is multiplied by is a sensitivity gain, comparable to the gain that multiplies the pitch attitude error in the forward loop. These gains play together, and as can be seen in figure 14-2, beginning at VREL = 547 ft/s, the pitch attitude error gain decreases and the acceleration gain increases. This is how load relief works; increasing sensitivity to acceleration error and decreasing sensitivity to attitude error so that the vehicle will actually steer off course in order to reduce acceleration error if the unanticipated wind shear is strong enough (i.e., the wind shear that is not accounted for in the DOLILU design).

The pitch forward loop also contains an autotrim loop. It consists of an integrator and a limiter that act on the pitch attitude error. The integrated error, limited to $\pm 1.41^\circ$, is summed downstream of the forward loop gain so as not to lose the maximum trim value in the gain multiplications (the forward loop gains are decreasing with time because of decreasing inertia). Autotrim is provided to correct for thrust misalignment and small steady state errors. Note that the software switch indicates that autotrim is cut off when VREL = 547 ft/s (the beginning of load relief). The output of the integrator is held constant through load relief to help reduce loads by not "trimming," then the signal is ramped to zero prior to SRB separation (SEP). The logic is then turned on again 12 seconds after SRB separation and continues until main engine cutoff (MECO).

The feedback loop output is then filtered to form a pitch rate error. This error and the integrator output are sent to the appropriate Command Processors. If the Orbiter is still in first stage, then both the SRB and MPS Command Processors receive these data; otherwise only the MPS Command Processor receives the signals. In either case, the rate error signal is multiplied by a gain that is a function of time, a forward loop gain. Allowing the gain to change with time is an accurate way of tracking the greatly changing inertia of the vehicle. These forward loop gains, one for MPS, the other for SRB's, change the units of the signal from degrees per second to degrees, thus making the signal a body axis attitude error signal to be flown out by the MPS and SRB's.

The yaw axis control loop, shown in figure 14-3, is very similar to the pitch loop. One difference occurs in the module that compensates for the accelerometers' being offset from the center of

gravity. Yaw and roll rates are both needed to correct for this. One other difference occurs when mission elapsed time (MET) reaches 90 sec. The yaw accelerometers are used to aid in controlling the vehicle in yaw as it goes through SRB tailoff and prepares for separation. This signal, called lateral acceleration feedback (Ny FDBK), is then ramped to zero when MET reaches 124 sec or when SRB chamber pressure (Pc) becomes less than 110 psi. Also a Tower Clearance Bias is included into the yaw rate error if the left SSME fails between liftoff and MET 2.5 seconds. The bias of -1.5 degrees is added to yaw to keep the left SSME away from structure as the Orbiter clears the launch tower. Normally the SSMEs are sent to a failed position that is away from the other engine bells, but this is a necessary performance sacrifice, that yaws over all the SSMEs for a short time.

The roll axis loop, shown in figure 14-4, is different from the pitch and yaw loops in that accelerations are not used for load relief. Roll has a unique way of performing load relief. The goal of load relief in the roll channel is to maintain the thrust vector in the desired trajectory plane while the yaw channel is simultaneously trying to reduce the sideslip angle. The yaw channel becomes more sensitive to the yaw acceleration error (by increasing the gain) and less sensitive to yaw attitude error (by decreasing the gain) during the load relief region. By reducing loads, the orbiter flies into the winds that are causing the wing loading problem. But, by flying into the winds, (shears, gusts, etc.), the end result is that the orbiter actually flies out of the planned trajectory plane. In other words, if the orbiter first yaws the vehicle toward the wind, it reduces loads but rotates the thrust vector out of the desired plane. However, if the software subsequently rolls the vehicle, it is able to place the thrust vector back into the desired plane. In the load relief region (between a VREL of 547 and 2628) the yaw attitude error is gained higher than the roll attitude error so that the vehicle will roll the thrust vector back into the desired plane proportional to how much it had yawed it out of the plane.

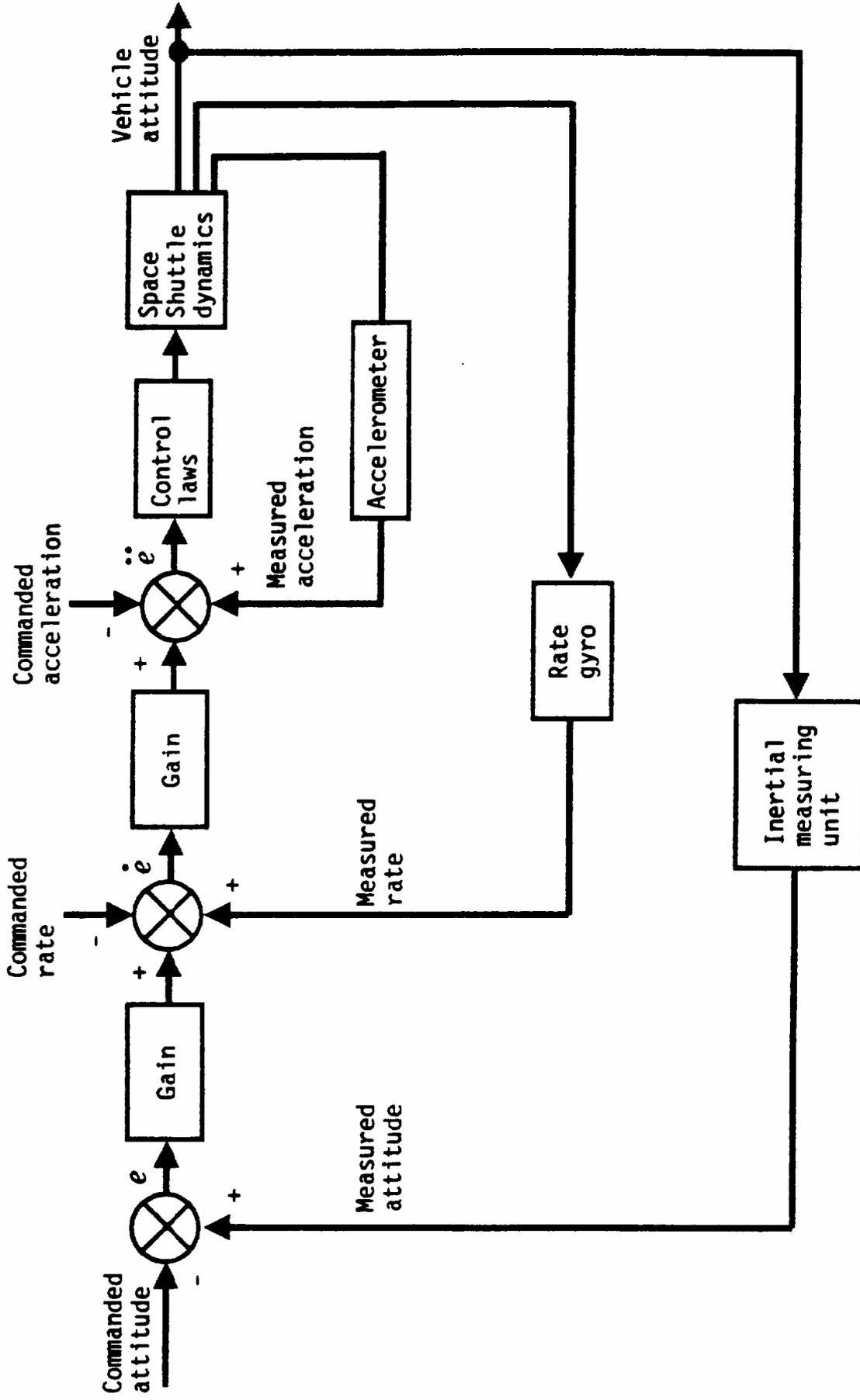


Figure 14-1. Generic control loop

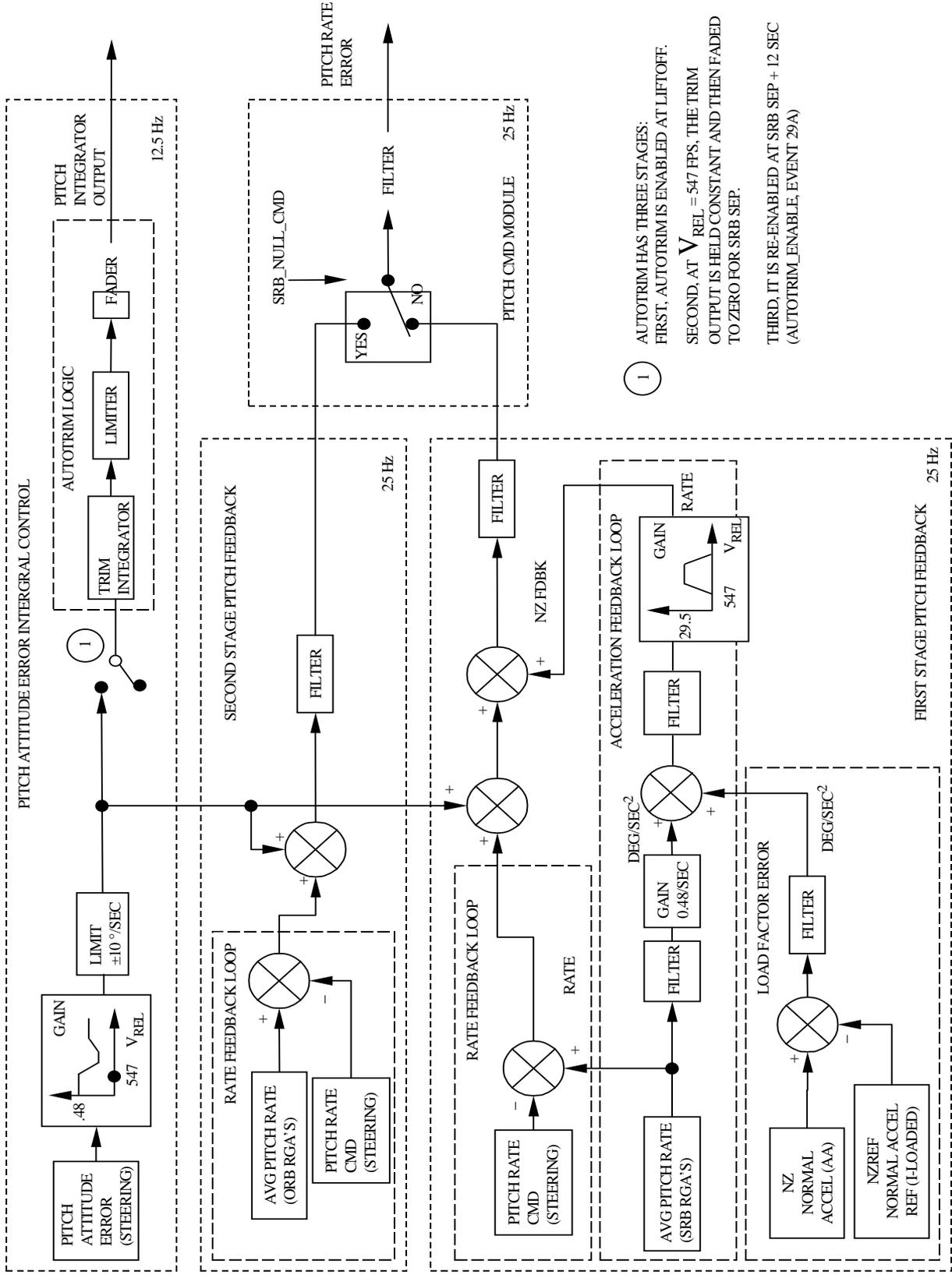


Figure 14-2. Pitch control loop

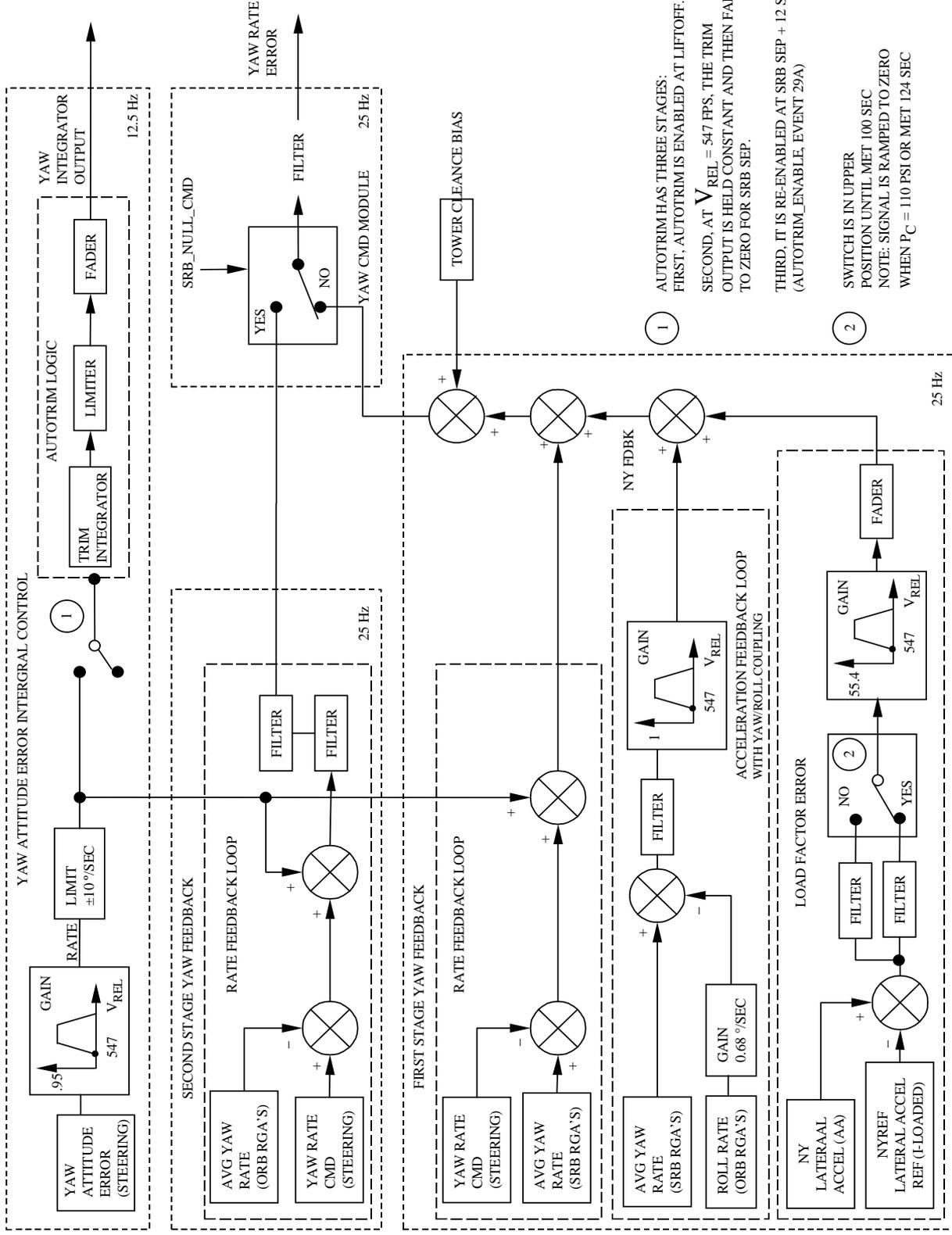


Figure 14-3. Yaw control loop

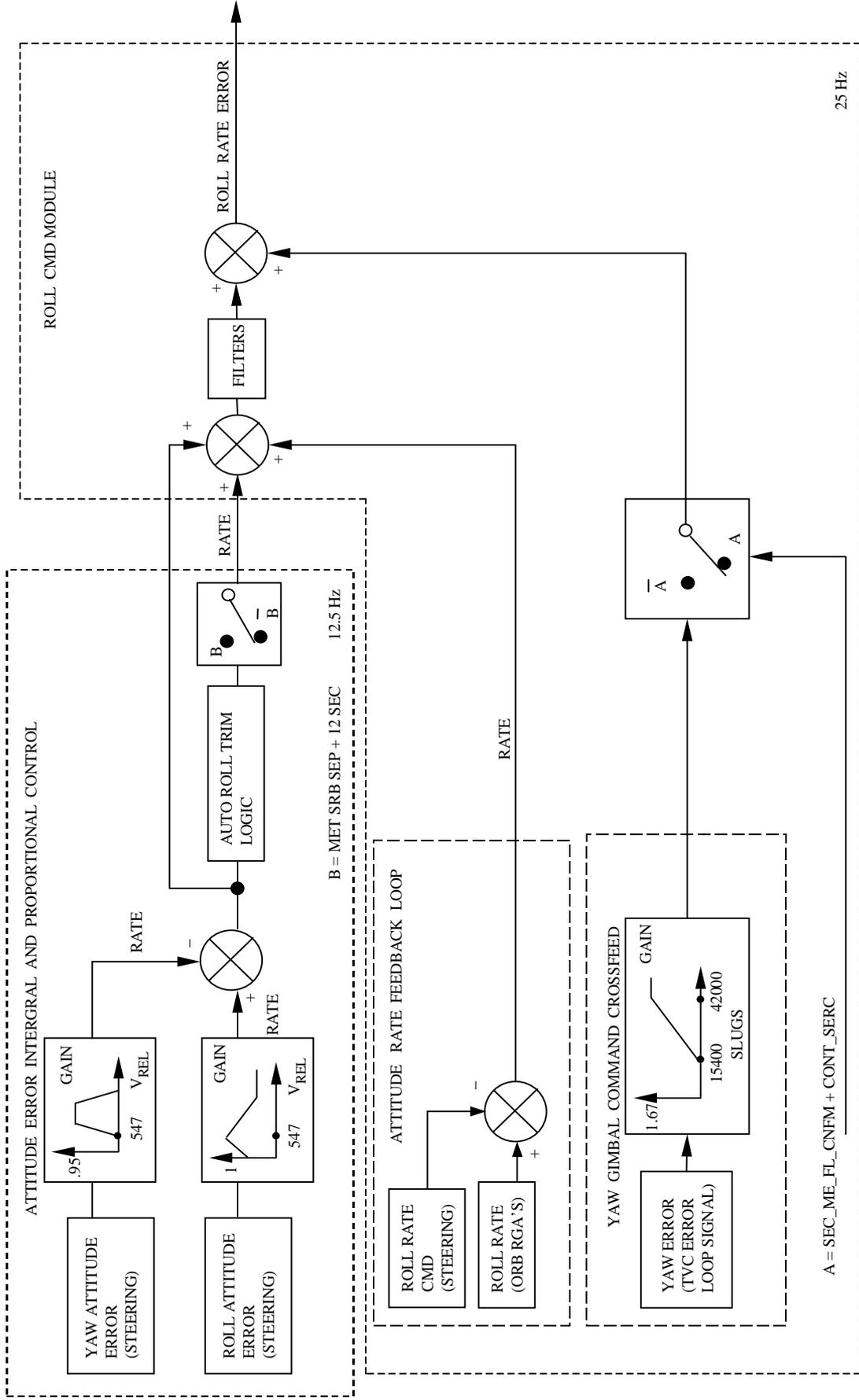


Figure 14-4. Roll control loop

14.2.2 MPS Command Processor

The major function of the MPS command processor is to convert the body axis attitude error signals that are generated in TVC into pitch and yaw engine bell deflection commands. Pitch and yaw here describe engine bell deflection axes, not body axes. In figure 14-5 a top, rear, and side view of the main engines (ME's) is shown. In the rear view, the arrows shown correspond to engine bell deflection in response to positive or plus actuator commands. Note that the pitch sign convention for the left ME is the opposite of the other two. This is because of the physical location of the actuator. It is mounted so as to allow the ground crew to access the plumbing and other hardware in the rear of the vehicle. The sign convention problem is rectified by changing the sign (multiplying by a minus sign) in the MPS command subsystem operating procedure (SOP). It is the center and right engines, however, that have their sign changed; and one can see why it is done that way when one has learned a little more about the command processor.

Figure 14-6 is a functional view of the MPS command processor. The errors on the left are from TVC. Addressing the lettered boxes in order, starting with box A1, we see that the roll body axis error passes through mixing gains and limits. The mixing is required because to generate a vehicle roll, the engines must be deflected in some combination of (engine) pitch and yaw deflection. Therefore, the logic in box A1 converts the body roll error into the appropriate combination of engine bell pitch and yaw errors. This appropriate combination is dependent on the number of SSME's that are operating (i.e., not failed) and multiple hydraulic system failures. When all three are operating, the roll error is converted to a left and right ME pitch error and a center ME yaw error.

This is done by multiplying by various gains. This concept is best explained in chart form (fig. 14-7). The roll mixing gains on the left column each multiply the roll error from TVC. The resulting signals become pitch and yaw engine bell deflection errors due to body axis roll error. Looking back at figure 14-6, the body axis pitch and yaw errors come in from TVC and pass through as ME deflection errors in pitch and yaw after being limited (box B1). The limits are there to keep the amplitude of the error signal within a specified region. Limits are given on page 14-16.

The engine bell deflection errors are defined for body axis roll, pitch, and yaw errors; a pitch and yaw error for body pitch and yaw error, and a pitch and yaw error for body roll error. These errors must be combined (mixed) to form engine bell pitch and yaw deflection commands, box C1. This is done with the following matrix:

$$\begin{bmatrix} 1 & 0 & 0 & 0 & 1 & 0 \\ 0 & -1 & 0 & 0 & 0 & 1 \\ 0 & 0 & -1 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 & 0 & 1 \\ 0 & 0 & 1 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 & 0 & 1 \end{bmatrix}$$

The matrix relates the computed errors to ME deflection commands. However, the sign notation of the matrix is such that the signs are reversed so that when the matrix is used, deflection commands are generated. In order to compute the six (engine bell) deflection commands, the matrix is multiplied by the six error signals. It looks like this:

$$\begin{array}{|l}
 \text{CENTER ME PITCH CMD} \\
 \text{CENTER ME YAW CMD} \\
 \\
 \text{LEFT ME PITCH CMD} \\
 \text{LEFT ME YAW CMD} \\
 \\
 \text{RIGHT ME PITCH CMD} \\
 \text{RIGHT ME YAW CMD}
 \end{array}
 =
 \begin{array}{|cccccc}
 1 & 0 & 0 & 0 & 1 & 0 \\
 0 & -1 & 0 & 0 & 0 & 1 \\
 \\
 0 & 0 & -1 & 0 & 1 & 0 \\
 0 & 0 & 0 & 1 & 0 & 1 \\
 \\
 0 & 0 & 1 & 0 & 1 & 0 \\
 0 & 0 & 0 & 1 & 0 & 1
 \end{array}
 \begin{array}{|l}
 \text{III} \\
 \text{I} \\
 \\
 \text{II} \\
 \text{IV} \\
 \\
 \text{V} \\
 \text{VI}
 \end{array}$$

The errors (designated by Roman numerals) correspond to those shown in figure 14-6.

These six errors are combined appropriately and the proper sign convention applied. That is where the matrix comes in. There are six equations generating the appropriate deflection commands. Below is an example.

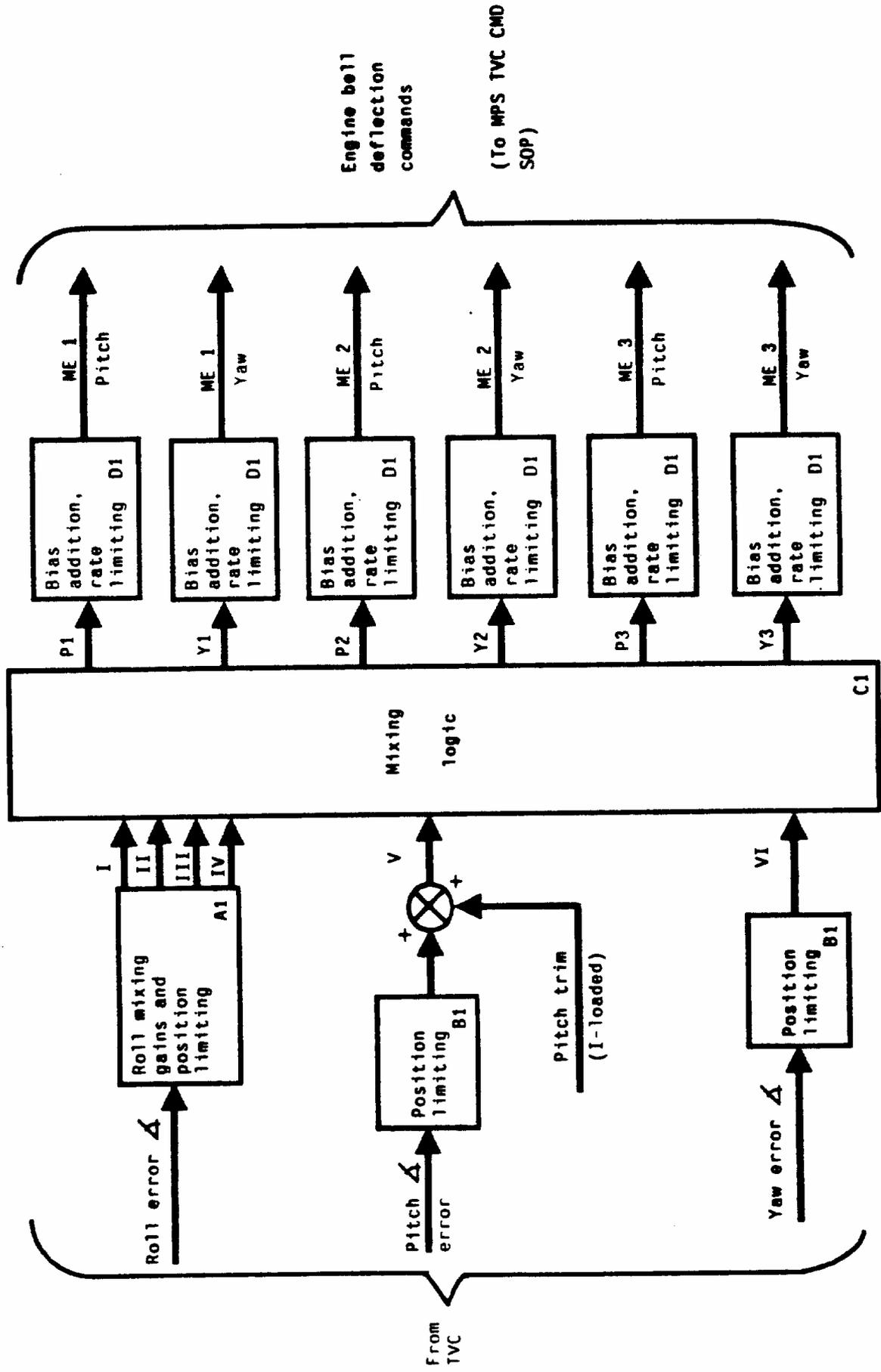


Figure 14-6. Main propulsion system command processing.

EXAMPLE: Assume that the roll and yaw error from TVC are both zero. Next assume that the actual vehicle attitude in pitch is +2°. The commanded vehicle pitch attitude is +6°. Therefore, the body axis attitude error in pitch, assumed generated from TVC, is actual minus commanded or 2 - 6 = -4°.

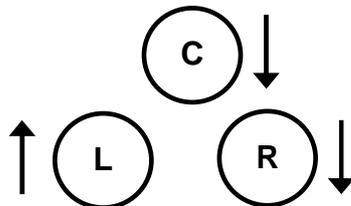
So the pitch error coming into MPS command processor is -4°. The mixing logic applies as follows:

$$\begin{array}{l}
 \left| \begin{array}{l} \text{CENTER ME PITCH CMD} \\ \text{CENTER ME YAW CMD} \\ \\ \text{LEFT ME PITCH CMD} \\ \text{LEFT ME YAW CMD} \\ \\ \text{RIGHT ME PITCH CMD} \\ \text{RIGHT ME YAW CMD} \end{array} \right| = \left| \begin{array}{cccccc} 1 & 0 & 0 & 0 & 1 & 0 \\ 0 & -1 & 0 & 0 & 0 & 1 \\ \\ 0 & 0 & -1 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 & 0 & 1 \\ \\ 0 & 0 & 1 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 & 0 & 1 \end{array} \right| \left| \begin{array}{l} 0 \\ 0 \\ \\ 0 \\ 0 \\ \\ -4 \\ 0 \end{array} \right|
 \end{array}$$

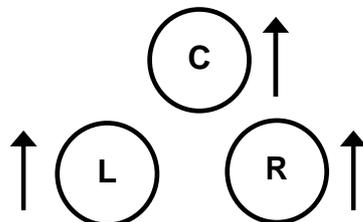
Therefore:

CENTER ME PITCH CMD = -4°
LEFT ME PITCH CMD = -4°
RIGHT ME PITCH CMD = -4°

These deflection commands correspond to the following motions.

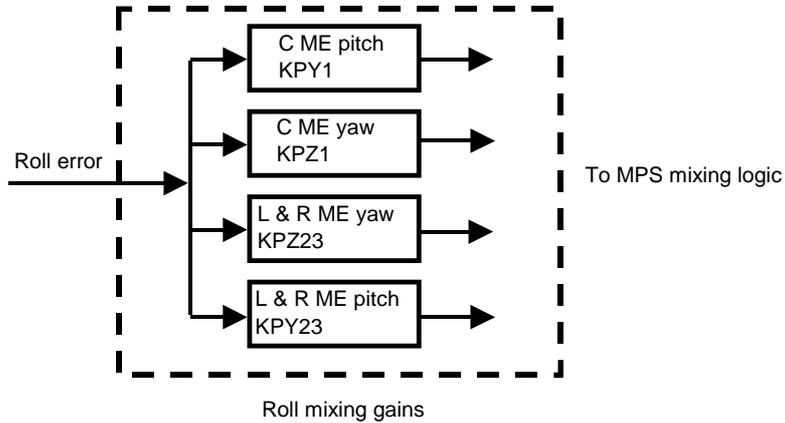


But recall that center and right pitch commands have the sign changed in the SOP; therefore, the deflections become:

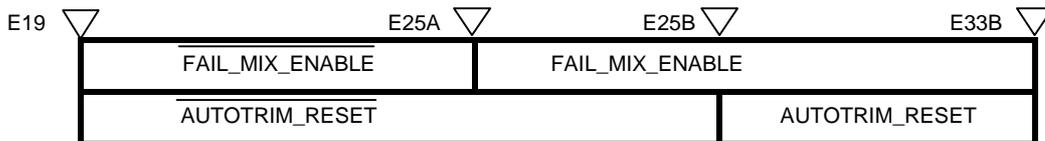


This creates a vehicle pitch up, which is the correct response to the guidance command.

The final column in figure 14-7 gives the roll mixing gain values when two ME's have failed. Center Pitch is zero, while all other gimbals have a very reduced roll gain. This means that while the roll error signal still goes to the remaining ME, it is a greatly reduced magnitude. The RCS command processor mainly handles the roll error, with this small amount of help from the last running ME (See Section 14.2.4, RCS Command Processor for more details).



| ROLL MIXING GAINS | $\overline{(\text{FAIL_MIX_ENABLE OR AUTOTRIM_RESET})}$ AND NO SSME FAILS AND HYD FAILS < 2 | (AUTOTRIM_RESET) AND NO SSME FAILS AND HYD FAILS < 2 | (FAIL_MIX_ENABLE) AND (SSME FAILS > 0 OR HYD FAILS > 1) | | | 2ND SSME FAIL OR SERC ENA |
|-------------------|--|---|---|--------------------------|--------------------------|------------------------------|
| | | | C SSME FAIL OR LOCKUP | L SSME FAIL OR LOCKUP | R SSME FAIL OR LOCKUP | |
| KPY1 | 0.0 | 0.0 | 0.0 | -2.5 | 2.5 | 0.0 |
| KPZ1 | 0.75 | 0.75 | 0.75 | 2.0 | 2.0 | 0.35 |
| KPZ23 | 0.0 | 0.0 | 0.0 | 2.0 | 2.0 | -0.35 |
| KPY23 | 1 | 2.5 | 2.5 | 2.5 | 2.5 | 0.35 |



- Event 19 LIFTOFF
- Event 25A PRE_SEP : MET > 107 SEC
- Event 25B AUTOTRIM_RESET : BOTH SRB PC'S < 380 PSI OR MET > 120 SEC
- Event 33B ME_ZERO_THRUST

Figure 14-7. Roll mixing gains

The blocks labeled D1 in figure 14-6 provide a bias addition and rate limiting of each of the six deflection commands.

One can observe what happens (in box D1) to the pitch command of the center ME and use that as the model since all the other commands are handled in the same manner. Refer to figure 14-8.

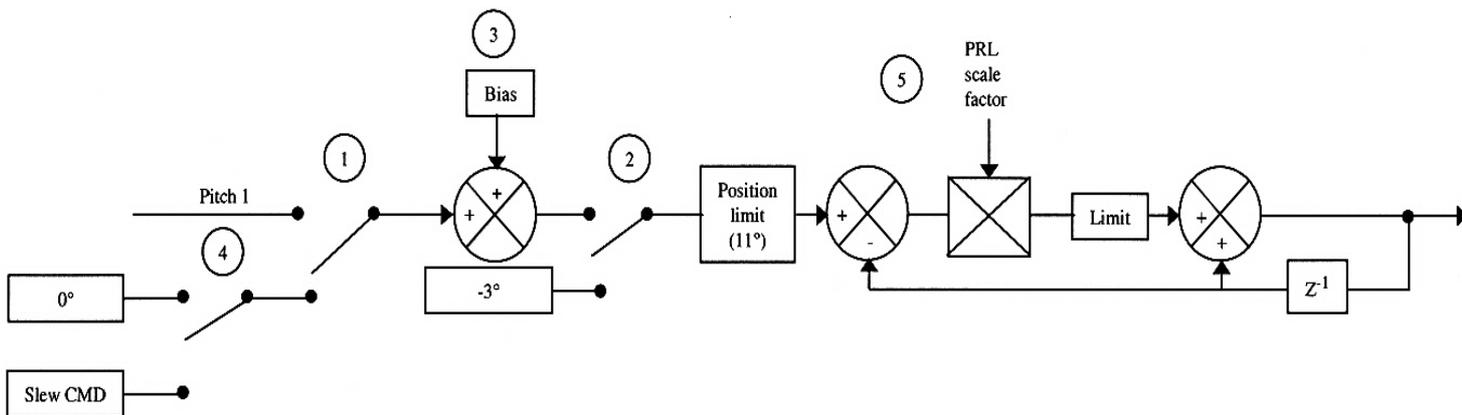


Figure 14-8. Command biasing and limiting.

Switch 1 is closed (control loop closure) when lift-off has occurred, approximately MET = 0.32 sec. (These are software switches.)

Switch 2 is normally in the upper position. If the center ME were to fail, the switch would go to the lower position which would send a -3° pitch position command (Failed ME's are put in the failed position, (-3, 0), (3, 3), (3, -3) for ME's 1-3, respectively, for the first failed ME only). Switch 3 will be discussed later.

The slew command, switch 4, is used prior to lift-off to test the actuators. The slew command is +2° and -2° deflection.

Eventually a pitch command comes out of switch 2 and into the logic labeled 5. This logic limits the rate of change of the command. Here is an example. Assume the previous pitch command was +7°. A second later, the command is -4° (fig. 14-9). (For simplicity, 1-Hz cycle time is assumed; however, the software is actually cycled at 25 Hz.)

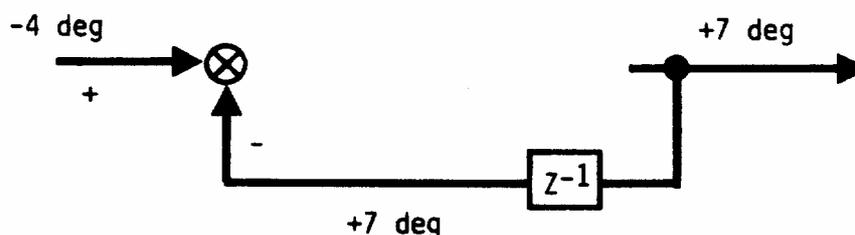


Figure 14-9. Command change

At the first junction is: $-4 - 7 = -11^\circ$.

See figure 14-10. (The priority rate limiting (PRL) scale factor is 1.0 in first stage.)

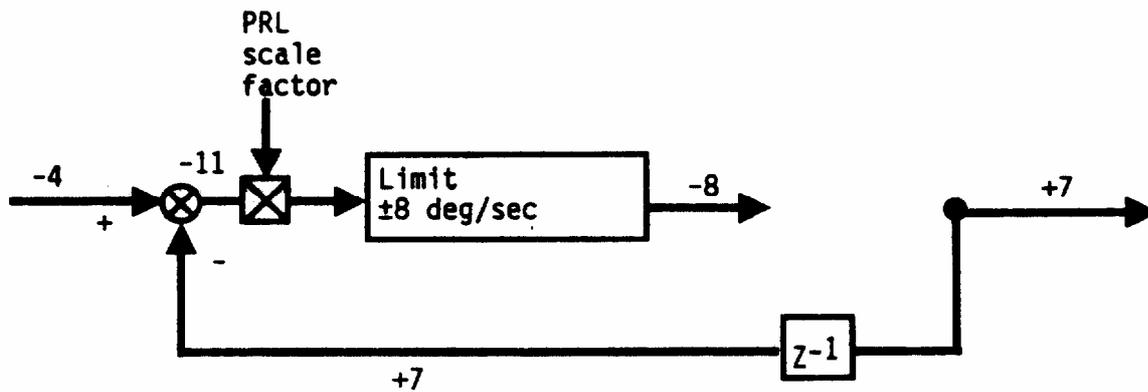


Figure 14-10. Command limiting

Then the rate limit is applied (limits depend on number of ME's failed and hydraulic system failed). Nominally, the limit is ± 8 deg/s. Since the -11° command exceeds the rate limit, only a -8° is passed through the limiter.

Finally, the signal is summed with the past value to generate the new command (fig. 14-11).

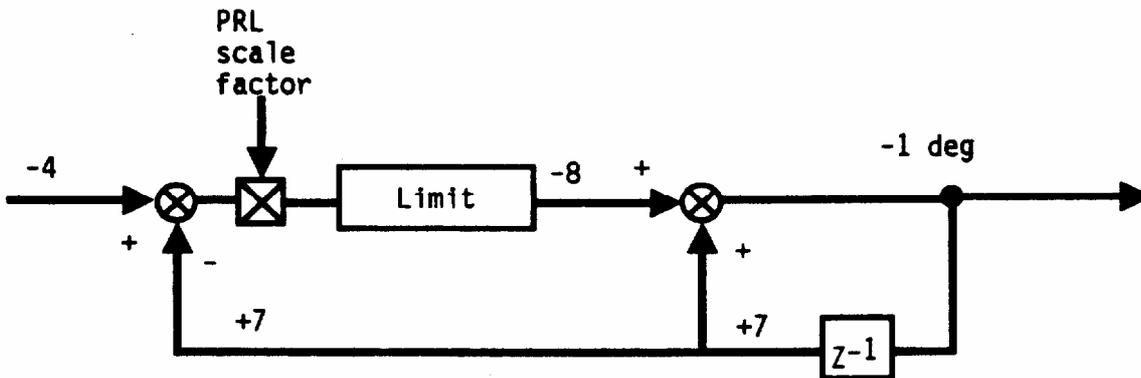


Figure 14-11. Generating new command

The new signal is -1° deflection in pitch, not the total -4° command since it would exceed limits. The slack in the command would be picked up in the next second. As was mentioned earlier, the rate limits depend on failures. The limits are:

8 deg/s -- No failures

5 deg/s - Single hydraulic system failure

4 deg/s - Two hydraulic system failures

2 deg/s - In event of ME failure, limit failed engine only

No software limits on engine deflection are necessary to ensure that the commands do not exceed the maximum size (10 bits in 2's complement) that can be used by the multiplexers/demultiplexers (MDM's). The software limits are $\pm 11^\circ$ in pitch, $\pm 9^\circ$ in yaw; the hardware limits are $\pm 10.5^\circ$ in pitch, $\pm 8.5^\circ$ in yaw.

The biases labeled 3 in figure 14-8 are used prelaunch to command the ME's to the start position. They are also used after lift-off through MECO in order to parallel or unparallel the null axes in pitch and yaw.

Note in figure 14-5 that the yaw null positions are offset from a line parallel to the body X-axis by $\pm 3.5^\circ$. To aid in thrust performance, a bias of 3.5° is added from the time the ME's are put in launch position until the MECO command. However, if the left or right ME fails, the bias is not added in second stage (between SRB SEP + 12 sec and MECO) so that total control authority is available. For any hydraulic failures or multiple SSME failures, these biases are removed. Additionally, in the no failure case, the bias is removed at MECO command.

The pitch axes (fig. 14-5) are not parallel to the body X-axis; they are canted up (Canting helps rid heating problems on the belly of the Orbiter and also c.g. tracking problems.) A small amount of thrust performance, therefore, is lost due to the angle generated by the null axis, during first stage. Twelve seconds after SRB SEP, if there are no SSME or hydraulic failures, all three engines are biased such that they are all paralleled to 12° above the Orbiter X-axis ($+4^\circ$ for C ME, and -2° for L & R ME). In the event of a left or right ME failure, and when in second stage, a bias is added to the center and remaining lower ME to keep the engines paralleled and burning through the Orbiter center of gravity, thus maintaining the performance increase gained by burning parallel (bias is $+3^\circ$ for the top ME and -3° for the lower ME.) The bias is then ramped to zero when the vehicle mass gets below an I-loaded value, which occurs near MECO command (See Appendix A, Extend Pitch Parallel to MECO).

One other function within the MPS command processor is the addition of an I-load pitch trim (fig. 14-6). The trim is referenced to VREL. The trim values are generated from simulation data. They take into account c.g. travel, winds, elevon movement, aerodynamic data, etc.

The pitch and yaw engine bell deflection commands for the three ME's are now generated. The commands next go to the MPS TVC command SOP where the sign convention in pitch is rectified and the commands are scaled properly and converted to voltages for use by the actuators.

14.2.3 SRB Command Processor

This processor works in exactly the same fashion as the MPS command processor. Functionally, it takes the body axis roll, pitch, and yaw errors from TVC, adds a pitch trim (I-loaded), applies SRB mixing logic to generate SRB rock and tilt commands. The SRB's gimbal axes are at 45° angles to the Orbiter body axes; therefore, the deflections are not pitch and yaw but are tilt and rock (fig. 14-12).

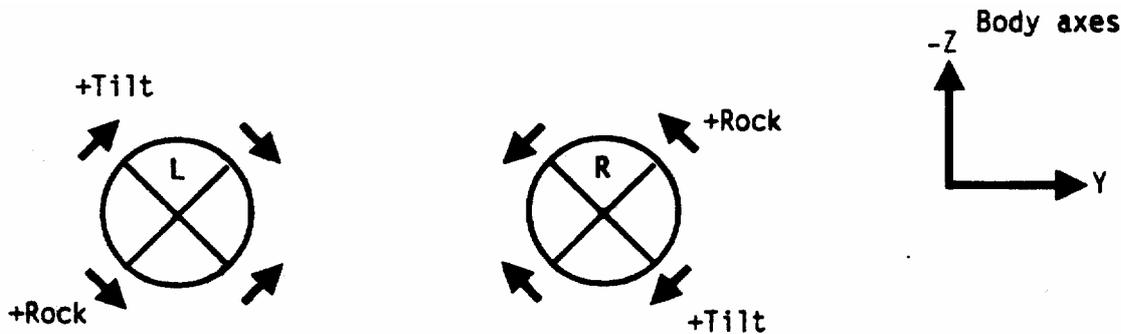


Figure 14-12. SRB rock and tilt commands.

Figure 14-13 is a functional view of the SRB command processor. Since it is so similar to the MPS command processor, a detailed discussion will not be presented. The DAP generates Orbiter body axis deflection commands, which are converted to actuator deflections in the Rock/Tilt frame (taking into account the 45° gimbal axis rotation) just before sending the commands to the actuator. The mixing matrix is shown in figure 14-14.

The rate limit on the command signal (in body axes) is 7 deg/s. This limit effectively limits the rock/tilt movement to 5 deg/s. The SRB TVCs are limited to 2° of movement until MET of 2.5 seconds, to prevent contact with the tower during liftoff. After this point, the limit is increased to 4° of deflection, per axis. These limits translate to 2.83° and 6.36° software limits. The difference is because of the rotated gimbal position.

There is a part of the first stage software that is in place to limit SRB TVC deflections near SRB SEP. Previous versions of this document (circa 1988) indicated that this was in place to alleviate structural contact concerns, but provided no supporting documentation of this concern. While this part of the software still exists, the limits are set to their nominal values, essentially making no change to the deflection limit. It is possible, however unlikely, that this limit could be changed in the future, as it is based on an I-loaded value. The following information is to assist in verification if it becomes necessary: CGCS_SLIM_TAIL are the SRB nozzle limits that are set at the COMLIM4/COMLIM5 INIT event (Event 25C&D).

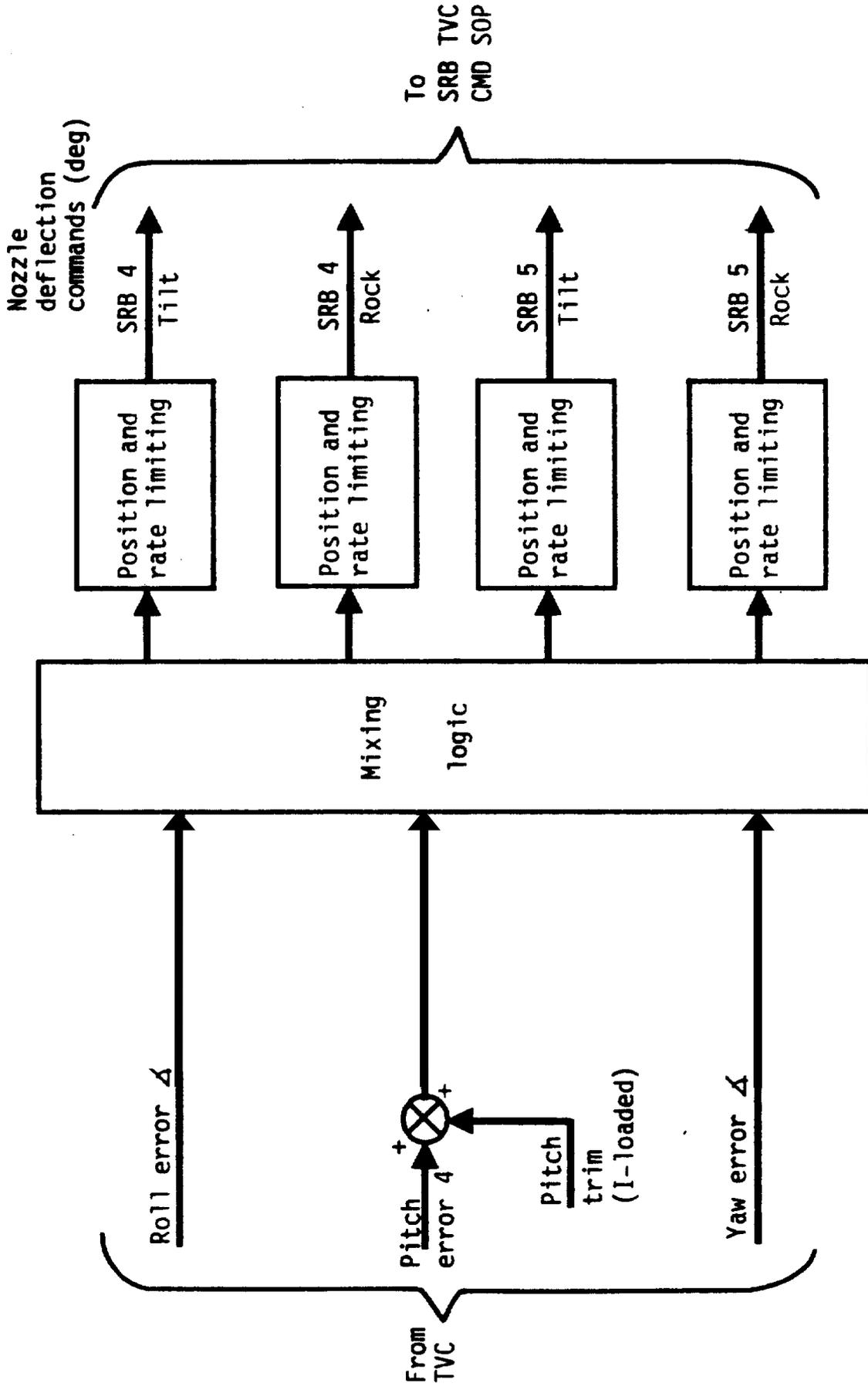
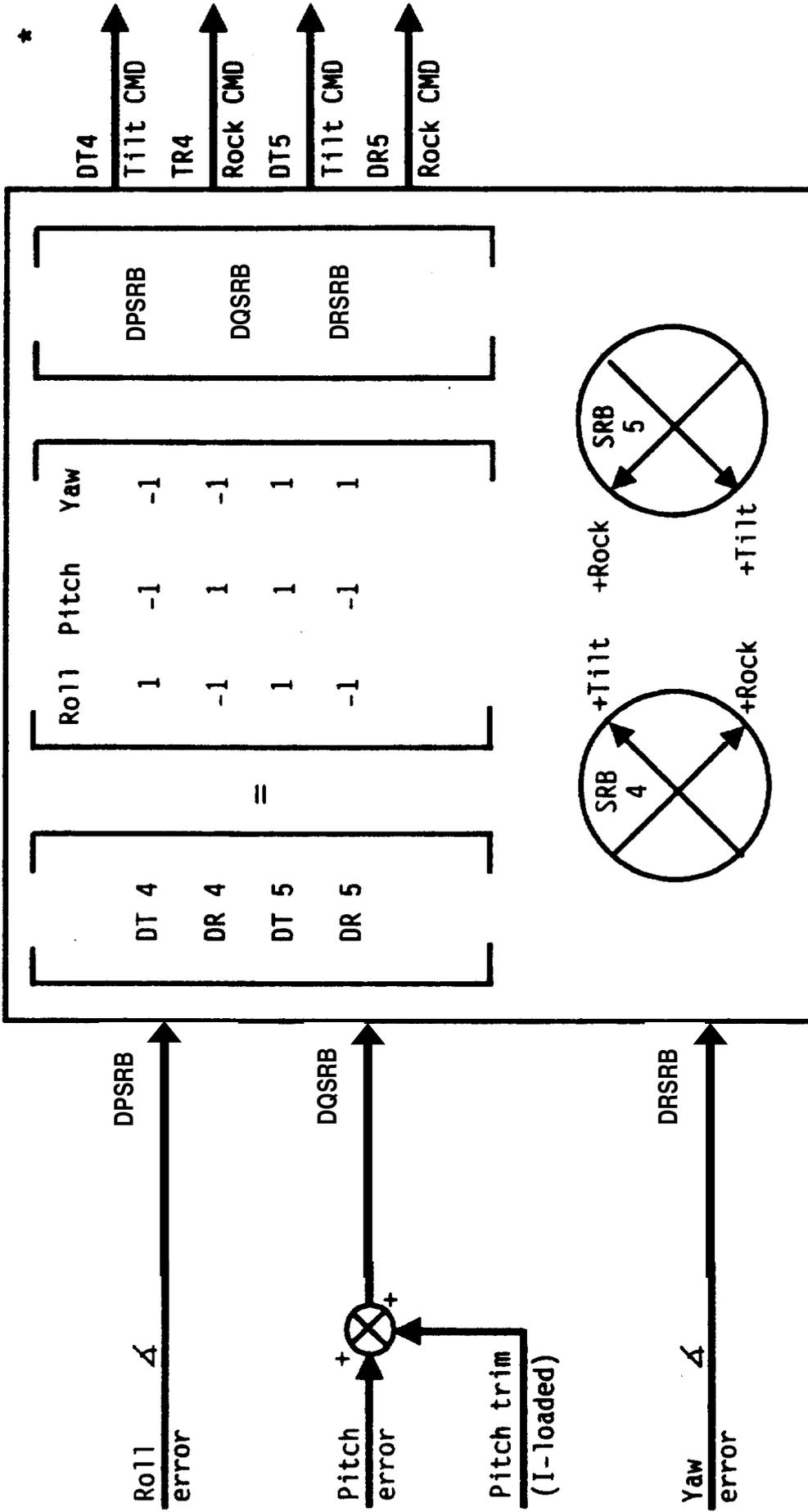


Figure 14-13. SRB command processing



*Outputs go through limiting then go to SRB TVC CMD SOP.

Figure 14-14. SRB command processing (mixing logic)

14.2.4 RCS Command Processor

The RCS is used to provide roll control during a single SSME maneuver (two SSME's failed in second stage). The roll error comes in from the MPS command processor and goes through a hysteresis loop with a deadband. If the error is outside the deadband, the appropriate RCS jets are commanded ON in order to null the roll error. It should be noted however, that the TVC generated roll error is still throughput to the MPS Command Processor and the SSMEs will still actuate in response to a roll error, albeit in a much more limited manner than normal SSME roll actuation (see figure 14-7). The RCS logic is shown in figure 14-15.

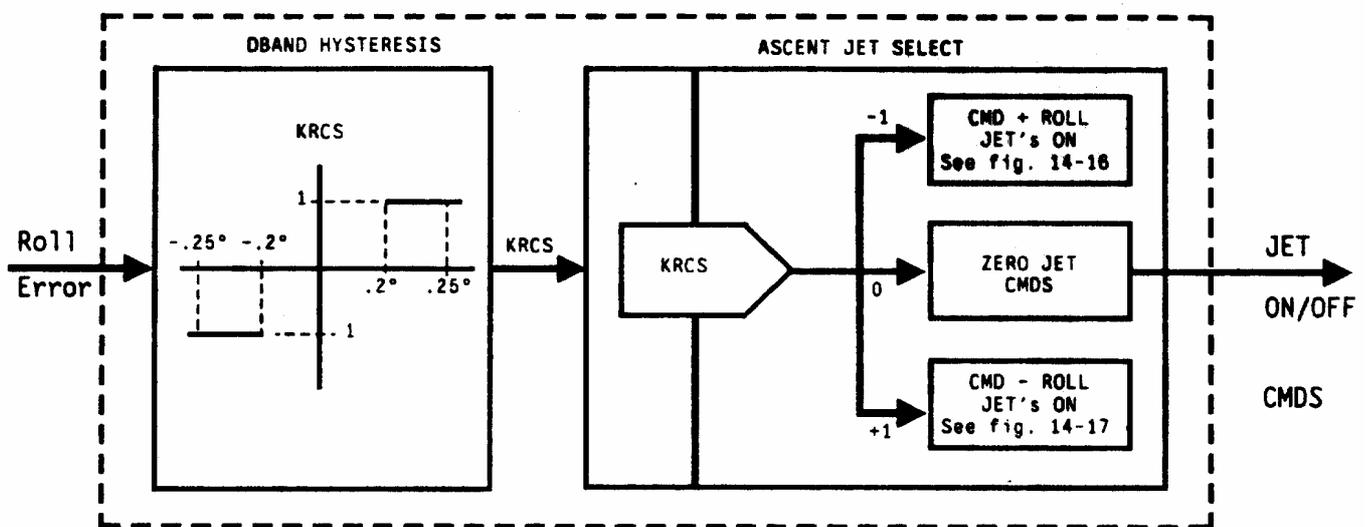


Figure 14-15. RCS roll logic

KRCS is computed as follows:

$$\begin{aligned}
 & \text{IF } |\text{Roll Error}| \geq |.25| \\
 & \quad \text{or} \\
 & \quad (\text{KRCS} = \text{Sign}(\text{Roll Error})) \text{ and } (|\text{Roll Error}| > |.20|) \\
 & \quad \text{then} \\
 & \quad \text{KRCS} = \text{Sign}(\text{Roll Error}) \times 1 \\
 & \quad \text{otherwise} \\
 & \quad \text{KRCS} = 0
 \end{aligned}$$

The RCS jet requirements for adequate Single Engine Roll Control (SERC) are defined in the Shuttle Operations Data Book (SODB). Section 4.5.1.1.1.3f, (Guidance, Navigation, and Control Subsystem, Vehicle Control Requirements) "Single engine roll control", identifies two separate sets of conditions for SERC capability. The first set of conditions is SRB SEP, and RTLS and Contingency Aborts. The second set is Post-SRB SEP, nominal insertion, TAL, and AOA Aborts. Below are the requirements for each:

SRB SEP, and RTLS and Contingency Aborts - 24 roll control jets are required

- 12 aft pitch jets
- 8 aft yaw jets
- 4 forward yaw jets

Post-SRB SEP, nominal insertion, TAL, and AOA Aborts - 12 of the 24 roll control jets are required

- 6 plus roll
- 6 minus roll

SERC can be activated two different ways. The first is via a crew item entry on the ASCENT TRAJ specialist page (ITEM 6). The second method is the automatic activation performed by the software. The following is a discussion of the methodology and impacts of each separate manner of activation.

The crew has the ability to execute the item entry to activate SERC at any point during ascent, regardless of whether or not there currently exists a control requirement for the RCS jets to assist with controlling the orbiter's roll axis. When the ITEM 6 is executed, the software determines the current phase of flight. If it is still in the first stage, until slightly before SRB SEP, the software will do nothing, because the SRB's are able to provide roll control authority. When the SRB P_C's on both the left and right reach 380 psi, or at an MET of 120 seconds, and any time thereafter until MECO, the software will perform the following steps:

- disable the SSME's yaw to roll gimbal command crossfeed
- enable the OMS to RCS interconnect
- zero the SSME biases (pitch and yaw paralleling)
- set the SERC roll axis MPS mixing gains
- enable the RCS jets to handle all roll errors generated by the Ascent DAP TVC module

All actions but the first item, disabling the yaw to roll command crossfeed, are performed after a set delay (ranging from 2 to 6 seconds) to allow for the interconnect to execute successfully. It should be noted that everything except the interconnect can be reversed if the item entry is executed again, without the conditions for SERC being present.

The ASC_RECON module constantly monitors the SSME failure status throughout all of powered flight. At SRB PC of 380 psi, approximately 14 seconds before SRB SEP and through MECO, the software will enable SERC if sufficient SSME fail flags are present (i.e., Guidance recognized SSME failures). Additionally, during second stage, after SRB SEP, the DAP will perform the Second Stage Auto Recognition of Second Engine Out module. This software monitors changes in the acceleration of the orbiter if one SSME has failed or there is a data path failure for a SSME. If the orbiter acceleration changes by a certain threshold, the software assumes that a second SSME has failed and will engage SERC. This acceleration check is made every 0.02 seconds by the software. The threshold is approximately one and a half engines worth of rated SSME thrust at vacuum; meaning if the sensed acceleration drops below ~1.5 SSME thrust output, SERC will be engaged.

14.2.5 Aerosurface Control

This logic is responsible for holding the body flap and rudder/speedbrake in place during ascent. It also controls the elevons, which are driven during the load relief region to reduce hinge moments. Since all surfaces but the elevons are held in place during ascent, only the elevons will be discussed here.

There is an I-loaded profile the elevons follow during first stage. The elevons do not move in second stage. There are also primary delta pressure (ΔP) feedbacks that are utilized to perform real-time monitoring of the hinge moments just in case the I-loaded profile is not accurate. The elevons, when they are moved, are slaved. This means the left and right inboards are driven together, and the left and right outboards are driven together. The inboards are driven independent of the outboards, and vice versa. Whichever side, left or right, is experiencing the greatest load (evident by the higher primary ΔP), will determine if those elevons will enter the load relief logic.

Figure 14-18 is a functional view of the elevon logic. The same logic is done for the inboards and the outboards separately.

The left and right ΔP 's go through a hysteresis loop with a deadband. If the ΔP 's grow larger than the drive limit (2052 psi for inboard, 2226 psi for outboard) the scheduled elevon command is biased so as to relieve the hinge moment. When the loads reduce, the ΔP 's will reduce as well. While the ΔP 's are lowering, the bias will remain constant, until the return limit is reached (1723 psi for inboard, 1934 psi for outboard) and the bias is integrated out. To illustrate this, below is an example, for the Inboard elevons.

Referring to the lower portion of figure 14-18, assume one or both ΔP 's are large enough to be out in the drive region (>2052 psi). Therefore, a bias is added to the scheduled command to reduce ΔP . As long as one or both of the ΔP 's remain greater than the drive limit of 2052 psi, the bias is integrated over time. Finally, when both ΔP 's are in the dead zone, the integration stops. When the ΔP 's lower into the return region, the bias begins being subtracted from the scheduled command. A similar example would apply to outboard ΔP 's.

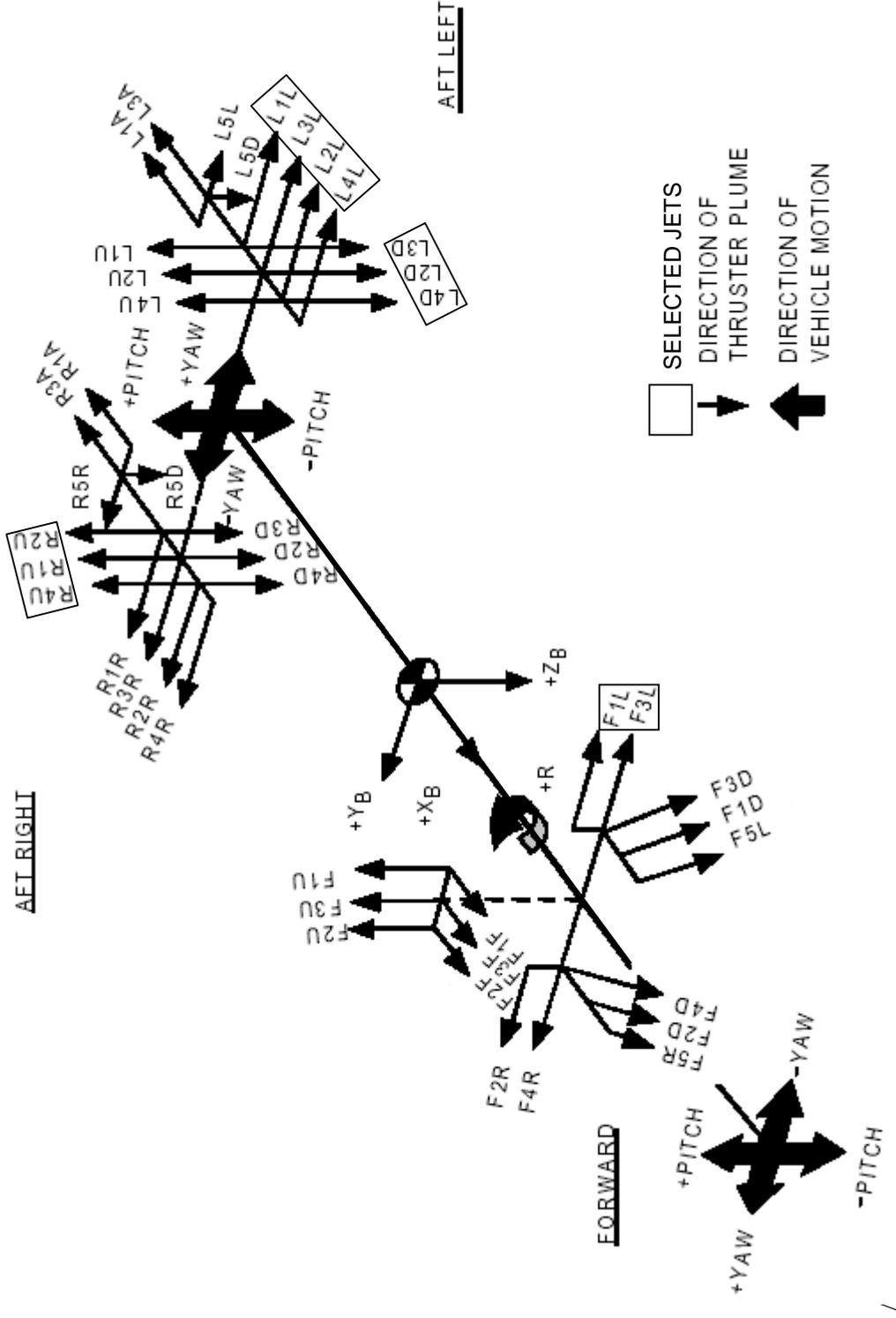


Figure 14-16. RCS plus roll jets for ascent

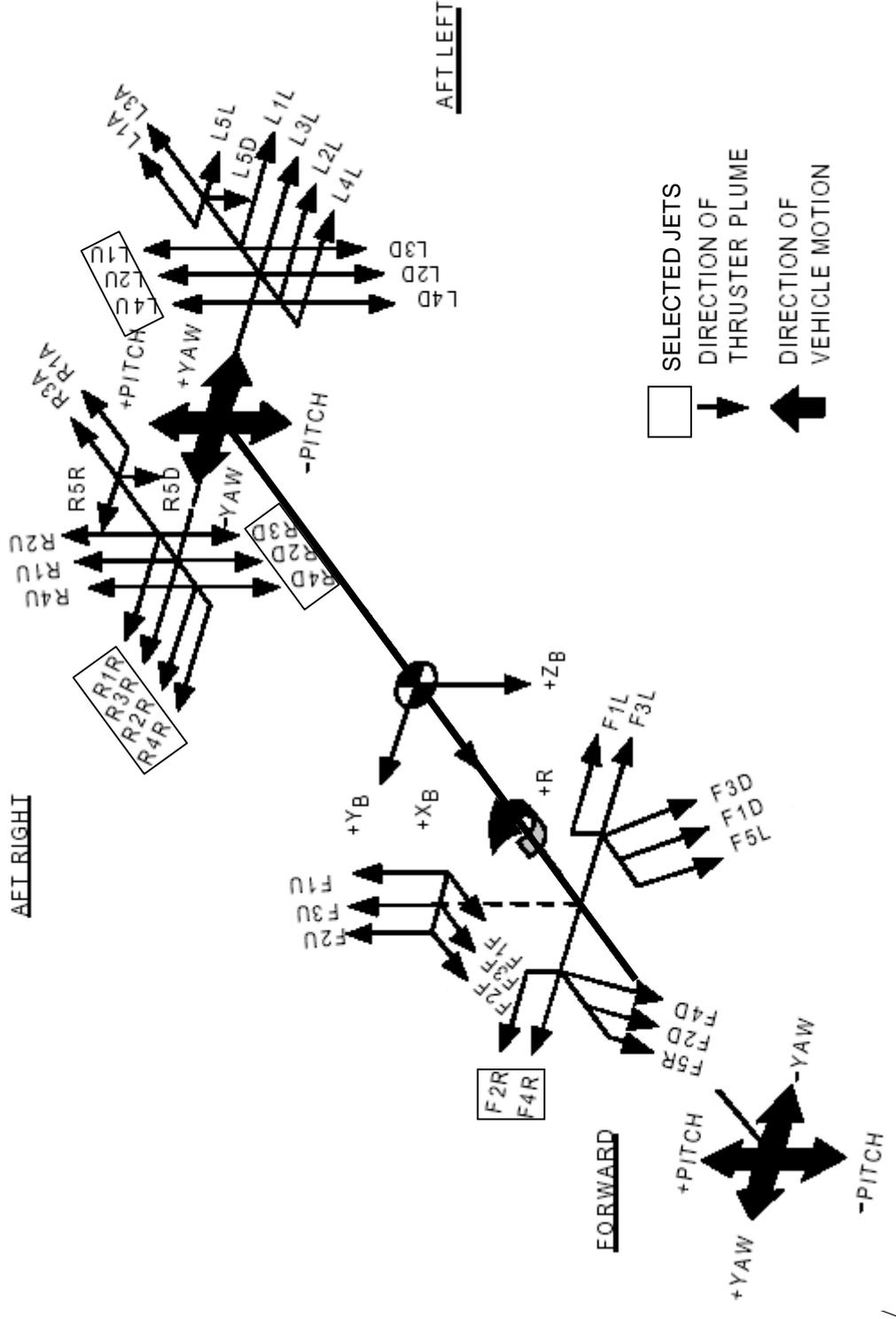


Figure 14-17. RCS minus roll jets for ascent

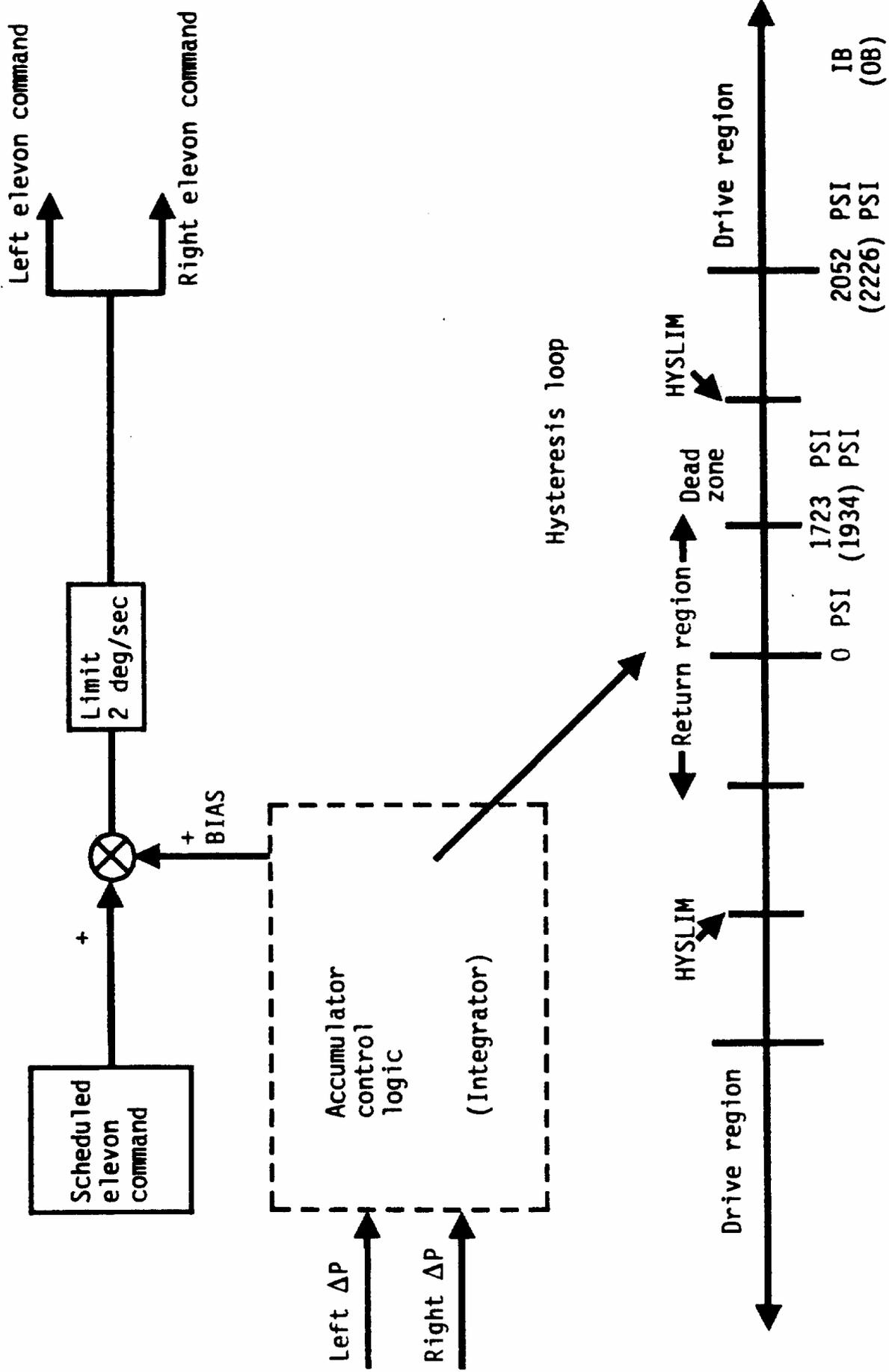


Figure 14-18. Active evelon hinge moment load relief

SHUTTLE OPS
GNC/JSC-18863

ORIG: D. WEILER
7/31/03: BASIC, REV C

ASCENT DAP
SB 14.0

Appendix A: OI-26A Performance Enhancements

The OI-26A Flight Software (FSW) build included Ascent Performance Enhancements (PEs) designed to increase performance during ascent. These FSW changes were implemented over the course of three Shuttle flights, STS-85, -86, and -87. STS-88 was the first flight of all the PEs as well as the first flight of the Super Lightweight Tank. The PEs included changes to the Ascent Flight Control and Guidance software and are detailed below. Note that "Second Stage Pitch Gimbal Relief (CR 91042)" is not a Performance Enhancement, but a necessary change brought out by the introduction of the Superlight Weight External Tank, that was included in the OI-26A FSW changes.

Constant Pitch Rate at SRB SEP (CR 91015)

The pitch steering command is propagated at a constant rate starting at 100 seconds MET in order to maintain an angle of attack near the design value in the SRB tailoff region. At MET of 100 seconds, the current guidance commanded pitch angle is captured and then propagated at a constant pitch rate (-0.44 deg/sec nominally and -0.50 deg/sec for an engine out prior to 100 seconds) for the next 4 seconds, effectively decreasing the pitch command (NOTE: 4 seconds is an I-loaded value, CGGS_EV25E_DELAY, that can change from flight to flight). After the I-load delay, the software will hold the pitch command constant, through staging, for increased performance and a more benign transient at second stage guidance initiation.

Extend Pitch Parallel to MECO (CR 91016)

Before the PE changes in OI-26A, pitch parallel biases were added to the SSME commands approximately 12 seconds into second stage and removed approximately 60 prior to MECO to avoid TVC saturation. This PE CR changes how the biases are removed. The pitch bias is gradually reduced in a linear fashion without sacrificing gimbal authority through the use of a fader. The bias plus the trim on the center engine is constrained by an I-loaded value and the biases for the lower engines are half of the center engine bias during this time frame in order to maintain moment balance. The biases are removed for any SSME or hydraulic failures.

Ascent DAP Trim for Performance (CR91018)

This CR adds the capability to reduce the included angle between the SSME and SRB thrust vectors during first stage. The SSME trim is scheduled versus time and an increment is added to the SRB trim to maintain moment balance. The SSME thrust vector is moved by adding a pitch trim increment to the main engines, PBIAS_E1 for the top and PBIAS_E23 for the side SSMEs. The SRB trim increment is added in the opposite direction. The trim is faded out prior to the SRB separation or immediately following an SSME or hydraulic failure.

Second Stage Pitch Gimbal Relief (CR 91042)

This provides automatic throttle down logic in major mode 103 to permit increased pitch control authority when a pitch gimbal command approaches full deflection. The throttle down reduces the deformation of the thrust structure and permits an increased effective thrust angle. This throttle down will occur if the following are true: (1) Manual throttles not selected (2) Vehicle mass less than an I-Loaded value (12,000 slugs, which is close to MECO) (3) SSME #1 has failed and (4) The pitch gimbal commands of either SSME #2 or #3 are exceeding an I-Loaded threshold (10°).

The Superlight Weight Tank causes the vehicle center of gravity to move about 6 inches further up into the orbiter (Z-axis) at MECO. This higher cg makes controlling the vehicle difficult with

SSME #1 failed. Without the change, large uncommanded pitch rates can occur near MECO when the orbiter cg is near the top of the allowable envelope.

SRB Sep Yaw Gain Enhancement (CR 91054)

This CR increases the separation clearances between the SRBs and the ET during staging. There is a possibility of negative clearance between the aft SRB strut and the External Tank when rates at SRB separation approach the limits set forth in Volume X. The Ascent DAP yaw forward loop gain, KN_MPS, is initialized at zero at SRB separation and then faded to the full value over four cycles of INTERPS. This effectively holds the SSME yaw command at its SRB SEP value while the SRBs move away from the External Tank.

SECTION 15 TRANSITION DIGITAL AUTOPILOT

15.1 OVERVIEW

15.1.1 Transition Digital Autopilot Functional Description

The transition digital autopilot (TDAP) is the module which controls the orbiter in response to either automatic or manual commands during the insertion (OPS 1) and deorbit (OPS 3) flight phases. Vehicle control during these phases of flight is maintained by the orbital maneuvering engines' (OME) thrust vector controller (TVC) systems and primary reaction control system (PRCS) jet firings. TDAP determines vehicle state by using attitude rate feedback from the rate gyros, and attitude feedback and velocity increments from the IMU via the attitude processor module.

Figure 15.1 shows a high-level diagram of the TDAP's major sections. TDAP consists of two major sections: the reconfiguration logic (TRNS_RECON) and the control laws. TRNS_RECON controls the moding, sequencing, and initialization of the control law modules by processing input received from other modules as well as orbiter hardware configurations. TRNS_RECON also sets gains, deadbands, and rate limits for the control law module. The control law section is composed of the steering, RCS, and OMS processors.

The steering processor is responsible for processing manual command inputs, providing commands for the -Z-axis separation maneuver at external tank separation (ET_SEP), and providing attitude error display information for manual steering in OMS TVC and guided manual RCS delta-V burns. The RCS processor generates RCS jet commands for desired vehicle translations and rotations using attitude and rate error signals from the steering processor or by using manual inputs from the commander's and pilot's hand controllers. The OMS processor generates the OMS engine actuator commands used to gimbal the OME's to produce the desired engine position/thrust direction using attitude rate commands from the steering processor, rate gyro feedback signals and initial engine trim values.

15.1.2 TDAP Availability

TDAP becomes active while the orbiter and external tank are still mated at main engine cut-off (MECO) confirmed in MM 103. The crew can verify TDAP is active by confirming the DAP panel lights on panel C3 are lit. The TDAP initializes in the RCS AUTO rotation mode with the discrete rate pbi's selected. The initial translation option is manual acceleration.

TDAP then holds attitude using RCS jets until external tank separation, executes the separation burn, then controls the orbit insertion OMS burns. TDAP is deactivated upon transition to MM201, and then later reactivated for the deorbit phase of flight (MM 301 - 303).

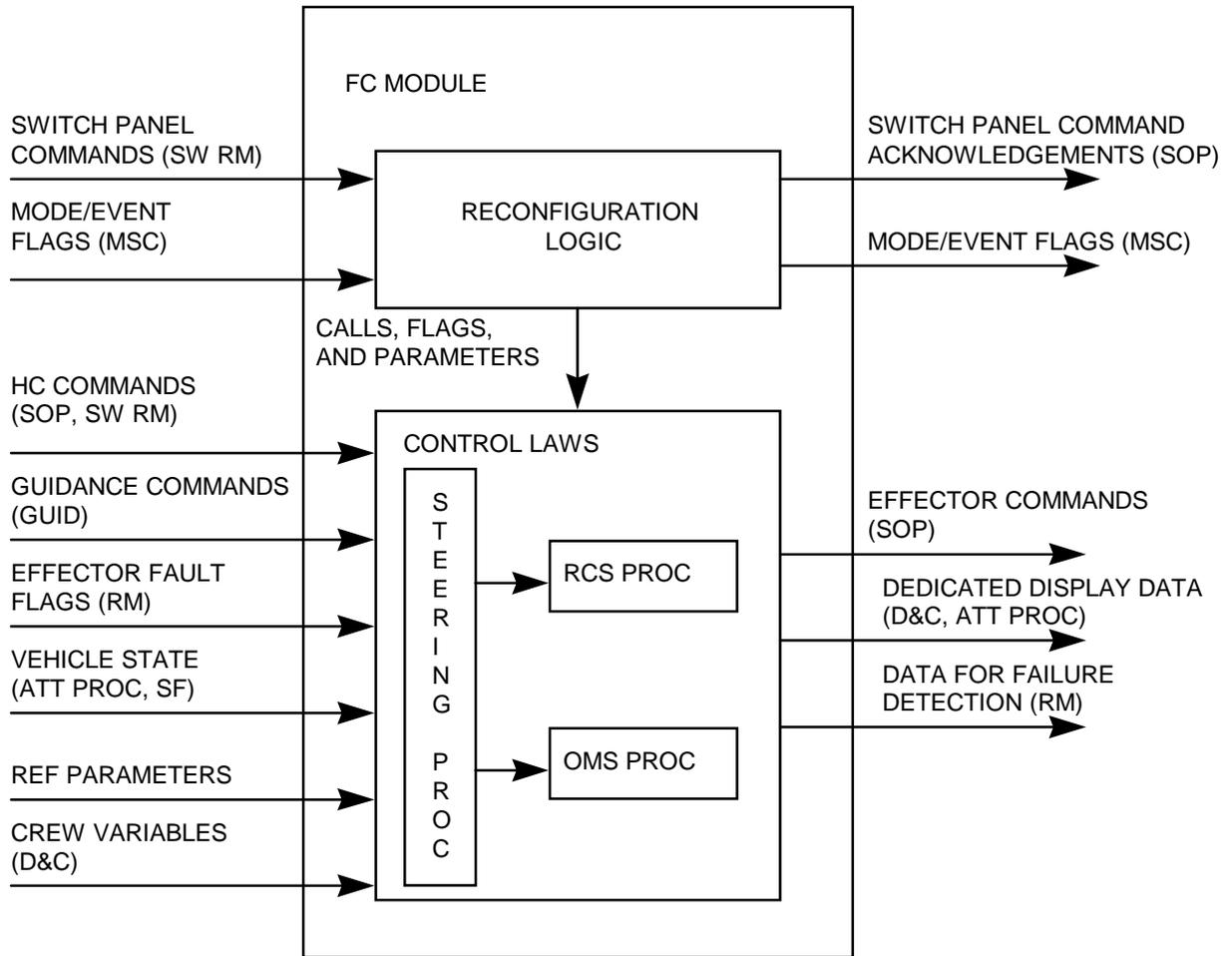


Figure 15-1. Major sections of the TDAP

15.1.3 TDAP Module Interfaces

The TDAP interfaces with the modules listed in Table 15-1.

Table 15-1. TDAP input and output functional parameters.

| Input Functional Parameters for TRANS DAP | Module | Output Functional Parameters from TRANS DAP |
|---|------------------|---|
| X | ASC MNVR DIP | X |
| | ASC RCS CMD SOP | X |
| X | A/E ATT PROC | X |
| | D&C PANEL | X |
| X | DEORB GUID | X |
| X | DEORB MNVR DIP | |
| X | ET SEP SEQ | |
| X | GN&C SW RM | |
| | HARDWARE (lamps) | X |
| X | MSC | X |
| X | OMS FIRE SEQ | |
| | OMS TVC CMD SOP | X |
| X | ORB INS GUID | X |
| X | ORB DAP | X |
| X | 3-AX RHC SOP | |
| | RCS ACT LT | X |
| X | RCS/RCS XFEED | |
| X | RCS RM | |
| X | SF | |
| X | XXXXX MNVR YYYYY | X |
| X | UL | |

15.2 TDAP DETAILED FUNCTIONAL DESCRIPTIONS

15.2.1 TRNS_RECON Logic

TRNS_RECON is responsible for calling the various modules of the TDAP in their proper sequence.

15.2.1.1 Moding and Sequencing

The TDAP logic is configured in various modes that are set by either the crew (manually) or the software (automatically). The software applies control discretets to the DAP at certain mission events to enable moding changes automatically. Manual control is activated by pressing the inertial (INRTL) pushbutton indicator (pbi) or deflecting the translation hand controller (THC) or the rotation hand controller (RHC). Illumination of the appropriate pbi(s) by the DAP indicates the mode that is currently active or (if applicable) preselected. AUTO, INRTL, DISC RATE (3), and PULSE (3) are the only pbi's that are active in the TDAP. Figure 15-2 shows the pbi layout and indicates which pbi's interface with the TDAP.

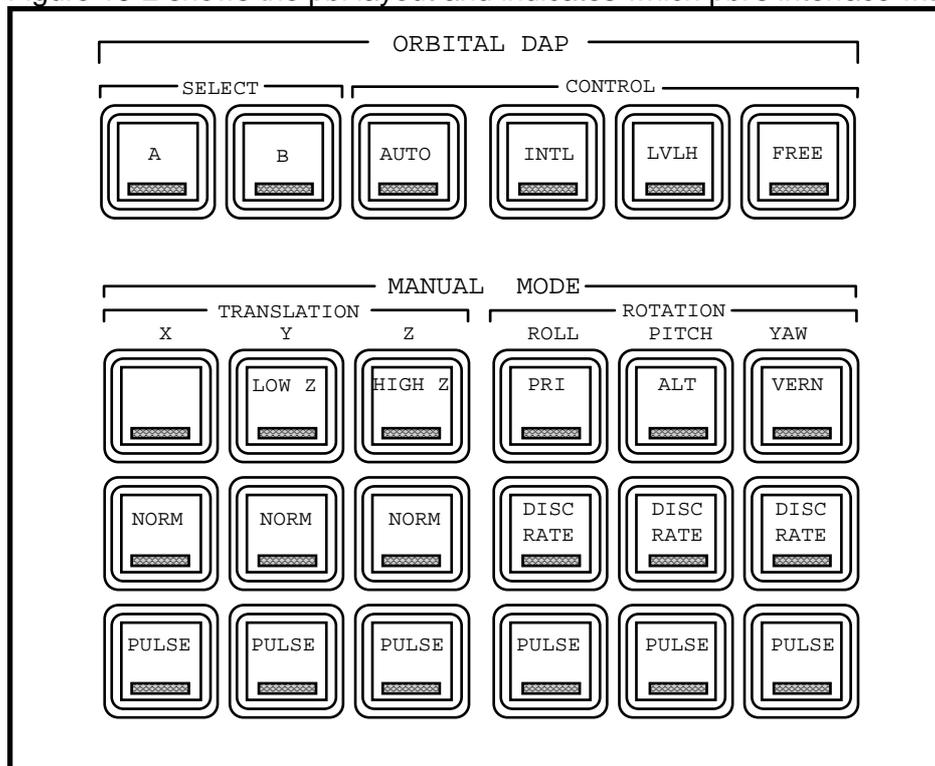


Figure 15-2. DAP panel

Attitude control in TDAP is performed using TVC attitude or RCS attitude or translation control modes. **The RCS processor runs during both modes, and translation control by the RCS processor is active at all times.**

15.2.1.1.1 TVC Attitude Control Mode

The TVC mode is active when an ON command is present for one or both OME's. If an ON command is present for both OME's, and one of those ON commands is turned off for any reason during the burn, the TDAP will reconfigure to a single OME burn status. Automatic and manual submodes are available in the TVC mode by depressing the AUTO/INRTL pushbutton indicators (pbi's), respectively, at any time. Additionally, the manual submode may be achieved by deflecting the RHC. TDAP will remain in the manual mode until the RHC is returned to detent AND AUTO is selected on the DAP panel.

The automatic submode is dependent on OME status (one or two engines burning) and AUTO pbi selection. The manual submode is dependent on OME status, INRTL pbi selection, and deflected RHC position, which generates the proportional rate command to the OME TVC. A proportional rate is commanded anytime the RHC is out of detent but inside the soft stop in all three control axes for a two OME burn, and in pitch and yaw for a single OME burn. These proportional-rate commands drive both the RCS rotation control and the OMS TVC. If the RHC is deflected past the soft stop in any axis, then that axis is downmoded to the ACCEL submode (other axes are not affected). RHC commands are passed directly to the JET_SELECT_TRANSITION module (part of the RCS processor), which generates the commanded primary jet firings. When in the ACCEL mode, the phase plane module (PHASE_PLANE_TRANSITION; also part of the RCS processor), is bypassed to preclude attitude hold jet firings.

During single OME burns, discrete rate command attitude hold will be provided for the roll axis only. If the RHC is deflected past the soft stop in the roll axis, it is downmoded to the ACCEL mode as described above. When the RHC is in the range between the soft stop and detent (upon returning from the ACCEL mode), free drift will be invoked about the roll axis until the stick is returned to detent and deflected again.

NOTE: The RCS processor will run during all active submodes of the OMS processor. Table 15.2 summarizes the TVC modes.

Table 15.2 TVC Attitude Control Modes Summary

| RHC Status (by axis) | Manual | | Automatic | |
|------------------------------|--|---|--|---|
| | 2 OME | 1 OME | 2 OME | 1 OME |
| In detent | Attitude hold | P/Y/R: Attitude hold | P/Y: Cross product steering R: Attitude hold | P/Y: Same as 2 OME R: Attitude hold |
| Between detent and soft stop | Rate commanded in proportion to RHC deflection | P/Y: Same as for 2 OME R: DSC RT attitude control | Mode becomes Manual | Mode becomes Manual |
| Past soft stop | OMS TVC command deg/s, RCS in ACCEL mode | P/Y/R: ACCEL mode | Mode becomes Manual | Mode becomes Manual |
| Between soft stop and detent | Rate commanded in proportion to RHC deflection | P/Y: Same as for 2 OME R: Free drift | Mode becomes Manual | Mode becomes Manual |

15.2.1.1.2 RCS Attitude Control Mode

The RCS attitude control (or rotation) mode is active in the absence of an OME ON command. Automatic and manual submodes are available in the RCS mode by depressing the AUTO or INRTL pushbutton indicators (pbi's), respectively, at any time. Additionally, the **manual** submode may be achieved by deflecting the RHC, which causes the illumination of the INRTL pbi. TDAP will remain in the manual mode until the RHC is returned to detent AND AUTO is selected on the DAP panel. The rotation mode pbi's (discrete rate and pulse) may be configured at any time, whether AUTO or INRTL is selected, however, will only affect the attitude control mode when in INRTL. When in **automatic** RCS rotation mode, TRNS_RECON selects a target attitude, and AUTO_MNVR (a steering processor module) generates the control signals to cause an attitude hold or maneuver to the selected attitude followed by an attitude hold.

When in the RCS attitude control mode, the TDAP will set the target attitude equal to the existing attitude when the mode changes from manual to automatic or vice versa if:

- a guided RCS burn is not occurring
- it is first pass insertion or deorbit flight phases
- ET_SEP is initiated
- an OMS burn has just been completed.

The reference attitude is expressed in the Mean of 1950 inertial reference frame, and is used in computing total roll, pitch, and yaw attitude errors for display on the ADI error needles. The error(s) is the difference between the reference attitude and the current measured vehicle attitude. Normally, the crew uses ADI error needle indications to guide manual maneuvering or steering, or to monitor the equivalent automatic operation.

For a set time interval (DRIFT_TIME) after the ET_SEP command is issued, there is no RCS attitude control (all three body rotation axes are in free drift). Attitude control is resumed at the expiration of DRIFT_TIME, and the attitude hold reference is the attitude of the vehicle at the time the separation was issued.

AUTO_MNVR also provides the coordinated three-axis maneuver to burn attitude (ITEM 24, 25, and 26 are R, P, and Y, respectively) function on the XXXXX MNVR YYYYY display (ITEM 27), as well as the ensuing attitude hold once in attitude. If the targeted burn attitude is modified or manual mode is entered (INRTL pbi selected, RHC moved out of detent) while the maneuver is in progress, the maneuver is terminated and an inertial attitude hold is implemented. To resume/initiate the maneuver, the AUTO pbi must be selected and ITEM 27 must be executed again. This operation is different from the orbit DAP where an ITEM 27 Exec while in INRTL will cause a maneuver to commence on transition to AUTO.

AUTO_ALPHA_BETA_MANAGEMENT software provides automated Alpha and Beta attitude management after MECO on nominal uphill, ATO abort and TAL Abort. In the case of a TAL abort the auto alpha/beta management sequence has two parts to it, a mated coast auto function, and a post ET_SEP auto function. Nominally, the auto alpha/beta management sequence will initiate after MECO when an I-loaded timer starts and the sequence attempts to achieve a targeted alpha and beta ($\alpha=20$ deg, $\beta=0$ deg +/- the TRANS DAP attitude deadbands) at a maneuver rate of 1.5 deg/sec. The timer is 8.62 seconds long and starts after MECO_CONFIRM and terminates (damps rates) after 8.67 seconds to allow ET_SEP to occur. At the expiration of the 8.62 second timer, if the desired alpha and beta have not been achieved, the sequence will pause and damp rates (to allow ET_SEP to occur). If the desired alpha/beta have been achieved during mated coast, the sequence waits until after ET_SEP, and checks that the vehicle is now at a target alpha and beta ($\alpha=28$ deg, $\beta=0$ deg +/- the TRANS DAP attitude deadbands). If it is at the desired alpha/beta, then the sequence terminates. If it is not at the desired post ET_SEP alpha/beta, it will perform a second auto maneuver at 5.0 deg/sec to achieve the target alpha/beta. Once the target alpha/beta is achieved, the sequence terminates and the TRANS DAP holds that desired attitude profile until the OPS 3 transition. In the Nominal or ATO case the Auto Maneuver attempts to achieve a pre ET sep Alpha of -2 deg and Beta of 0 deg at a maneuver rate of 0.8 deg/sec. This is designed to improve the ET sep clearance margins. No maneuver is performed post ET sep on uphill cases. There are two separate flags that enable this auto maneuver, one for TAL (I-load V97U2290C) and one for uphill (I-load V97U2269C). Currently (STS-121 time frame) they are both activated.

If either the OPS 3 transition is commanded, or CSS is selected prior to achieving the targeted alpha and beta on the TAL transition, then the auto maneuver sequence will terminate, and the current inertial attitude will be held by the TRANS_DAP. At this point, the auto sequence is no longer available and the crew must ensure that the required alpha and beta are achieved via manual maneuver on the RHC. Even if AUTO is reselected after termination of the auto alpha/beta maneuver sequence (via CSS downmode), the current Inertial attitude is held (i.e., reselecting AUTO after CSS downmode, does not override the crew downmode to CSS).

It is important that the crew ensures that the required alpha/beta targets have been achieved prior to the OPS 3 transition. The consequences of being outside the desired alpha/beta targets at the OPS 3 transition are 3-fold and include: (1) an increased risk of ET recontact/debris, or (2) loss of control due to a roll off once in OPS 304 under Aerojet DAP control (as documented in FSW DR 035047), or (3) delayed/incomplete OPS 3 OMS dump (CG/landing weight problems).

In the **manual** RCS rotation mode, TDAP may be configured axis by axis in the discrete rate attitude hold or pulse submodes. If the RHC is deflected past the soft stop, the ACCEL mode is entered for that axis. The previously selected mode will still be selected upon return of the RHC to detent, providing free drift about the downmoded axis while the RHC is between detent and the soft stop.

15.2.1.1.3 RCS Translation Control Mode

The TDAP is in a manual translation acceleration command submode at all times except during an automatic -Z maneuver (active from ET_SEP command (SEP_MINUSZ_CMD) to separation complete). If the THC is deflected during this maneuver, the automatic -Z commands will no longer be issued, and the manual acceleration mode will take over. The ET_SEP_PROCESSOR will run until the SEP_COMPLETE flag is set (desired separation velocity achieved). If the THC is returned to detent before the SEP_COMPLETE flag is set, TRNS_RECON will set the SEP_COMPLETE flag at the start of the ensuing OMS burn.

The manual acceleration mode is always available to the crew and generates a jet command for a given axis (as long as the THC is out of detent in that axis) in the direction of THC deflection.

Table 15.3 summarizes the RCS modes.

Table 15.3 RCS Attitude Control/Translation Modes Summary

| Option | How Selected | Effect |
|--|--|--|
| Rotation ACCEL mode | Anytime RHC is past soft stop, by axis | Rotation acceleration commanded as long as RHC deflected by axis; free drift when RHC moved back within soft stop but not to detent; return to pulse or discrete when in detent |
| Rotation pulse mode | Pulse mode selected by axis, RHC deflected but not past soft stop and INRTL selected | If RHC deflected, a preselected rotational rate impulse is produced (80 ms) by axis; free drift after pulse is completed |
| Rotation discrete rate mode | Discrete rate button selected by axis, INRTL selected, and RHC not past soft stop | If RHC is deflected, a pre-selected rotation rate is obtained (0.5°/s - I-load); attitude hold when RHC returned to detent |
| Automatic rotation maneuver | RHC in detent and AUTO selected (may also load desired attitude in maneuver display) | If a desired attitude has been loaded, or guidance is active, vehicle is rotated to desired attitude at predetermined rate (0.5°/s I-load) and then holds desired attitude, otherwise only holds attitude |
| Translation acceleration mode | Active at all times except during -Z external tank separation maneuver | Translation acceleration commanded as long as THC deflected by axis |
| Automatic -Z maneuver | ET_SEP -Z maneuver commanded by software | -Z acceleration until sensed separation velocity greater than or equal to the desired value or OMS-1 starts, whichever first. Any deflection of the THC during the Automatic -Z command will permanently inhibit the -Z command. |
| Automatic TAL Abort Alpha/Beta Attitude Management | Automatically occurs after ET Sep on a TAL. Commanded by software. | Auto sequence starts timer after MECO_CONFIRM and attempts to maneuver to target alpha/beta during mated coast. After timer expires, the sequence pauses to allow ET_SEP to occur. After ET_SEP, the auto sequence resumes auto maneuver if the target alpha/beta have not been achieved. After completion of auto maneuver, software holds target attitude to within desired attitude deadbands until OPS 3 transition. |

Table 15.4 RCS Rotation Modes

| RHC status (by axis) | Discrete rate | Pulse |
|--|---|---|
| In detent | Phase plane attitude hold | Free drift |
| Out of detent but not past soft stop | Phase plane attitude hold at predetermined rate trajectory (0.5°/s) | Predetermined rotation impulse provided (80ms); then free drift |
| Past soft stop | Continuous acceleration response | Continuous acceleration response |
| Between soft stop and detent, after having been past soft stop | Free drift | Free drift |
| Back in Detent | Phase plane attitude hold | Free drift |

15.2.1.2 Other TRNS RECON Functions

TRNS_RECON sets module (AUTO_MNVR, PHASE_PLANE, ROT_DISC, TVC_LAW_ROLL, etc.) initialization flags used by the control law modules; causes OMS gimbal trim; sets gains, deadbands, flags, etc.; keeps “clocks” which determine the proper time for calls to modules; and provides an interface to the DAP panel to determine pbi selection

15.2.2 Control Law Module

The configuration of the control law module is determined by inputs from the supervisory module TRNS_RECON. The control law function modules are grouped into three categories, steering, RCS, and OMS processors. The general sequence of processing is 1) steering processor and 2) RCS, or RCS and OMS, processor(s). Figure 15-12 shows the TDAP Control law module layout.

15.2.2.1 Steering Processor

The steering processor converts a variety of input commands and data to forms acceptable by the RCS and OMS processors. The steering processor consists of the following modules:

- One module for *automatic RCS rotation* (**AUTO_MNVR**)
- One module for *automatic RCS translation* and ET separation monitoring (**ET_SEP_PROCESSOR**)
- Three *manual RCS rotation* (**ROT_ACCEL**, **ROT_DISC_TRANSITION**, and **ROT_PULSE_TRANSITION**)
- One module for *manual RCS translation* (**TRANS_ACCEL_TRANSITION**)
- One module for *RCS error determination and attitude data processing* (**RCS_ERRORS_TRANSITION**)
- Two modules for *automatic OMS steering* (**OMS_STEERING_TRANSITION** and **GUIDANCE_COMPENSATION**)
- One module for *manual OMS steering* (**MANUAL_TVC**)
- One module for *rate gyro output filtering* (**RATE_FILTER**)

A brief description of each module is listed below.

AUTO_MNVR provides rate commands (0.5°/s) and attitude errors to the RCS error determination module (**RCS_ERRORS_TRANSITION**) to perform either an inertial attitude hold or a maneuver to a commanded pre-burn attitude followed by an inertial attitude hold. Maneuver or hold mode is determined by the magnitude of the difference between the actual vehicle attitude and the target attitude. If the initial mode is maneuver, the module computes the time needed to complete the maneuver, then modes to attitude hold at the end of this time. The **AUTO_MNVR** module supplies the RCS processor with a **FORCE_FIRE** flag, which acts as a rate damping command, in all three axes when starting or ending a maneuver to reduce angular rate error. It causes the jets to fire *immediately* upon an auto maneuver request and damps rates when in attitude. This immediate jet response is a result of the phase-plane switching logic initiating a firing to reduce the angular rate error to an acceptably small magnitude, at which time the phase-plane switching logic resumes normal limit cycle operation.

ET_SEP_PROCESSOR processes translation in only the vehicle Z-axis and monitors the separation velocity in both manual and automatic translation modes. Once the desired separation velocity has been achieved, the -Z translation command is zeroed and the **SEP_COMPLETE** flag is set. The module computes -Z translation velocity changes until a threshold of 4 ft/s is exceeded nominally or 11 ft/s for the transoceanic abort landing mode. The sensed velocity change is checked at 0.96-second intervals. The ET SEP is started automatically, and upon completion of the desired velocity change, -Z translation is terminated if the automatic mode has not been overridden.

ROT_ACCEL, **ROT_DISC_TRANSITION**, and **ROT_PULSE_TRANSITION** are modules used when in *manual RCS rotation mode*. Control modes (acceleration, discrete rate, and pulse) are selectable axis by axis. See Table 15.4 for a summary of RCS rotation modes.

ACCEL rotation mode will fire jets continuously as long as the RHC is deflected past the soft stop. **TRNS_RECON** makes the **FORCE_FIRE** request if the RHC was moved beyond soft stop and then back to detent.

Discrete rate rotation mode will fire jets, bypassing the phase plane, to generate a predetermined vehicle angular rate ($0.5^\circ/\text{sec}$) about that axis in the same sense of deflection while the RHC is out of detent in that axis. Normally, in the disturbance hysteresis region, a jet firing will not be initiated but only continued if jets were ON before entering it. Should the RHC be deflected or returned to detent when no jets are firing, and the rate error is less in magnitude than the rate limit, the state could lie in the disturbance hysteresis region, and no jets would fire until the state drifted out of that region (which would require minutes). The crew would then perceive the mode as having failed to respond. The **ROT_DISC_TRANSITION** module supplies the RCS processor with a **FORCE_FIRE** flag, causing the phase plane to immediately command jet firings opposite to the rate error, regardless of where in the phase plane the vehicle state is. This rate damping command is requested each time the RHC is deflected from detent, and each time the RHC is returned to detent (attitude hold).

The **ROT_DISC_TRANSITION** module also initializes the **RCS_ERRORS_TRANSITION** module at three events in the discrete rate attitude hold mode, setting the desired attitude equal to the present attitude. The three events are: 1) when the mode is first entered, 2) when the RHC goes out of detent, and 3) when the RHC returns to detent.

Pulse rotation mode will fire jets (80 milliseconds), bypassing the phase plane, to generate a predetermined vehicle angular rate *increment* about that axis while the RHC is out of detent in that axis.

OMS_STEERING_TRANSITION and **GUIDANCE_COMPENSATION** are modules used for automatic OMS steering. **OMS_STEERING_TRANSITION** commands a steering angle equal to the cross product of the desired and actual vehicle velocity vectors (vector difference (magnitude and direction) between the two vectors). The output of this module is scaled, filtered, and limited by **GUIDANCE_COMPENSATION**. The manual steering module, **MANUAL_TVC**, simply commands a rate proportional to the RHC deflection. A detailed discussion of OMS steering is included in 15.2.2.3 RCS and OMS Processor.

RCS_ERRORS_TRANSITION generates attitude error and angular rate error signals based on angular rate commands from steering processor modules, an attitude error signal from AUTO_MNVR, and vehicle measurement data. These attitude and rate errors are then sent to the RCS processor.

RCS_ERRORS_TRANSITION receives rate gyro data and attitude increment information through the attitude processor module, A/E_ATT_PROC. Commanded body rates and attitude are received from the DAP, and attitude and rate errors are formed for the PHASE PLANE.

attitude error =

old attitude error + attitude increment - commanded rate · DAP cycle time

rate error = measured rate - commanded rate

RATE_FILTER provides filtering of the rate gyro data for the RCS section of the TDAP after SEP_COMPLETE.

TRANS_ACCEL_TRANSITION allows the crew to command a constant linear acceleration along each of three body axes by deflecting the THC.

The steering processor provides the drive signals to the OMS and RCS processors. For the OMS processor, these drive signals are body rate commands. For the RCS processor, they are either attitude error and angular rate error signals derived from body rate commands, or RCS jet commands.

15.2.2.2 RCS Processor

The RCS Processor consists of the **JET_SELECT_TRANSITION** and **PHASE_PLANE_TRANSITION** modules.

JET_SELECT_TRANSITION generates jet firing commands required to implement commanded translations and rotations. The logic enables selection of primary jets only. Jet failures, ET SEP, RCS propellant crossfeed valve position, and tank constraints are all taken into account. The logic processes commands emanating from the PHASE_PLANE, hand controllers, pbi's, and CRT inputs, and determines which RCS jets to fire on each DAP cycle. Thrusters failed by RM are never commanded. Flight phase and propellant tank flow-rate constraints affect the control authority provided by the logic. The primary jets are selected using a table lookup scheme. Vernier jets are not an available option for the TDAP.

This module also includes software to dump FRCS propellant for vehicle mass and center of gravity control (OPS 3 only). The FRCS "dump" is selected by a sequence of item entries on the XXXXX MNVR YYYYY display, and causes opposing forward Y-jets to fire, expending propellant with negligible impulse to the Orbiter. All translation (but not rotation) commands are ignored during the dump and attitude control is by aft primary jets only.

Propellant valve states can be manually changed affecting OMS/RCS inter-connect and RCS/RCS crossfeed. Constraints exist on the number of simultaneous aft jet firings that can occur as a function of valve states (see SODB XXXX for details). The discrettes set by the valve switches automatically feed to the jet selection logic (JSL). Translation control authority can automatically downmode as a result of the valve positions (particularly +X, permitting adequate attitude control).

Jet selection is according to a table look-up scheme applied to each axis individually. Appropriate jet selection is performed by the JET_SELECT_TRANSITION logic such that attitude hold is provided during translational acceleration. A detailed table of jet selection for various failure cases may be found in the Flight Control-Ascent FSSR 8 Volume 2. A summary of this jet selection is provided in Table 15.5

The principle behind the table lookup scheme is that all jets that fire in the Z-axis can give Z-translation and/or roll rotation and/or pitch rotation control. These 19 jets are considered independent of other axes, however. The 12 jets that fire in the Y-axis are considered to give Y-translation and/or yaw rotation only. The 7 jets that fire in the X-axis are considered to give X-translation control only. Interactions between axes are handled by phase-plane feedback and/or closed-loop feedback.

Within each axis, jets are divided into directional groups and prioritized (priority of jets is flight specific and may be modified real-time). Once the appropriate line in the table for the required translation and/or rotation is identified, the highest priority available jet(s) for the groups listed are commanded to fire.

Tank Constraint Logic

Single axis translation logic is implemented to prevent jets from being commanded simultaneously in the X and Z translation axes in the event of simultaneous translation commands. The priorities are Z axis first and X axis second through ET separation, and X axis first and Z axis second after ET separation.

Y translation default condition inhibits Y-translation if either the left or right forward Y-jet clusters have dual failures.

Aft Z-rate default condition degrades -Z translation if +X and -Z translations occur simultaneously post ET SEP with open crossfeed or interconnect valve to keep within the 5 jet limit.

Degraded +X-translation. After ET SEP, if +X is commanded simultaneously with roll or pitch with the RCS/RCS crossfeed in use, no more than one jet per +X jet cluster is commanded.

Aft pitch default during -X. If -X is commanded, forward pitch jets are not commanded.

Mated coast high-level roll command logic increases the number of roll jets used, when possible, if a yaw rotation is commanded simultaneously with roll in a direction that couples unfavorably in the roll axis.

High yaw command logic increases the number of aft yaw jets used if:

- -Z translation is commanded
- both yaw jets and both +pitch jets on the same side of the nose have failed
- and the commanded yaw opposes the failure-induced disturbance

Pitch/roll rate reduction logic reduces the number of jets used for combined -pitch and roll maneuvers after ET SEP.

Forward propellant dump logic determines if a 2 or 4 jet forward propellant dump will occur and which jets to use. Attitude control is provided by the aft primary jets only (forward jets are turned "off"). During nose dump (firing of opposing nose Y-jets) all translation is inhibited.

Table 15-5.- TRANSITION DAP JET SELECTIONS

(a) TDAP Z-axis jet selections

| Axes with commands | Number of jets per group | | | | | | | |
|--------------------|--------------------------|-----|-----|-----|------|------|------------|------------|
| | Case | G-4 | G-5 | G-6 | G-11 | G-12 | G-13 | G-14 |
| -Z, +ROLL, +PITCH | A | | 2 | 2 | | | 3 | 1 |
| | B,C | | 1 | 1 | | | 2* | 0 |
| -Z, +ROLL | A | | 1 | 1 | | | 3 | 1 |
| | B,C | | 1 | 1 | | | 2 | 0 |
| -Z, +ROLL, -PITCH | A | | 0 | 0 | | | 3 | 1 |
| | B,C | | 0 | 0 | | | 2 | 0 |
| -Z, +PITCH | A | | 2 | 2 | | | 1 | 1 |
| | B | | 1 | 1 | | | 1* | 1* |
| | C | | 1 | 1 | | | 1 | 1 |
| -Z | A | | 2 | 2 | | | 3 | 3 |
| | B | | 2 | 2 | | | 2 Δ | 2 Δ |
| | C | | 1 | 1 | | | 2 | 2 |
| -Z, -PITCH | A | | 1 | 1 | | | 3 | 3 |
| | B | | 0 | 0 | | | 2 | 2 |
| | C | | 0 | 0 | | | 2 | 2 |
| -Z, -ROLL, +PITCH | A | | 2 | 2 | | | 1 | 3 |
| | B,C | | 1 | 1 | | | 0 | 2* |
| -Z, -ROLL | A | | 1 | 1 | | | 1 | 3 |
| | B,C | | 1 | 1 | | | 0 | 2 |
| -Z, -ROLL, -PITCH | A | | 0 | 0 | | | 1 | 3 |
| | B,C | | 0 | 0 | | | 0 | 2 |
| +Z, +ROLL, +PITCH | | 0 | | | 0 | 1 | | |
| +Z, +ROLL | | 1 | | | 0 | 1 | | |
| +Z, +ROLL, -PITCH | | 1 | | | 0 | 1 | | |
| +Z, +PITCH | | 0 | | | 1 | 1 | | |
| +Z | | 1 | | | 1 | 1 | | |
| +Z, -PITCH | | 1 | | | 0 | 0 | | |
| +Z, -ROLL, +PITCH | | 0 | | | 1 | 0 | | |
| +Z, -ROLL | | 1 | | | 1 | 0 | | |

| | | | | | | | |
|-------------------|---|----------|----------|---|---|---|---|
| +Z, _ROLL, -PITCH | 1 | | | 1 | 0 | | |
| +ROLL, +PITCH | | 1 Ψ | 1 Ψ | 0 | 1 | | |
| +PITCH | | 1 Ψ | 1 Ψ | 1 | 1 | | |
| -ROLL, +PITCH | | 1 Ψ | 1 Ψ | 1 | 0 | | |
| +ROLL, -PITCH | 1 | | | | | 2 | 0 |
| -PITCH | 1 | | | | | 1 | 1 |
| -ROLL, -PITCH | 1 | | | | | 0 | 2 |
| +ROLL | | | | 0 | 1 | 1 | 0 |
| -ROLL | | | | 1 | 0 | 0 | 1 |

A- Through separation, no jet failures

B - Through separation, with one jet failure in applicable groups

C - After separation

* Value is 1 less if either G-5 or G-6 has dual failures

^t Value is 1 if either G-13 or G-14 has dual failures

Δ Value is 1 if at least three jets are lost in G-5 and G-6. Value is 3 if only one -Z jet failure exists and it is in G-13 or G-14

Ψ Value is zero if either G-5 or G-6 has dual failures

(b) TDAP Y-axis jet selections

| Axes with commands | Number of jets per group | | | |
|--------------------|--------------------------|-----|-----|------|
| | G-2 | G-3 | G-9 | G-10 |
| +Y, +YAW | 1 | 0 | 0 | 0 |
| +Y | 1 | 0 | 1 | 0 |
| +Y, -YAW | 0 | 0 | 1 | 0 |
| -Y, +YAW | 0 | 0 | 0 | 1 |
| -Y | 0 | 1 | 0 | 1 |
| -Y, -YAW | 0 | 1 | 0 | 0 |
| +YAW | 1 | 0 | 0 | 1 |
| -YAW | 0 | 1 | 1 | 0 |

(c) TDAP X-axis jet selections

| Axes with commands | Number of jets per group | | |
|--------------------|--------------------------|-----|-----|
| | G-1 | G-7 | G-8 |
| +X | 0 | 2 | 2 |
| -X | 2 | 0 | 0 |

PHASE_PLANE_TRANSITION determines whether an angular rate change is needed and the sign of the desired rate change (by axis).

The PHASE_PLANE logic operates during automatic attitude hold and rate controlled flight phases using RCS control. It processes information about attitude rate and position errors, and, through a series of switching curves defined for each rotational axis, determines whether rotation commands should be generated and sent to the jet selection module. Figure 15-3 shows a view of the entire phase plane. The horizontal axis represents the attitude position error. The vertical axis represents a function of attitude rate error. A term combining gain and the latest rotation command value is added to the rate error to provide margin to the phase plane control law. Specific actions in the different regions are defined. With large errors in rate, acceleration is always commanded. Coast zones are provided to permit fuel-efficient state correction, and hysteresis is provided to allow time-efficient state correction and limit cycling. The hysteresis zone holds rate error commands ON until the horizontal axis is crossed, and allows coast otherwise.

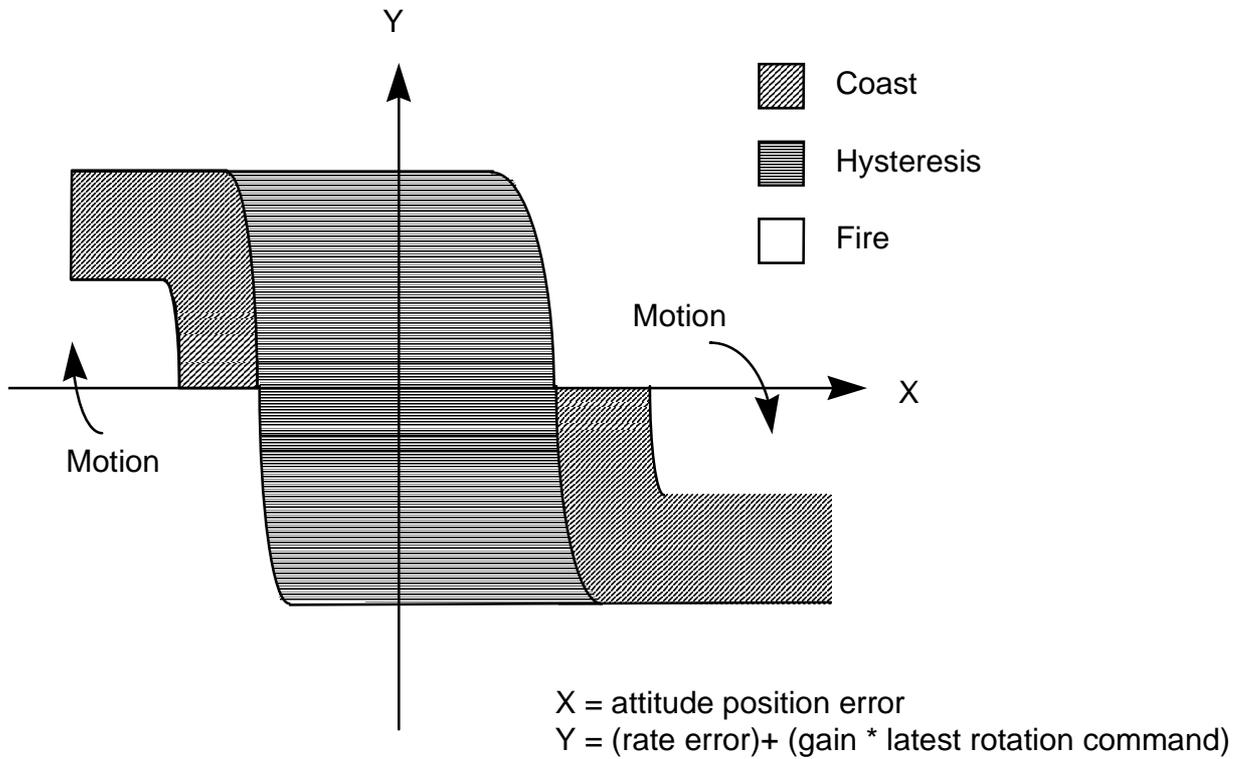


Figure 15-3. Phase plane regions

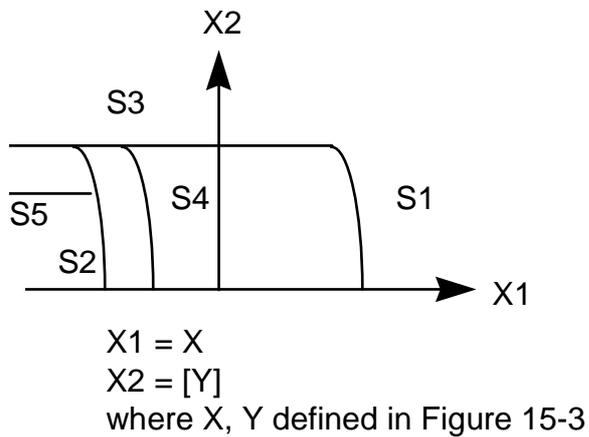


Figure 15-4. Phase plane switch lines

Table 15-6. Switch curve equations

| Line | Definition |
|------|---|
| S1 | $X1 = -\frac{X2^2}{2U_C} + DB + \frac{0.2 DB k}{KSTEP}$, $0 \leq X2 < +RL$ |
| S2 | $X1 = -\frac{CX2^2}{2U_C} - 1.2 DB$, $0 \leq X2 < +RL$ (see note) |
| S3 | $X2 = +RL$, bounded on the right by S1 |
| S5 | $X2 = LRL$, bounded on the right by S2 |
| S4 | $X1 = \frac{-X2^2}{2U_C} - DB$, $0 \leq X2 < +RL$ |
| | 1.25, $ ROT_JET_CMD \neq 1$ Note: $C = 1.0$, $ ROT_JET_CMD = 1$ |

DB = attitude position deadband
 RL = attitude rate limit (upper)
 LRL = lower rate limit
 UC = control acceleration
 KSTEP = marching switch curve constant
 k = marching switch curve counter

Figure 15-4 shows the switch curve designations in the phase plane with Table 15-6 showing the line definitions. The following actions are taken:

- a. In the fire region, command rotation to accelerate in the direction of the motion arrow.
- b. In the coast region, no control action is commanded.
- c. In the hysteresis region, continue commands to accelerate if they were issued before the region was entered and the horizontal axis was not yet crossed. Otherwise, no control action is commanded.

A modification was made to the TDAP phase plane to reduce jet cycling frequency in the presence of rate gyro (RG) and multiplexer/demultiplexer (MDM) noise. The modification is called a marching switch curve because switching line S1 is moved to the right (upper half plane) in a series of steps. Figure 15-5 illustrates the process. Whenever the attitude error becomes larger than the DB, the stepping process begins. One step of size 0.2 DB/KSTEP is added to the X1 position of S1 on each of KLIM passes until line S1 has reached 1.17 DB. When the attitude error is less than 0.9 DB, K is set to zero. The mechanism by which the marching switch curve prevents unnecessary jet firings caused by noise is illustrated in Figure 15-6. The series of numbered encircled points represent sampled state values while the dashed line is the rigid body motion. At sample point 4, the state has moved beyond the switch curve S1 and a ROT_JET_CMD is issued which drives the rate error beyond null.

After the firing, the state drifts towards zero attitude error; however, at 7, a noise spike causes a perception of reentry into the firing zone, and another firing commences. At the end of this second firing, the true rate error is larger than it would have been had there been no second firing, and the drift rate will be larger across the deadband thus increasing the limit cycle frequency. The noise spike, therefore, adds a second firing at sample 7 and decreases the limit cycle period. If the S1 switch line has been moved outward by one step after sample point 4 occurred, the firing due to noise would not have occurred.

If a noise spike in the negative direction occurs at sample point 5, the burn could terminate early leaving a growing attitude error; however, since S1 has moved, no firing will occur until the state sample has an attitude error beyond the new S1 position. At that time, another S1 step will be invoked.

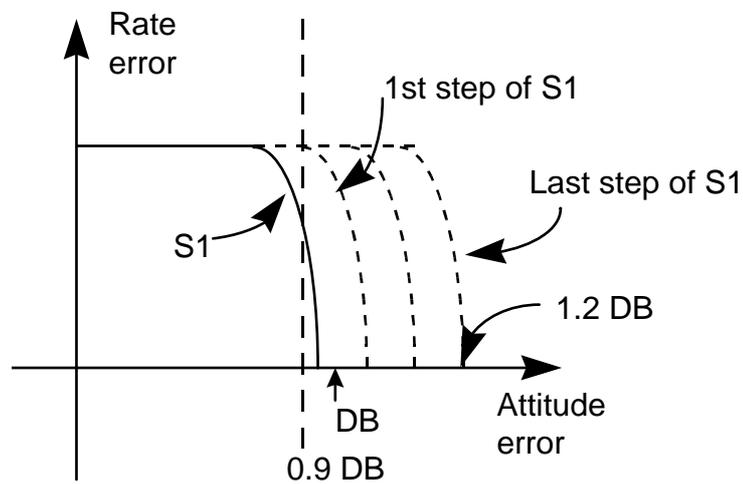


Figure 15-5. Marching switch curve S1

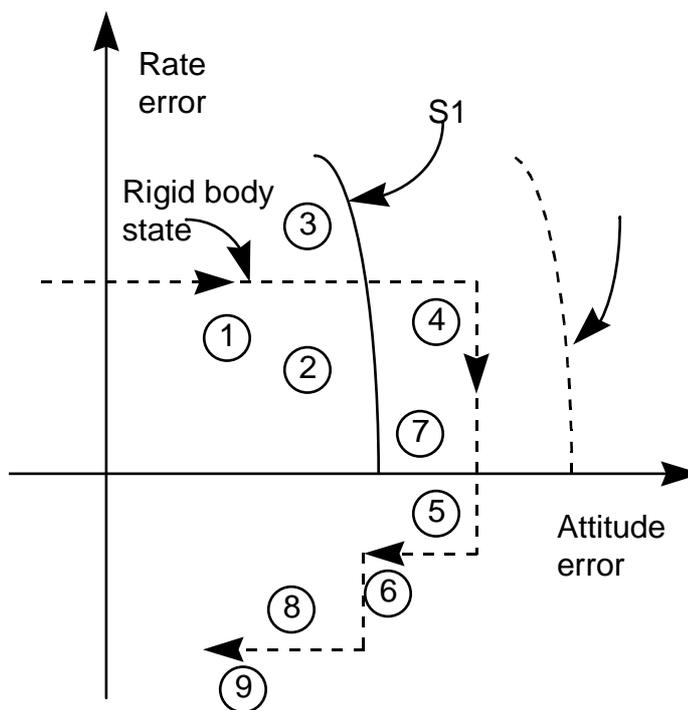


Figure 15-6. Extra jet firing condition

The phase-plane angular rate limit, lower angular rate limit, and deadband (DB) are dependent on mode or mission event. Table 15-7 correlates the constants with the applicable mission segment or mode.

Table 15-7. Phase plane boundary selection

| Condition | Phase-plane boundary | | |
|---|----------------------------|---------------------------|---|
| | Rate limit | Lower rate limit | Deadband |
| First pass insertion phase until automatic -Z (ET SEP) translation commanded; i.e., mated to ET | 0.5°/sec/axis RL_SEP | 0.1°/sec LRL_SEP | 10°/axis DB_MATED_COAST |
| ET SEP translation through completion | | | 3°/axis DB_SEP |
| OMS-with-RCS mode | 2.05°/sec/axis RL_TVC | 0.3°/sec LRL_TVC | 5° Roll axis 10° Pitch/ yaw axes DB_TVC |
| Automatic +X (abort) in RCS-only mode, or RCS backup OMS burn | 0.5°/sec/axis RL_PLUSX_RCS | 0.1°/sec LRL_PLUSX_RCS | DB_PLUSX_RCS |
| All other times | 0.5°/sec/axis RL_COAST | 0.1°/sec LRL_COAST | DB_COAST |

15.2.2.3 RCS and OMS Processor

In TVC operation, the OMS and RCS processors are driven by three-axis commands from the steering processor. For axes in which the OMS processor has control capability (roll, pitch, and yaw in a two OME burn; pitch and yaw in a single OME burn), it has the primary control responsibility, while the RCS processor operates as a wraparound controller, exercising RCS control only if the TVC control authority is inadequate (excessive attitude or angular rate errors). In a single OME burn, roll is controlled exclusively by the RCS processor, operating with the same attitude and angular rate error tolerances as in the two OME burn case.

15.2.2.3.1 OMS Processor

The **OMS processor** consists of the **TVC_LAW_PITCH_YAW**, **TVC_LAW_ROLL**, and **TVC_MIXER** modules. A brief description of each module is provided below.

TVC_LAW_PITCH_YAW closes the inner loop of the TVC DAP. Commanded pitch and yaw rates and the corresponding desired rate errors are subtracted from the vehicle rates to form rate errors, which are processed by compensation filters and passed through proportional-plus-integral paths. Outputs are limited and passed on as position commands to the *OMS engine actuator servos*.

TVC_LAW_ROLL provides control of the vehicle in roll during two-engine TVC burns. Commanded roll rates and the corresponding desired rate errors are subtracted from the vehicle rates to form rate errors, which are processed by compensation filters and

passed through proportional-plus-integral paths. Output is passed on as a roll command to the *TVC mixer module*.

TVC_MIXER derives the left and right OMS pitch actuator servo commands as the difference and sum respectively of the pitch and roll TVC commands. The roll command is limited as a function of the pitch command, so that the total pitch command cannot exceed the desired limit.

15.2.2.3.2 OMS Steering

The basic elements of the OMS TVC are shown in Figure 15-7. The diagram shows, in a simplified block representation, the main features of the TVC loop. The OMS TVC consists of OMS steering guidance compensation blocks and the inner rate loop. This inner loop is a rate loop consisting of (1) the DAP whose output is the engine servo

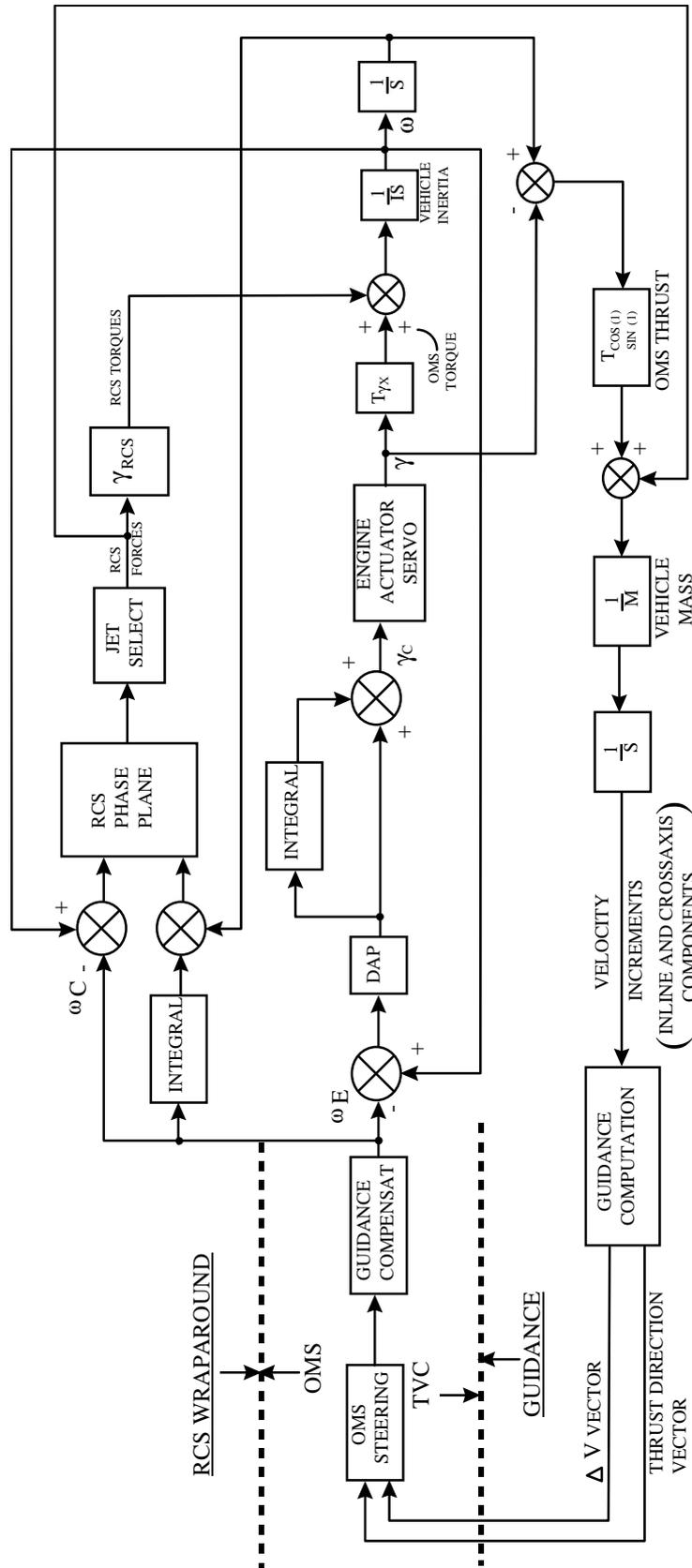


Figure 15-7. - Simplified TVC block diagram (pitch or yaw).

position command, (2) the engine actuator servo which positions the thrust vector as commanded, and (3) the vehicle dynamics.

The inner loop has an additional parallel control path which is the RCS wraparound. This path is largely inactive except when the control authority of OMS TVC is limited and rate commands become large enough to activate RCS jets.

Finally, in the lower part of the diagram, the elements of the guidance (or outer) loop are outlined. The outputs of guidance computation are two vectors: Delta velocity (ΔV) acquired over the guidance sampling interval of .96 sec and the thrust direction vector. These are based on measured velocity increments which depend mainly on the OMS thrust force time cosine and sine functions of vehicle attitude q minus OMS engine position angle d . These are basically components in the desired (in-line) thrust direction and cross axis direction, respectively. The RCS forces (if any) also contribute to velocity increments.

The block diagram in Figure 15-7 is valid for pitch and yaw control because the two loops are similar. For roll control, the simplified diagram in Figure 15-7 reduces to the inner loop and the RCS wraparound only since the roll component of rate command from OMS steering is zeroed. Thus roll rate is maintained at zero within the loop's capability to do so. For single engine burns where the OMS has no roll control authority, the roll control is done by RCS wraparound only.

The OMS TVC (pitch/yaw) is a control system consisting of two control loops: inner and outer. The inner loop contains an engine actuator servo which is also a closed loop system. Thus, a logical procedure was used to maximize the crossover frequencies (and thus bandwidths) for the sequential loop closures. Noting that a linearized analysis of the actuator servo yields a bandwidth of about 3 rad/s, the inner loop crossover was selected at about 0.8 rad/s. This crossover permitted a significant amount of attenuation for vehicle bending frequencies which start at about 25 rad/s. The final loop closure is the outer loop. Here a simple gain compensation was found to be inadequate. The loop was unstable with the desired steering gain of 0.1 (desired from performance point of view). It was necessary to introduce the guidance compensation block which stabilized the loop. The crossover frequency of this loop is about 0.1 rad/s.

The RCS wraparound is an on-off type controller called upon to assist the inner loop in control limited situations and no linear bandwidth characteristics have been attributed to it. A possible RCS-TVC force fight was inhibited by greatly expanding the attitude DB's in pitch/yaw, effectively eliminating them, and thus achieving the RCS-TVC control objective of zeroing rate errors.

The OMS steering uses a cross product of two vectors' DVB-SEC and thrust direction LAMC. This can be understood by a simplified single plane description where V_g is similar to the thrust direction vector LAMC, and DV is similar to the vector DVB-SEC.

During external DV burns, the guidance or steering controls the burn by applying appropriate commands to the inner autopilot loop. In the case of cross-product steering, guidance computation results in a rate command $\vec{\omega}$ defined by

$$\vec{\omega} = \frac{\vec{\Delta V}_x \vec{V}_g}{|\Delta V| |V_g|} = \vec{1} \sin(\alpha_{\Delta V} - \alpha_{V_g}) \quad (1)$$

$\alpha_{\Delta V}$ is the angle of the ΔV vector or velocity increment achieved over the past guidance sample period T_S , and α_{V_g} is the angle of the velocity-to-be-gained vector.

The derivation of guidance loop relationships, given below, considers the guidance problem in a single plane. Thus, in the coordinate system of a plane shown in Figure 15-8, the X-axis is defined as the desired thrust direction, and components of velocity in the Y-axis direction are the cross-axis or undesirable velocity errors.

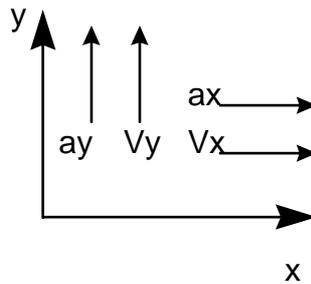


Figure 15-8. Coordinate plane system

During a burn, guidance takes snapshots of velocities, as measured by the accelerometers, at sampling intervals T_S . A typical velocity vector diagram is shown in Figure 15-9. From Figure 15-9 the angle $\alpha_{\Delta V}$ of the ΔV vector is given by

$$\alpha_{\Delta V} = \tan^{-1} \frac{\Delta V_y}{\Delta V_x} = \tan^{-1} \left[\frac{V_{y_n} - V_{y_{(n-1)}}}{V_{x_n} - V_{x_{(n-1)}}} \right] \quad (2)$$

and the angle α_{V_g} of velocity vector V_g is given by

$$\alpha_{V_g} = \tan^{-1} \frac{V_{g_y}}{V_{g_x}} = \tan^{-1} \left[\frac{-V_{y_n}}{V_{x_n}} \right] \quad (3)$$

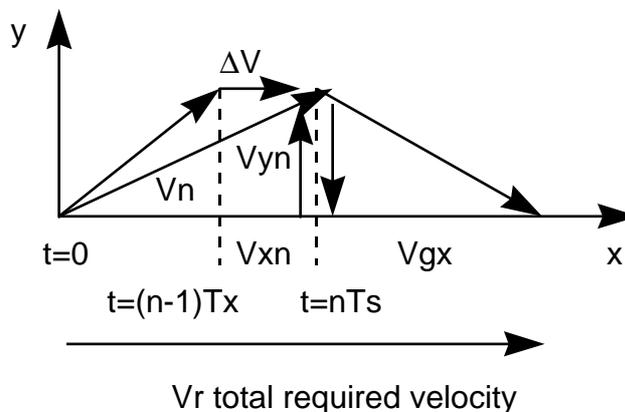


Figure 15-9. Velocity vector diagram.

In equations (2) and (3), the subscript n represents the present sample period and $(n-1)$ the previous sample period of the guidance loop. Note that in Figure 15-9 and in equation (3) the velocity-to-be-gained vector in the Y direction (the undesirable velocity component) is exactly equal and opposite to the present cross-axis velocity.

Now, it will be shown that these velocity components, measured by guidance, are related to dynamic quantities of the vehicle attitude θ and the engine angle δ . Figure 15-10 shows the vehicle in the X - Y plane.

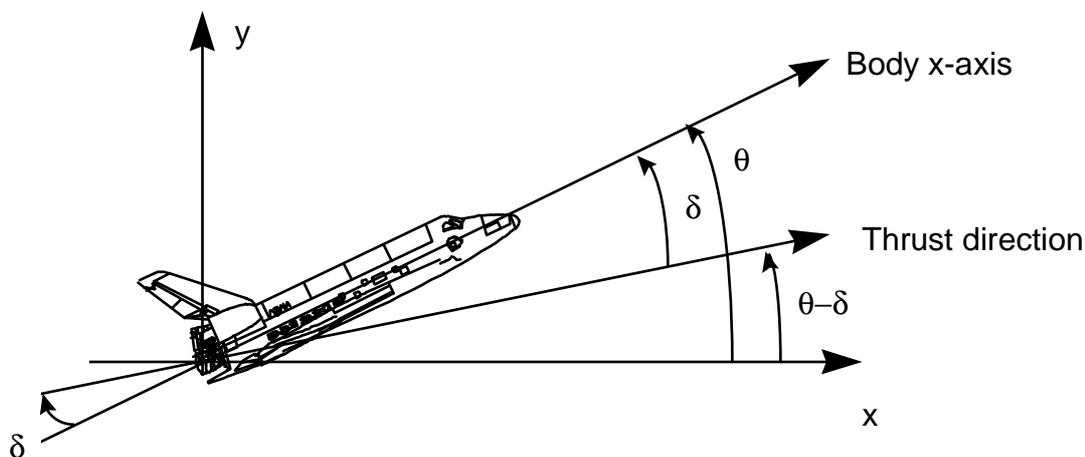


Figure 15-10. Vehicle in X - Y plane.

In Figure 15-10, the resultant of the two OMS engine angles is shown by the engine angle δ . It is measured from body X -axis and is defined positive when it generates a positive moment and a positive angular acceleration θ . Then, the thrust direction angle is the difference between the attitude angle θ and the engine angle δ . With linear acceleration of the vehicle equal to thrust T divided by mass M , the velocity components in the X - and Y -directions at time $= t$, expressed in terms of θ and δ , are given by

$$\begin{aligned} V_x &= \frac{T}{M} \int_0^t \cos(\theta - \delta) dt \\ V_y &= \frac{T}{M} \int_0^t \sin(\theta - \delta) dt \end{aligned} \quad (4) \text{ and } (5)$$

(The burn starts at $t = 0$).

For small values of $(\theta - \delta)$

$$\cos(\theta - \delta) \approx 1$$

$$\sin(\theta - \delta) \approx \theta - \delta$$

and at $t = nT_s$ equations (4) and (5) become

$$V_{x_n} \approx \frac{T}{M} n T_s \quad (6)$$

$$V_{y_n} \approx \frac{T}{M} \int_0^{nT_s} (\theta - \delta) dt \quad (7)$$

Similar expressions are obtained at $t = (n-1)T_s$. By substituting equations (6) and (7) into equations (2) and (3), the angles $\alpha_{\Delta V}$ and α_{V_g} are expressed in terms of $(\theta - \delta)$:

$$\alpha_{\Delta V} = \tan^{-1} \left[\frac{\frac{T}{M} \int_0^{nT_s} (\theta - \delta) dt - \frac{T}{M} \int_0^{(n-1)T_s} (\theta - \delta) dt}{\frac{t}{m} [nT_s - (n-1)T_s]} \right] \quad (8)$$

$$a_{V_g} = \tan^{-1} \left[\frac{\frac{-T}{M} \int_0^{nT_s} (\theta - \delta) dt}{\frac{T}{M} (t_{\max} - nT_s)} \right] = \tan^{-1} \left[\frac{- \int_0^{nT_s} (\theta - \delta) dt}{t_{\max} - nT_s} \right] \quad (9)$$

The denominator of equation (9) is the difference between the burn time required to achieve the desired velocity and the elapsed burn time, or time-to-go (t_{go}). Using small angle approximation, the arctangent functions are eliminated and equations (8) and (9) become

$$\alpha_{\Delta V} = \frac{1}{T_s} \left[\int_0^{nT_s} (\theta - \delta) dt - {}^{(n-1)T_s} \int_0 (\theta - \delta) dt \right] \quad (10) \text{ and } (11)$$

$$\alpha_{\Delta V_g} = -\frac{1}{t_{go}} \int_0^{nT_s} (\theta - \delta) dt$$

Substituting equations (10) and (11) into equation (1) and again invoking small angle approximation, the result of guidance computation, ω , is obtained:

$$\omega \frac{1}{T_s} \left[\int_0^{nT_s} (\theta - \delta) dt - {}^{(n-1)T_s} \int_0 (\theta - \delta) dt \right] - \left[-\frac{1}{t_{go}} \int_0^{nT_s} (\theta - \delta) dt \right] \quad (12)$$

A block diagram of equation (12) is given in Figure 15-11.

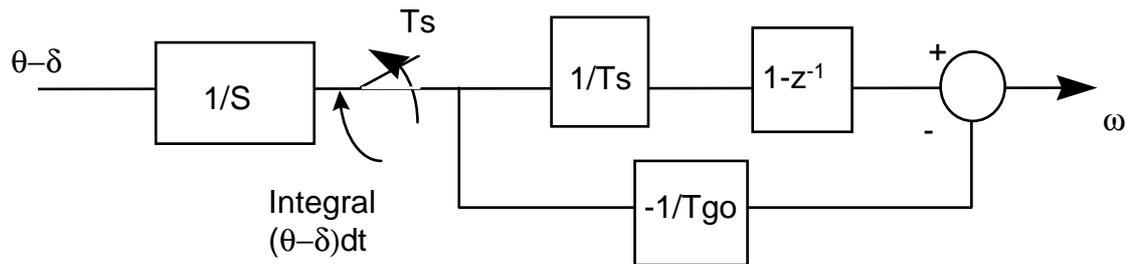


Figure 15-11. Block diagram of guidance command computation (OMS steering simplified).

SECTION 16
ORBIT DAP

16.1 GENERAL DESCRIPTION 2

16.2 UNIVERSAL POINTING SOFTWARE 7

 16.2.1 MNVR Option 8

 16.2.2 TRK Option 8

 16.2.3 ROT Option 11

 16.2.4 CNCL Option 12

16.3 UNIVERSAL POINTING DISPLAY (OPS DISPLAY) 12

16.4 DAP CONFIGURATION DISPLAY (SPEC 20) 15

16.5 DAP PANEL PUSH BUTTON INDICATORS (PBI) 18

 16.5.1 Function of DAP Panel PBIs 18

 16.5.2 DAP Panel PBI Redundancy Management 20

 16.5.3 DAP Panel PBI Prioritization 21

16.6 MANEUVER DISPLAY (OPS DISPLAY) 21

16.7 DETAILED DESCRIPTION OF SOFTWARE 22

 16.7.1 On-Orbit DAP RECON module (GCQ_ORB_DAP_RECON_25HZ) 23

 16.7.2 On-Orbit DAP Critical Processor module (GFF_ORB_DAP_CP_25HZ) 27

 16.7.3 Post Contact Thrusting Arm/Activate Logic module (GC1_ORB_DOCK_ARM_ACT) 41

 16.7.4 Post Contact Docking Jet Selection module (GFD_ORB_DOCKING_JS) 41

 16.7.5 On-Orbit DAP OMS Steering module (GC8_ORB_OMS_STEERING) 41

 16.7.6 On-Orbit State Estimator Part 1 Filter module (GC7_ORB_P1FILT_6HZ) 42

 16.7.7 Auto Maneuver/Track module (GC9_ORB_AUTO_MNVR_TRACK_1HZ) 42

 16.7.8 Orbit DAP LVLH Track module (GC5_ORB_LVLH_TRACK) 43

 16.7.9 Orbit OMS Processor module (GCR_ORB_OMS_PROCESSOR) 43

 16.7.10 DAP Load Processor module (GC6_ORB_DAP_LOAD_PROC) 43

16.8 NOTES OF INTEREST 43

16.9 AUTOMATIC REBOOST 44

 16.9.1 Background 44

 16.9.2 OI-28 Automatic Reboost 44

16.10 REFERENCES 48

SECTION 16
ORBIT DAP

16.1 GENERAL DESCRIPTION

The orbit Digital Auto Pilot (DAP) is the flight control software used on orbit. The orbit DAP consists of several software algorithms which provide attitude maintenance, rotational control for maneuvers, and translation control of the orbiter while in operational sequence (OPS) 2 and OPS 8. The orbit DAP can be divided into two nominal submodes: Reaction Control System (RCS) DAP and Orbital Maneuvering System Thrust Vector Control (OMS TVC) DAP. The off-nominal submode is the Inertial Measurement Unit (IMU) Fail mode. It exists because it is not possible to run normal flight control software without good IMU data.

While the RCS DAP is under closed-loop control, attitude maintenance is done primarily by comparing Inertial Measurement Unit (IMU) output data (via the state estimator software) with the desired attitude and rates (from guidance software) to determine attitude errors and rate errors in each orbiter body axis (roll, pitch, and yaw). These errors are compared with crew selectable attitude and rate deadbands (DB) to determine whether the errors are within acceptable limits (the acceptable attitude and rate limits make up the phase plane). If the errors are within the DBs, no action is taken. If the errors are too large, the phase plane software will issue a command to bring the errors back within the acceptable limits by firing the appropriate jets. A rate estimate for those jets is then sent to the state estimator. Since the IMU data are fed into the estimator at a rate of 6.25 Hz, and the phase plane and jet select software run at 12.5 Hz, the rate estimate is used to extrapolate the actual vehicle attitude change and rate change between IMU data takes. The loop then repeats with the state estimator derived rate and attitude being compared with the desired to determine errors. Figure 16-1 shows the simplified process. A more detailed illustration of the orbit DAP functions can be found in Figure 4.2.2-1 of the Orbit DAP Functional Subsystem Software Requirements (FSSR) document (STS 83-0009). A more detailed explanation of the DAP software can be found in Section 16.7.

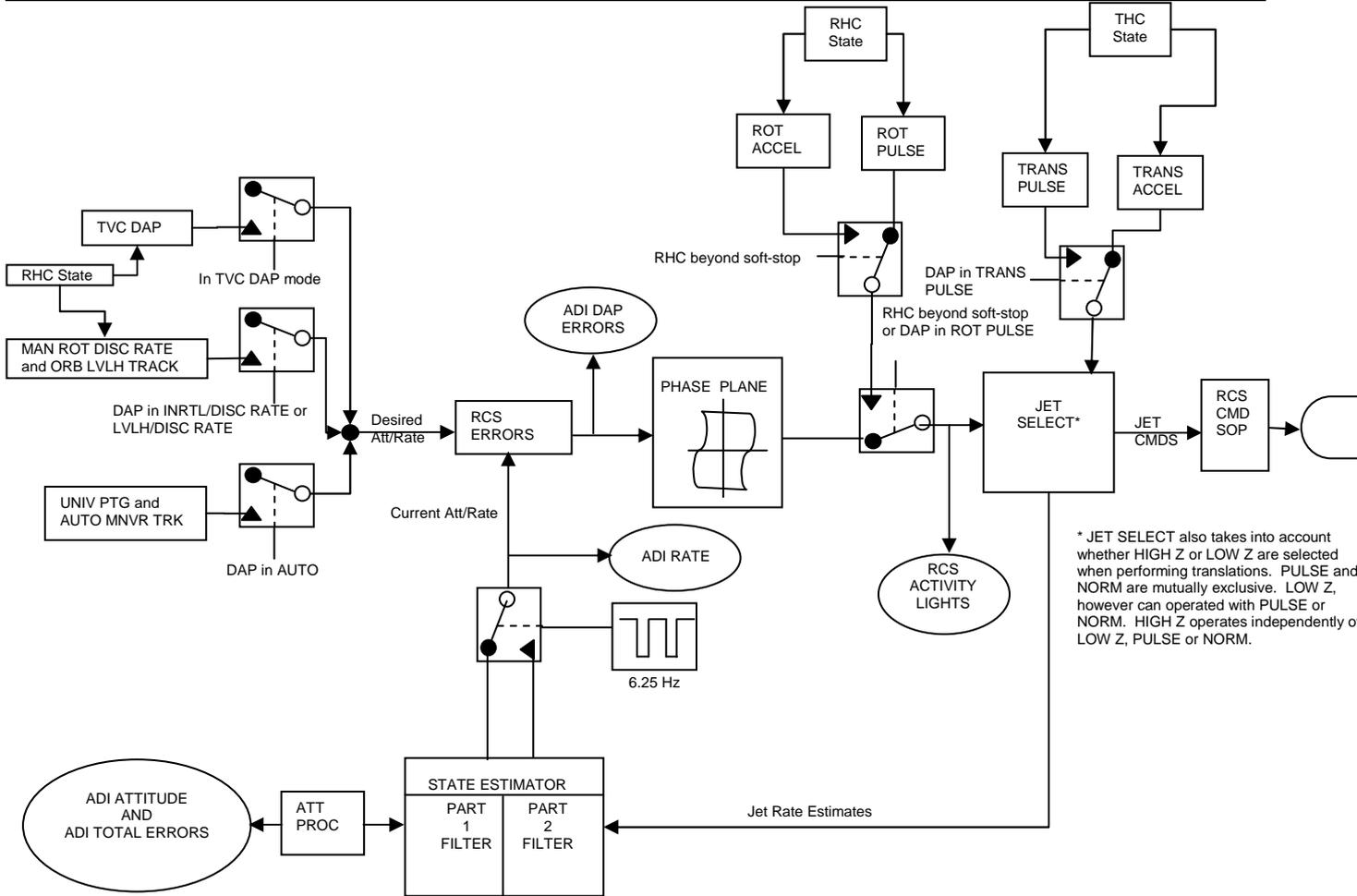


Figure 16-1: Orbit DAP Overview

The DAP panels (Figure 16-2) located on orbiter panels C3 and A6 are used by the crew to select one of the three closed-loop attitude control modes (AUTO, INRTL, LVLH) or free drift (FREE). They also allow for the selection of several different translation and rotation control modes. A detailed description of the DAP panels can be found in Section 16.5.

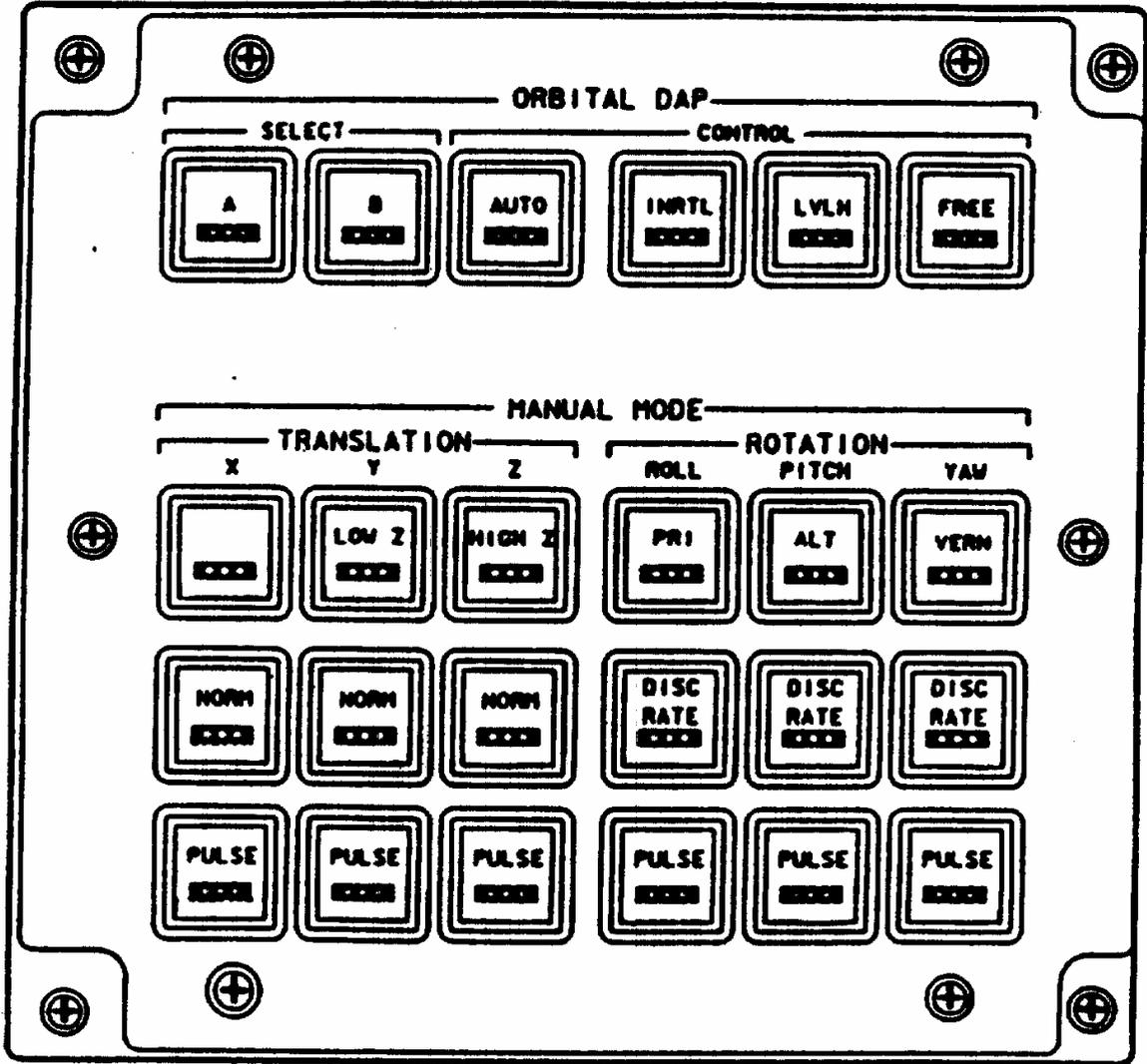


Figure 16-2: Orbiter DAP Panel

The RCS DAP will utilize the values I-loaded or updated by the crew on SPEC 20 DAP CONFIG display (Figure 16-3) that correspond to the DAP configuration selected on the DAP panel. A detailed description of SPEC 20 can be found in Section 16.4.

| XXXX / 020 / | | DAP CONFIG | | | | XX X DDD/HH:MM:SS DDD/HH:MM:SS | |
|--------------|----|------------|----|----------|------|-----------------------------------|------------|
| XXX | 1 | XXXXXXXX | 2 | XXXXXXXX | PRI | DAP EDIT | |
| ROT RATE | 10 | X.XXXX | 30 | X.XXXX | 50 | X.XXXX | 3 DAP AXX |
| ATT DB | 11 | XX.XX | 31 | XX.XX | 51 | XX.XX | 4 DAP BXX |
| RATE DB | 12 | X.XX | 32 | X.XX | 52 | X.XX | 5 XXXX |
| ROT PLS | 13 | X.XX | 33 | X.XX | 53 | X.XX | |
| COMP | 14 | .XXX | 34 | .XXX | 54 | .XXX | |
| P OPTION | 15 | XXXX | 35 | XXXX | 55 | XXXX | |
| Y OPTION | 16 | XXXX | 36 | XXXX | 56 | XXXX | NOTCH FLTR |
| TRAN PLS | 17 | X.XX | 37 | X.XX | 57 | X.XX | ENA 6X |
| XXX | | | | | ALT | | |
| RATE DB | 18 | X.XXX | 38 | X.XXX | 58 | X.XXX | XJETS ROT |
| JET OPT | 19 | XXXX | 39 | XXXX | 59 | XXXX | ENA 7X |
| # JETS | 20 | X | 40 | X | 60 | X | |
| ON TIME | 21 | X.XX | 41 | X.XX | 61 | X.XX | REBOOST |
| DELAY | 22 | XX.XX | 42 | XX.XX | 62 | XX.XX | 8 CFG X |
| XXXX | | | | | VERN | | 9 INTVL |
| ROT RATE | 23 | X.XXXX | 43 | X.XXXX | 63 | X.XXXX | XXX.XX |
| ATT DB | 24 | XX.XXX | 44 | XX.XXX | 64 | XX.XXX | |
| RATE DB | 25 | .XXX | 45 | .XXX | 65 | .XXX | |
| ROT PLS | 26 | X.XXX | 46 | X.XXX | 66 | X.XXX | |
| COMP | 27 | .XXX | 47 | .XXX | 67 | .XXX | |
| CNTL ACC | 28 | X | 48 | X | 68 | X | |

Figure 16-3: DAP Configuration SPEC

If the DAP is in the automatic (AUTO) mode, the desired attitude is determined by the Universal Pointing option (Items 18-21) that is selected at the time on the UNIV PTG display (Figure 16-4). A detailed description of the various Universal Pointing options can be found in Section 16.2. Details of the display itself can be found in Section 16.3.

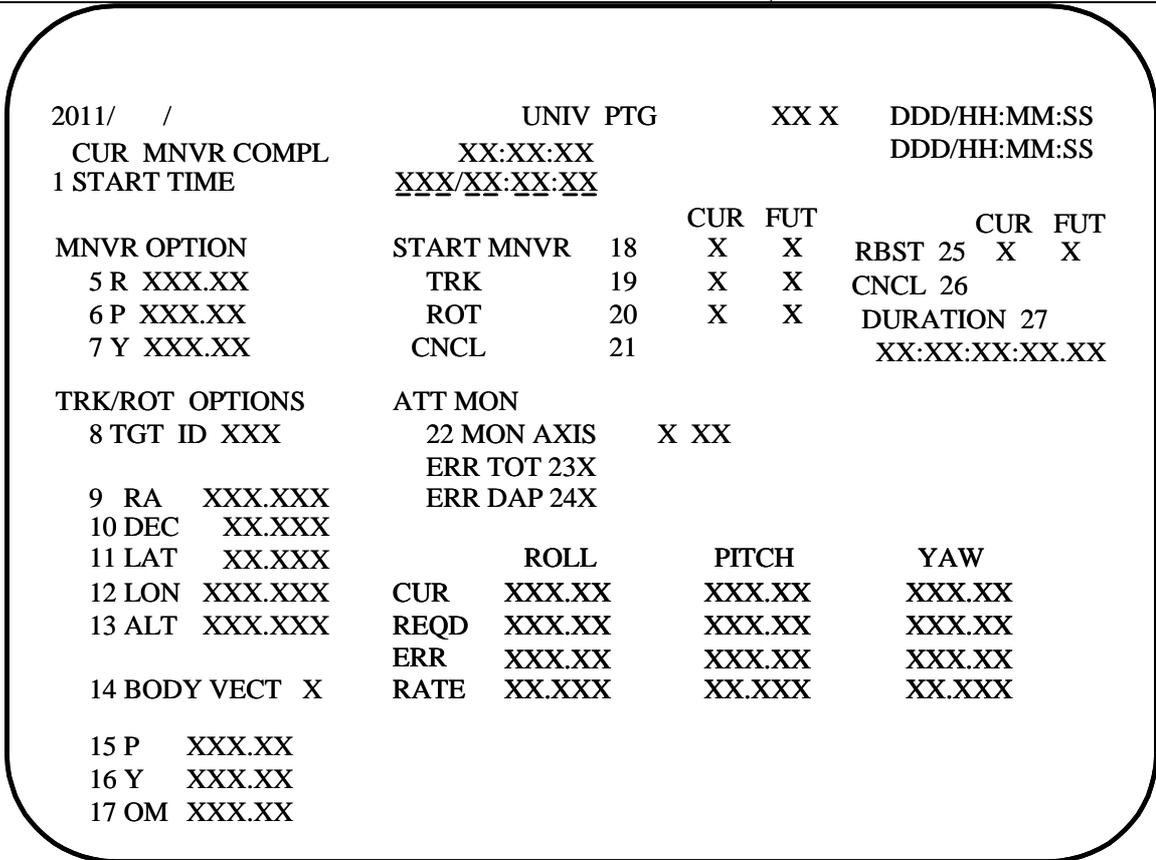


Figure 16-4: Universal Pointing Display

When in Major Mode (MM) 202, after a preburn maneuver has been executed (Item 27 on Maneuver (MNVR) display) or OMS ignition has been executed, orbit DAP controls attitude based on data found on the MNVR display (Figure 16-5). The TVC DAP mode is activated when either one or two OMS engines is commanded on and if the IMU Data Good flag is on. TVC DAP has two submodes; automatic mode (DAP is in AUTO and RHC is in detent) and manual mode (DAP in INRTL). FREE and LVLH are not supported in TVC DAP mode. After TVC DAP is activated, pitch and yaw rate commands are generated based on the difference between desired and actual vehicle velocity vectors (Delta Vs) to drive the OMS TVC gimbals (if DAP in AUTO). Roll rate commands are generated and incorporated in the pitch commands for each engine in order to maintain a commanded roll rate of zero. If control authority of the gimbals is not enough to maintain the commanded rate, RCS jets may be used to augment the gimbals and correct errors. This is called RCS wraparound and will be used if the following deadbands are exceeded:

OMS TVC DAP - RCS Wraparound deadbands

- Roll = 5 deg
- Pitch and Yaw = 10 deg
- Rate = 2.05 deg/sec (in all axes)

During a single engine burn, RCS jets will fire to maintain the roll rate of zero. Only the primary jets are used in the TVC DAP mode. The DAP will upmode to PRI from ALT or VERN when the TVC DAP mode is entered. TVC DAP is active until OMS engine cutoff, when RCS DAP is again activated. While in TVC DAP, THC inputs are ignored and any current or future reboost will be canceled.

In manual TVC DAP, RHC deflections supply the rate commands to gimbal the OMS engines. When the RHC is between detent and the softstop, a proportional rate command is sent to the appropriate gimbals. During a two engine burn, RCS wraparound is still available for all axes. If the RHC is deflected beyond the softstop, a maximum rate of 2.0 deg/sec is commanded to the appropriate gimbals and RCS jets fire continuously. For a single engine burn, Pitch and Yaw axes perform exactly as in the two engine case. The Roll axis is held in the 5 deg deadband with RCS jets when the RHC is in detent. When the RHC is out of detent but within the softstop in the Roll axis, DISC RATE is used by the RCS jets, regardless of the DAP selection (if PULSE is selected on the DAP, it will remain illuminated). If the RHC is deflected beyond the softstop, the RCS jets will fire continuously.

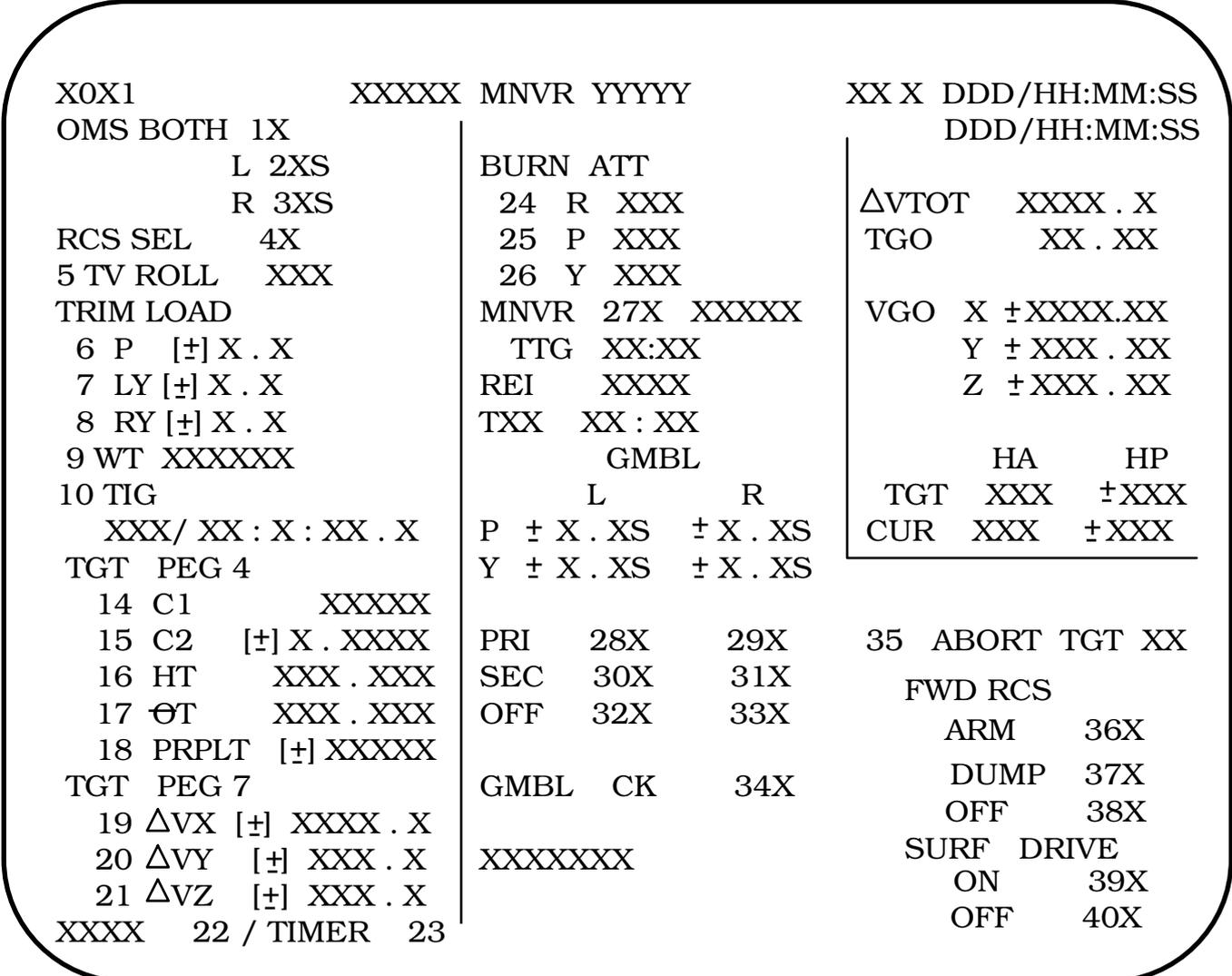


Figure 16-5: Maneuver Display

16.2 UNIVERSAL POINTING SOFTWARE

Universal Pointing software is the primary orbital guidance software used to control the Orbiter attitude when the DAP is in AUTO. The crew/software interface for this function is the MM 201 OPS display, Universal Pointing (UNIV PTG), which will be discussed later.

The Universal Pointing Processor software module (GVU_UNIV_POINT_PROC) has a processing rate of 0.52 Hz. It is active throughout OPS 2 and OPS 8. Upon transition to MM 202, Universal Pointing software is used to control Orbiter attitude until a preburn maneuver (Item 27 on MNVR Display) occurs or OMS ignition is commanded. Upon return to MM 201, Universal Pointing software will resume its last executed maneuver. Universal Pointing will also control Orbiter attitude during Ops 8, as long as it was active on the transition to Ops 8. The UNIV PTG display is not available in either MM 202 or Ops 8 and therefore cannot be modified. The UNIV PTG display is available only in MM 201.

On the first pass through the software module (except when coming from OPS 8 or MM 202), QBI (current inertial attitude quaternion) is snapshot. That is, the required body quaternion (REQD on UNIV PTG display) is set equal to QBI and Item 21 CNCL is selected on UNIV PTG display. The DAP will therefore hold the current inertial attitude.

There are four options available to the crew through Universal Pointing.

- Item 18 MNVR - Maneuver
- Item 19 TRK - Track

Item 20 ROT - Rotation
Item 21 CNCL - Cancel

For each option there is the capability to execute a current option or load a START TIME (See Figure 16-4) in the future and reserve the option (except CNCL) for that START TIME. An asterisk appears under current (CUR) or future (FUT) for the particular option executed. One current and one future option can be selected at any particular time. When a future option is loaded, the current option will not be affected until the specified START TIME is reached. At this time the future option becomes the current option (the asterisk moves from FUT to CUR). During the time between 30 seconds prior to the scheduled initiation of a future option and initiation of that option, the software will check to see whether or not the DAP is in AUTO. If the DAP is in INRTL, LVLH, or FREE, a message (SEL AUTO) will be driven on the scratch pad line to indicate the need to mode to AUTO on the DAP. If a future option has been selected and then data related to that particular option is changed prior to the START TIME via keyboard inputs, the future option will be deselected (the asterisk next to FUT will disappear).

Universal Pointing software also provides the Orbiter body attitude rate vectors when a Local Vertical/Local Horizontal (LVLH) attitude is requested. An LVLH attitude will be held either when a TGT ID of 2 is selected on UNIV PTG display or LVLH is selected on the DAP panel.

Note: The ability for the Orbiter to maintain an accurate LVLH attitude is dependant on an accurate state vector. If the Orbiter state vector is inaccurate, the LVLH attitude held will incorrect. Also, if the onboard state vector is updated while an LVLH attitude is being held, a maneuver may take place due to the update of the LVLH reference frame.

16.2.1 MNVR Option

The function of this option is to maneuver the vehicle to the inertial roll, pitch, yaw attitude in M50 coordinates displayed on the UNIV PTG display (MNVR OPTION, Items 5, 6, and 7). The maneuver begins at the START TIME (Items 1 to 4). If the time is current or in the past when MNVR (Item 18) is executed, the maneuver begins immediately. When MNVR is initiated, the REQD attitude is set equal to the MNVR OPTION attitude. The maneuver is an eigen rotation about the eigen vector formed by the CUR attitude and the REQD attitude. The maneuver rate about the eigen vector is at the ROT RATE on SPEC 20 (Items 10, 30, 23, or 43 depending on the DAP panel configuration). To maneuver the Orbiter, the DAP software rotates the phase plane by changing the desired attitude and rate (center of the phase plane) of each Orbiter body axis appropriately each software cycle to maintain the eigen maneuver at the ROT RATE. The phase plane is defined by the ATT DB on SPEC 20 (Items 11, 31, 24, or 44 depending on the DAP panel configuration) and the RATE DB on SPEC 20 (Items 12, 32, 18, 38, 25, or 45 depending on the DAP panel configuration). The maximum acceptable attitude error from the current desired attitude in any axis at a given time during the maneuver is the ATT DB. The maximum acceptable rate error from the currently commanded rate about any axis during the maneuver is the RATE DB. When these DAP errors exceed the deadbands, jets will fire to start and maintain the Orbiter maneuver. Total errors are always the difference between REQD and CUR attitudes on UNIV PTG. When the AUTO_MNVR_TRACK software module determines the vehicle is at the REQD attitude, an attitude hold is initiated. The asterisk, however, will remain by CUR MNVR on the UNIV PTG display. This means that if the crew manually maneuvers away from this commanded attitude, the DAP will return the vehicle to it when AUTO is reselected.

16.2.2 TRK Option

This algorithm is by far the most complex of the four options and comprises the major portion of Universal Pointing software. There are five subcases to this option. Each subcase represents a Target Identification (TGT ID) on UNIV PTG (Item 8). These subcases (TGT ID's) are:

- Orbiting Vehicle (TGT ID = 1)
- Center of the Earth (TGT ID = 2)
- Earth target (TGT ID = 3, requires LAT, LON, ALT entry – Items 11, 12, and 13)
- Center of the Sun (TGT ID = 4)
- Celestial target (TGT ID = 5, requires RA, DEC data entry – Items 9 and 10)
- Navigation (NAV) Star target (TGT ID = 11-110)

The TGT ID is simply a line of sight (LOS) or target vector from the Orbiter c.g. to a target defined by TGT ID. TGT ID = 1 requires Rendezvous Navigation enabled (SPEC 33) and a current target State Vector (SV). TGT ID = 3 or 5

requires a manual entry of the appropriate information defined above. TGT ID = 11-110 correspond to the NAV stars already in the general purpose computer (GPC).

The TRK option is used in conjunction with the body vector (Item 14 BODY VECT on UNIV PTG display). This vector is a direction vector centered at the Orbiter c.g. and is defined by the pitch (P), yaw (Y) combination (Items 15 and 16, respectively). The P, Y are referenced to the Orbiter +X axis. It is important to note that the body vector is referenced to the Orbiter only and not any external coordinate systems. It is, therefore, by itself, attitude independent. The TRK software will point the body vector at the defined TGT ID. Now the Orbiter attitude is somewhat constrained. That is, there is a requirement to point the body vector at a target (TGT ID) and these constraints will determine to some degree the Orbiter attitude when this pointing task occurs. The attitude will not be completely constrained, however, until OMICRON (Item 17) is specified. By simply pointing the body vector at the target, no body orientation about that vector has been specified. The roll orientation about the body vector (not to be confused with Orbiter body roll) is the value, OMICRON. OMICRON can be simply defined as the angle between two quarter planes. One quarter plane is defined by the pointing vector (the term pointing vector is simply the body vector already pointing at its target) and the + Y Orbiter axis. The second quarter plane is defined by the pointing vector and the negative angular momentum vector (-H) of the current orbit (Figure 16-6).

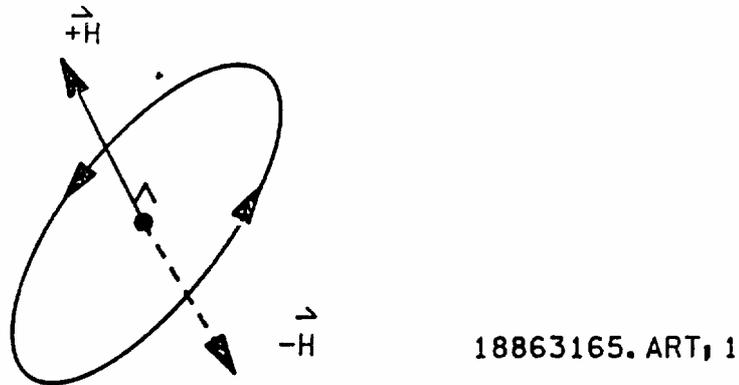


Figure 16-6.- Orbital angular momentum vector.

When the Orbiter orientation about this pointing vector is such that the two quarter planes are coincident (when the Orbiter's right wing is pointing toward the southern pole of the orbit), then OMICRON is zero degrees. As the two planes are rotated apart in a direction using the right hand rule with the thumb along the pointing vector, OMICRON increases positively. Examples are shown in Figures 16-7 to 16-10.

1. Body vector = +X axis (P = 0, Y = 0)
Target vector = 2 (Center of Earth)
OMICRON = 0°

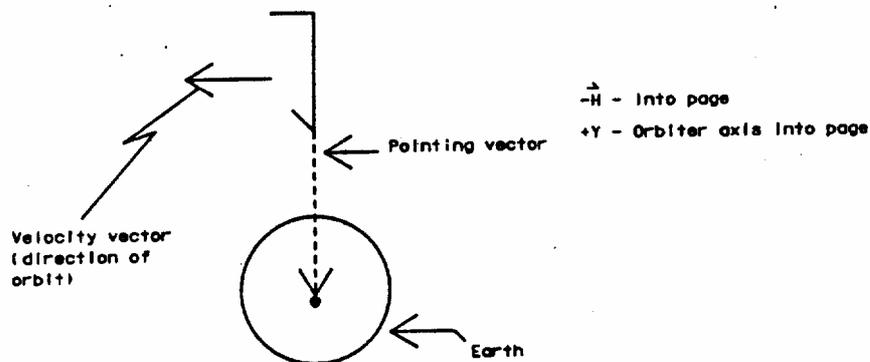


Figure 16.7: Body Vector along +X, OM=0.

2. Body vector = +Z axis (P = 270, Y = 0)
Target vector = 2 (Center of Earth)

OMICRON = 0°

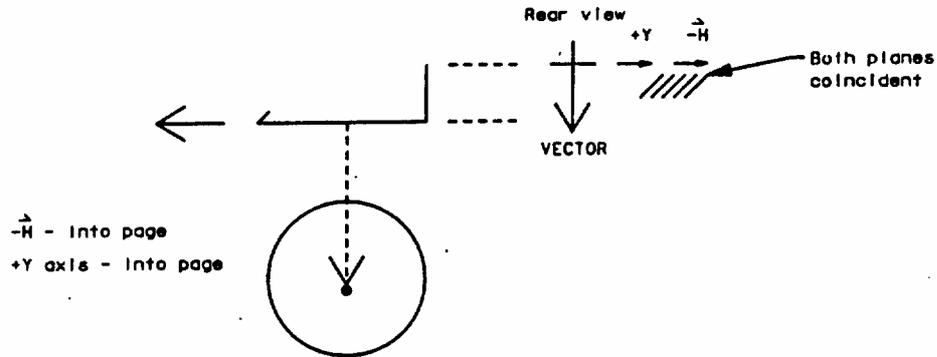


Figure 16-8: Body Vector along +Z, OM=0.

- 3. Body vector = P = 45, Y = 0
Target vector = 2 (Center of Earth)
OMICRON = 180°

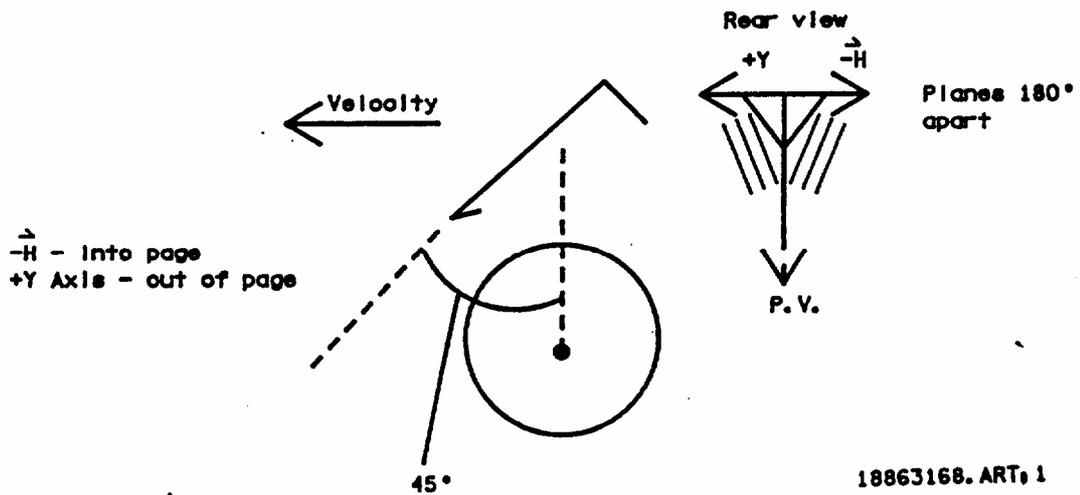


Figure 16-9: Body Vector in X-Z plane, OM=180.

There is a problem, however, if the body vector is defined along the Y axis. In this case, one of the conditions, which define one of the quarter planes, must change, since only one plane is definable (it takes two non-coincident lines to define a plane). The -Z Orbiter axis replaces the +Y axis if the body vector is within 1° of the Y-axis.

- 4. Body vector = -Y axis (P = 0, Y = 270)
Target vector = 2 (Center of Earth)
OMICRON = 0°

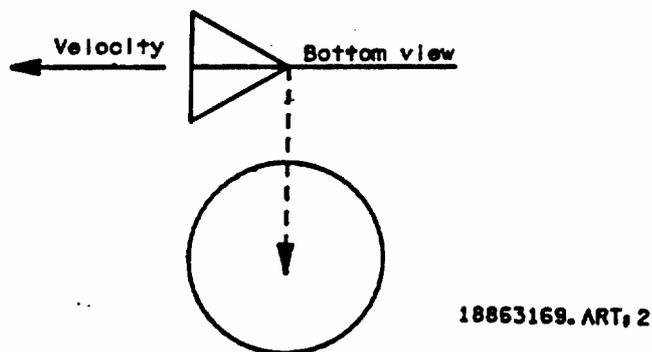


Figure 16-10: Alternate OMICRON definition.

Another special case is when the target vector is coincident (within 1°) with the negative or positive momentum vector. In this case, a vector from the center of the earth through the descending node takes the place of the negative momentum vector in order to measure OMICRON.

When the TRK option is initiated, the required attitude is determined based on the body vector (including OMICRON if it has been specified) and the target vector. This attitude is transformed to inertial M50 coordinates, displayed as the REQD attitude on the Universal Pointing display, and continuously updated as the required attitude moves in inertial space. To maneuver the Orbiter to the required attitude, the DAP software performs up to a three-axis eigen maneuver to the attitude, and once in attitude begins active track of the target. The maneuver about the eigen vector is performed in the same manner as when using the MNVR option. If OMICRON is not specified, the DAP software performs a two-axis eigen maneuver to the required attitude, computing the minimal maneuver necessary to point the Body Vector at the target. Flight data has shown that during the transition from maneuvering to tracking the target, it is possible for the attitude errors to temporarily exceed the attitude deadband. This is nominal DAP behavior and the vehicle will quickly maneuver back within the deadbands.

16.2.3 ROT Option

The ROT (rotation) option is a fairly simple routine. This logic takes the body vector defined by Items 15 and 16 and initiates a rotation rate about it, using ROT RATE on SPEC 20 (Items 10, 30, 23, or 43 depending on the DAP panel configuration). It does this by taking a snapshot of QBI at ROT initialization and propagating an eigen angle from this QBI at the ROT RATE. The propagation is performed each cycle (0.52 Hz) by multiplying the time from ROT initialization to present time by the ROT RATE. Whenever the ROT RATE changes, the propagation must be reinitiated by snapping QBI. Several conditions cause this to occur (see Table 16-1 for summary).

1. Initialization of the ROT option. This occurs when the crew selects a current ROT or when a future ROT becomes current.
2. Transitioning to AUTO while the ROT option is active.
3. Changing the ROT RATE on SPEC 20. This will occur when:
 - a. A new ROT RATE is entered in the present DAP while ROT is active.
 - b. A new DAP load (A/B,PRI/ALT/VERN) is crew selected (if the new ROT RATE is different) while ROT is active.

Note: No capability exists to alter ROT RATE or recenter the phase plane automatically in OPS 8. This is due to the fact that neither UNIV PTG nor DAP CONFIG (SPEC 20) exists in this OPS. For these reasons, care should be taken prior to OPS 8 (with an active ROT) to ensure the same ROT RATE exists in all DAP loads to be used while there. In OPS 8, however, the QBI will be snapped if a new DAP is selected with a different ROT RATE or if AUTO is selected (when previously in a manual DAP mode) and the ROT option is active.

Transitioning from MM201 to MM202 will neither snap QBI nor change the DAP settings. An active ROT option will continue uninterrupted until the MM 202 preburn maneuver is initiated (Item 27 EXEC) or OMS ignition. From this time until return to MM201, ROT is inhibited. Upon return to MM201, a new snapshot of QBI will only occur if the ROT RATE on SPEC 20 has been changed by Item entry or DAP reconfiguration.

TABLE 16-1: QBI SNAP DURING ROTATION

| Event | QBI snap? | Note |
|--|-----------|---------------------------------|
| Crew selects current ROT or future ROT becomes current | Yes | Nominal rotation initialization |
| DAP is switched to AUTO while ROT is active | Yes | None |
| New ROT RATE is entered by crew on SPEC 20 in current DAP with an active ROT (DAP in AUTO) | Yes | None |

| | | |
|---|------------------|---|
| New DAPLOAD (A/B or PRI/ALT/VERN) entered by crew while ROT is active | No (See Note) | Switching DAPLOADS will not snapshot QBI unless the new DAPLOAD has a different ROT RATE from the old DAPLOAD |
| MM 201 → 202 transition | No | |
| MM 202 → 201 transition | No (See Note) | QBI will be snapped at 202 → 201 transition if the ROT RATE in the current DAP is different from the ROT RATE prior to preburn maneuver or OMS ignition |

16.2.4 CNCL Option

This option is the simplest of the four options. There is no future CNCL capability. CNCL does two things: snapshots QBI and maintains that inertial attitude (re-centers the phase plane), and terminates any other current or future option (MNVR, TRK, ROT). Asterisks by all options CUR and FUT will disappear when CNCL is executed. Also, each time the DAP is moded to AUTO with the CNCL option active, QBI will be snapshot and an inertial attitude hold will be initiated. This option is selected automatically when entering OPS 2 from any OPS other than OPS 8.

Other ways to stop a Universal Pointing current option include:

- Select MNVR (Item 27) in the MNVR display in MM 202
- Select LVLH, INRTL, or FREE on the DAP panel
- Take a powered RHC out of detent (DAP will downmode to INRTL)

Other ways to stop a Universal Pointing future option include:

- Any of the above methods, provided that the DAP is not moded back to AUTO in MM 201 prior to the start time
- Change any parameter (Item 1 through 17) that affect the future option

16.3 UNIVERSAL POINTING DISPLAY (OPS DISPLAY)

This subsection presents a detailed outline of the UNIV PTG display (Figure 16-4).

CUR MNVR COMPL – shows in hrs:mins:secs of MET the estimated completion time of the current MNVR or TRK option based on current DAP selections, or the execution time of the current ROT option. This time is frozen when the total attitude errors are within a given tolerance of the DAP attitude deadband or when LVLH is selected on the DAP panel. When INRTL or FREE is selected and the attitude errors are within the attitude deadband, the completion time updates with the current MET each cycle. If the attitude error exceeds the deadband, the completion time will reflect the maneuver time back to the REQD attitude.

START TIME (Items 1 through 4) – used as time (days/hrs:mins:sec) to initiate auto maneuvers (MNVR, TRK, ROT). START TIME in the past or present will cause a maneuver to begin immediately (if DAP is in AUTO). An asterisk will be driven by the selected option under CUR. A START TIME in the future will cause an auto maneuver to begin when START TIME becomes current. An asterisk will be driven next to the selected option under FUT until the maneuver begins at which time the asterisk moves to CUR and blanks any other CUR asterisks.

MNVR OPTION (Items 5 through 7) – used for MNVR (Item 18) only. Provides inertial roll, pitch, yaw attitude in M50 coordinates, which become the REQD attitude when MNVR (Item 18) is executed. This is commonly used for star tracker aligns, crew optical alignment sight (COAS) calibrations, and pointing to other inertial targets. (R and P limits: 0 to 359.99 deg, Y limits: 0 to 90 and 270 to 359.99 deg)

TRK/ROT OPTIONS (Items 8 through 17) – used for TRK (Item 19) or ROT (Item 20) only.

1. TGT ID (Item 8) – used in combination with the body vector – to define the TRK attitude. The TGT ID's are:
 - a. TGT ID = 1: Orbiting vehicle – used for tracking an orbiting target. Rendezvous navigation must be enabled on SPEC 33 and a propagated target state vector is required for this TGT ID. The target LOS vector is calculated from state vectors of the Orbiter and the other vehicle. It is important to disable this TGT ID prior to entry to OPS 8. While Universal Pointing remains active in OPS 8, REL NAV does not. Excessive fuel usage could occur due to Universal Pointing chasing a stale target state vector.
 - b. TGT ID = 2: Center of the Earth – used to maintain an LVLH attitude. TGT ID is initialized to this selection.

- c. TGT ID = 3: Earth relative target – must be accompanied by LAT (Item 11), LON (Item 12), and ALT (Item 13). These parameters will flash when a 3 is entered for TGT ID, indicating manual crew inputs are required.
 - d. TGT ID = 4: Center of the Sun – the Sun RA (Item 9) and DEC (Item 10) will appear when 4 is entered for TGT ID. RA and DEC will be driven dynamically as the Sun moves in inertial space (although the outputs to the UNIV PTG display are not dynamically updated) and will be used to determine LOS vector to the center of the Sun.
 - e. TGT ID = 5: Celestial Target – used to point body vector at any celestial target. It must be accompanied by RA (Item 9) and DEC (Item 10). These parameters will flash when a 5 is entered for TGT ID, indicating manual crew inputs are required.
 - f. TGT ID = 11 to 110: Navigation stars – used to point the body vector at star already defined in GPC NAV table. The selected star RA (Item 9) and DEC (Item 10) will appear when 11-110 is entered for TGT ID.
- 2. RA (Item 9) – Right Ascension (0 to 359.99°)
 - 3. DEC (Item 10) – Declination (-90 to 90°)

Note: The RA and DEC fields are the only “hot” fields on the left side of the Universal Pointing display. This means that if a track option is current with a TGT ID of 5, then any changes made to RA or DEC via an Item 9 or 10 will immediately cause the DAP to maneuver to the new RA or DEC without another Item 19.

- 4. LAT (Item 11) – Latitude (-90 to 90°, + = North, - = South)
- 5. LONG (Item 12) – Longitude (-180 to 180°, + = East, - = West)
- 6. ALT (Item 13) – Altitude (-3444 to 20000 nm)
- 7. BODY VECT (Item 14) – defines LOS vector centered at the Orbiter c.g. to be pointed at TGT ID. It is defined by the pitch (Item 15) and yaw (Item 16) values, referenced to the Orbiter +X axis. Defined body vectors are:
 - a. BODY VECT = 1: +X AXIS – initialized to this selection. This entry points Orbiter +X axis at TGT ID. When 1 is entered; P = 0, Y = 0 will appear (Items 15 and 16). OMICRON will flash indicating crew input is optional.
 - b. BODY VECT = 2: -X AXIS – points Orbiter -X axis at TGT ID. When 2 is entered; P = 180, Y = 0 will appear (Items 15 and 16). OMICRON will flash indicating crew input is optional.
 - c. BODY VECT = 3: -Z AXIS – points Orbiter -Z axis at TGT ID. When 3 is entered; P = 90, Y = 0 will appear (Items 15 and 16). OMICRON will flash indicating crew input is optional.
 - d. BODY VECT = 4: -Y Star Tracker – points Orbiter -Y Star Tracker axis at TGT ID. When 4 is entered; P = 0, Y = 280.57 will appear (Items 15 and 16). OMICRON will flash indicating crew input is optional.
 - e. BODY VECT = 5: Selectable – points any crew-selected Orbiter BODY VECT at TGT ID. P, Y, and OMICRON will flash indicating crew input is required for pitch and yaw, and optional for OMICRON.
- 8. P (Pitch - Item 15) – value of 0 to 359.99° must be entered if BODY VECT = 5. If BODY VECT = 1 through 4, pitch value will be static as defined above.
- 9. Y (Yaw - Item 16) – value of 0 to 90° or 270 to 359.99° must be entered if BODY VECT = 5. If BODY VECT = 1 through 4, yaw value will be static as defined above.
- 10. OM (OMICRON - Item 17) – initialized at a value of 0 (three-axis tracking initialized). A value of 0 to 359.99° may be entered for any BODY VECT selection. If OMICRON is left blank, the Orbiter will not be constrained about the defined body vector when the maneuver is executed (two-axis tracking). In this case, a required body quaternion will be calculated that will result in a minimum attitude maneuver to get from the current attitude to the required attitude. Once the Orbiter is in the required attitude, it will be maintained until another maneuver is executed. To blank the OMICRON when a value has already been selected, a new TGT ID (Item 8) must be entered.

Note: Once flashing, RA, DEC, LAT, LONG, ALT will continue to flash until:

- 1. New TGT ID is entered
- 2. Data is entered into the field
- 3. Maneuver is executed

Once flashing, P (Item 15) and Y (Item 16) will continue to flash until:

- 1. BODY VECT is changed
- 2. Data is entered into the field
- 3. Maneuver is executed

Once flashing, OM will continue to flash until:

- 1. Data is entered into the field
- 2. Maneuver is executed

START MNVR (Item 18) – if START TIME is current or in the past, a maneuver to the defined inertial attitude will be initiated (asterisk will be displayed under CUR). If START TIME is in the future (asterisk will be displayed under FUT), the current option will be active until the START TIME, at which time the maneuver will be initiated (asterisk will move to CUR).

START TRK (Item 19) – if START TIME is current or in the past, a maneuver to the attitude defined by the TGT ID and BODY VECT will be initiated (asterisk will be displayed under CUR). If START TIME is in the future (asterisk will be displayed under FUT), the current option will be active until the START TIME, at which time the maneuver will be initiated (asterisk will move to CUR).

START ROT (Item 20) – if START TIME is current or in the past, the current BODY VECT orientation in inertial space is snapshot, and an Orbiter rotation about the defined BODY VECT begins (asterisk will be displayed under CUR). If START TIME is in the future (asterisk will be displayed under FUT), the current option will be active until the START TIME, at which time the BODY VECT orientation in inertial space is snapshot, and an Orbiter rotation about the defined BODY VECT begins (asterisk will move to CUR) .

CNCL (Item 21) – initialized to this selection. When executed, all current and future options cease processing (all asterisks extinguish) and an inertial attitude hold is initiated.

ATT MON (Items 22-24) options – Selects reference for display of attitude angles, errors, and rates (CUR, REQD, ERR, RATE).

1. MON AXIS (Item 22) – An Item entry of 1 (initialized to this selection) selects the + X Axis for display. Quantities are displayed with respect to the standard Orbiter body axis coordinate system. An Item entry of 2 selects the –X Axis for display. Quantities are displayed relative to a coordinate system rotated 180° about the Orbiter Z axis (yaw).
2. ERR TOT (Item 23) – Causes errors to be displayed which are defined as the difference between the CUR and REQD attitude (initialized to this selection). Errors are eigen axis errors and are displayed in the “fly-to” sense. This item also affects ADI error needles (which are also displayed in the “fly-to” sense) and is mutually exclusive with Item 24.
3. ERR DAP (Item 24) – Causes errors to be displayed which are defined as the difference between CUR attitude (as determined by State Estimator) and desired attitude (which is the center of the phase plane). Errors are displayed in the “fly-to” sense. This item also affects ADI error needles (which are also displayed in the “fly-to” sense) and is mutually exclusive with Item 23.

NOTE: Entry into MM 202 will cause the attitude errors displayed on the ADI to be total errors regardless of the selection on UNIV PTG (the downlisted errors will continue to be what was selected on UNIV PTG prior to the transition). Upon return to MM201, the errors displayed on the ADI will return to the UNIV PTG selection. The attitude errors displayed on the ADI (and the downlisted errors) will continue to be what was selected on UNIV PTG after transitioning to OPS 8.

CUR – Display of current attitude in M50 coordinates using pitch, yaw, roll Euler sequence (although data is displayed in the order of roll, pitch, yaw for crew convenience).

REQD – Display of required attitude (See Table 16-2). If MNVR (Item 18) is executed, REQD is set equal to MNVR OPTION: if TRK or ROT (Item 19 or 20) is executed, REQD is set equal to a rotating inertial quaternion.

ERR – Difference between CUR and REQD if ERR TOT (Item 23) is selected; otherwise ERR DAP (Item 24) is displayed.

RATE – Orbiter body rate (deg/sec) as determined by State Estimator.

When attitudes are displayed on the UNIV PTG display using the pitch, yaw, roll Euler sequence, a yaw near 90° or 270° will cause multiple solutions to be possible for pitch and roll. The software solves the solution ambiguity by setting roll to 0 and solving for pitch. This may cause an apparent discontinuity in the data, but the displayed Euler sequence will always accurately represent the quaternion. See Table 16-2 for the values for required attitude, DAP errors, and total errors based on selected DAP and Universal Pointing inputs.

TABLE 16-2: Required Attitude and Error Values

| DAP selection | Required Attitude | DAP Errors | Total Errors |
|---|--|--|---|
| Active Universal Pointing attitude control (DAP in AUTO) | Based on CUR selection (Items 18-20) or FUT selection (Items 18-19) if no CUR selection | Difference between Current attitude and DAP Desired attitude (center of phase plane) | Difference between Current attitude and Required attitude |
| CNCL (Item 21) selected in Universal Pointing (DAP in AUTO) | Inertial attitude at time of CNCL selection or inertial attitude when returning to AUTO if CNCL selected | Difference between Current attitude and DAP Desired attitude (center of phase plane which equals Required attitude) | Difference between Current attitude and Required attitude |
| DAP in LVLH | Inertial attitude at time of LVLH selection, will update with Current attitude (Required = Current) | Difference between Current attitude and manual DAP desired attitude (center of phase plane which updates as Orbiter orbits Earth) | Zeros are displayed |
| DAP in INRTL | Based on Items 18-21 on Universal Pointing display | Difference between Current attitude and manual DAP desired attitude (center of phase plane which is QBI snapshot at time of INRTL selection) | Difference between Current attitude and Required attitude |
| DAP in FREE | Based on Items 18-21 on Universal Pointing display | Zeros are displayed | Difference between Current attitude and Required attitude |

The Reboost section of the Universal Pointing display will be covered in Section 16.9.

16.4 DAP CONFIGURATION DISPLAY (SPEC 20)

The DAP CONFIG display (Figure 16-3) provides a means for the crew to review or change the parameters that the DAP utilizes when carrying out the functions specified by Universal Pointing (reference Sections 16.2 and 16.3) and the DAP panel (reference Section 16.5). A table of 30 DAP loads is stored in GPC memory as I-loads. This table is split into 15 DAP A loads (A1 through A15) and 15 DAP B loads (B1 through B15). Each DAP load consists of 19 parameters, which can be viewed, temporarily modified, or permanently modified on the DAP CONFIG display. The DAP A loads can be displayed in the first column and the DAP B loads can be displayed in the second column of the display. The DAP CONFIG display is available in MM 201 and 202 and is not available in OPS 1, 3 or 8.

The following is a description of each item entry, along with the Universal Pointing and RCS modes (PRI, ALT or VERN) in which the parameter applies. More information on these parameters can be found in the Data Processing System Dictionary (DPS DICT, JSC-48017) and the Displays and Control FSSR (STS 83-0020).

DAP A and DAP B (Items 1 and 2) – permit a DAP A load and a DAP B load to be selected from the DAP table and displayed (DAP A will be displayed next to Items 10 through 28, DAP B will be displayed next to Items 30 through 48). The DAP load that is displayed will be used by the orbit DAP software when the corresponding DAP button (A or B which are mutually exclusive) is selected on the DAP panel (the active DAP load will be overbright on the display). An integer value of 1 through 15 can be entered for either item to display the desired DAP load. A1 and B1 are the initial settings. If any item in the respective column is modified, the field after Item 1 or Item 2 will blank (a 0 will be displayed on the MCC ground displays), indicating a temporary edit in that column has taken place. If the DAP with a temporary edit is selected on the DAP panel, the value entered for the parameter will be used by the DAP software, however, it will not be saved in the DAP table. The next time the DAP is selected on SPEC 20, the edited parameter will revert to the DAP table value.

DAP A EDIT and DAP B EDIT (Items 3 and 4) – mutually exclusive entries which allow the crew to load and permanently edit a DAP load from the DAP table in the third column of SPEC 20 (Items 50 through 68). An integer value of 1 through 15 can be entered for either item to indicate the desired DAP load, Item 3 corresponding to DAP A and Item 4 corresponding to DAP B. A value of 0 entered for either Item 3 or 4 will cause both DAP EDIT values to blank and Items 50 through 68 to blank.

DAP EDIT LOAD (Item 5) – used to permanently store any DAP EDIT changes made via Items 50 through 68. Any changes made in the DAP EDIT column are not stored in the DAP table until and Item 5 is executed. If the DAP EDIT ID is the same as one of the selected DAP IDs (Item 1 or 2), then when the Item 5 is executed, the selected DAP parameters will immediately change to the new values. When any item in the edit column (Items 50 through 68) is modified, LOAD will flash until an Item 5 is executed.

PRI section (PRI will be overbright if PRI is selected on the DAP panel):

ROT RATE (Items 10, 30, 50) – specifies the rate at which the vehicle will rotate for any Universal Pointing maneuver, whether it is commanded by the MNVR, TRK, or ROT options (DAP panel must be in AUTO/PRI or ALT). It also specifies the rate at which the vehicle will maneuver if the RHC is deflected between detent and the soft-stop while the DAP is in INRTL/PRI or ALT or LVLH/PRI or ALT for the rotational axes in which DISC RATE is selected. (Allowable range of entries: .05 to 2.0 deg/sec)

ATT DB (Items 11,31,51) – specifies how far the Orbiter can drift from the desired attitude (per axis) before the DAP will fire jets to reduce the attitude error. This parameter determines the width of the orbit DAP phase plane and is used anytime PRI or ALT is selected and the DAP is not in FREE. (Allowable range of entries: .1 to 40.0 deg)

RATE DB (Items 12,32,52) – specifies the rate error (per axis) from the desired rotational rate that the orbiter is allowed before the DAP will fire jets to more accurately achieve the desired rate. This parameter determines the upper and lower boundaries of the orbit DAP phase plane and is used anytime PRI is selected and the DAP is not in FREE. (Allowable range of entries: .1 to 5 deg/sec)

ROT PLS (Items 13,33,53) – specifies the change in rotational velocity (via an open loop calculation) that the DAP will try to achieve if the RHC is deflected (for a single pulse) between detent and the soft-stop while the DAP is in INRTL/PRI or ALT, LVLH/PRI or ALT, or FREE/PRI or ALT for the rotational axes in which PULSE is selected (PULSE is the only rotational selection available when in FREE). The DAP will always assume that full rotational control authority is available when calculating the jet on-time required to achieve the rate change (e.g. it will assume that both forward and aft pitch jets will fire for a positive pitch command even when the DAP is configured for tail-only control). (Allowable range of entries: .001 to 1.0 deg/sec)

COMP (Items 14,34,54) – specifies the rotation rate about a given axis that is allowed as the result of jet commands in another axis. If cross-coupling results in rates above those specified by this parameter, the jet commands will be altered. If the values entered for this parameters are below I-loaded values that correspond to the minimum impulse of the RCS jets, this parameter will be ignored and no compensation will be performed. A value of zero specifies that no compensation is desired. This parameter is used when AUTO/PRI or ALT is selected on the DAP. (Allowable range of entries: 0 to .999 deg/sec)

P OPTION (Items 15,35,55) – allows the crew to select ALL (nose/tail), NOSE (nose-only), or TAIL (tail-only) control in the pitch axis. Executing the item entry toggles through the choices. These control options are active any time that PRI is selected on the DAP. If nose-only control is selected for pitch, the DAP will still command aft jets to fire for roll control. If Lo Z is selected, it will take precedence over nose-only or tail-only pitch options.

Y OPTION (Items 16,36,56) – allows the crew to select ALL (nose/tail), NOSE (nose-only), or TAIL (tail-only) control in the yaw axis. Executing the item entry toggles through the choices. These control options are active any time that PRI is selected on the DAP. If nose-only control is selected for yaw, the DAP will still command aft jets to fire for roll control.

TRAN PLS (Items 17,37,57) – used any time the THC is deflected while translation PULSE is selected on the DAP panel for the given axis. It specifies the change in translation velocity that will be achieved by a single THC deflection. If ALT or VERN is selected on the DAP panel, the DAP will upmode to PRI (THC hot-stick downmode) when the THC is deflected and mode back to ALT or VERN when the translation jet firings are complete. (Allowable range of entries: .001 to 5.0 ft/sec)

ALT section (ALT will be overbright if ALT is selected on the DAP panel):

RATE DB (Items 18,38,58) – specifies the rate error (per axis) from the desired rotational rate that the orbiter is allowed before the DAP will fire jets to more accurately achieve the desired rate. This parameter determines the upper

and lower boundaries of the orbit DAP phase plane and is used anytime ALT is selected and the DAP is not in FREE. (Allowable range of entries: .05 to 5 deg/sec)

JET OPT (Items 19,39,59) – allows the crew to select ALL (nose/tail) or TAIL (tail-only) control for the pitch and yaw axes (nose-only is not supported in ALT). Executing the item entry toggles between the choices. These control options are active any time that ALT is selected on the DAP. No matter the selection, the DAP will still command aft jets only to fire for roll control. If Lo Z is selected, it will take precedence over the tail-only option.

JETS (Items 20,40,60) – allows the crew to specify the maximum number of jets (1, 2 or 3) that the DAP is allowed to fire simultaneously when ALT is active. There is no scenario in which more than the specified number of jets will fire, but the DAP may fire less than the number specified if doing so satisfies the rotational commands that have been issued. Also, as is discussed in the JET_SELECT software section, any jets that do fire simultaneously will be from different PRCS jet groups. ALT DAP can never fire more than one jet from a group at the same time.

ON TIME (Items 21,41,61) – allows the crew to specify how long jets are allowed to continuously fire when ALT is selected. If a one-second on time is specified and # JETS is set to three, then the DAP is allowed to fire up to three jets simultaneously and continuously for up to one second (they may fire for a shorter period of time if the full second is not required). After the jet firing has completed, the DAP will impose a mandatory delay before another firing is allowed. The ON TIME parameter will be ignored if the DELAY is set to zero (e.g. if no delay is required between firings, then the DAP can command jet firings as long as required to satisfy the phase plane constraints). (Allowable range of entries: .08 to 9.99 sec)

Note: .08 seconds is the minimum jet on time due to RCS hardware constraints (determined by the time required to open and close jet fuel and oxidizer valves).

DELAY (Items 22,42,62) – allows the crew to specify the minimum delay time that should be imposed between jet firings when ALT is active. For example, if a delay of seven seconds is specified, then a minimum of seven seconds will pass between the time that one jet firing ends and the next jet firing begins (note that "jet firing" here could refer to one, two or three jets firing simultaneously, depending on what is specified in # JETS). If the delay is set to zero, then the DAP is allowed to fire jets at any time. (Allowable range of entries: 0 to 99.99 sec)

VERN section (VERN will be overbright if VERN is selected on the DAP panel):

ROT RATE (Items 23, 43, 63) – specifies the rate at which the vehicle will rotate for any Universal Pointing maneuver, whether it is commanded by the MNVR, TRK, or ROT options (DAP panel must be in AUTO/VERN). It also specifies the rate at which the vehicle will maneuver if the RHC is deflected between detent and the soft-stop while the DAP is in INRTL/VERN or LVLH/VERN for the rotational axes in which DISC RATE is selected. (Allowable range of entries: .002 to 1.0 deg/sec)

ATT DB (Items 24,44,64) – specifies how far the Orbiter can drift from the desired attitude (per axis) before the DAP will fire jets to reduce the attitude error. This parameter determines the width of the orbit DAP phase plane and is used anytime VERN is selected and the DAP is not in FREE. (Allowable range of entries: .01 to 40.0 deg)

RATE DB (Items 25,45,65) – specifies the rate error (per axis) from the desired rotational rate that the orbiter is allowed before the DAP will fire jets to more accurately achieve the desired rate. This parameter determines the upper and lower boundaries of the orbit DAP phase plane and is used anytime VERN is selected and the DAP is not in FREE. (Allowable range of entries: .01 to .5 deg/sec)

ROT PLS (Items 26,46,66) – specifies the change in rotational velocity (via an open loop calculation) that the DAP will try to achieve if the RHC is deflected (for a single pulse) between detent and the soft-stop while the DAP is in INRTL/VERN, LVLH/VERN, or FREE/VERN for the rotational axes in which PULSE is selected (PULSE is the only rotational selection available when in FREE). (Allowable range of entries: .001 to 0.5 deg/sec)

COMP (Items 27,47,67) – specifies the rotation rate about a given axis that is allowed as the result of jet commands in another axis. If cross-coupling results in rates above those specified by this parameter, the jet commands will be altered. If the values entered for this parameters are below I-loaded values that correspond to the minimum impulse of the RCS jets, this parameter will be ignored and no compensation will be performed. A value of zero specifies that no compensation is desired. This parameter is used when AUTO/VERN is selected on the DAP. (Allowable range of entries: 0 to .999 deg/sec)

CNTL ACC (Items 28,48,68) – allows the crew to select 1 of 10 sets of Control Acceleration I-loads (represented by an integer ranging from 0 to 9) that are used to calculate MAG_CONT_ACCL in the DAP RECON software. MAG_CONT_ACCL affects the shape of the phase plane, jet on time while in rotational PULSE mode for PRI, ALT, and VERN, and jet select logic when ALT or VERN is selected. Control accelerations are table references that contain mass property/moment of inertia information and notch filter data. Data can be uplinked by MCC to slots 1 through 9 to update the I-loads during a flight (slot 0 contains default Orbiter mass properties and no notch filters). An uplink to a CNTL ACC slot may contain: orbiter c.g., moments of inertia, products of inertia, and six notch filters (an uplink to a CNTL ACC slot can only be performed if the CNTL ACC is not currently active). Control Accels are varied throughout a flight for different docked configurations or payload orientations on the RMS. A higher moment of inertia in a certain axis would cause a lower acceleration/rotation rate about that axis.

Note: When PRI is selected on the DAP panel, CNTL ACC 0 is used. For default values used for rotation maneuvers in PULSE mode, see Section 16.7.2 paragraph 1d.

NOTCH FLTR ENA (Item 6) – allows the crew to enable or disable a series of filters (max of 6 per Control Accel) used by the state estimator to attenuate frequency response and structural loads. A notch filter damps out adverse structural frequencies that could be induced by jet firings during different payload orientations on the RMS or docked configurations. Notch filters contribute to flight control system stability and minimize deadband violations which minimizes propellant use. If a CNTL ACC is selected that has notch filters, Item 6 will be initialized at the enabled state. If CNTL ACC 0 (which does not have notch filters) or a CTNL ACC is selected that has no notch filters, Item 6 will be initialized at the disabled state.

X JETS ROT ENA (Item 7) – allows the crew to enable or disable the ability of the Orbiter to control the pitch axis using the -X and +X PRCS jets when ALT is selected. For ISS flights, if the combined Orbiter/ISS center of mass is sufficiently displaced from the X-Y plane of the orbiter, the moment arm will enable a large enough torque to be produced when the X jets are fired to allow pitch control.

The Reboost section of Spec 20 will be covered in Section 16.9.

16.5 DAP PANEL PUSH BUTTON INDICATORS (PBI)

The DAP panels are located on orbiter panels C3 and A6 (Figure 16-2). These panels each contain 23 crew-selectable pushbutton indicators (PBIs). Each panel also contains one spare (blank) PBI, which is currently used to activate Post Contact Thrusting during the docking sequence with the ISS. An input on either panel will be reflected on both panels.

16.5.1 Function of DAP Panel PBIs

The OPS 2 and 8 function of each of these momentary PBIs is summarized below. Not all PBIs are available in OPS 1 and 3, and those that are available do not necessarily function the same as in OPS 2 and 8. Reference the TRANS DAP System Brief for OPS 1 and 3 functionality. Since this section focuses mainly on the function of the PBIs themselves, more details of the various modes invoked by the PBIs can be found in other sections.

DAP A/B PBIs – mutually exclusive PBIs allow the crew to select the control parameters from the SPEC 20 DAP CONFIG display (Figure 16-3) that are loaded in the first column (DAP A) or the second column (DAP B).

AUTO PBI – mode that is dependent on UNIV PTG display or Orbit MNVR display information for its operation (See Sections 16.3 and 16.6). There are no active rotation submodes (i.e. PULSE or DISC RATE) in AUTO, although the submode PBIs will not be extinguished when AUTO is selected. AUTO is selected via PBI only. Selection of AUTO will extinguish the PBIs with which it is mutually exclusive (INRTL, LVLH, FREE), but will not affect any of the other PBIs on the DAP panel. When in RCS DAP, any downmode of the DAP to INRTL, LVLH, or FREE and subsequent re-selection of AUTO will cause the maneuver or requested attitude from Universal Pointing to resume automatically. In addition to the selection of INRTL, LVLH, and FREE on the DAP, an RHC deflection past the softstop will downmode the DAP out of AUTO.

INRTL (Inertial) PBI – selection will automatically mode the rotation submodes to DISC RATE and will place the vehicle in inertial attitude hold at the current inertial attitude. At the same time, the PBIs that are mutually exclusive with INRTL (AUTO, LVLH, FREE) will be extinguished. The rotation PULSE PBIs will also be extinguished, since they are mutually

exclusive with DISC RATE. When INRTL is selected, the desired vehicle rates will be zero unless the rotational hand controller (RHC) is deflected. After INRTL has been selected, the crew is free to select PULSE in any or all rotation axes. Any axis that is in INRTL/PULSE will be in free drift.

If the AUTO mode is active, deflecting the RHC out of detent will mode the DAP to INRTL. In this case, no other DAP panel PBIs will be affected. The rotation submodes (PULSE or DISC RATE) that were previously active will continue to be active.

LVLH (Local Vertical/Local Horizontal) PBI - selection will automatically mode the rotation submodes to DISC RATE and will place the vehicle in LVLH attitude hold at the current LVLH attitude. At the same time, the PBIs that are mutually exclusive with LVLH (AUTO, INRTL, FREE) will be extinguished. The rotation PULSE PBIs will also be extinguished, since they are mutually exclusive with DISC RATE. After LVLH has been selected, the crew is free to select PULSE in any or all rotation axes. Any axis that is in LVLH/PULSE will be in free drift.

FREE (Free Drift) PBI – selection will automatically mode the rotation submodes to PULSE and will place the vehicle in free drift. At the same time, the PBIs that are mutually exclusive with FREE (AUTO, INRTL, LVLH) will be extinguished. The DISC RATE PBIs will also be extinguished, since they are mutually exclusive with PULSE. While FREE is active, DISC RATE cannot be selected in any axes. FREE can only be deactivated by selecting a mutually exclusive mode. All other DAP panel PBIs are available for selection.

PRI (Primary DAP mode) PBI – selection will result in the use of the Primary Reaction Control System (PRCS) and Primary jet select logic. The PBIs associated with its mutually exclusive modes (ALT and VERN) will be extinguished, but the DAP panel will be otherwise unaffected. When PRI is active, the appropriate parameters from SPEC 20 DAP CONFIG (Figure 16-3) will be utilized for rotation and translation control (See Section 16.4). THC hot-stick capability also allows PRI to be activated by deflecting THC when ALT or VERN is active, although the DAP will mode back to its previous configuration as soon as the PRCS translation jet firings are complete.

ALT (Alternate DAP mode) PBI – selection will result in the use of the Primary Reaction Control System (PRCS) along with the ALT jet select logic. The PBIs associated with its mutually exclusive modes (PRI and VERN) will be extinguished, but the DAP panel will be otherwise unaffected. When ALT is active, the appropriate parameters from SPEC 20 DAP CONFIG (Figure 16-3) will be utilized for rotational control (See Section 16.4).

VERN (Vernier jets) PBI – selection will result in the use of the 6 jets in the Vernier Reaction Control System (VRCS) and Vernier jet select logic. The PBIs associated with its mutually exclusive modes (PRI and ALT) will be extinguished, but the DAP panel will be otherwise unaffected. When VERN is active, the appropriate parameters from SPEC 20 DAP CONFIG (Figure 16-3) will be utilized for rotational control (See Section 16.4).

Note: If one of the Vernier jets fails (or is manually deselected by the crew) while VERN is selected, the DAP will downmode to FREE and a DAP RECONFIG message will be annunciated. In this situation, the AUTO, INRTL, or LVLH PBI can be reselected, so care must be taken to determine if attitude control authority was lost (See Flight Rule A8-60). If PRI or ALT is selected and then VERN is reselected, the DAP will again downmode to FREE and the DAP RECONFIG message will be re-annunciated. If a Vernier jet fails while PRI or ALT is selected, the DAP RECONFIG message will be annunciated, but the DAP downmode to FREE will not occur.

DISC RATE (Rotation Discrete Rate) PBI – selection for any axis will extinguish the rotation PULSE PBI for that axis. When AUTO is active, DISC RATE can be selected but will not be utilized. When INRTL or LVLH are active, any axes in DISC RATE will be in attitude hold. When a powered RHC is taken out of detent (and not beyond the soft stop), the Orbiter will rotate about the commanded axis at the ROT RATE on the DAP CONFIG display for the selected DAP. When the RHC is back in detent, the Orbiter will maintain the new attitude. When FREE is active, DISC RATE cannot be selected.

PULSE (Rotation Pulse) PBI – selection for any rotational axis will extinguish the DISC RATE PBI for that axis. When AUTO is active, PULSE can be selected but will not be utilized. When INRTL or LVLH are active, any axes in PULSE will be in free drift. When a powered RHC is taken out of detent (and not beyond the soft stop), the Orbiter will fire jets for an open loop calculated duration to rotate about the commanded axis at the ROT PLS rate on the DAP CONFIG display for the selected DAP. When the RHC is back in detent, the Orbiter will again be in free drift. Rotation PULSE is forced in all axes when the FREE PBI is active.

Note: The position of the Aft Sense switch (-X/-Z) on panel A6 affects the Rotation DISC RATE and PULSE PBIs on the aft DAP panel. Changing the position of the switch effectively swaps the Roll and Yaw columns. No other portion of the aft DAP panel is affected by the position of the Sense switch. If the -Z sense is selected, the Roll column on the aft DAP panel will equal the Yaw column on the forward DAP panel, and vice versa.

NORM (Translation Normal) PBI – selection in the X and Y axes will extinguish the translation PULSE PBI for that axis. Selection in the Z axis will extinguish both the PULSE and HIGH Z PBIs. LOW Z will not be affected. With NORM selected, PRCS jets will fire continuously with a powered THC out of detent in the appropriate axis.

PULSE (Translation Pulse) PBI – selection in the X and Y axes will extinguish the NORM PBI for that axis. Selection in the Z axis will extinguish both the NORM and HIGH Z PBIs. LOW Z will not be affected. With PULSE selected, PRCS jets will fire to achieve TRANS PLS rate on the DAP CONFIG display with a powered THC out of detent in the appropriate axis.

LOW Z PBI – selection will not affect the NORM or PULSE PBIs. If HIGH Z is active (neither NORM nor PULSE active in Z axis) when LOW Z is selected, the HIGH Z PBI will be extinguished and NORM will become active in the Z axis, along with LOW Z. All upfiring RCS jets will be inhibited while LOW Z is active, regardless of whether a translation or rotation is being performed. This mode responds to THC deflections in the +Z body axis direction by commanding ON two forward-firing jets (group 1) and two aft firing jets (one in each of groups 7 and 8). Since each of these jets has a small component of thrust in the -Z direction, there is a net +Z acceleration. If a nose-only or tail-only jet option is selected while LOW Z is active, the DAP will default to nose and tail control in the pitch axis when PRI is selected. If the tail-only option is selected in ALT while LOW Z is active, the DAP will default to nose and tail control in all axes. LOW Z can be deactivated either by depressing the LOW Z PBI or by selecting HIGH Z.

Since the up-firing jets remain disabled the entire time LOW Z is in effect, it is important to disable the mode prior to an OMS burn in MM202. If for any reason a single OMS engine is used for a burn while LOW Z is enabled, roll control will be sloppy. Also, LOW Z cannot be disabled while in the OMS TVC DAP mode, even though the DAP lights change with an alternate mode selection. Upon completion of the burn and re-entry into the RCS DAP mode, however, any new mode selection made during the burn will be enabled.

It is also important to note that LOW Z is not smart enough to know that group 1, 7 or 8 jets have failed. In other words, a LOW Z, +Z THC deflection will command jets in groups 1, 7 and 8 without regard to the availability of the jets. For example, if all jets in groups 1 and 7 have failed, one jet in group 8 will fire if a +Z THC command is issued.

HIGH Z PBI – selection will extinguish the LOW Z PBI, as well as the Z axis NORM and PULSE PBIs (PULSE Z, NORM Z and HIGH Z are mutually exclusive). If a +Z translation is performed with HIGH Z active, all nine upfiring PRCS jets will fire. -Z translations are not affected by selection of HIGH Z (jets will fire as in NORM mode although the NORM PBI is not selected).

16.5.2 DAP Panel PBI Redundancy Management

The six PBIs on the top row of each DAP panel are all three-contact pushbuttons and use standard three-level switch redundancy management (RM). The remaining 18 PBIs are two-contact pushbuttons and utilize standard two-level RM. This processing is performed at 6.25 Hz. More details on switch RM can be found in section 4.12 of the RM Functional Subsystem Software Requirements (STS 83-0010).

All the DAP panel PBIs also utilize a function known as leading edge detection, which is implemented in the orbit DAP UNPACK_SW module rather than in the switch RM module. Switch RM processes each contact on each PBI for both DAP panels. If the selection filter (SF) determines that a new mode is being selected (2-of-2 contacts at the two-level; at least 2-of-3 at the three-level) on either DAP panel, the discrete associated with that mode on that DAP panel will be sent to the Flight Control Switch (FC_SWIT) processor (See Figure 16-11). This discrete will remain ON as long as the selected output from SF is set ON. FC_SWIT will OR the forward and aft DAP panel inputs together from switch RM and send them as a single discrete input for each available DAP mode to UNPACK_SW. This discrete also is set high for as long as the SF output is ON. If UNPACK_SW receives any input discrettes from FC_SWIT that have changed from OFF to ON, that DAP mode will be activated. If no inputs have changed from OFF to ON from the previous cycle to the current cycle, then no DAP modes will be changed on the current cycle.

For an example of how the leading edge detection logic is utilized, assume that when the forward VERN PBI is depressed, both contacts stick ON. VERN will become active on the cycle that the contacts are set high. On each

subsequent cycle, UNPACK_SW will see the flag associated with the VERN PBIs set high. If PRI or ALT is selected, that mode will become active, VERN will be deactivated and the VERN PBI will be extinguished even though its contacts are still ON. Unless the VERN PBI failure is cleared, there is now no way to select VERN via PBI. If the forward PBI is depressed, no change in state will be seen since the contacts are already failed high. If the aft PBI is depressed, SF will recognize the change in state. However, since the forward and aft outputs from SF are OR'd together before being sent to UNPACK_SW, the flag sent to UNPACK_SW will already be set high due to the forward PBI failure, UNPACK_SW will not see a change in state, and VERN will not be selected. In summary, if a DAP PBI is failed high, any mutually exclusive mode can be selected nominally. However, doing so will not allow the mode associated with the failed PBI to be reselected. (An OPS 3 to 2 transition will reinitialize the DAP panel. Any mode that has a PBI failed high will then be seen as a new input, and that mode will automatically be selected by UNPACK_SW.)

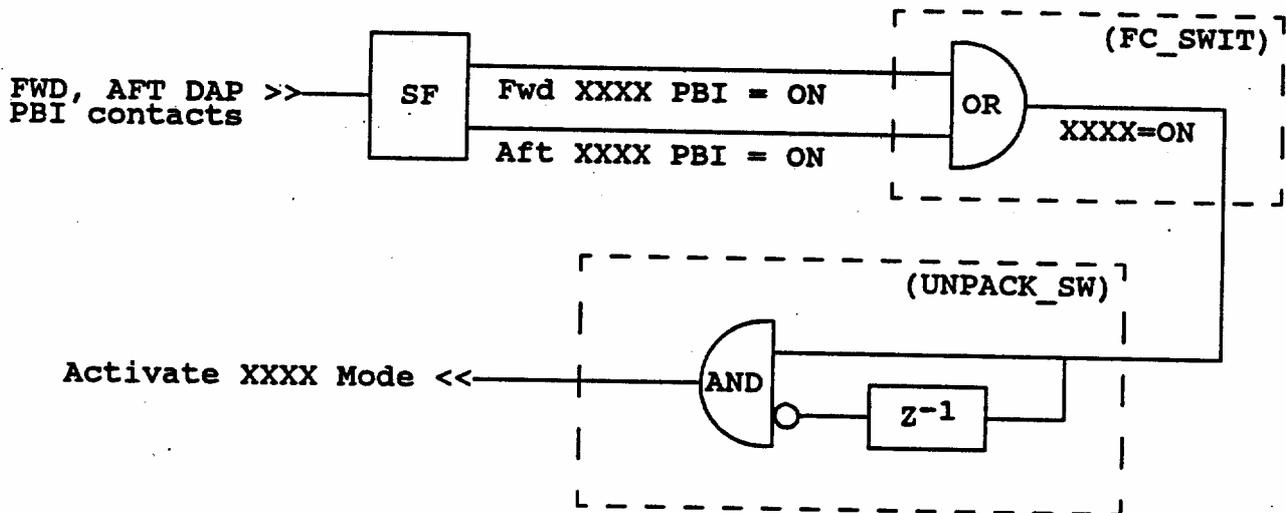


Figure 16-11: Leading Edge Detection Logic

16.5.3 DAP Panel PBI Prioritization

Even with leading edge detection, prioritization among mutually exclusive DAP panel PBIs is still necessary. However, the only time this logic is used is when two mutually exclusive PBIs are depressed on the same cycle. In this case, the higher priority mode will be activated. If two mutually exclusive PBIs are depressed on different passes and held in, leading edge detection logic will activate the mode that was selected last, regardless of the prioritization. Table 16-3 contains the PBI prioritization; the highest priority is listed first.

TABLE 16-3: DAP Panel PBI Prioritization

| | |
|------------------------|-------------------------|
| DAP Select | DAP A, DAP B |
| DAP Control | INRTL, AUTO, FREE, LVLH |
| Rotation Mode | PRI, VERN, ALT |
| Rotation Submode | DISC RATE, PULSE |
| Translation (X,Y) | NORM, PULSE |
| Translation (Z) Part 1 | NORM, PULSE, HIGH Z |
| Translation (Z) Part 2 | HIGH Z, LOW Z |

16.6 MANEUVER DISPLAY (OPS DISPLAY)

This subsection presents a detailed outline of the MNVR display (Figure 16-5), which is the Ops display in MM 202. Only the sections of the display that are relevant to orbit DAP will be discussed.

OMS BOTH, L, R (Items 1, 2, 3) – used to select the engine configuration for an OMS burn. For an OMS engine to be commanded on by Moding, Sequence, and Control (MSC) software, either Item 1, 2, or 3 must be selected, targets must be loaded (via Item 22) and the EXEC key must be pressed after EXEC flashes (15 seconds prior to TIG) on the MNVR display, and TIG time must be reached.

RCS SEL (Item 4) – used to select an RCS burn. RCS burns can be done with or without guidance computed data (RCS burns can be done without entering and loading targeting data). RCS burns have no automatic flight control (THC must be deflected by the crew to execute an RCS burn).

Note: On the subsequent call up of the MNVR display, the previously selected mode (Item 1 through 4) will be displayed. Item 1 through 4 are mutually exclusive.

TV ROLL (Item 5) – roll about the thrust vector (Orbiter head's up relative to the local horizon is 0 degrees) used by guidance to calculate the burn attitude. Many OMS burns are executed with a TV ROLL of 180, so the crew can use the horizon as a reference. TV ROLL will be displayed on the subsequent call up of the MNVR display. This value can only be changed in OPS 2; therefore, it must be set up as desired for the deorbit burn prior to leaving OPS 2. (0 to 359 deg)

TRIM LOAD P, LY, RY (Items 6, 7, 8) – pitch (for both OMS engines), left and right yaw gimbal trim loads used by guidance to calculate the burn attitude. After the values are input and loaded (via Item 22), the OMS gimbals will maneuver to the trim positions 15 seconds prior to TIG (if OMS BOTH, L, or R was selected via Items 1, 2, or 3). Trim values are recalculated and updated throughout an OMS burn and will be displayed on the subsequent call up of the MNVR display. (-9.9 to 9.9 deg)

WT (Item 9) – Orbiter weight, which is updated during an OMS burn as propellant is used and will be displayed on the subsequent call up of the MNVR display. (100000 to 999999 lbs)

TIG (Items 10 through 13) – time of ignition in MET. The entry is constrained to within 5 minutes in the past to 12 hours in the future. (0 to 365 days, 0 to 23 hours, 0 to 59 minutes, 0 to 59.9 seconds)

TGT PEG 7 ΔV_X , ΔV_Y , ΔV_Z (Items 19, 20, 21) – targeted change in velocities in the LVLH reference frame for the burn. When values are entered, TGT PEG 4 data fields are blanked. (X: -9999.9 to 9999.9 ft/sec, Y and Z: -999.9 to 999.9 ft/sec)

LOAD (Item 22) – sends targeting data to guidance in order to compute a burn solution and attitude. When Items 1 through 21 are changed (except Item 18), LOAD will flash to indicate that an Item 22 must be executed for any of the entries to be recognized by guidance.

TIMER (Item 23) – starts the timer, which counts down to TIG.

BURN ATT R, P, Y (Items 24, 25, 26) – burn attitude in M50 inertial coordinates. These data fields are blanked when display is called up and are updated after Item 22 is executed. Any attempt to enter data directly into these field in OPS 2 will result in an ILLEGAL ENTRY message. During a burn, the data fields are updated cyclically. (R and P: 0 to 359 deg, Y: 0 to 90 and 270 to 359 deg)

Note: If the computed burn attitude has a yaw component near 90 or 270 (attitude singularity), then yaw will be displayed, but roll and pitch will be blanked. Guidance will still function normally and the ADI error needles will still be valid.

MNVR (Item 27) – causes an automatic maneuver to the burn attitude if the DAP is in AUTO. While maneuvering, an asterisk appears next to the Item 27. In OPS 2 when the attitude is reached, the asterisk disappears. When an Item 27 is executed, the ADI error needles will be referenced accurately to the inertial burn attitude. Once the Item 27 is executed, Universal Pointing software will no longer be used to control Orbiter attitude until the return to MM201. The field next to the Item 27 contains the current selected DAP mode (AUTO, FREE, INTRL or LVLH).

TTG (TIME-TO-GO) – if the DAP is in AUTO and the maneuver to a burn attitude has been initiated with an Item 27, this field will display the amount of Time-to-Go (TTG) to complete the maneuver in the current DAP. This field will be displayed over bright if the pre-burn maneuver will not be complete prior to Tig - 30 seconds.

16.7 DETAILED DESCRIPTION OF SOFTWARE

Since the orbit DAP consists totally of software, the specific modules making it up will be discussed in this section. Because the orbit DAP software is extremely involved, this section will not give a complete description of its contents.

For more a more detailed explanation of a specific software module, see the Orbit DAP FSSR (STS 83-0009) or the Detailed Design Specification (DDS). Also, software functionality that has been covered in previous sections of this brief may not be covered in this section. The software modules and their submodules are outlined as follows.

16.7.1 On-Orbit DAP RECON module (GCQ ORB DAP RECON 25HZ)

This module controls the DAP configuration by performing DAP parameter initializations, interpreting crew switch, RHC and THC inputs, and monitoring the validity of the IMU data entering the DAP. The module is activated on transition to OPS 2 or 8 and is active until transition out of OPS 2 or 8. The RECON module (GCQ) contains four internal procedure service modules.

1. UNPACK_SW (GCX) – This service module performs crew switch input interpretation (DAP panel inputs) so GCQ can make the appropriate mode change. This module contains the leading edge detection logic (Section 16.5.2) and the DAP PBI prioritization logic (Section 16.5.3). This module also resets the Reboost Enable flag and sets the Reboost Cancel flag when the FREE PBI is selected.
2. DAPLOAD_SELECT (GCY) – This service module is used to assign DAP control parameters and set flags based on DAP panel selections and the DAP CONFIG (SPEC 20) display. GCY is called by the GCQ module for the following reasons:
 - a. DAP parameter initialization
 - b. Change of DAP A/B
 - c. Change of PRI/ALT/VERN
 - d. Changes to Orbit DAP parameters or Control Accels via SPEC 20
 - e. Change of RCS attitude control mode (AUTO/INRTL/LVLH/FREE) or TVC attitude control mode

This module computes the magnitude of the average angular acceleration (MAG_CONT_ACCL) per orbiter body axis, based on the current jet mode. It also determines whether the criteria has been met for minimum angle jet selection logic to be used by the GFF (DAP critical processor) software module. The current I-load is set such that whenever ALT is selected on the DAP panel, the minimum angle jet select flag is set and GFF uses the minimum angle jet select logic. When VERN is selected, the minimum angle jet select flag will be set if a control accel (CNTL ACC on SPEC 20) other than 0 is loaded.

3. ROT_TRANS_PROCESSOR (GCZ) – This service module is used to implement auto and manual rotation and manual translation options. The outputs of this module will be used to activate/deactivate the appropriate software in various other modules composing the orbit DAP including AUTO_MNVR_TRACK (GC9), MANUAL_ROT_DISC (GC4), and the DAP Critical Processor module (GFF) which contains ROT_PULSE, ROT_ACCEL, TRANS_ACCEL, TRANS_PULSE, and RCS_ERRORS.

When the DAP is in AUTO, GCZ sets a flag to activate the AUTO_MNVR_TRACK module. GCZ sends the parameter AUTO_ATT_OPTION to AUTO_MNVR_TRACK so that it knows which automatic rotation option has been selected. The automatic rotation options are: 1 = pre-burn maneuver (in MM202), 3 = maneuver/track, 4 = attitude hold (2 is not an option). GCZ logic will also determine which attitude quaternion to use as the required attitude maneuver quaternion. This information is also sent to the AUTO_MNVR_TRACK module for implementation.

AUTO_ATT_OPTION will be set to 1 whenever a pre-burn maneuver is initiated in MM202 via an Item 27 on the MNVR display. It will be set to 3 if Universal Pointing is providing the required attitude quaternion and the current option is MNVR or TRK. It will be set to 4 for all other cases.

Manual rotation and manual translation have been previously discussed and no further details are included here.

4. THC_HOT_STICK – This service module allows the DAP to automatically upmode to PRI mode for the duration of a translation command whenever a powered THC is deflected. When the command is discontinued, the DAP will mode back to either VERN or ALT mode, whichever it was in prior to the upmode. The translation command is discontinued when the THC is back in detent when the translation submodule is NORM and when the jet pulse is complete (regardless of THC position) when the translation submodule is PULSE. If the PRI, ALT, or VERN pbi is pushed during a THC hot stick upmode, the upmode will be

terminated. ALT or VERN selection will cause the translation to also be terminated. PRI selection will allow the translation to continue, however PRI will remain selected after the translation is complete.

Specific GCQ module subfunctions are as follows:

1. IMU failure processing (GCQ.1) - When the IMU attitude (ATT) data good (DG) flag equals 0, GCQ will downmode the DAP to FREE, PULSE submode will be selected in all rotation and translation axes, and LO Z will be deselected. AUTO, INRTL and LVLH are not crew selectable, nor are the DISC RATE rotational submodes. As always, the ACCEL mode is available via RHC deflections past the soft stop. GCQ allows the DAP A or B to be selected, as well as PRI, ALT, or VERN. LO Z and translation NORM can be re-selected after the first software pass in GCQ.1.

When the IMU ATT DG flag equals 0, GCQ also terminates any reboost in progress, and selects the RCS control mode if the TVC control mode is active. If a maneuver is in progress in MM202, as indicated by an asterisk by Item 27 on the MNVR display, then the asterisk will be removed when the IMU ATT DG flag goes to 0.

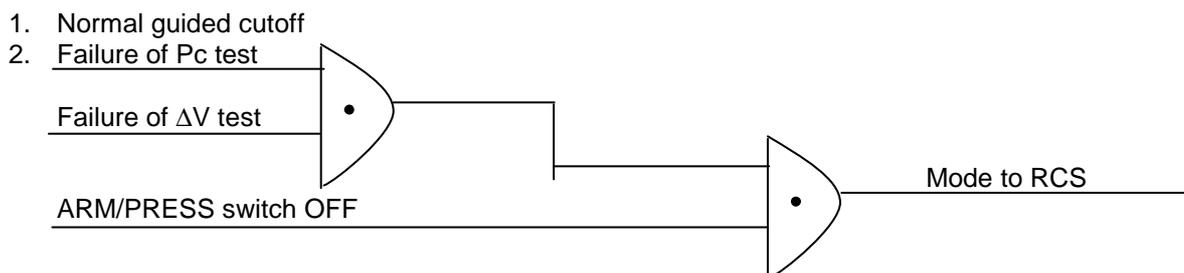
When at least one IMU becomes available and the IMU ATT DG flag equals 1, the DAP will remain in the downmoded configuration until crew action is taken on the DAP panel.

Note: If conditions exist such that GCQ.3 thinks that the current mode should be the TVC mode (during an OMS burn or if stuck in the TVC mode), after at least one IMU becomes available, the DAP will mode back to the TVC DAP mode. If this happens, INRTL will be selected on the DAP panel (because the TVC DAP only has two submodes, AUTO and INRTL), however PULSE will remain selected and the Orbiter will remain in freedrift. The crew will have to take action on the DAP panel to regain attitude control.

2. Orbit DAP initialization (GCQ.2) - On the first pass in OPS 2 (due to an OPS 1 to 2 or OPS 3 to 2 transition, re-IPL of all GNC GPCs, or any time the GNC GPCs go through OPS 0), the DAP will initialize to A/AUTO/PRI, DISC RATE (Rotation submode), PULSE (Translation submode), Lo Z off, and A1 and B1 loaded on SPEC 20 DAP CONFIG display. Also, all current and previously selected DAP panel switch options are reset, so that the configuration can be changed in the next software cycle (leading edge detection in reset). The following subsequent transitions will not affect the DAP configuration:
 - a. OPS 2 to 8
 - b. OPS 8 to 2
 - c. Any major mode (MM) recall which changes the redundant set

3. Control mode selection (GCQ.3) – The available DAP control modes are TVC DAP, RCS DAP, and IMU Fail. If at least one orbital maneuvering system (OMS) engine is being commanded ON (only possible in MM202) and the IMU Fail flag is OFF, then GCQ sets the current mode indicator to TVC. Otherwise, the mode indicator is set to RCS (if the IMU Fail flag is OFF). This mode indicator will be used in the reconfiguration processing, GCQ.6, to be discussed later. It is also used to select gains and deadbands for the TVC and RCS DAP modes. When in TVC DAP mode, this submodule will also determine whether one or two OMS engines are being used.

Once in the TVC DAP mode there are four ways to mode back to RCS DAP mode:

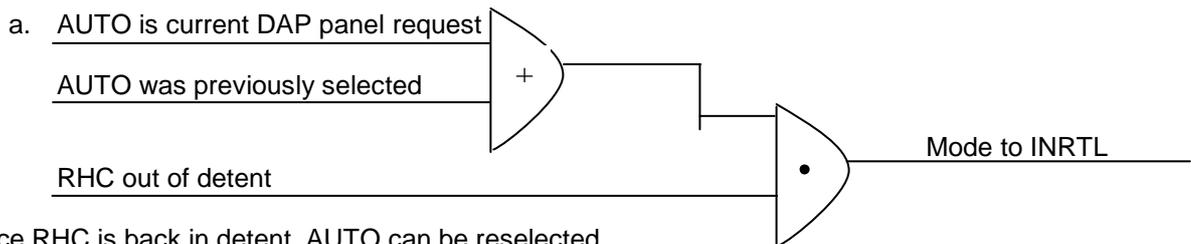


Note: Pc indications are routed through FA3 for the L OMS engine and FA4 for R OMS to OMS FDIR software to perform the Pc test.

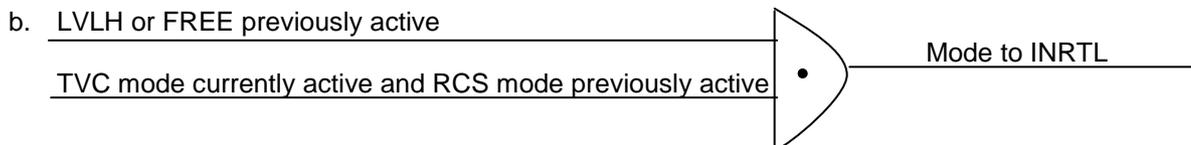
3. Transition to MM201
4. Maneuver 90 degrees out of computed burn attitude (causes the OMS cutoff flag to be set in Guidance software—See FSSR STS 83-0003-29 Paragraph 4.2.5)

It is possible to get stuck in the TVC DAP while doing a single OMS engine burn with two subsequent failures. An FA3 failure before or during a left OMS burn (or an FA4 failure before or during a right OMS burn) or Pc failed high on the burning engine during the burn will inhibit the ability of GCQ to recognize an OMS failure or early OMS engine shutdown by the crew (see 2nd way to mode back to RCS DAP, listed above). If there is an OMS failure or the crew shuts down the OMS engine before the burn is complete, normal guided cutoff will not be reached and the Orbiter will be stuck in the TVC DAP mode (even though the OMS engine is no longer burning). To recover, the crew can maneuver 90 degrees out of the computed burn attitude or transition to MM201 and then back to MM202. Impacts of being stuck in the TVC DAP include:

- a. THC inputs will not be recognized (the burn will not be able to be completed until back in the RCS DAP)
 - b. RCS will be holding attitude using RCS wraparound deadbands (10 degrees in Pitch and Yaw, 5 degrees in Roll)
 - c. OMS gimbals will continue to move after burn cutoff in an attempt by the DAP to null the VGOs
4. Forced manual Inertial moding (GCQ.4) - This logic forces the DAP to INRTL when (in this logic, "previous" refers to last software cycle and "current" refers to current software cycle):



Once RHC is back in detent, AUTO can be reselected.



Once the TVC mode is active, GCQ does not allow the selection of LVLH, FREE, VERN, or ALT (PRI jets are selected at TVC mode activation).

5. Maneuver (MNVR) execute processing (GCQ.5) - This submodule only functions during OPS 202. Its purpose is to set latch parameters, determine which attitude quaternion is selected as the required attitude while in OPS 202, and process OMS gimbal trims.

There are two parameters that must get set in order for the calculated Preburn Attitude Quaternion (Items 24, 25, and 26 on the Orbit MNVR display) to be selected as the required attitude. The Universal Pointing lockout (UP_LOCKOUT) parameter gets set when a preburn maneuver is executed (Item 27 on MNVR display) or if the DAP control mode has changed from RCS DAP mode to TVC DAP mode (OMS burn has begun). The PREBURN_LATCH parameter gets set when a preburn maneuver is executed (Item 27 on Orbit MNVR display) or during the first pass of GCQ since the start of an OMS burn, provided a maneuver is not already in progress. Once these parameters are set, the Preburn Attitude Quaternion will be selected as the required attitude (if the DAP is in AUTO) and the ADI errors will be referenced to this attitude. The Universal Pointing attitude will not be used again until return to OPS 201. Once ignition occurs, the required attitude is driven dynamically by guidance, as are the ADI error needles. The attitude in Items 24, 25, and 26 on the Orbit MNVR display will also be dynamic during the burn. When the burn is complete, the DAP will select the current M50 inertial attitude as the required attitude (if the DAP is in AUTO) and hold that attitude until transition to OPS 201 occurs. This aspect of MM202 processing is the same as the operation of CNCL (Item 20 on UNIV PTG display) in OPS 201.

Any DAP moding to LVLH, INRTL, or FREE during a preburn maneuver will reset PREBURN_LATCH. When moding back to AUTO, the current M50 inertial attitude will be held until another Item 27 is executed (and PREBURN_LATCH is set again).

If UP_LOCKOUT or PREBURN_LATCH are not set (i.e. when a multi-axis RCS burn is executed without a preburn maneuver) the Universal Pointing attitude is the required attitude before, during, and after the completion of the burn (if the DAP is in AUTO).

Upon return to OPS 201, UP_LOCKOUT and PREBURN_LATCH are reset and Universal Pointing will automatically resume (if the DAP is in AUTO at the OPS transition and CNCL was not selected prior to the OPS 202 transition).

Finally, if a request for gimbal trims is received, the active flags for Orbit OMS Processor (GCR) and TVC_LAW_PITCH_YAW (in GCR) are set. If TVC_LAW_PITCH_YAW is currently active and RCS DAP is selected, both flags are reset.

6. DAP reconfiguration (GCQ.6) - This logic uses the DAP mode indicator from GCQ.3 to reconfigure the DAP. Configuration dependant parameters are initialized, software modules are activated or deactivated, and processing of selected rotation and translation options (via GCZ) are initiated, based on the current desired DAP configuration and the previous DAP configuration. There are 20 defined DAP configuration transitions which can occur. They are as follows:

- Case 1: Manual RCS to Auto RCS
- Case 2: RCS to TVC Manual (one engine)
- Case 3: RCS to TVC Manual (two engines)
- Case 4: RCS to TVC Auto (one engine)
- Case 5: RCS to TVC Auto (two engines)
- Case 6: RCS Auto to RCS Auto or RCS Manual to RCS Manual
- Case 7: RCS Auto to RCS Manual
- Case 8: TVC to RCS
- Case 9: TVC Manual (two engines) to TVC Manual (two engines)
- Case 10: TVC Manual (two engines) to TVC Manual (one engine)
- Case 11: TVC Manual (two engines) to TVC Auto (one engine)
- Case 12: TVC Manual (two engines) to TVC Auto (two engines)
- Case 13: TVC Manual (one engine) to TVC Manual (one engine)
- Case 14: TVC Manual (one engine) to TVC Auto (one engine)
- Case 15: TVC Auto (one engine) to TVC Manual (one engine)
- Case 16: TVC Auto (one engine) to TVC Auto (one engine) or TVC Auto (two engines) to TVC Auto (two engines)
- Case 17: TVC Auto (two engines) to TVC Manual (one engine)
- Case 18: TVC Auto (two engines) to TVC Manual (two engines)
- Case 19: TVC Auto (two engines) to TVC Auto (one engine)
- Case 20: First-pass Orbit DAP initialization

7. Module status update (GCQ.7) - This algorithm checks for a change in the required configuration modules from the previous cycle. If a change has occurred, then the high frequency executive dispatcher is invoked to obtain the correct set of modules to execute. This logic is preformed regardless of the state of the IMU Data Good flag.
8. Set lamps (GCQ.8) - This logic sets/resets the appropriate C3 and A6 DAP panel lights per the current requests. When a change is detected in DAP A/B or PRI/ALT/VERN PBIs, then a flag is set to update the DAP CONFIG display (SPEC 20). This logic is preformed regardless of the state of the IMU ATT DG flag.

Note: Flight Critical MDM failures will affect the lights on the C3 and A6 DAP panels. FF1 provides the light commands for the C3 DAP panel and FF3 provides the commands for the A6 DAP panel. Depending on the specific failure of the MDM, the lights on the corresponding DAP panel may be blanked or frozen in the configuration at the time of the failure (depressing the PBI will still mode the DAP appropriately, however the lights on the affected panel will not change in this failure mode). With such an MDM failure, the unaffected DAP panel must be used to verify DAP panel reconfigurations.

9. State Estimator Part 2 filter (GCQ.9) - This logic is the half of the state estimator which provides the extrapolation necessary to match the 12.5 Hz frequency of the DAP critical processing. It is therefore computed at the end of a flight control cycle for use on the next 80 ms cycle. This filter incorporates the rate estimates (DELTA_OMEGA_RCS) issued from JET_SELECT (GFF.2) as discussed in Section 16.7.2 and undesired acceleration from the State Estimator Part 1 Filter software module to compute body angle estimates and body angle rate estimates. This logic will not be performed if in the IMU Fail mode.
10. CGQ continued processing (GCQ.10) - This module resets the Reboost Enable flag and sets the Reboost Cancel flag if the RHC is moved beyond the softstop.
11. IMU Fail = Off continued processing (GCQ.11) – This is the module that downmodes the DAP to FREEDRIFT if VERN is selected and a vernier jet fails.

16.7.2 On-Orbit DAP Critical Processor module (GFF ORB DAP CP 25HZ)

At the heart of the orbit DAP is GFF, the orbit DAP critical processor. This module processes data from guidance subsystems, vehicle sensors, and crew inputs to control vehicle translation and rotational acceleration by utilizing the RCS jets. The processor is activated on entry to OPS 2 and is active throughout OPS 2 and 8. GFF is broken down into two major segments, the steering processor (GFF.3 and GFF.4) and the RCS processor (GFF.2). The RCS portion of GFF is shown in Figure 16-1.

1. Steering Processor

a. OMS TVC control (GFF.3)

GUIDANCE_COMPENSATION - This algorithm is active while in the OMS TVC DAP mode with AUTO selected. It provides the filtered and limited vehicle angular rate commands in the pitch and yaw axes. It also forces the commanded roll rate to zero during OMS burns.

MANUAL_TVC - This logic transforms the RHC deflections into angular rate commands while in INRTL during OMS burns. These commands will be used by the OMS and RCS processors. If only one OMS engine is being used, roll rate commands will not be processed by this algorithm.

RCS_ASSIST – This algorithm stores the commanded vehicle angular rate computed in GUIDANCE_COMPENSATION. The desired attitude increment during one flight control cycle is then derived by multiplying the commanded angular rate by the duration of that flight control cycle (80 ms). This desired attitude increment will be used by the OMS and RCS processors.

b. RCS Rotational Error determination (GFF.3)

RCS_ERRORS – This logic takes the desired attitude and rate from various command modules (GC9_AUTO_MNVR_TRK, GVU_UNIV_PTG, GC4_ROT_DISC and GC5_LVLH_TRACK) depending on the selected DAP mode and compares them with the current attitude and rate, as determined with IMU data from the state estimator. The calculated attitude and rate errors are then sent to the phase plane software.

c. Manual RCS translations (GFF.3)

TRANS_ACCEL - This logic generates a command (TRANS_JET_CMD), which is sent to jet select software to achieve a constant translation acceleration of the vehicle as long as the forward or aft THC is out of detent (when NORM is selected on the DAP panel in the selected translation axis).

JET_SELECT will determine which jets will fire. The magnitude of the acceleration is dependent on the axis being commanded. When the THC is returned to detent, the acceleration request is zeroed and the jets will cut off.

TRANS_PULSE - This logic will calculate and command (via TRANS_JET_CMD) a vehicle translation velocity change each time the THC is taken out of detent (when PULSE is selected on the DAP panel in the selected translation axis), regardless of the duration of the THC deflection. The commanded

velocity change is defined by the TRAN PLS value on SPEC 20 (Items 17 or 37). The velocity is attained without the use of IMU feedback (it is an open loop command based on jet on times). The equation used to determine the number of cycles that the jet ON command is sent to the JET_SELECT module is:

$$\text{CGCV_REMAINING_PASSES} = \text{CGCV_REMAINING_PASSES} + \text{CGPV_ARY_THC} \times (\text{TRUNCATE}(\text{CGCV_TRANS_PULSE_SIZE} \times \text{K_T_TEMP} + 1))$$

where:

CGCV_REMAINING_PASSES = The number of 80 ms jet ON commands necessary to attain the desired vehicle rate by axis.

CGCV_ARY_THC = Sign of THC command by axis (+1, 0, -1)

CGCV_TRANS_PULSE_SIZE = Value (feet/second) on SPEC 20 (Item 17 or 37). This is the change in the translation velocity that the DAP will try to attain for each THC deflection.

K_T_TEMP = (CGCV_ORB_DAP_MASS_HFE x E) / (CGCK_MAG_CONTROL_FORCE x TDAP)

where:

CGCV_ORB_DAP_MASS_HFE = I-loaded estimate of vehicle mass (slugs). This parameter can be updated by loading a new weight (Item 9) on the MNVR display in MM 202 and executing the LOAD (Item 22) on the same display.

E = 11 for a Low Z translation, 1 for any other translation

CGCK_MAG_CONTROL_FORCE = I-loaded force (lbs) in each axis during translation firings. Since +Z fires three jets and -Z fires six jets, the average value for the two directions was used for the Z axis. This results in different rate changes in each of these directions for a given pulse size. (X(V97U2957C)=1760 lbs, Y(V97U2958C)=1760 lbs, Z(V97U2959C)=3055 lbs)

CGCV_TDAP = Duration of one flight control cycle (80 ms).

K_T_TEMP is a temporary variable, which is computed each time the TRANS_PULSE subroutine is initialized. It is computed for each axis and accounts for the control force available in each axis. There are five occasions in which K_T_TEMP is recomputed using the most recent vehicle mass (if a vehicle translation is completed in the PULSE mode without an updated vehicle mass, the actual velocity change will most likely be more than the expected velocity change due to the continuous decrease of orbiter mass during a mission caused by consumable use) :

1. OPS 2 to 8 transition: K_T_TEMP recomputed for all axes.
2. OPS 8 to 2 transition: K_T_TEMP recomputed for all axes.
3. OPS Mode recall in MM 201 or MM 801: K_T_TEMP recomputed for all axes.
4. If PRI is active, K_T_TEMP is recomputed for a given axis when a transition from NORM to PULSE is made in that axis. Other axes will not be affected and will retain their old value of K_T_TEMP.
5. If any translation axis is in PULSE, K_T_TEMP is recomputed for that axis when a transition to PRI (from either ALT or VERN) is made. This includes occasions in which PRI is activated by deflecting the THC while VERN or ALT is active (THC hot-stick).

If LOW Z is selected and a +Z translation is commanded, the value of E becomes eleven. This will cause the ±X jets to be commanded ON longer to attain the desired change in velocity.

The remaining passes term is decremented each pass after a THC command until it is zero. The jets will then cut off if the THC has not been subsequently deflected. The routine is additive; therefore, if the THC is deflected five times, the number of remaining passes is five times as great as one THC deflection. If the THC is deflected five times one direction then five times the other direction, the net vehicle motion will be close to zero regardless of whether the first five deflections have decremented or not. It must be noted that the THC must be returned to detent between each successive deflection in the same axis in order to increment the remaining passes term.

As with the TRANS_ACCEL mode, THC hot-stick capability exists for the TRANS_PULSE mode, meaning that the mode can be entered by deflecting the THC, regardless of whether PRI, ALT or VERN is selected. However, the hot-stick capability was not implemented exactly as desired. Under normal circumstances, with PRI selected, jet firings will occur if the THC is deflected and held out of detent. After the jets firings are complete, if the THC is then deflected in another axis without first passing through detent, jet firings will occur in that axis. However, due to a deficiency in the THC hot-stick CR requirements, if this same scenario is carried out while ALT or VERN is selected, the latter THC command will not result in jet firings. The only way to ensure that subsequent jet firings will occur in this mode is to take the THC back to detent between each pulse command. Additionally, if a THC input is made while ALT or VERN is selected, the translation jets will fire and the DAP will mode back to its previous configuration as soon as the jet firings are complete, even if the THC is held out of detent.

d. Manual RCS Rotations (GFF.4)

ROT_ACCEL - This routine generates a command (ROT_JET_CMD), which is sent to jet select software to cause jets to fire any time the RHC is deflected past the soft-stop. The jets will continue to fire until the RHC is brought back within the soft-stop. Once within the soft-stop, the vehicle will be in free drift until the RHC is returned to detent. The DAP rotational submodes will then mode back to the configuration that was active before the RHC was deflected past the soft-stop (DISC RATE or PULSE).

ROT_PULSE - Like TRANS_PULSE, this routine is an open loop algorithm which determines the number of 80 ms jet ON cycles that are needed to attain the desired vehicle angular rate change as defined by the ROT_PLS value on SPEC 20 (Items 13, 33, 26, or 46) for each RHC deflection. This logic is active when PULSE is selected on the DAP panel in a particular rotational axis and the RHC is moved out of detent in that axis, provided the Post Contact Docking sequence is inactive. The algorithm generates the ROT_JET_CMD and sends it to jet select software after it has determined that jets should be fired during the current software cycle. The equation used in this algorithm is similar to the translation equation:

$$\text{PASS_TEMP} = \text{TRUNCATE} (1 + (\text{PULSE_SIZE} / (\text{MAG_CONT_ACCL} \times \text{TDAP})))$$

where:

PASS_TEMP = The number of 80 ms jet ON commands necessary to attain the desired vehicle angular rate.

PULSE_SIZE = Value (deg/sec) loaded by the crew on SPEC 20 (Items 13, 33, 26, or 46). This is the change in rotational velocity that the DAP will try to attain for each RHC deflection.

MAG_CONT_ACCL = average angular acceleration (deg/sec²) by axis (R, P, Y). The ROT_PULSE routine is sloppy in pitch since four jets fire for +P and three jets for -P, and only one value of MAG ACCEL exists for the pitch axis. The ROT_PULSE routine also does not take into account the DAP configuration (it uses the same values for Norm Z, Low Z, tail-only, etc.). This value is calculated onboard and can be found in the downlist. Nominally for Control Accel = 0, the values are as follows: (Roll(V90U2798C)=.8 deg/sec², Pitch(V90U2828C)=.9 deg/sec², Yaw(V90U2829C)=.6 deg/sec²)

Note: When the Control Accel does not equal 0, the MAG_CONT_ACCL variable becomes critically important in determining the jet on time and the jet selection (for ALT and VERN) for a

rotation command in the rotational PULSE mode. Because of unpredictability of jet firings and jet firing durations, the use of the rotational PULSE mode while ALT is selected with non-zero Control Accel is considered uncertified by Draper Labs (reference "Alternate DAP Mode Constraints, Guidelines, and Performance Assessment" memo, CSDL-R-2380).

TDAP = Duration of one flight control cycle (80 ms)

The PASS_TEMP term is decremented each cycle and when it reaches zero, the jets cut off if no more deflections of the RHC have occurred. Also, if an RHC deflection in the opposite direction occurs before PASS_TEMP = 0, jets will fire in the opposite direction until the change in vehicle angular rate is near zero.

Note: If the DAP configuration is changed (between PRI, ALT, or VERN) before the PASS_TEMP term has decremented to zero, the rotational burn will cease until a new RHC deflection occurs (the PASS_TEMP term will effectively reset to zero).

Manual Rotational Discrete module (GC4_ORB_MAN_ROT_12HZ) - This is a module external to GFF, but is included here since it is a rotational submodule. GC4 is a closed-loop routine, which is activated via the DISC RATE PBIs and functions with either the INRTL or LVLH mode.

When the DAP is in INRTL/DISC RATE, it commands zero vehicle angular rate (inertial attitude hold) while the RHC is in detent. While the RHC is out of detent, the DAP commands a vehicle rate according to the ROT RATE value on SPEC 20 (Items 10, 20, 23, or 43). Jets will be commanded ON or OFF as necessary to maintain the desired rate within the tolerances of the phase plane. When the RHC is released, the rate command goes to zero and jets will fire, if necessary, to achieve and maintain inertial attitude hold.

LVLH processing is the same as inertial processing except the LVLH reference frame is used. When the DAP is in LVLH/DISC RATE, the vehicle's attitude will be held constant with respect to the LVLH reference frame. If only two of the three axes are in DISC RATE, a plane is established which is, initially, orthogonal to the third (free) axis. The DAP will then generate the necessary jet commands to constrain the two axes in DISC RATE in relation to that plane.

With all three axes in DISC RATE, any RHC inputs will continually reinitialize the desired attitude until the hand controller is returned to detent. At that point the current LVLH attitude becomes the desired and an LVLH hold about that attitude occurs.

If two axes are in DISC RATE, RHC inputs in these axes continuously reinitialize the plane until the RHC is returned to detent, at which time the plane becomes fixed in LVLH. RHC inputs in free (PULSE) axis set up a rate orthogonal to the plane.

2. RCS Processor

a. PHASE_PLANE (GFF.2)

This submodule uses the desired attitude and rate errors computed in RCS_ERRORS to determine whether or not there is a need for automatic jet firings. The DAP must be in INRTL/DISC RATE, LVLH/DISC RATE or AUTO for the phase plane logic to operate. If it is determined that jet firings are necessary, this routine sets a jet command (CGCV_ROT_JET_CMD) for a given rotational axis (PHASE_PLANE is not used for translation jet firings) to either ± 1 , depending on the required rate change. The ROT_JET_CMD is sent to the JET_SELECT software, which will determine which jet(s) to fire. It must be noted that ROT_JET_CMD is a scalar value between ± 1 and values that are less than a magnitude of 1 are used by JET_SELECT software when ALT or VERN is selected. Only a value of ± 1 , however, will cause jets to fire, regardless of whether PRI, ALT, or VERN is selected.

This algorithm begins by using the attitude and rate errors to define X1 and X2 for each rotational axis as follows (see Figure 16-12):

$$X1 = \text{SIGN}(\omega_e) * \theta_e$$

$$X2 = \text{ABS} (\omega_e)$$

where:

$$\omega_e = \text{rate error}$$

$$\theta_e = \text{attitude error}$$

Next, the switching lines S1 through S5 are defined (see Figure 16-12 and Table 16-4). The size and shape of the phase plane (the design of the switching lines) is analyzed before each mission by Draper Laboratories to determine the proper trade-off between high propellant usage (tight deadbands for precise attitude control) and lower propellant usage (wider deadbands for coarse attitude control). Next, the software uses X1 and X2 to determine which region the vehicle is in for each axis. The following defines the actions taken depending on the Region in which X1 and X2 fall:

Region 1: This is the acceleration region where ROT_JET_CMD will get set such that jets will fire to decrease the rate error. The equation: $\text{ROT_JET_CMD} = -\text{SIGN} (\omega_e)$ The resulting motion is shown in Figure 16-12. A desired angular rate error is also computed for use if in the TVC DAP mode.

Region 2: This is the Disturbance Hysteresis region. It will be discussed later

Region 3: This is another acceleration region where jets will fire to increase the rate error and push the vehicle toward the drift channel (Regions 4 and 5). The equation: $\text{ROT_JET_CMD} = \text{SIGN} (\omega_e)$ The resulting motion is shown in Figure 16-12. A desired angular rate error is also computed for use if in the TVC DAP mode.

Region 4: This is part of the drift channel or coast region. This region was designed to allow the vehicle to drift towards the Hysteresis region without the need to fire jets. If PRI or ALT is selected, then $\text{ROT_JET_CMD} = 0$. If VERN is selected, then the command from the previous software cycle is compared to the rate error in the following manner: If $\text{ROT_JET_CMD} (\text{previous cycle}) = -\text{SIGN} (\omega_e)$, then leave the ROT_JET_CMD as is. Basically, this means that if the vehicle has been driven into Region 4 from Region 1, then keep firing the appropriate jets until the vehicle reaches the S4 switching line (in VERN only). If the vehicle is in Region 4 (with VERN selected) and the above statement does not apply, compute the ROT_JET_CMD using the following equation (the value will be somewhere between -1 and +1 and will be used by JET_SELECT software as explained later):

$$\text{ROT_JET_CMD} = \text{SIGN} (\omega_e) * \text{WFRATE} * ((\text{S4_SCALE} * \text{RL}) - \text{X2}) / ((1 - \text{S4_SCALE}) * \text{RL})$$

where:

$$\text{WFRATE} = \text{scale factor for off-axis vernier performance (I-load)}$$

$$\text{S4_SCALE} = (1 + \text{S5_SCALE}) / 2$$

$$\text{S5_SCALE} = \text{I-load used to define the width of the drift channel (usually .6)}$$

$$\text{RL} = \text{Rate deadband}$$

Region 5: This is the other half of the drift channel. As in Region 4, if PRI or ALT is selected, then $\text{ROT_JET_CMD} = 0$. If VERN is selected, then the command from the previous software cycle is compared to the rate error in the following manner: If $\text{ROT_JET_CMD} (\text{previous cycle}) = \text{SIGN} (\omega_e)$, then leave the ROT_JET_CMD as is. This means that if the vehicle has been driven into Region 5 from Region 3, then keep firing the appropriate jets until the vehicle reaches the S4 switching line (in VERN only). If the vehicle is in Region 5 (with VERN selected) and the above statement does not apply, compute the ROT_JET_CMD using the same equation as in Region 4.

If the vehicle was determined to be in Region 2, the Disturbance Hysteresis logic is used. This algorithm begins by using the attitude and rate errors to define Y1 and Y2 for each axis as follows (See Figure 16-13):

$$Y1 = \text{SIGN} (U_{\text{DR}}) * \theta_e$$

$$Y2 = \text{SIGN}(U_{DR}) * \omega_e$$

where:

$$U_{DR} = U_D + (U_R * \text{RBST_PP_SF})$$

U_D = undesired accelerations (from State Estimator Part 1 Filter module)

U_R = Reboost disturbance accelerations (from Jet Select)

RBST_PP_SF = scale factor for reboost accel compensation (I-load)

Next, the switching lines are defined (see Figure 16-13 and Table 16-4). $Y1$ and $Y2$ are then used to determine which region the vehicle is in for each axis. The following defines the actions taken depending on the Region in which $Y1$ and $Y2$ fall:

CS Region: If in the TVC DAP mode, the ROT_JET_CMD is set equal to 0. If in the RCS DAP, the following equation is used to calculate ROT_JET_CMD :

$$\text{ROT_JET_CMD} = \text{SIGN}(U_{DR}) * \text{WFRATE} * ((S11(Y1) - Y2) / (RL + S11(Y1)))$$

where:

$S11(Y1)$ = the value (in the $Y2$ axis) of the $S11$ switching line at $Y1$

Again, the value for ROT_JET_CMD will be less than the magnitude of one in this case and jets will not fire.

HS1 Region: If in the TVC DAP mode, the ROT_JET_CMD is set equal to 0. If in the RCS DAP and the ROT_JET_CMD (previous cycle) = $-\text{SIGN}(U_{DR})$, then leave the ROT_JET_CMD as is. This means that the vehicle has been driven into the HS1 Region from Region 1 and there are undesired accels (or accels due to an active reboost) that are working in the opposite direction of the jet firings. The jets will keep firing until the vehicle reaches the $S11$ switching line, which will set the vehicle up for single sided deadbanding in this axis. If the FORCE_FIRE flag is set (in the Manual Rotational Discrete module) then set the ROT_JET_CMD to fire jets in the opposite direction of the undesired accels. The equation: $\text{ROT_JET_CMD} = -\text{SIGN}(U_{DR})$ Otherwise, compute the ROT_JET_CMD using the following equation:

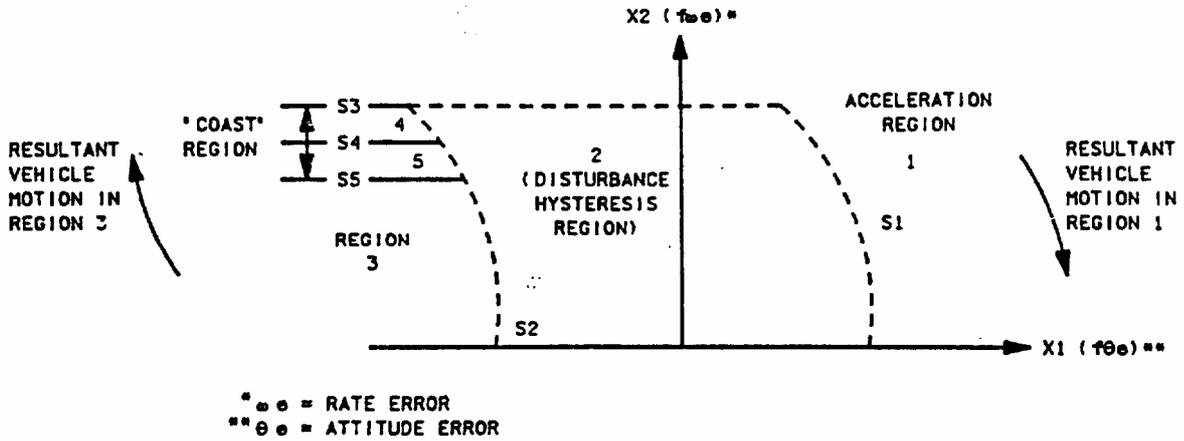
$$\text{ROT_JET_CMD} = \text{SIGN}(U_{DR}) * \text{WFRATE} * ((Y2 - S11(Y1)) / (RL - S11(Y1)))$$

HS2 Region: If in the TVC DAP mode, the ROT_JET_CMD is set equal to 0. If in the RCS DAP and the ROT_JET_CMD (previous cycle) = $\text{SIGN}(U_{DR})$, then leave the ROT_JET_CMD as is. This means that the vehicle has been driven into the HS2 Region from Region 1 and there are undesired accels (or accels due to an active reboost) that are working in the same direction as the jet firings. The jets will keep firing until the vehicle reaches the $S11$ switching line, which will set the vehicle up for single sided deadbanding in this axis. If the FORCE_FIRE flag is set (in the Manual Rotational Discrete module) then set the ROT_JET_CMD to fire jets in the same direction as the undesired accels. The equation: $\text{ROT_JET_CMD} = \text{SIGN}(U_{DR})$ Otherwise, compute the ROT_JET_CMD using the following equation:

$$\text{ROT_JET_CMD} = \text{SIGN}(U_{DR}) * \text{WFRATE} * ((S11(Y1) - Y2) / (RL + S11(Y1)))$$

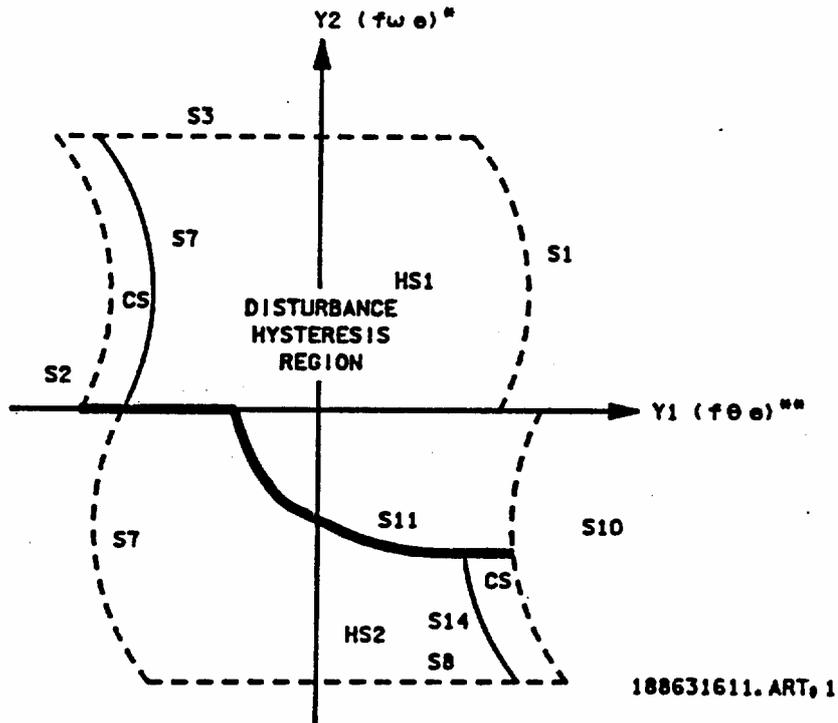
The parameter U_c (control acceleration available in a particular axis) is an I-load with the HAL name of $\text{CGCK_MAG_CNTL_ACCEL}$. This parameter helps shape the phase plane as seen in Table 16-4. In theory when in the acceleration region, the motion of the error state will parallel the outer switching lines. In order for this to occur, the MAG_CNTL_ACCEL must accurately represent actual vehicle acceleration due to nominal jet firings. If Orbiter inertia changes significantly (e.g., ejection of a large payload, a large payload attached to an extended remote manipulator system (RMS), or docking to ISS), then the parallel motion will not occur, causing inefficient RCS activity. This increased activity is especially true for Vernier RCS (VRCS) due to their lack of control authority. For this reason, the set of vehicle mass configuration values that are used to compute MAG_CNTL_ACCEL are changeable

by the crew via the CNTL ACC items on SPEC 20 (Items 28 and 48, Figure 16-3) to ensure proper phase plane/vehicle mass configuration match.



188631610.ART, 1

Figure 16-12: Phase plane for function P-P.



188631611.ART, 1

$*\omega e$ = Rate error
 $**\theta e$ = Attitude error

Figure 16-13: Phase plane for disturbance hysteresis logic.

| Line | Definition |
|------|--|
| S1 | $S1(X2) = -\frac{X2^2}{2U_{CP}} + DB$ $S1(Y2) = -\frac{Y2^2}{2U_{CP}} + DB$ |
| S2 | $S2(X2) = -\frac{CX2^2}{2U_{CP}} - 1.2 DB - KL$ $S2(Y2) = -\frac{CY2^2}{2U_{CP}} - 1.2 DB - KL$ |
| S3 | S3 = +RL |
| S4 | If PRIMARY_VERNIER_SW = OFF then S4 = S4_SCALE RL else not used |
| S5 | If RCS then S5 = LRL_TVC else if PRIMARY_VERNIER_SW = OFF then S5 = S5_SCALE RL else S5 = min (S5_SCALE RL, max (RL - 2W _{min} , 0.02)) |
| S7 | $S7(Y2) = K \frac{Y2^2}{2U_{CP}} - DB, K = -1, Y2 \geq 0$ $K = +1, Y2 < 0$ |
| S8 | S8 = -RL |
| S10 | $S10(Y2) = \frac{CY2^2}{2U_{CP}} + 1.2 DB + KL$ |
| S11 | $S11(Y1) = 0$ $= -\sqrt{2 U_{DR} (Y1 + 0.5 DB) + W_{MIN}},$ if S11(Y1) > 0, then S11(Y1) = 0 if S11(Y1) < -RL + W _{MIN} , then S11(Y1) = -RL + W _{MIN} $-\frac{CX2^2}{2U_{CP}} - 1.2 DB - KL \leq Y1 < -0.5 DB \text{ or } \overline{RCS}$ $-0.5 DB \leq Y1 \leq S10 \text{ and } RCS$ |
| S14 | $S14(Y2) = +\frac{Y2^2}{2U_{CP}} + DB$ <p>Notes: S4_SCALE = (1 + S5_SCALE)/2 If PRIMARY_VERNIER_SW = ON Then KL = KLEDGE Else KL = 0.0 C = 1.25, ROT_JET_CMD ≠ 1 1.0, ROT_JET_CMD = 1 U_{CP} = U_C - SIGN(W_o)U_D</p> |

TABLE 16-4: DEFINITION OF SWITCHING LINES

Note: For further information concerning the phase plane, refer to the Orbit DAP FSSR, STS 83-00098.

b. JET_SELECT (GFF.2)

The basic function of this algorithm is to command on and off the appropriate jets according to ROT_JET_CMD (received from the PHASE_PLANE, ROT_ACCEL, or ROT_PULSE), TRANS_JET_CMD (received from TRANS_ACCEL or TRANS_PULSE), and rotation compensation. The JET_SELECT software uses multiple inputs from various software modules to determine which particular jet or jets should be fired. The remainder of this section will cover the specifics of the JET_SELECT software, but a few specific requirements need to be emphasized:

1. PRI, ALT, and VERN are mutually exclusive.
2. PRI can accommodate simultaneous rotation and translation commands (translation by THC only).
3. VERN and ALT control rotation only.

The JET_SELECT algorithm begins with the failure recognition submodule. If a jet status change flag is received from RCS_RM, then failure recognition is run to update the failure status of the primary and vernier jets and to update jet group failure flags for all 14 primary jet groups and verniers. Any jet or group of jets that have been determined by RCS_RM to be failed will not be commanded to fire by JET_SELECT.

Next, the algorithm sets reboost flags depending on whether or not a reboost has been enabled. These flags are used by GFT_REBOOST to perform an automatic reboost of the Orbiter and ISS.

The next submodule is rotation compensation. It provides open loop rotation compensation to help minimize undesired cross-axis accelerations due to jet firings in other axes. First, a crew modifiable I-load comp value (Item 14, 34, 27, or 47 on SPEC 20, depending on selected DAP and whether primary jets or vernier jets are being used) is compared to the minimum impulse for primary or vernier jets. The HAL names and values for the minimum impulse parameters (I-loads) are:

CGCV_PRI_ROT_MIN_IMPULSE for PRI and ALT (0.112 deg/s)
CGCS_VERN_ROT_MIN_IMPULSE for VERN (0.002 deg/s)

If the comp value on SPEC 20 is less than the minimum impulse for the selected jets, then rotation compensation will not occur (i.e. if the comp value is zero, rotation compensation does not occur). Also, for any axis that is in closed loop attitude control (attitude is being controlled by PHASE_PLANE), rotation compensation will not occur. A comp value (as long as it is higher than the minimum impulse) indicates the maximum allowable rate (deg/sec) in an off axis due to a jet firing. If this rate is exceeded, jets will fire in the off axis to minimize the rate change in the off axis.

VERN or ALT jet select:

Next, the jet selection process begins. If ALT or VERN is selected, jet select is accomplished by a propellant efficient routine, which seeks the jets with the largest contribution to the desired angular velocity (dot product method) or minimum angular error between the angular acceleration and the ROT_JET_CMD (min angle method). These routines are necessary for vernier jets since six jets must provide the entire spectrum of attitude maneuvers. ALT was designed to take advantage of the vernier jet select logic, which is more propellant efficient than the primary jet select look up tables.

If VERN or ALT is selected, VERN_CMD is calculated based on ROT_JET_CMD and any rotation compensation that is required (via COMP_CMD). The VERN_CMD is a command vector (R,P,Y) that represents the desired vehicle rotation and is defined as follows:

R = Roll value between -1 (negative roll) and +1 (positive roll)
P = Pitch value between -1 (negative pitch) and +1 (positive pitch)
Y = Yaw value between -1 (negative yaw) and +1 (positive yaw)

The VERN_CMD will only be set to +1 or -1 in a rotational axis if COMP_CMD or ROT_JET_CMD has a value of ± 1 (a value of less than a magnitude of 1 will be stored in VERN_CMD in a particular axis if

the ROT_JET CMD is less than a magnitude of 1 in that particular axis and COMP_CMD is zero in the same axis. This case will not cause jet firings).

At this point in the software, the VERN algorithm branches off from ALT algorithm, although the actual jet select logic is nearly the same. If VERN is selected, the jet select algorithm will be run if the VERN_CMD has changed, if a vernier jet has failed, or if the same VERN_CMD has been issued for the last 5 software cycles. If the VERN_CMD has not changed from the previous cycle and the repeat counter has not become greater than 5, then the jets that were selected to fire last cycle will continue to fire this cycle (NEW_JONLST = OLD_JONLST). If none of the VERN_CMD axes are +1 or -1, the angular rate increment estimate for each axis (DELTA_OMEGA_RCS), which is sent to the State Estimator Part 2 software, is set to zero. Otherwise, jet select is performed using either the dot product method or the min angle method.

Dot Product method:

The dot product method will command either one, two, or three jets to fire. First, AUX_ANG_INCS are received from DAP_SELECT in the RECON software module for each of the six vernier jets. AUX_ANG_INCS is a vector (R,P,Y), which specifies the resultant vehicle acceleration (deg/sec change per 80 msec cycle) for the particular vernier jet firing based on the selected Control Accel (Item 28 or 48 on SPEC 20). Test 1 of the dot product method is to perform a dot product of VERN_CMD and AUX_ANG_INCS for each vernier jet that has not failed or been deselected. The largest value will be saved (VAL1) and the corresponding jet will be commanded to fire (via NEW_JONLST which is sent to RCS CMD SOP software). In Test 2, the remaining dot product values are compared to VAL1. If the next largest value is greater than 50 percent of VAL1, the associated jet will also be commanded to fire. If Test 2 resulted in a second jet being commanded to fire, then jet select will perform one last test, Test 3. The third highest dot product value will be found and, if it is greater than 40 percent of VAL1, the associated jet will be fired along with the first two. Note that if Test 2 did not result in a second jet being commanded to fire, then Test 3 will not be performed. Finally, for each jet that is commanded to fire, DELTA_OMEGA_RCS is updated for each axis and sent to the State Estimator Part 2.

Min Angle method:

The min angle method will also command either one, two, or three jets to fire. Minimum angle jet selection was designed and implemented in OI-24 in order to meet loads constraints for the proposed Space Station Freedom. During testing, however, the min angle method showed propellant savings in nearly every test scenario. It differs from the dot product method in two significant ways. First, it selects jet combinations that minimize the angle between the acceleration vector and the command vector (VERN_CMD). Second, the min angle method evaluates combinations of jets, where the dot product method operates on each jet individually. The min angle software module (GFF_MINA_JS) is used for both VERN and ALT modes.

The min angle software module begins by determining the dot products between the AUX_ANG_INCS vectors and the VERN_CMD vector for all vernier jets that are not failed or deselected and determining which is the largest. Once the maximum dot product is determined, the dot products for the other jets are compared to it and will only be used as candidates for the min angle calculations if they exceed a certain I-load percentage of the maximum dot product (The current I-load for vernier jets is -.2 and for ALT is 0.0). Next, the software limits the number of candidate jets to 8 to minimize calculations. If there are more than 8 jets that exceed the I-load percentage of the maximum dot product, the largest 8 dot products will be used (this does not come into play with verniers, but may with ALT jet groups).

Once the candidate jets are determined, the candidate jet with the maximum change in angular velocity is determined and multiplied by a magnitude threshold (I-load value which is currently .15 for the 1 jet magnitude test) and the result is squared. The square of the change in the angular velocity for all of the other candidate jets is compared with above result. If the change in the angular velocity for a jet does not exceed 15% of the maximum change in angular velocity, then it is dropped from the candidate list. This test is done to exclude jets that may oppose each other or that provide insignificant acceleration in the desired direction. Next, two and three jet combinations from the updated jet candidate list are tested to see if their combined change in angular velocity is a certain

percentage of the maximum change in angular velocity (I-load values which are currently .24 for the two jet magnitude test and .30 for the three jet magnitude test). All the comparisons are done using squared values as with the single jet magnitude test. The two jet magnitude test will only be done if the two jet test threshold (T1) is less than 1 (T1 will be less than 1 if VERN is selected or if ALT is selected and # JETS on SPEC 20 for the selected DAP is 2 or 3). The three jet magnitude test will only be done if T1 and the three jet test threshold (T2) are both less than 1 (T2 will be less than 1 if VERN is selected or if ALT is selected and # JETS on SPEC 20 for the selected DAP is 3).

Finally, from the list of possible 1 jet, 2 jet combinations, or 3 jet combinations that passed the magnitude tests above, the jet or jet combination with the minimum angle between the VERN_CMD and the AUX_ANG_INCS will be chosen to fire. The minimum angle is determined by maximizing the dot product of the unitized VERN_CMD and AUX_ANG_INCS divided by the magnitude of AUX_ANG_INCS.

The ALT jet select process is nearly the same as VERN, therefore, only the differences will be discussed here. The ALT jet commands will be modified to limit the number of jets that fire simultaneously (1, 2, or 3), limit continuous jet on time, and incorporate delays between firings based on the values loaded on SPEC 20. None of these parameters can be altered in the VERN jet select logic. ALT jet select logic will be run if one of the VERN_CMD axes has a value of +1 or -1.

Since ALT jet select logic uses the primary jets, the jet select logic is performed by group, rather than by individual jet. In other words, the ALT logic will select one, two or three jet groups to fire (using the same dot product or min angle logic as described by the VERN jet select paragraphs above) and will then command the highest priority jet in each of those groups to fire (see Figure 16-14 for arrangement of jet groups). It will never fire more than one jet from the same group at the same time. The AUX_ANG_INCS used by ALT jet select logic are the average accelerations for all the jets in the group, so there is one AUX_ANG_INCS vector (R,P,Y) for each of the 14 jet groups based on the selected Control Accel on SPEC 20.

The ALT jet select logic will also take into account whether the DAP is configured for LOW Z, Tail-only control, or X jet pitch rotation enable. The jet select logic will only consider the groups that are available for firing in these configurations. The following Table contains a list of jet groups available (See Figure 16-14) depending on the DAP configuration:

TABLE 16-5: ALT Jet Select Jet Groups Available

| DAP Configuration | Jet Groups Available |
|--|--------------------------------------|
| X jet pitch rotation inhibited, no Low Z, Nose and Tail | All Groups but 1,7,8 (11 groups) |
| X jet pitch rotation inhibited, no Low Z, Tail only | Groups 9 - 14 (6 groups) |
| X jet pitch rotation inhibited, Low Z, Nose and Tail or Tail only (Low Z takes precedence) | Groups 2,3,5,6,9,10,13,14 (8 groups) |
| X jet pitch rotation enabled, no Low Z, Nose and Tail | Groups 1 - 14 (14 groups) |
| X jet pitch rotation enabled, no Low Z, Tail only | Groups 7 - 14 (8 groups) |
| X jet pitch rotation enabled, Low Z, Nose and Tail or Tail only (Low Z takes precedence) | All Groups but 4,11,12 (11 groups) |

PRI jet select:

If PRI is selected, the jet select process will choose the highest priority jet or jets per group, based on the DAP selections and ROT_JET_CMD and TRANS_JET_CMD. The required jet groups will be determined from a look up table (See Orbit DAP FSSR, STS 83-0009, Tables 4.2.2.2.1-1 through 4.2.2.2.1-3). Any PRCS jet (whether PRI or ALT is selected) that has been deselected by RM or the crew will not be commanded to fire by the jet select logic. If the highest priority jet in a group is unavailable when a command is issued, then the second priority jet will be commanded to fire, if available. If it, too, has been deselected, then the third priority jet (if the group has three jets) will be commanded to fire. The priorities of the PRCS jets within each group can be modified on SPEC 23, and these priorities will be utilized by both the PRI and ALT jet select logic.

Z translation, roll rotation, or pitch rotation Default logic and look up table:

PRI jet select logic begins by combining the ROT_JET_CMD and TRANS_JET_CMD into a twelve bit array (ACOM) where the even bits are translation or rotation commands (the order is X, Y, Z, R, P, Y) and the odd bits are the sign of the proceeding command (1 = negative, 0 = positive). If a Z translation, roll rotation, or pitch rotation is being commanded, Pitch Rate and Z Translation Rate Default logic and table look up are performed to determine how many jets in Groups 4, 5, 6, 11, 12, 13, and 14 (See Figure 16-14) to fire. There are 45 cases (or tables) that can be selected based on crossfeed/interconnect configuration, High Z selection (on DAP panel), number of jet failures, Norm Z or Low Z selection (on DAP panel), high or low rate pitch rotation (see explanation below), and the differing combinations of Z translation, pitch rotation, and roll rotation (See Orbit DAP FSSR, STS 83-0009, Table 4.2.2.2.1-1). High or low pitch rotation is determined based on the following equation:

$$\text{HI_PITCH} = 1 \text{ If Low Z selected OR (Nose and Tail selected AND ((No failure of jet groups (2 OR 3) AND (5 OR 6)) OR - pitch rotation commanded))}$$

Basically, with no jet failures, high rotation rate will be used when Nose and Tail (ALL on SPEC 20 for Item 15, 35, 19, or 39) is selected or Low Z is selected (Low Z overrides Nose only or Tail only). Low rotation rate will be used when Nose only (NOSE on SPEC 20) or Tail only (TAIL on SPEC 20) is selected with no jet failures.

If a +Z translation is commanded while Low Z is selected, the logic sets the X_JETS_PLUS_Z flag to ON and changes the Z component of ACOM to 0. The X_JETS_PLUS_Z flag will tell the X Translation Default logic to fire appropriate + and -X jets to accomplish the +Z translation. Also, if a -Z translation is commanded and either Group 5 or 6 has failed, the translation will be locked out if a yaw rotation is being commanded at the same time. If there is no yaw rotation being command, then a -Z translation will be allowed with Group 5 or 6 failed.

The next portion of the logic masks the appropriate forward or aft pitch jets if low rotation rate is selected (Nose only or Tail only selected on SPEC 20, based on the equation above) and no Z translation is taking place. The logic also takes into account whether a roll rotation command is present or not. If a roll rotation command is present and Nose only is selected, aft jets will be used and forward jets will be masked (roll rotation can only be accomplished by firing aft jets).

Y translation or yaw rotation Default logic and look up table:

Next, if a Y translation or yaw rotation is required, Yaw Rate and Y Translation Default logic and table look up are performed to determine how many jets in Groups 2, 3, 9, and 10 to fire. There are 9 cases (or tables) that can be used based on high or low rate yaw rotation (see explanation below), jet failures, and differing combinations of yaw rotation and Y translation (See Orbit DAP FSSR, STS 83-0009, Table 4.2.2.2.1-2). High or low rate yaw rotation is determined based on the following equation:

$$\text{HI_YAW} = 1 \text{ If Nose and Tail selected OR ((Group 5 or 6 failed) AND (Group 5 or 6 jets commanded to fire))}$$

High yaw rate will be used if Nose and Tail (ALL on SPEC 20 for Item 16, 36, 19, or 39) is selected or if either Group 5 or 6 has failed and a jet in Group 5 or 6 is being commanded to fire. The latter part of this equation is necessary due to the yaw cross coupling that exists if a single Group 5 or 6 jet is firing without the balancing effect of a jet in the other group.

Next, the Y translation component of ACOM will be set to 0 if Group 2, 3, 5, or 6 jets are failed. Finally the jet mask logic will mask the appropriate forward or aft yaw jets if low rotation rate is selected (Nose only or Tail only selected on SPEC 20, based on the equation above) if no Y translations are taking place.

X translation Default logic and look up table:

Next, if a Z translation is commanded and High Z is selected on the DAP, the X translation component of ACOM will be set to 0.

Finally, if an X translation or +Z translation using X jets (X_JETS_PLUS_Z flag is ON) is required, then X Translation Default logic and table look up are performed to determine which jets in Groups 1, 7, and 8 to fire. If a +Z translation is being commanded and 4 or more forward jets are being commanded to fire this cycle, then 1 jet from Group 1 and 1 jet each from Groups 7 and 8 (as long as 1 jet is available in each group) will be commanded to fire. Otherwise, 2 jets from Group 1 (as long as 2 jets are available) and 1 jet each from Groups 7 and 8 will be commanded to fire for a +Z translation in Low Z. If a -X translation is being commanded and 4 or more forward jets are being commanded to fire this cycle, then 1 jet from Group 1 will be commanded to fire. Otherwise, 2 jets from Group 1 will be commanded to fire for a -X translation (as long as 2 jets are available). If a +X translation is being commanded, 1 jet each from Groups 7 and 8 will be commanded to fire (as long as 1 jet is available in each group).

As in ALT and VERN, PRI jet select logic will create NEW_JONLST using the jets that it is commanding to fire and update DELTA_OMEGA_RCS based on these jets. If the jet was fired during the last cycle, DELTA_OMEGA_RCS will be updated base on the AUX_ANG_INCS value. If the jet was not fired during the last cycle, a minimum impulse scale factor will be multiplied by AUX_ANG INCS for the particular jet before it is added to DELTA_OMEGA_RCS. This is to take into account the time that it takes the jet to reach its full impulse once it is commanded to fire.

If PCT is active, Post Contact Docking Jet Selection module (GFD_ORB_DOCKING_JS) will be called and it will overwrite NEW_JONLST.

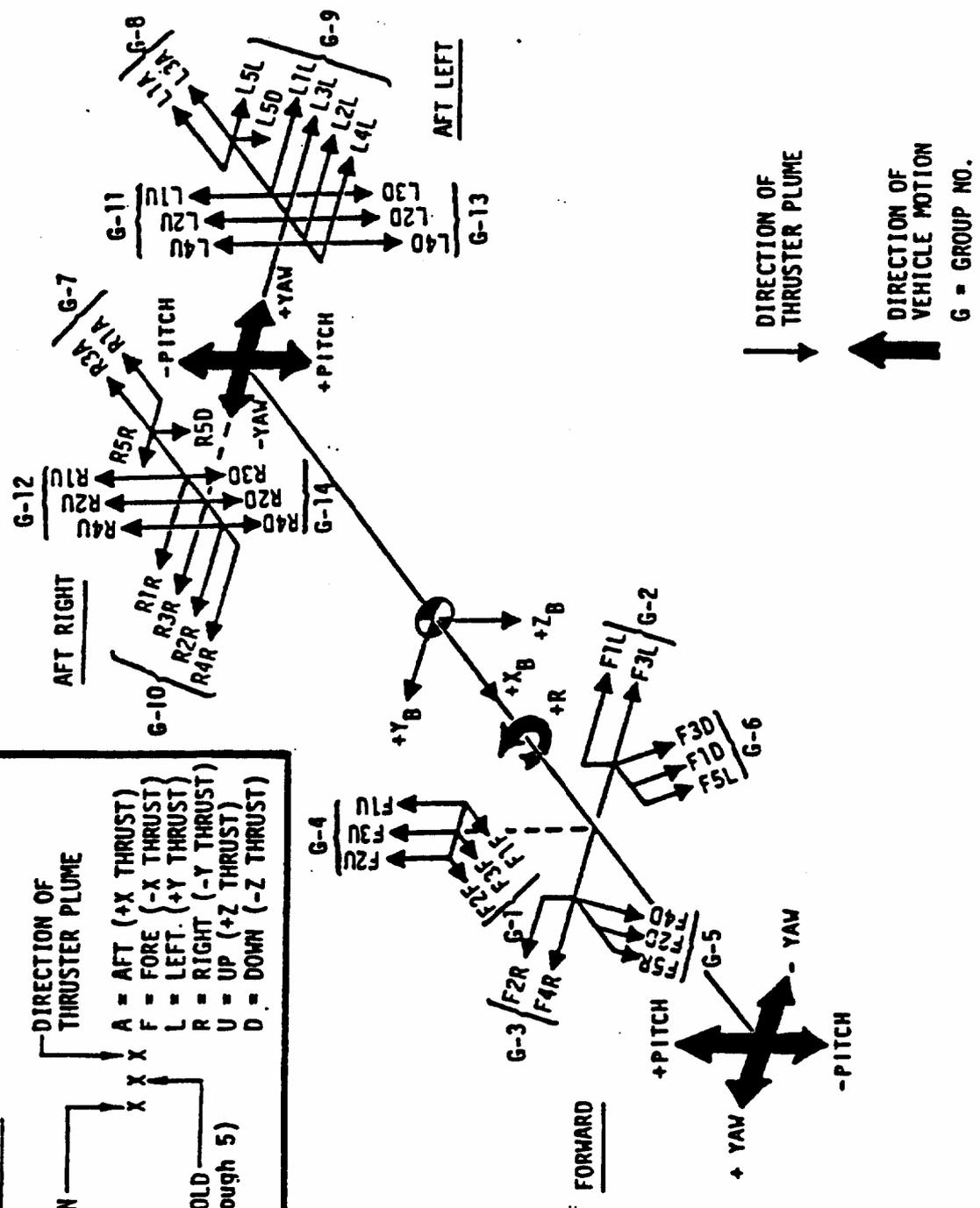
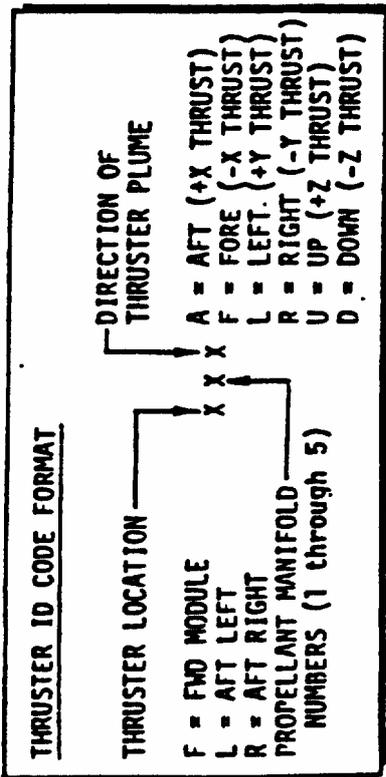


Figure 16-14: RCS jet locations and plume directions.

16.7.3 Post Contact Thrusting Arm/Activate Logic module (GC1_ORB_DOCK_ARM_ACT)

This software module is used to arm/disarm and activate/deactivate PCT during the docking timeframe. It also calls the DAP Load Processor module (GC6) to reconfigure the DAP loads once the docking sequence is completed or if it is aborted.

The module begins by setting the PCT_DAP, which is a 24 bit array, defined from I-loads. The PCT_DAP will be sent to the RECON software module (UNPACK_SW submodule) to reconfigure the DAP settings after the PCT sequence is complete, or has timed out or been aborted. The PCT_DAP current I-loads will reconfigure the DAP to A/FREE/VERN, PULSE in all translation axes, and no LOW Z.

Next the module checks the output of the SPDBK/THROT AUTO/MAN push buttons on F2 and F4. If either button is pushed and was not pushed the previous cycle (leading edge detection), then the software will set DOCKING_ARM to 1 (arms PCT) if it was previously 0 (not armed) and sets it to 0 (disarms PCT) if it was previously 1 (armed). The module now checks the output of the BODY FLAP AUTO/MAN push button on F2 or the spare PBI on either DAP panel. Using leading edge detection, the software activates PCT (sets a flag which activates Post Contact Docking Jet Selection module) if any of these 3 push buttons are pushed, DOCKING_ARM =1 (PCT is armed), and PCT was not previously active. If PCT was previously active, pressing BODY FLAP AUTO/MAN push button on F2, SPDBK/THROT AUTO/MAN push buttons on F2 and F4, pressing the spare PBI on either DAP panel, taking the RHC or THC out of detent, or selecting FREE on either DAP panel will stop the PCT sequence and send a request to load PCT_DAP to the RECON module and DAP A9/B9 to the DAP Load Processor module. Once the PCT sequence times out, PCT_DAP is sent to the RECON module and DAP A9/B9 is sent to the DAP Load Processor module. While PCT is active, the CONTACT_DAP will be sent to the RECON module. The CONTACT_DAP is defined as FREE/PRI (neither DAP A or B is requested) with PULSE selected for all translation and rotation axes, and no LOW Z.

TABLE 16-6: PCT Quicklook

| PBI | When PCT is disarmed... | When PCT is armed... | When PCT is active... |
|---|-------------------------|--|-----------------------------|
| L or R SPDBK/THROT AUTO/MAN PBI (AUTO lit on PBI when PCT armed/active) | Arms PCT | Disarms PCT | Disarms and Terminates PCT* |
| L BODY FLAP AUTO/MAN PBI or DAP: Spare PBI (PBI lit when PCT active) | No Effect | Modes to DAP: FREE/PRI and activates PCT | Terminates PCT* |
| DAP: FREE PBI or THC/RHC out of detent | Normal Function | Normal Function | Terminates PCT* |

* PCT DAP will be loaded including: A9/B9 will be loaded, A/FREE/VERN, PULSE in all translation axes, no LOW Z

16.7.4 Post Contact Docking Jet Selection module (GFD_ORB_DOCKING_JS)

The Post Contact Docking Jet Selection module is used to select primary RCS jets to fire in an I-loaded open loop sequence when PCT is active. The purpose of PCT is to drive the Orbiter docking mechanism into the target mechanism to achieve acceptable capture conditions.

This software module is activated when an activation and initialization flag is set in the PCT Arm/Activate Logic module. Once this module is activated, it modifies the NEW_JONLST from JET_SELECT using jets in Groups 5, 6, 13, and 14 based on jet availability. I-loads define the firing sequence and the number of jets in each group fired. The sequence is broken down into 2 phases. Current I-loads define the firing sequence in the following manner:

- Phase 1: 4 jets (1 jet each from Group 5, 6, 13, 14) fire 3 80 msec pulses, each followed by a 160 msec delay
- Delay between phases: 800 msecs
- Phase 2: 2 jets (1 jet each from Group 5 and 6) fire 4 160 msec pulses, each followed by an 80 msec delay

16.7.5 On-Orbit DAP OMS Steering module (GC8_ORB_OMS_STEERING)

This software module provides a commanded vehicle steering angle in degrees (per Orbiter body axis) to the Orbit DAP Critical Processor module (GUIDANCE_COMPENSATION submodule), which is used to calculate the vehicle

angular rate commands during an OMS burn. The commanded vehicle steering angle is calculated by taking the cross product of the commanded thrust direction and the sensed vehicle acceleration during the burn.

16.7.6 On-Orbit State Estimator Part 1 Filter module (GC7_ORB_P1FILT_6HZ)

This software module runs at 6.25 Hz and works with the State Estimator Part 2 Filter (in RECON module running at 12.5 Hz) to provide estimates of Orbiter attitude and rate using measured and extrapolated values. Just like the Part 2 Filter, this module is bypassed if the DAP is in the IMU Fail mode.

Attitude is determined by integrating body attitude increments from Orbiter Attitude Processor software. This is MEASURED_ATTITUDE and computed every cycle. If a Control Accel is loaded that contains notch filters, and NOTCH_FLTR_ENA is selected on SPEC 20, the MEASURED_ATTITUDE will be run through the Notch Filter logic submodule. The purpose of this submodule is to attenuate frequency response and loads. If the Notch Filter logic computes a change in attitude which it attributes to a natural vibration frequency (defined by the upper and lower limits of each notch filter), it modifies MEASURED_ATTITUDE in an attempt to take out that particular attitude change. The new attitude is set equal to the parameter ATTITUDE. ATTITUDE is what is used by RCS_ERRORS (in the DAP Critical Processor module) to determine attitude errors for PHASE_PLANE. If the Notch Filter Logic does not identify vehicle movement within a defined notch or if NOTCH_FLTR_ENA is not selected on SPEC 20 (or if a Control Accel is selected that has no notch filters), then ATTITUDE is set equal to MEASURED_ATTITUDE.

The ATTITUDE parameter is then filtered with attitude and attitude rate extrapolations (from the State Estimator Part 2 Filter) in two parallel filters (a "rate" filter and an "acceleration" filter) to obtain attitude and rate estimates. The rate filter creates the rate estimate that is used by many other software modules. The acceleration filter uses the attitude residual (difference between the ATTITUDE parameter and the attitude estimate) to obtain a time-averaged estimate of any undesired accelerations (which are used by the State Estimator Part 2 Filter and PHASE_PLANE).

16.7.7 Auto Maneuver/Track module (GC9_ORB_AUTO_MNVR_TRACK_1HZ)

This software module produces Orbiter desired attitude and rates (when the DAP is in AUTO) which it sends to the DAP Critical Processor (RCS_ERRORS submodule). It receives the parameter AUTO_ATT_OPTION and the required attitude quaternion from GCZ in RECON. AUTO_ATT_OPTION can have one of three values: 1 = pre-burn maneuver (in MM202), 3 = maneuver/track, 4 = attitude hold (2 is not an option). If AUTO_ATT_OPTION has a value of 1 or 4, the required attitude quaternion and current attitude quaternion are multiplied together to determine the total error. If AUTO_ATT_OPTION has a value of 3 (MNVR or TRK option is currently selected on UNIV_PTG display), then the required attitude quaternion is sent to a quaternion integrator. This integrator takes into account any differences between the current attitude timetag (T_NOW from Attitude Processing) and the required attitude timetag (T_CMD from Universal Pointing) and uses the required rate (from Universal Pointing) to appropriately modify the required attitude. This modified required attitude quaternion and the current attitude quaternion are multiplied together to determine the total error.

The total error computed earlier is used to determine whether the module will run in the track mode or the maneuver mode. The current attitude deadband (based on the current DAP settings) is multiplied by KH (an I-load which currently has a value of 4) and compared to the total error. If the total error is larger than 4 times the deadband, the maneuver mode is selected. Once maneuver mode is selected, it will remain selected until the total error is less than the deadband value, at which time the track mode will be selected. Track mode will remain in affect until the total error again exceeds 4 times the deadband.

On the first pass in the maneuver mode, the current attitude is snapshot, summed with a bias (the bias is calculated to attempt to compensate for the lag in achieving the ROT_RATE at the beginning of a maneuver. The bias should allow the DAP errors to move into the hysteresis region of the phase plane during the maneuver), and sent to RCS_ERRORS as the desired attitude. RCS_ERRORS propagates the desired attitude using the desired attitude increment parameter supplied by AUTO_MNVR_TRACK every cycle. The desired attitude increment is simply the desired rate multiplied by .08 seconds (the period of one DAP cycle). Desired rate is calculated by summing the ROT_RATE (from the DAP CONFIG display for the currently selected DAP) and the required rate (from Universal Pointing). It is sent to RCS_ERRORS every cycle also.

Note: The bias summed with the current attitude during the first pass of a maneuver may be incorrectly calculated when the DAP during the previous software pass was in manual LVLH. This is due to stale data being used to calculate the bias. The effects of this were seen during STS-108 (Reference User Note 50112).

If the track mode is selected, the desired attitude is calculated by summing the total error with the current attitude. Therefore, in track mode, the desired attitude is essential set equal to the required attitude. The desired rate is set equal to the required rate in the track mode.

16.7.8 Orbit DAP LVLH Track module (GC5_ORB_LVLH_TRACK)

This module produces the Orbiter desired attitude for RCS_ERRORS when manual LVLH is selected on the DAP panel. It operates in one of three modes, depending on the rotation submodes selected on the DAP panel. The modes are:

1. All axes in DISC RATE
2. Only one axis in PULSE
3. Two or three axes in PULSE

If the submode for all axes is DISC RATE, then LVLH_TRACK calculates the desired attitude in order to maintain the initial reference Orbiter attitude with respect to the LVLH reference frame. The reference LVLH attitude will reset each cycle that the RHC is deflected beyond detent. When the RHC is back in detent, the last reference LVLH attitude will be used to calculate the desired attitude.

If only one axis is in PULSE, the module establishes a plane normal to the axis in PULSE and produces a desired attitude, which constrains the axes in DISC RATE in that plane. RHC inputs about the axis in PULSE will not affect plane in which the attitude is constrained. RHC inputs about one of the axes in DISC RATE will reset the plane until the RHC is back in detent.

If two or three axes are in PULSE, then the module sets the desired attitude to the current attitude (essentially placing the Orbiter in free drift).

16.7.9 Orbit OMS Processor module (GCR_ORB_OMS_PROCESSOR)

This module produces OMS gimbal position commands that it sends to the OMS TVC CMD SOP in order to drive the OMS gimbals. There are three submodules that accomplish this.

TVC_LAW_PITCH_YAW receives vehicle pitch and yaw rate commands from GUIDANCE_COMPENSATION in GFF and compares them to estimated vehicle rates from the State Estimator Part 1 and Part 2 Filter and the desired rate errors from PHASE_PLANE. The calculated rate errors are used to produce the pitch and yaw gimbal position commands. Gimbal trim values are also calculated in this submodule during an OMS burn.

TVC_LAW_ROLL provides roll control of the Orbiter during two OMS engine burns. Roll rate commands are compared to estimated vehicle roll rates and desired roll rate errors to determine a calculated rate error. This error is used to produce a roll component of the pitch gimbal position command, which is sent to TVC_MIXER.

TVC_MIXER derives the pitch gimbal position command as a function of the pitch gimbal position command from TVC_LAW_PITCH_YAW and the roll component from TVC_LAW_ROLL. The resultant command is limited so that the total pitch command will not exceed pitch command limits.

16.7.10 DAP Load Processor module (GC6_ORB_DAP_LOAD_PROC)

This module manages the DAP A and B load tables. It controls the transfer of parameter values between the selected DAP loads, the DAP CONFIG (SPEC 20) display and the 30 load tables.

16.8 NOTES OF INTEREST

A. RMS/FCS Instability During Payload-Extended Operations

The following information is a select portion of that provided by the Charles Stark Draper laboratory (CSDA) in Memo SSV-88-49, dated 10/21/88. The reader should refer to this memo for a more detailed discussion of this subject.

The interaction between the orbit digital autopilot (DAP) and the remote manipulator system (RMS) during payload extended operations can potentially result in the formation of RCS-pumped short-period limit cycles (SPLC). This type of behavior is highly undesirable due to the excessive fuel consumption and jet duty cycle rates incurred. In addition, there is the possibility that sufficient energy may be 'pumped' into the system by the RCS jets to induce divergent orbiter body-rate oscillations, or excessive loads or payload motion.

Although RMS/DAP operations are currently restricted to planned positions that have been analyzed for VERN and ALT DAP stability, and have had mission-specific DAP I-loads assigned, the formation of SPLC's may still follow the occurrence of certain unanticipated contingencies. In particular, if a failure (e.g., of a joint) results in a payload/RMS configuration that lay outside regions previously studied, then there maybe strong reason for concern. The probability that VRCS stability problems may arise at a contingency position increases as the payload c.g. position moves in the -X and -Z Orbiter body-axis reference frame (note that most brake slip results in payload motion toward the payload bay).

In the event that such a contingency does occur, it will be very important to be able to recognize the formation of a DAP/FCS instability during its early stages. Caught quickly, SPLC should result only in moderate increases in fuel consumption, and should not threaten the success of the mission. Given rate-error, phase-plane, and jet firing plot information, an instability is very easy to detect. However, on the mission consoles, only downlink numerical data is available. Hence instability trends must be detected by careful observation of the data. In the absence of post processed phase-plane data, the formation of SPLC may not be immediately apparent. In order to detect the forming instability from the numerical data, several trends must be observed.

During the coast phase of a maneuver, or during attitude hold after termination of a maneuver, the DAP rate error will oscillate between +/- rate deadbands. In "normal" stable configurations, the rate error should damp with a time constant corresponding to the RMS mode in use at the time.

B. Shelf Pulsing

Shelf pulsing is a phenomenon where long periods of jet cycling in two axes can be seen in either ALT or VERN. Cross-coupling from jets being selected to fire could cause attitude and rate errors in two axes to simultaneously be driven to the phase plane shelf. When this occurs, jet cycling (and high RCS propellant usage) will continue, usually until a jet firing is performed in the third axis. The probability of occurrence and duration of shelf pulsing will increase when tighter attitude deadbands are selected with relatively high maneuver rates. Shelf pulsing normally occurs at the end of a maneuver.

To stop shelf pulsing if it occurs, action should be taken to re-center the phase plane. A manual DAP (LVLH or INTRL) could be selected on the DAP panel, followed by re-selection of AUTO. Also, Item 21 CNCL on the UNIV PTG display and then re-selection of the required option (Item 18 MNVR or Item 19 TRK) would stop the jet cycling.

16.9 AUTOMATIC REBOOST

16.9.1 Background

As the International Space Station (ISS) assembly progresses, the manual reboost procedures will not be certifiable for a number of reasons. Reboost events will become lengthier and structural constraints will become more restrictive for later flights. The reboost events for these flights will be too cumbersome for the crew to execute because of the length of the reboost and small pulse sizes of 80 milliseconds are too small for the crew to keep track of. Also, ISS assembly will make use of scavenged propellant from the Shuttle mostly in the aft. Aft only propellant requires coordinated THC/(RHC or moding) commands for long periods. Therefore, the required reboost for these later flights have been automated for the first time with the OI-28 Flight Software. The OI-28 Flight Software initially flew on STS-98/5A. Since that flight, upgrades have been made to automatic reboost with the OI-29 Flight Software version. In the OI-29 version, the capability to stagger jet firings from a jet group has been added for the reboost translation firings as discussed later in this brief.

16.9.2 OI-28 Automatic Reboost

The automatic reboost software consists of sequences of reboost translation jet firings and rotational control jet firings separated by time delays. For an automatic reboost sequence, reboost translation jet firings continue until attitude and rate errors exceed the attitude and rate deadbands. If this happens, a time delay (Trans to Rot Delay) will occur in order to damp out unwanted structural frequencies before rotational control jets are fired. The rotational control jets will then fire until the errors are within the deadbands and then another time delay (Rot to Trans Delay) will occur before reboost translation firings are resumed. The sequence above is summarized in the following:

AUTOMATIC REBOOST SEQUENCE

1. Reboost Translation Firings
2. Trans to Rot Time Delay
3. Rotational Control
4. Rot to Trans Time Delay

Up to four different automatic reboost configurations can be specified. Each configuration specifies which jets are to be fired and for how long during the reboost translation firing period. It defines the delay times between translation firings and rotational control firings. Each configuration design also dictates which rotational control mode (VERN or ALT) to use as shown below. Note that the delay times are not used if VERN is exclusively used for reboost translation firings and rotational control firings. As an example, the following configurations for STS-98/5A are shown below.

| Configuration | Reboost Translation Period | Rotational Control |
|---------------|-----------------------------------|--------------------|
| 1. +X/- Pitch | +X jets with down firing jets | ALT |
| 2. +X | +X jets only - no pitch balancing | ALT or VERN* |
| 3. -Z VRCS | Down vern jets | VERN |
| 4. +X/VRCS | +X jets w/ aft down vern jets | VERN |

*As of STS-102, Configuration 2 reboosts only use ALT for rotational control, due to VRCS duty cycle constraint violations.

An explanation of the trade-offs between each four configurations will be covered later in this section. First, a detailed explanation of each part of the automatic reboost sequence for a given configuration will be presented to aid in describing the different configurations.

Reboost Translation Firings

Reboost translation firings for each configuration is defined by four vector I-Loads. The first vector of six integers (Jet Groups) defines which jet groups are to be used for the reboost. The second vector of six integers (No. of Jets) defines how many jets from the groups chosen by the first vector are going to be fired. The third vector of six real values (ON Times) defines how long the jets will fire. The fourth vector of 6 real values (Jet Firing Start) defines firing starts in seconds per jet group. This feature was added in OI-29 to reduce loads from primary reboost translation impulses to extend the opportunity of using automatic reboost beyond station assembly sequence 12A. The fifth vector of ten real values (Reboost Interval Factor) helps define the time between each jet firing. For all the missions up to this date, the Reboost Interval Factor values have been 1.0. The Reboost Interval Factor is multiplied by the Reboost Interval, which the crew defines on SPEC 20 and the product of these two parameters is the time between jet firings.

The following example plot shows a sequence of reboost translation firings for Configuration 1 were the +X jets are used in combination with aft down firing jets for pitch balancing. The following example of vectors define the plot below:

Jet Groups [7, 8, 13, 14, 0, 0]
No. of Jets [1, 1, 1, 1, 0, 0]
ON Times [0.08, 0.08, 0.16, 0.16, 0.0, 0.0]
Jet Firing Start [0.0, 0.0, 0.240, 0.240, 0.0, 0.0]
Reboost Interval Factor [1.0, 1.25, 0.75, 0.75, 1.25, 1.4, 0.0, 0.0, 0.0, 0.0]

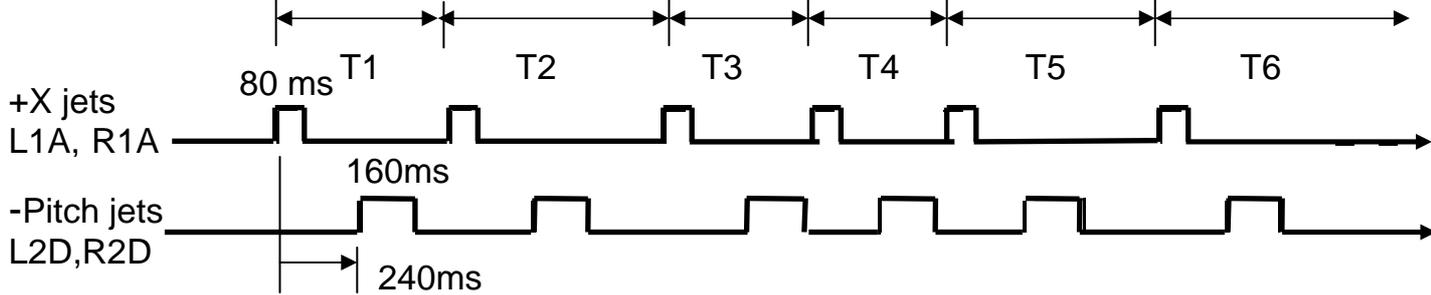


Figure 16-15: Reboost Translation Firings

As shown above, the +X jets will be pulsed for 80 ms on each firing and the aft pitch jets will be pulsed 160 ms. All jets will be fired at the same time and will have the same varying delay time between firings as defined by $T(x)$ which is the product of the Reboost Interval Factor times the Reboost Interval Item entry in SPEC 20. A zero in the vector I-Load simply means to cycle the vector again from the beginning of the vector.

Trans to Rot Time Delay

When the attitude or rate errors exceed the deadbands, a time delay will occur before rotational control firings begin. This time delay is defined by a vector I-load of one real value.

Rotational Control Firings

Rotational control firings are defined by the current mode that the DAP is using for attitude control (VERN or ALT) during reboosts. It is the same DAP mode used for nominal on-orbit control except for an additional change to the time between jets firings in ALT mode. An I-loaded vector of ten real values (Alt Delay Factor) helps define the time between jet firings during the rotational control period. When the ALT mode is active, the time between jet firings is the product of the Alt Delay Factor times the Alt Delay for the given DAP configuration being used. Note that this vector will be used for the entire on-orbit operations, even when not executing automatic reboost.

Rot to Trans Time Delay

When the attitude and rate errors get to within the deadbands during rotational attitude control, a time delay will occur before reboost translation firings begin. This time delay is defined by a vector I-load of one real value.

Automatic Reboost Sequence Plot

The following plots put all the elements described earlier together. During the reboost translation period the jets are fired for the set ON time defined in the I-loaded vector. The times between firings are defined as the Reboost Interval on SPEC 20 times the Reboost Interval Factor vector. There are no jets fired for rotational control at this time. When the attitude or rate errors go beyond the deadbands, Trans to Rot Time Delay occurs and no jets are fired to dampen out and unwanted frequencies. After the delay, jets for rotational control are fired to bring the errors to within the deadbands. As soon as the errors are within the deadbands another time delay (Rot to Trans Time Delay) is executed and no jets are fired. After the time delay the reboost translation period is resumed.

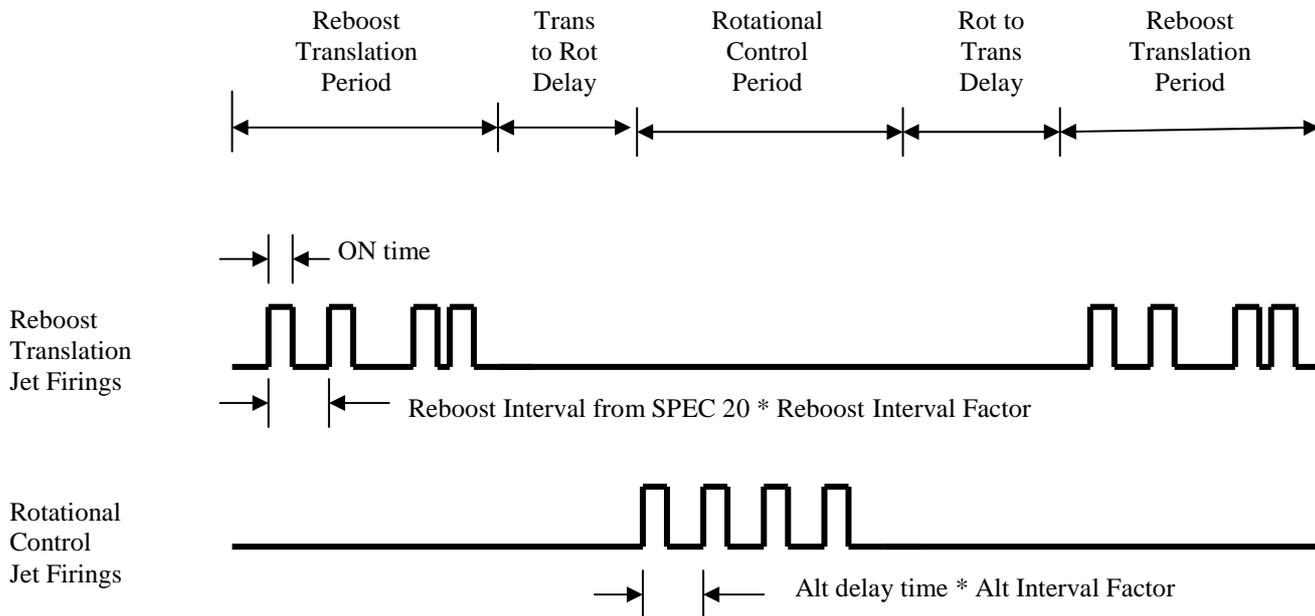


Figure 16-16: Reboost Translation and Rotational Firings

Crew Interfaces

Two crew displays, DAP CONFIG (Figure 16-3) and UNIV PTG (Figure 16-4) are used to configure and enable automatic reboosts. On the DAP CONFIG display, the crew can select which reboost configuration will be used from the four different configurations with an Item 8 entry (1-4). An Item 9 on this display defines the Reboost Interval time mentioned earlier in secs (0.08 – 999.99). On UNIV PTG, an Item 25 will enable the start time if the time duration does not equal zero. An Item 26 on this display will cancel automatic reboost and Items 27-29 defines the duration of automatic reboost sequence in hours, minutes, and seconds, respectively. This time counts down during reboost and includes the whole sequence of translation, rotation, and delay periods.

Other crew interfaces are the FREE PBI on the DAP panel which aborts reboost when pushed. The RHC deflected beyond the softstop and any deflection of the THC will also abort automatic reboost.

Aborts and Abort Overrides

Automatic reboost abort will occur if any of the conditions described above for the controllers are met. In addition, if all jets of any of the required jet groups used for reboost translation jet firings fail, the software will abort the automatic reboost sequence. An OPS transitions will also abort automatic reboost.

To override aborts, execute an Item 25 on UNIV PTG to enable automatic reboost again.

Automatic Reboost Configurations defined for STS-113/11A

1. +X/-Pitch (ALT)

Firing the primary +X aft jets result in a positive pitch motion since the CG of the combined Shuttle and ISS structures are well above the orbiter in the –Z direction. In order to balance this upward pitch motion the aft down primary jets are also fired to provide a negative pitch for pitch momentum balancing. The ALT DAP mode is used for rotational control with the use of the primary jets. This configuration results in higher structural loading than the other configurations.

2. +X (ALT)

The primary +X aft jets are used for reboost translation firings and the ALT DAP mode can be used for rotational control.

3. -Z VRCS (VERN)

Only the down firing vernier jets are used for reboost translation firings in the -Z direction. Rotational control is provided by verns. This is the best configuration for minimizing the loads on the payload structure.

4. +X/VRCS (VERN)

This configuration is a combination of the +X primary aft jets with aft down firing vern jets for pitch balancing. It uses verns for rotational control.

There are many things to consider in deciding which configuration is the best to use for automatic reboost. Things that have to be considered are structural life, propellant usage, time out of space station desired attitude for thermal and electrical concerns, and achieving the delta velocity necessary for reboost to be successful. The following table shows trade-offs of delta velocity achieved, time to gain a certain delta velocity, and total propellant used. This can be used as a quick reference to show configuration pros and cons. Extensive analysis is completed by Draper Laboratories pre-flight for each reboost configuration and any decisions made during a specific flight in regards to reboost configurations should be made using the flight specific analysis.

TABLE 16-7: Reboost Configuration Summary (STS-113)

| Reboost Configuration | Delta Velocity (ft/sec) per hr | | | | Time to gain 1ft/s | | Propellant (lbs) | | | | Result Vector Angle |
|-----------------------|--------------------------------|------|--------|-------|--------------------|-----|------------------|-------|-------|-----------|---------------------|
| | DVX | DVY | DVZ | Tot | Min | Sec | Total | Fwd | Aft | per 1ft/s | |
| 1 +X/-P(ALT) | 1.70 | 0.00 | -1.12 | 2.04 | 29 | 26 | 221.1 | 0.0 | 221.9 | 108.5 | -33.4 |
| 2 +X(ALT) | 0.96 | 0.01 | -0.74 | 1.22 | 49 | 16 | 134.1 | 2.8 | 131.1 | 110.1 | -37.6 |
| 3 -Z VRCS(VERN) | 0.73 | 0.01 | -10.37 | 10.40 | 5 | 49 | 1130.9 | 673.6 | 457.3 | 108.8 | -86.0 |
| 4 +X/VRCS(VERN) | 2.45 | 0.00 | -1.44 | 2.84 | 21 | 09 | 365.3 | 0.0 | 365.3 | 128.8 | -30.4 |

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SECTION 17 AEROJET DAP

17.1 GENERAL

The aerojet digital autopilot (DAP) is a set of general equations used to develop effector commands that will control and stabilize the Orbiter vehicle during its descent to landing from orbital altitude. The aerojet DAP resides in the primary redundant set during operations sequence (OPS) 3, but is used only during major mode (MM) 304 (entry) and MM 305 (terminal area entry management (TAEM) and approach/landing (A/L)). This is accomplished by using either manual control stick steering (CSS) commands or guidance derived (AUTO) commands as inputs to the equations. The solution of these equations results in deflection commands to the appropriate aerosurfaces or fire commands to the available reaction control system (RCS) jets.

A great deal of flexibility has been designed into the aerojet DAP software to accommodate its task of controlling the vehicle as a spacecraft during the ballistic region of entry and as a conventional aircraft during atmospheric maneuvering. This flexibility is achieved by reconfiguration logic which uses navigation (NAV)-derived and measured trajectory parameters to pick a particular set of equations for the immediate task.

A descriptive overview is presented in the following paragraphs and describes the type of flight control system (FCS), the various types of inputs it receives, and the various commands it generates. The remainder of this section is a presentation of the control laws that govern and control the vehicle about the three rotational axes. This also includes the priority rate limiting (PRL) logic and the jet select logic (JSL). The control laws are further broken down into the major entry, TAEM and A/L software divisions.

17.1.1 Description

The aerojet DAP is a rate command control system such that the automatic or manual commands request vehicle rotation rates about the three body rotational axes (roll, pitch, and yaw). Sensed body rate feedback is employed for stability augmentation in all three axes. This basic rate system is retained in a complex network of equations whose principal terms are constantly changing to provide the necessary vehicle stability while insuring that sufficient maneuver capability exists to follow the planned trajectory.

For conventional flight, the solution of the equations results in deflection commands to the elevator, ailerons, rudder, speedbrake, and body flap (BF). For exoatmospheric flight or flight during the trajectory where certain surfaces are rendered ineffective by adverse aerodynamics, a combination of aerosurface and RCS jet commands is issued. Inputs can consist of automatic commands from entry guidance in the form of attitude, angle of attack, surface position, and N_z commands and manual roll, pitch, and yaw rate commands

from crew-operated controllers or a combination of the two since the software channels may be moded independently.

Various peripheral software modules calculate terms for flight control dedicated displays and several pilot workload reduction functions. These modules, terms, and all other characteristics mentioned above will be emphasized in more detail in the appropriate control laws section.

17.1.2 Reconfiguration

This software module includes logic to configure the control laws based on current position along the entry trajectory. The reconfiguration logic checks the appropriate switching parameters to determine if a new configuration is warranted and provides the switching discrettes needed to add or delete the necessary modules. In the case of changing between AUTO and CSS modes, it also notifies the crew that reconfiguration has taken place by lighting the appropriate pushbutton indicators (pbi's) on panels F2 and F4.

17.1.2.1 AUTO/CSS pbi's

The failure modes and effects analysis (FMEA) and the critical items list (CIL) reviews of the FCS switches have reemphasized that the failure of a switch to transfer is a possibility. For entry flight control, failure of either the left or right pitch or roll/yaw AUTO pbi in the engaged (stuck closed) state could be catastrophic if no crew action is taken. Flight control will be in the AUTO mode whenever the AUTO pbi is engaged and the rotational hand controller (RHC) is in detent. This situation would make the vehicle manually unflyable. To negate the stuck closed failure, a crewman must continually depress either the left or right CSS pbi for the affected axis (pitch or roll/yaw). The flight control software will give the manual mode priority over the AUTO if both are requested concurrently. Backup flight system (BFS) engage will also circumvent the failure since only the CSS mode is available in the backup software. No other workaround has been identified as a viable remedy.

17.2 CONTROL LAWS

This section discusses the set of equations for vehicle control during the entry phase of the mission. A good deal of detail has been omitted for simplification, but sufficient information is provided to develop a functional understanding of the control laws and the capability they provide.

17.2.1 Entry (MM 304)

Contained in this section is the discussion of the equations that control the Orbiter during MM 304 or the entry portion of the descent flight phase. These are a subset of the complete system of equations; however, from a control standpoint, they stand alone as the controlling equations during MM 304.

17.2.1.1 Longitudinal Control Laws

For this document, the pitch channel, speedbrake channel, and BF channel are considered to be integral parts of the longitudinal control laws because of their interaction and because their function (control of vehicle longitudinal motion) is separate and distinct from the roll and yaw channels.

17.2.1.1.1 Pitch channel (fig. 17-1)

The pitch channel software issues two separate effector commands to control vehicle pitch motion. These are (a) the number of pitch jets to be fired to effect desired response, and (b) an elevator deflection command when in atmospheric flight. The effectiveness of the jets and elevator is a function of the dynamic pressure q with elevator effectiveness increasing with dynamic pressure and jets decreasing with dynamic pressure. Pitch jets provide forces for dynamic control before achieving a dynamic pressure of 40 lb/ft^2 where they are turned off, and the elevator is used for dynamic control after achieving a dynamic pressure of 2 lb/ft^2 . This overlap provides uninterrupted control in pitch when deleting the jets as a pitch effector.

The elevator is activated to provide aerodynamic trim as $q \geq 0.5 \text{ lb/ft}^2$ thereby relieving the pitch jets of this function and saving fuel. Prior to achieving $q = 40 \text{ lb/ft}^2$, elevator trim is calculated by numerically integrating the dynamic elevator and pitch jet commands. After $q = 40 \text{ lb/ft}^2$, elevator position feedback (PFB) is used as the elevator trim position.

The total elevator command is the sum of the elevator trim deflection and the dynamic deflection. The dynamic deflection is proportional to the pitch rate error signal which is produced by comparing the commanded pitch rate with the actual pitch rate (pitch rate feedback). Commanded pitch rate is either the auto guidance command or crew command from RHC and RHC trim.

The value of the elevator trim position is important in providing adequate roll response so that the elevator is driven to a desired trim schedule by BF activation and deployment of the speedbrake. The speedbrake is nominally deployed at $M = 10$ to induce a more positive elevator trim deflection. This is accomplished by biasing the elevator PFB during speedbrake activation. The elevator trim deflection is sent to the BF channel to facilitate the BF contribution to maintaining the proper elevator trim schedule.

An angle-of-attack error signal is produced by subtracting the guidance commanded angle of attack from the actual (NAV-derived) angle of attack. This signal is shipped to attitude director indicator (ADI) processing for display on the ADI pitch error needle. This occurs at all times while in MM 304 whether in AUTO or CSS.

17.2.1.1.2 Speedbrake channel (fig. 17-2)

Two outputs of the speedbrake software are provided during MM 304. These are (a) the commanded speedbrake deflection in hingeline degrees going to PRL and (b) the AUTO deflection command in hingeline degrees output to the surface position indicator (SPI) for processing. The AUTO deflection command is displayed on the SPI regardless of the speedbrake mode for crew reference and tracking.

When the speedbrake is in AUTO and Mach is greater than 0.95, a command is developed by using Mach numbers to select a deflection from a deflection vs. Mach number scale. Below Mach 0.95, the command is driven by guidance. A negative bias is added to this command at high Mach numbers to keep the speedbrake panels fully closed when the command is 0°. Finally, upper and lower software limits are imposed on the command before shipping the deflection command to PRL. The lower limit fluctuates depending upon the Mach number.

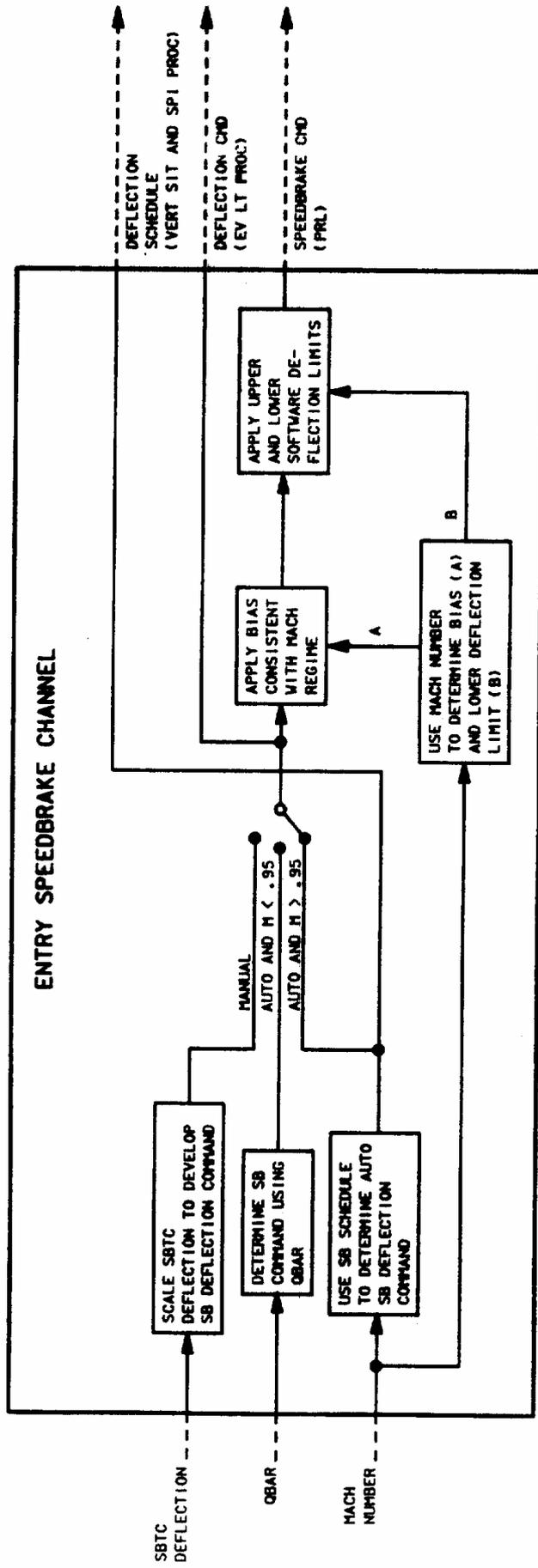
In the MANUAL mode, the speedbrake/thrust controller (SBTC) deflection is input from the selection filter, and a constant gain is applied to convert SBTC degrees to speedbrake hingeline degrees. This MANUAL speedbrake deflection command is then processed exactly as the AUTO command by applying the same bias and software limits.

17.2.1.1.3 BF channel (fig. 17-3)

The function of the BF is to provide the necessary pitch trim to keep the elevons on a preselected elevator trim deflection schedule. The MANUAL mode provides the capability to position the BF independent of the trim deflection schedule.

When the BF is in the AUTO mode, the elevator trim deflection command from the pitch channel is compared with a scheduled trim deflection selected by Mach number. Hysteresis allows up to 1° difference in these two deflections before commanding the BF to deflect at a rate that changes with Mach number. This rate command is integrated to produce a BF position command. A constant upward (-) deflection limit and a Mach variable downward (+) deflection limit are applied to the integrator output. This limited BF position command is then compared to the actual BF position feedback. Hysteresis is then applied to this difference and up to a 1° position error is allowed before a discrete is issued telling the BF to either drive up (-) or down (+) to null the error between the scheduled and actual elevator trim.

When in the MANUAL mode, the crew command to drive up or down is changed to a BF rate command that is a constant rather than Mach dependent. This means that the BF will deflect only when commanded by the crew without regard to the elevator trim position. The rate at which the integrator increments in the MANUAL mode is significantly greater because of slow BF movement and the size of the hysteresis deadbands. This allows the BF to respond instantly to a crew command regardless of where the position error is inside the deadband.



16863172. ART. 3

Figure 17-2. Entry speedbrake overview

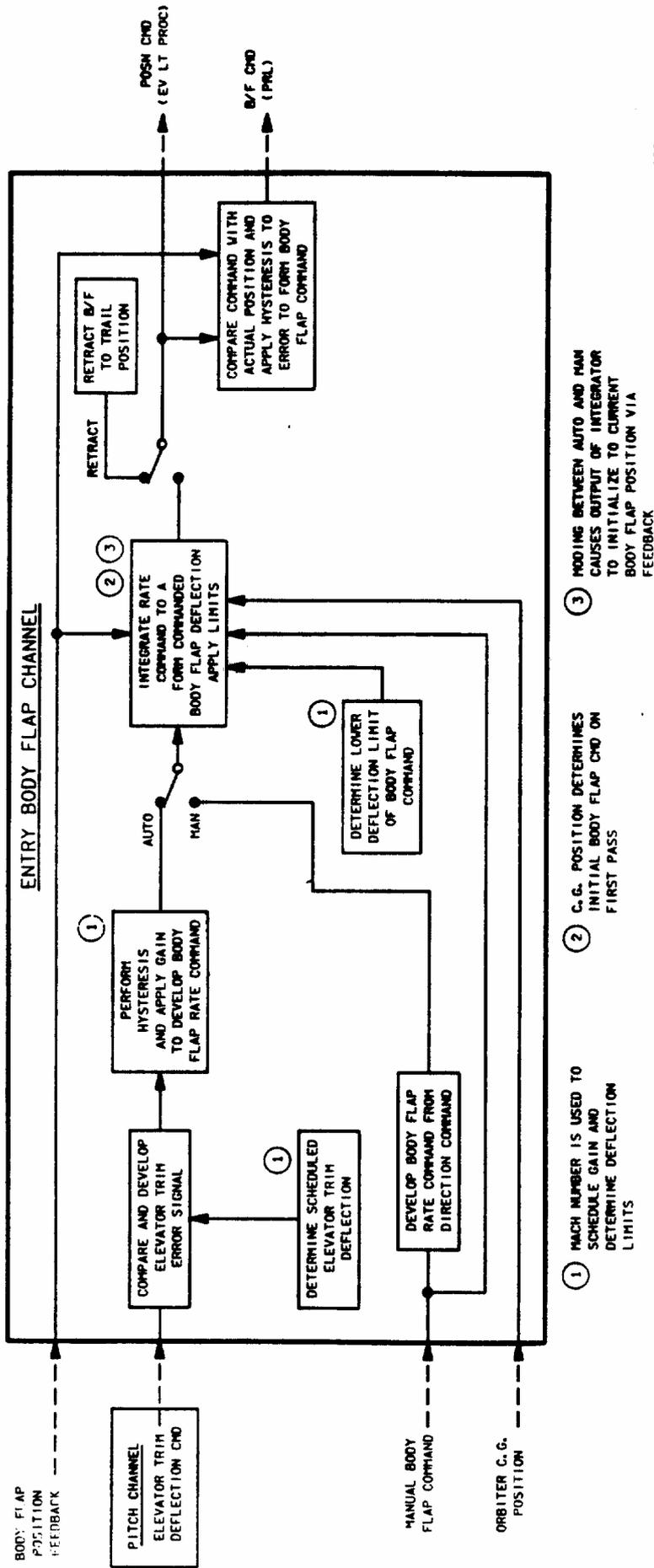


Figure 17-3. Body flap overview

Provision is made to initialize the position command (integrator output) as a function of the calculated center of gravity position. Also, because of the difference in the rates of incrementing the integrator, there can be a considerable mismatch between the commanded (integrated) position and the actual position when moding between MANUAL and AUTO. This is prevented by initializing the output of the integrator to the PFB value each time the output of the slew switch changes value. This automatically nulls the input to the final hysteresis.

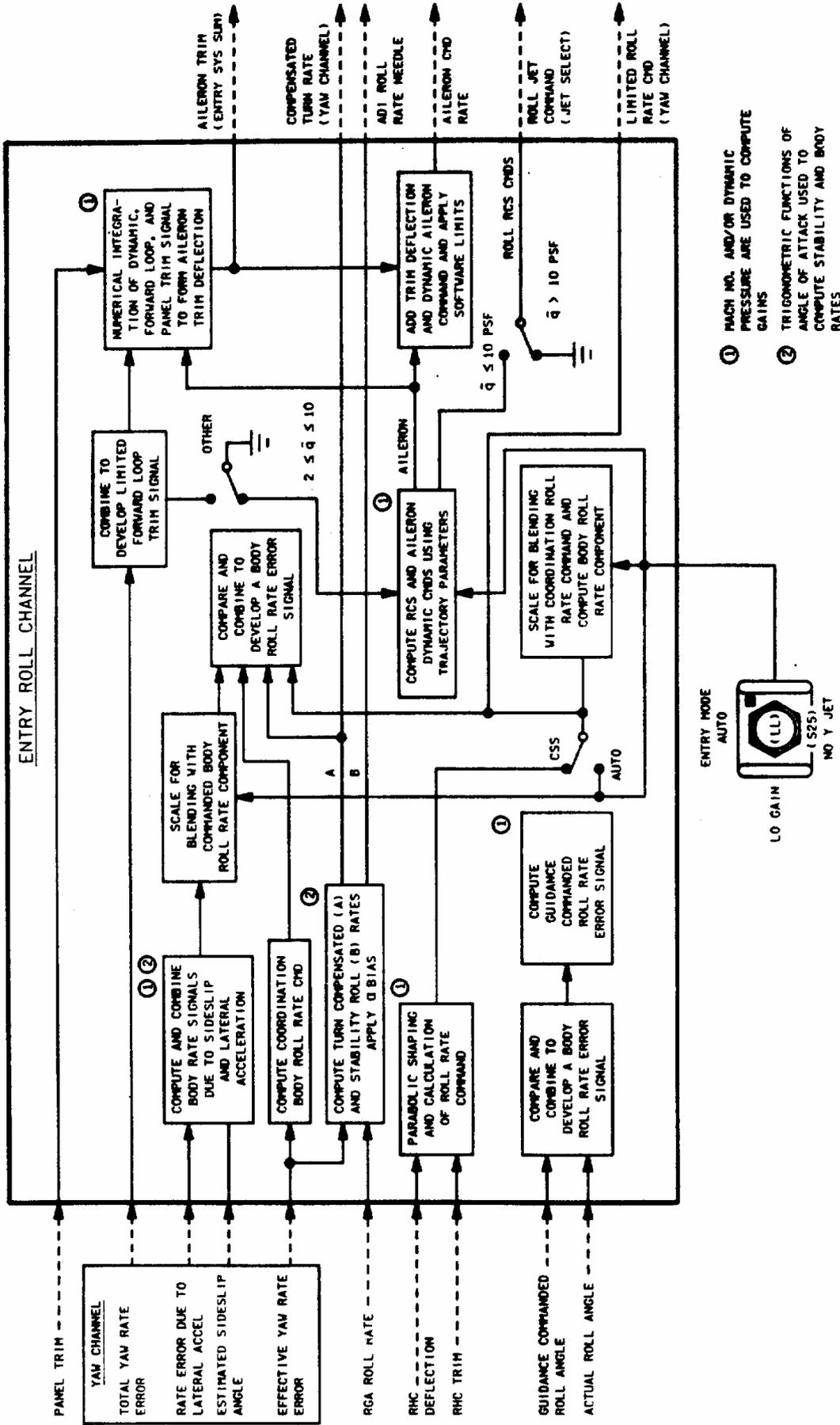
17.2.1.2 Roll/Yaw Control Laws

Refer to figures 17-4 and 17-5. The roll/yaw control laws issue a total of four effector commands to provide moments for maneuvering about the body roll and yaw axis. Because of the high angles of attack under which the Orbiter operates during entry, there is considerable interaction between the roll channel equations and the yaw channel equations.

As in the pitch channel, there is an overlap in commanding the RCS jets and aerosurface effectors in both the roll and yaw channels to provide uninterrupted control authority. The jets become ineffective with dynamic pressure, and the aerosurfaces become more effective. The ailerons are used for dynamic control and trim for dynamic pressures greater than 2.0 lb/ft² and the roll jets are turned off upon reaching a dynamic pressure of 10 lb/ft².

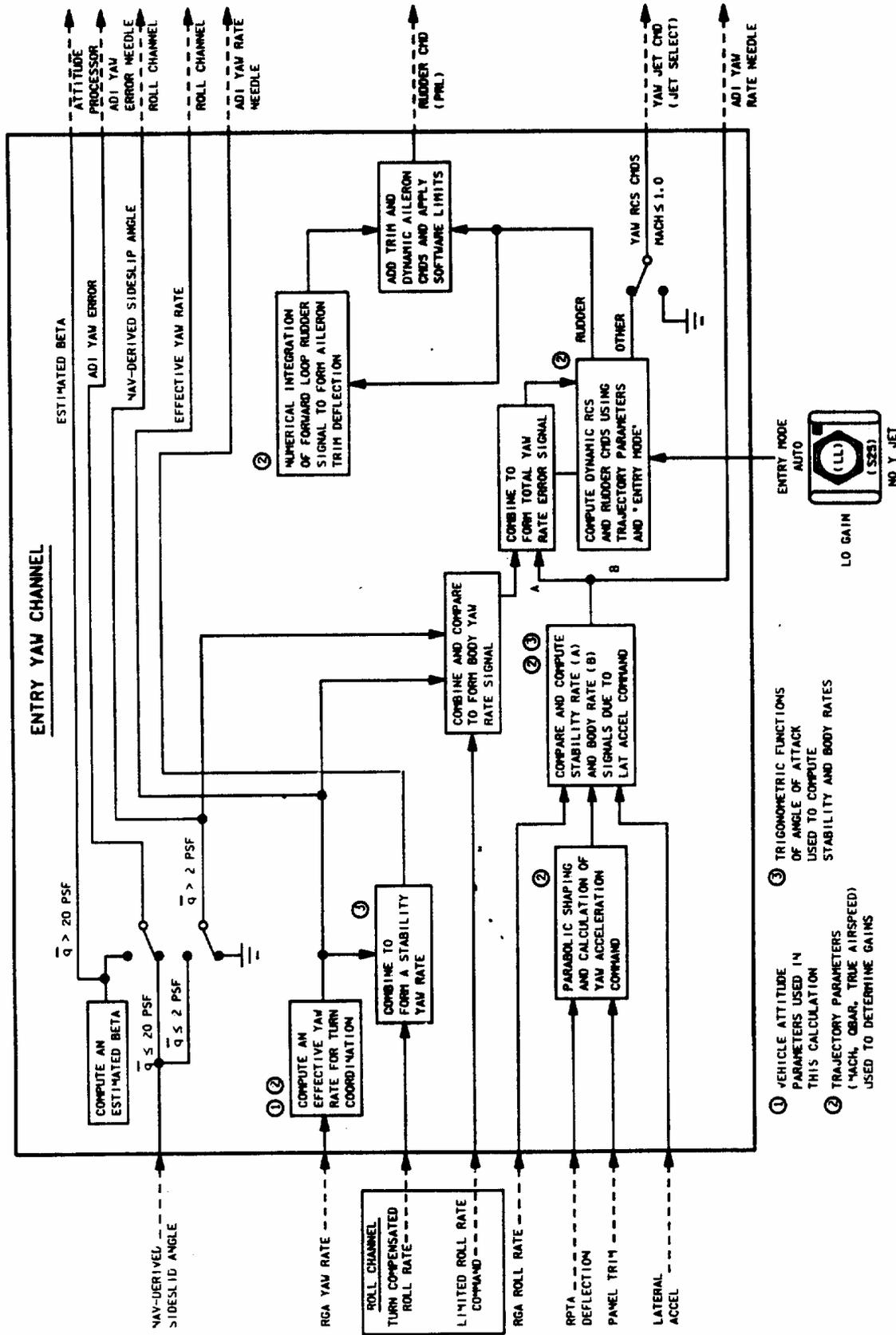
In the yaw channel, the rudder is not commanded to deflect for dynamic control or trim until reaching $M = 4.2$, and the yaw jets are used until $M = 1.0$.

For normal roll/yaw configuration (ENTRY MODE switch in AUTO), a roll maneuver is initiated by commanding the body yaw component of the stability roll rate (limited roll rate command). This body yaw component is a significant fraction of the stability roll rate because of the large angles of attack at which the Orbiter operates during entry. The basis for the roll signal is an effective yaw rate signal from the yaw channel. A roll rate command is calculated that will null the effective yaw rate signal, thereby controlling sideslip excursions. Other terms are added to provide coordinated maneuvers and rate feedback stability. As the Mach number decreases during entry, this configuration gradually changes to that of a conventional aircraft where the ailerons provide the control moments to initiate a roll rate, and the yaw channel provides a coordinated maneuver with rudder and/or yaw jets. These two configurations are normally referred to as the early and late systems, respectively, with transition occurring in two steps at $Mach = 3.0$ and $Mach = 0.9$. At the higher Mach numbers ($M > 3.0$), 10 percent of the limited roll rate signal is provided in the reverse direction to the ailerons to aid the yaw jets in achieving the desired body yaw component of this same signal. Subsequent to the initial command, the ailerons and roll jets provide sideslip dumping and coordination. During the transition, 75 percent of the limited roll rate command signal is used to establish a roll rate with ailerons, and only 25 percent of the sideslip damping term from the yaw channel is used. The yaw channel continues to provide the body yaw



10063174.ART.3

Figure 17-4. Entry roll channel



18863175. ART. 2

Figure 17-5. Entry yaw channel

component of limited roll rate command. Below Mach = 0.9, the roll channel provides all the effector power for initializing and sustaining roll maneuvers with the yaw channel providing the conventional coordination.

17.2.2 TAEM and A/L (MM 305)

This section contains the set of control laws that exist from the TAEM interface through rollout. The longitudinal control laws are a subset of the complete longitudinal control laws for the aerojet DAP. Although there is some commonality between entry and TAEM and A/L control laws, there are also several significant differences. These configurations for TAEM and A/L are shown in the overviews in figures 17-6, 17-7, and 17-8.

The changes occurring in the roll/yaw channel are more subtle than in pitch, but nonetheless significant. These changes primarily involve deleting certain terms at various times during the TAEM and A/L trajectory.

17.2.2.1 Longitudinal Control Laws

During TAEM and A/L, the speedbrake is used to modulate aircraft energy. This dynamic use of the speedbrake means that the speedbrake and BF will play a more active roll in determining and maintaining an elevator trim position.

17.2.2.1.1 Pitch channel

The basic structure of TAEM and A/L pitch channel corresponds to the entry configuration at the TAEM interface. Steady state and dynamic elevator trim deflection are provided by elevator PFB, speedbrake-induced pitch effects are trimmed out with speedbrake crossfeed, manual elevator return is provided by pitch panel trim, and RHC deflection and RHC trim inputs differenced from pitch rate feedback provide a manually commanded dynamic elevator deflection. The pitch channel outputs are the total elevator deflection command (trim deflection plus dynamic deflection) to priority rate limiting, the elevator trim deflection to BF channel, and an ADI pitch error signal.

The major difference from entry occurs in processing a pitch rate command in the AUTO mode. The command received from guidance is an incremental normal acceleration (N_z) instead of an angle of attack. This normal acceleration must be converted to a pitch rate command before pitch rate feedback can be subtracted for the necessary stability augmentation. An absolute N_z command is developed by adding the equilibrium glide load factor to the guidance commanded increment plus another fraction of the guidance-commanded incremental N_z . This fraction varies with true airspeed and reflects the changing relationship between normal acceleration and pitch rate. The actual sensed normal acceleration is then subtracted from this absolute command leaving a residual N_z which is converted to a pitch rate.

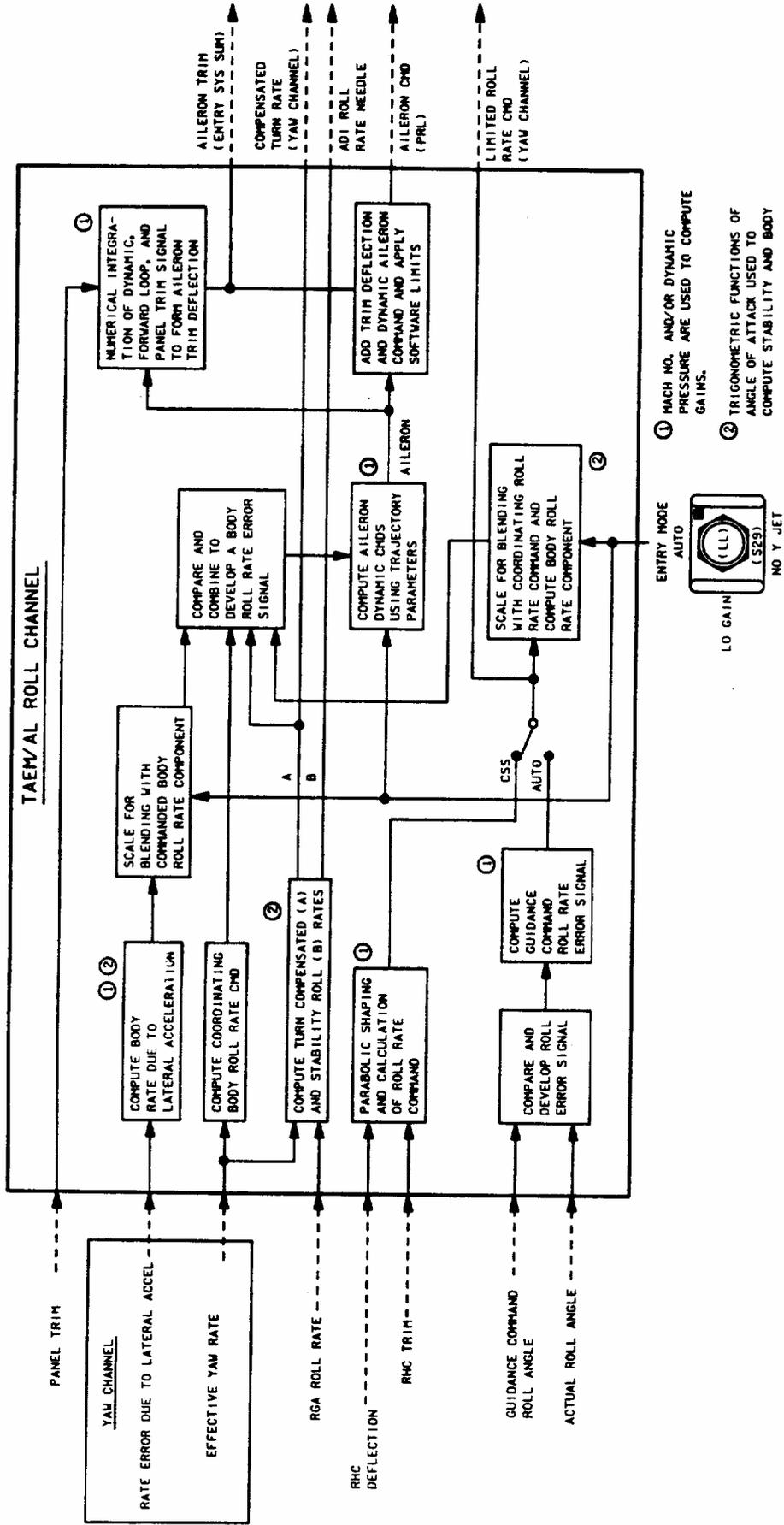


Figure 17-6. Roll TAEM overview

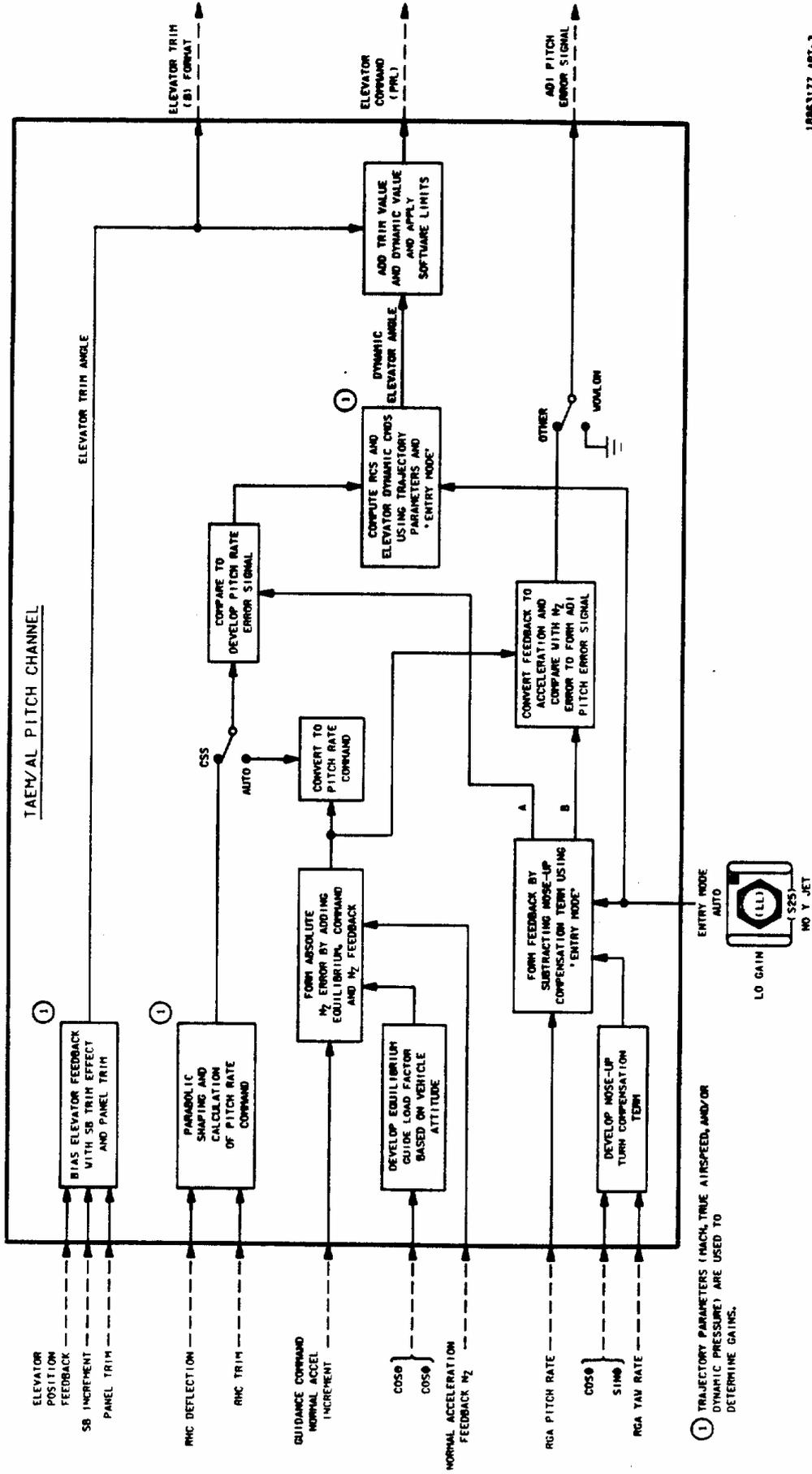


Figure 17-7. Pitch TAEM overview

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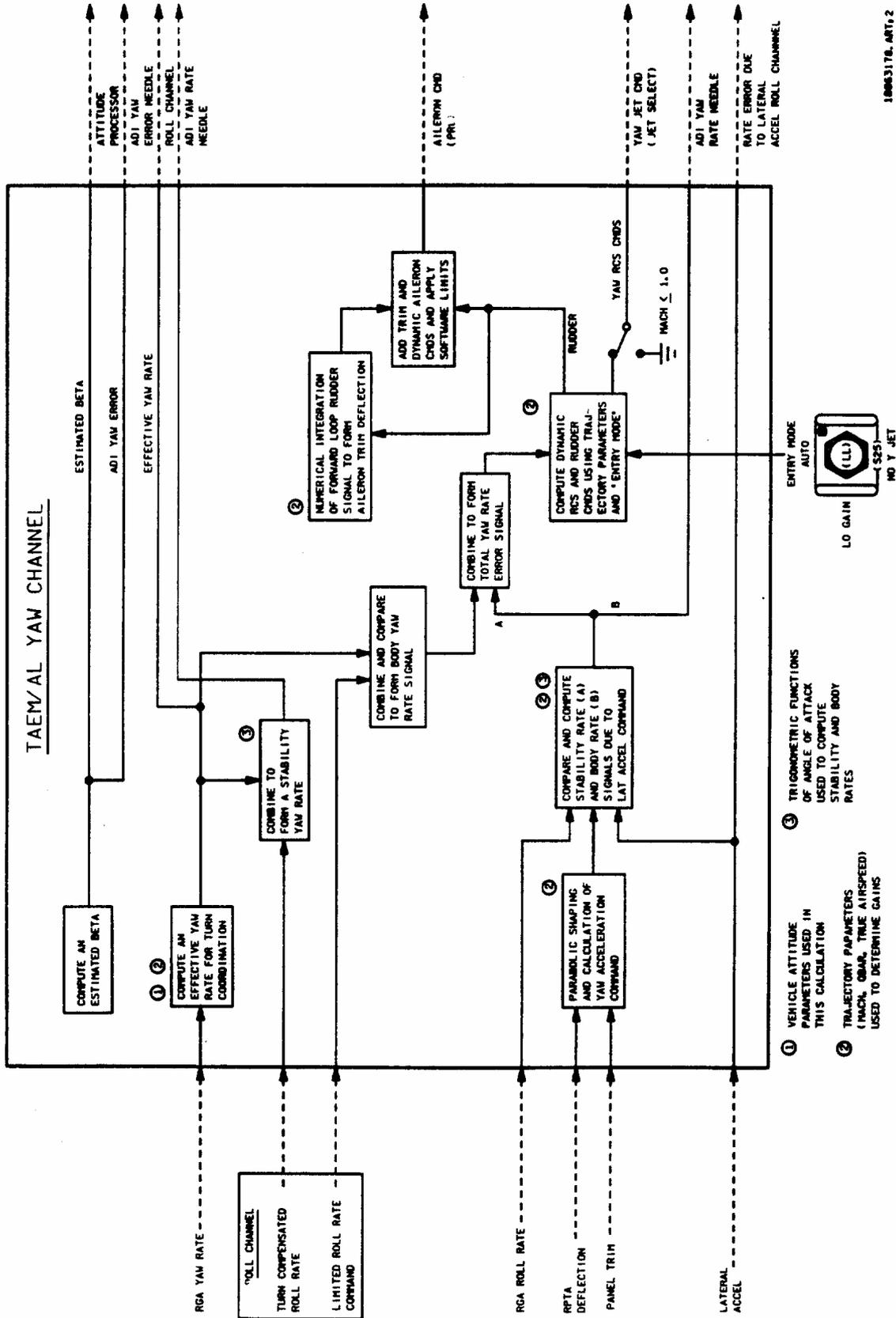


Figure 17-8. Yaw TAEM overview

This is the pitch rate command, or the pitch rate needed to satisfy the guidance increment. Pitch rate error signal is calculated by subtracting the body pitch rate feedback from this commanded rate. In any turn, a component of the turn rate will be sensed by the pitch rate gyros (nose-up turn compensation). To keep the nose from falling through during a turn, this component is calculated and subtracted from the sensed pitch rate.

The ADI pitch error signal during TAEM is developed in units of g's. This is done by taking the pitch command in g's just prior to conversion to a pitch rate, adding the nose-up compensation term in g's, and subtracting the pitch rate feedback in g's. This signal then is directly proportional to the pitch rate error signal, but differs in magnitude in association with the gain that converts pitch rate to normal acceleration. This signal is displayed on the ADI from TAEM interface until main gear touchdown.

17.2.2.1.2 Speedbrake channel

During TAEM and A/L, the speedbrake channel software follows closely that of entry except that the AUTO speedbrake command is formed in guidance and is used to modulate equivalent airspeed. The speedbrake is now used in a dynamic sense to modulate the TAEM trajectory. The speedbrake crossfeed into pitch channel will be critical in damping speedbrake-induced pitching moments.

Guidance logic determines when the speedbrake should be retracted (commanded CLOSED). This command is received by the speedbrake channel as a deflection command so that no special moding or logic must take place in speedbrake channel software to retract the speedbrake.

17.2.2.1.3 BF channel

The only BF logic peculiar to the TAEM and A/L MM is the BF retract logic. Upon receiving the BF retract discrete from A/L guidance, the BF is commanded to deflect to 0.0° , or the trail position. For the retraction to take place, the BF mode must be AUTO and the speedbrake deflection must be less than 50° . If the BF mode is manual or if the speedbrake is deflected 50° or more, the BF will not be retracted.

For slapdown, the pitch rate error command is composed of the difference between the pitch rate feedback and a desired constant pitch rate of -2 deg/s. This error is changed to an actual elevator command by a gain scheduled by pitch angle, such that the gain is reduced as the Orbiter achieves negative (nose down) pitch attitudes. This elevator command becomes the dynamic deflection command and is added to the trim deflection to compose the total elevator deflection command. The rate error signal is displayed on the ADI pitch error needle after receiving the weight on wheels latched on (WOWLON) discrete.

Upon commencing rollout (reception of nose gear contact discrete), the elevator trim deflection is set to zero and the dynamic elevator deflection command becomes a constant $+5^\circ$ (down). This down deflection is for main gear load alleviation during the rollout phase.

The above logic automatically performs the slapdown and rollout function when in AUTO pitch. When in CSS, the dynamic deflection is provided via RHC.

17.2.2.2 Roll/Yaw Channel

The most obvious differences may be noted by comparing the entry and TAEM and A/L diagrams. The TAEM and A/L roll and yaw channel signal flows as shown in figures 17-6 and 17-8. As the Mach number decreases below three, the yaw jet/rudder aileron blend gain decreases such that the ratio of aileron and yaw jet/rudder control approaches that of a conventional aircraft.

When in AUTO, the aileron trim angle is held constant upon receiving the flatturn discrete. Upon commencing rollout, the aileron trim angle is set to zero. Neither of these events occurs in CSS, and the trim function remains throughout rollout.

The downmode deadband module is bypassed whenever the late system is chosen or whenever $M \leq 0.9$. The latter condition, known as Mach, requires bank/roll rate crossfeed (MRB/RRXF).

Whenever the roll/yaw channel configuration is such that coordinated flight is not required, the steady state turn correction term is omitted. The two conditions that must exist for this to occur are AUTO mode and receipt of flatturn discrete, or CSS mode and receipt of rollout discrete.

The yaw jets are disabled below Mach = 1.0 as the rudder gains aerodynamic effectiveness. Shortly thereafter ($Mach \leq 0.9$), the LIMITED ROLL RATE command is deleted as a command to the yaw channel in favor of the TURN COMPENSATED ROLL RATE. This event effectively completes the transition to conventional aircraft control. Whereas, before attaining $Mach \leq 0.9$, the roll and yaw channels were simultaneously commanding deflections to achieve the commanded roll rate, the yaw channel is now responding to a feedback roll rate.

17.2.2.2.1 Flatturn and rollout

The roll/yaw channels have an extensive interface to provide coordinated flight during entry and TAEM maneuvering. This coordination occurs because some component of the roll command is taken as a yaw command and conversely the major component of the actual roll signal is designed to dampen sideslip rates. This complex interaction between roll and yaw channels precludes any independent operation of the two channels. During the landing/rollout sequence, however, there is a requirement for the roll and yaw channels to be independent. Aircraft geometry requires that the wings be level, or very nearly so, at touchdown. This requires groundtrack corrections to be made by yawing the vehicle in the desired direction. To provide for increased independence, the steady state turn correction terms are deleted from both channels. This is done upon receiving flatturn discrete when in AUTO or receiving rollout discrete when in CSS. The yaw command is set to zero until flatturn indication is received. At that point, A/L guidance calculates a yaw rate command proportional to the roll attitude command (same roll attitude command used previously in roll channel) and then sets the roll attitude command to zero. Also at flatturn, inputs to aileron trim are zeroed so that the aileron

trim is held constant during the landing. Upon receiving indication of rollout, the aileron trim is set to zero.

Also at flatturn, inputs to aileron trim are zeroed so that the aileron trim is held constant during the landing. Upon receiving indication of rollout, the aileron trim is set to zero.

The yaw rate command from A/L guidance is also used by nosewheel steering such that a deflection command to the nosewheel is sent out after receiving the ground-speed enable (GSENB). This occurs as ground speed decreases below 115 kts.

When the landing and rollout task is performed in CSS, the aileron forward loop trim is not disabled, and coordination between the roll and yaw channels is provided by the crew through the RHC and rudder pedal transducer assembly (RPTA). A different parabolic shape is applied to the RPTA command for determining nosewheel deflection. Once again nosewheel steering is not enabled until ground speed decreases below 115 knots. Unlike all other effector commands generated by the aerojet DAP, the nosewheel deflection command does not go to the PRL. It is sent directly to the aero-actuator subsystem operating program (SOP).

17.3 PRIORITY RATE LIMITING

The purpose of the PRL module is to provide the rate and acceleration limiting for both the elevator and the aileron commands. In addition, the module provides rate limiting for both the rudder and speedbrake with a special speedbrake slow closing rate for small speedbrake deflection angles. This slow closing rate is for protection against slamming the two rudder/speedbrake panels together. Another feature of the PRL module is the lower rudder rate limiting and maximum rudder/speedbrake closing rate when either the rudder or the speedbrake panels exceed a prescribed deflection limit. This is commonly referred to as the rudder panel margin.

17.3.1 PRL Overview

The inputs to PRL are the commanded positions for the elevator, aileron, rudder, and speedbrake from the flight control channels (fig. 17-9). The elevator and aileron commands are rate- and acceleration-limited for determining the position commands to each elevon as well as determining the amount of hydraulic flow used by the elevons. The rudder command is rate-limited for determining the rudder position command and the amount of hydraulic flow used by the rudder, which is summed with the hydraulic flow used by the elevons. This sum plus the hydraulic flow allowance for the BF and leakage is subtracted from the total amount of hydraulic flow available for the speedbrake, which is used for determining the speedbrake opening and closing rate limits. The speedbrake command is rate-limited for determining the speedbrake position command and the speedbrake increment. The rudder position and speedbrake position commands are combined and then compared to the rudder/speedbrake summed position limit to determine if the rudder/speedbrake panels are exceeding a prescribed panel limit (i.e., the rudder panel margin). If the rudder panel margin is exceeded, the rudder rate limits are changed, and the speedbrake closing rate limit is changed.

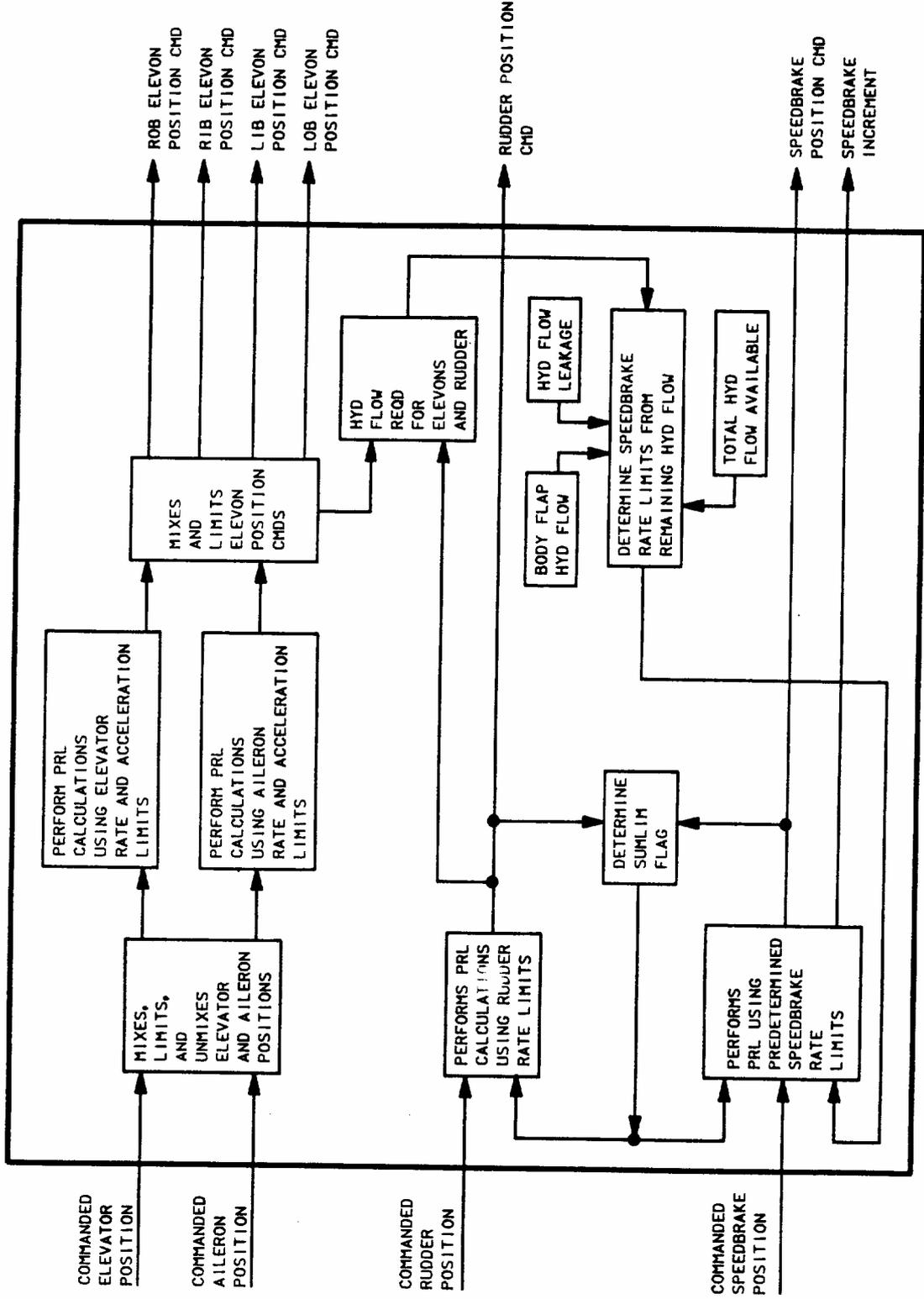


Figure 17-9. Priority rate limiting overview

In PRL, all rate-limited surface deflection commands are dependent upon the number of good hydraulic systems available. For the rudder/speedbrake section, the rudder panel margin is computed by the rudder/speedbrake position (limit 52.9°) minus the absolute value of the rate-limited rudder deflection command minus one-half the rate-limited speedbrake deflection command. In other words, rudder panel margin is equal to $52.9^\circ - \delta r - 1/2 \delta SB$. If this value is <0.0 , the sum limiting (SUMLIM) flag is set true. If the value is >0.0 , the SUMLIM flag is set false or SUMLIM (not sum limiting). This flag is utilized for determining the rudder and speedbrake rate limits as well as the amount of hydraulic flow available for the speedbrake.

Both the input elevator and aileron position command from the pitch and roll channels are mixed to determine the exact position command for the elevons which are checked against each elevon's position limit. The position limited elevon commands are then unmixed to form the position-limited elevator and aileron commands. The elevator position command and aileron position command are mixed to determine the right/left elevon positions which are position-limited and output as both the right and left elevons inboard and outboard position commands to the elevon actuators.

The rudder rate-limited error is converted to the hydraulic flow required for the rudder use. The elevator rate-limited command and aileron rate-limited command are mixed for determining the amount of hydraulic flow required for both the right and left elevons. The flow requirements for the rudder, right elevon, and left elevon are summed to form the total hydraulic flow required by the elevons and rudder.

The hydraulic flow required for the BF and the hydraulic flow allowed for leakage are subtracted from the total hydraulic flow available to give the remaining flow available. If the SUMLIM flag is set, the hydraulic flow required for closing the speedbrake is also subtracted from the total hydraulic flow available to give the remaining flow available. The actual hydraulic flow required by the elevons and rudder is subtracted from the hydraulic flow available which gives the hydraulic flow available for the speedbrake.

The hydraulic flow available to the speedbrake is converted to a speedbrake rate limit. The lesser of this rate limit and the speedbrake negative or closing rate limit is used for speedbrake negative or closing limit. The lesser of the rate limit and the speedbrake position or opening rate limit is used for the speedbrake positive or opening limit.

The speedbrake command is back-differenced with the postspeedbrake deflection command feedback to form a position error which is converted to a rate error. The rate error is then rate-limited according to the predetermined rate limit to form the rate-limited error. If the SUMLIM flag is true, the negative or closing rate limit for the speedbrake is fixed. Otherwise, the rate limit is the rate-limited error. Either way, the rate limit is used in the pitch channel.

The rudder position command and the speedbrake command are combined to form the rudder panel margin which was discussed previously.

17.4 JET SELECT LOGIC

The JSL determines which aft pod (right, left, or both) RCS jets must be turned on in response to the pitch, roll, and yaw jet commands from the entry flight control system.

For entry, JSL must adhere to the following constraints.

- a. Only the four Y-axis and six Z-axis RCS jets on each aft pod are utilized.
- b. No X-axis nor any vernier jets are utilized.
- c. Redundancy management (RM) sends the discrettes which designate which jets are available for firing.
- d. A maximum of four jets per pod may be fired simultaneously.
- e. During reconfiguration or when the crossfeed valve is open, the maximum combined total number of jets is four for both right and left pods. This action limits the number of yaw jets available during certain pitch and roll maneuvers.

The following functions are performed in each cycle through JSL:

- a. RM checks to see which RCS jets are good.
- b. The logic sets all roll and pitch jet commands to off.
- c. The logic computes the required roll and/or pitch JET ON commands. This gives a minimum pitch/roll jet-on time of one JSL minor cycle (0.08 sec).
- d. The logic computes the required yaw JET ON commands and the minimum number of JSL minor cycles the yaw jets must be on, based on the logic of the yaw RCS flag. If the flag is false, the yaw jets are on for one minor cycle (0.08 sec). If the flag is true, the yaw jets are on for four minor cycles (0.32 sec). This flag becomes true for yaw jet requirements below an altitude of 125,000 ft.
- e. The logic checks for reconfiguration or for the crossfeed valves being open. If either event is in process, the number of jets total in both aft RCS pods is four.
- f. The logic checks to see if the current firing jets will satisfy the requirements. If so, the current firing jets are issued another JET ON command for the next cycle.
- g. The logic issues JET ON commands to the appropriate jets.

During entry, the FCS uses the RCS jets in the following regimes.

TABLE 17-1. RCS USAGE

| RCS jets | Jets activated | Jets deactivated |
|----------|-----------------|--------------------------|
| Pitch | Entry interface | $q = 40 \text{ lb/ft}^2$ |
| Roll | Entry interface | $q = 10 \text{ lb/ft}^2$ |
| Yaw | Entry interface | $M = 1.0$ |

17.5 ENTRY FCS DOWNMODING

The STS entry/glide return to launch site (GRTLs) FCS is being designed to fly the nominal entry trajectory plus a large set of dispersions off-nominal. There are, however, potential flight conditions that the normal FCS may not handle very well or that may require a variation of the FCS to retain control. Preflight uncertainties or off-nominal conditions being considered at present are typical of the following: bending instabilities, large aerodynamic coefficient errors, angle of attack errors, anomalous RCS system malfunctions requiring minimum RCS fuel consumption, or control surface rate limiting.

The possibility of situations such as these led to the development of what is known as the ENTRY mode switch. The concept of downmoding is defined as the crew use of the cockpit panel-mounted ENTRY mode switch to selectively apply multipliers to the forward loop gains in the FCS pitch and roll/yaw axes and to effect reconfigurations in the roll/yaw channel control logic.

17.5.1 ENTRY Mode Switch

This switch has three positions, LO GAIN, NO Y JET, and AUTO.

17.5.2 LO GAIN

The LOW GAIN position decreases the forward loop gain on the pitch, roll, and yaw rate error by a factor of 0.5. Control response and damping will generally be slower with low gains as compared to nominal. So, if the FCS response appears to be overgained (high-frequency oscillatory) in pitch, roll, and yaw, selection of LOW GAIN should allow improved damping. If large amplitude diverging rates are induced as might be encountered in a large signal instability, the LOW GAIN may retard the vehicle accelerations, allowing recovery.

17.5.3 NO YAW JETS

If RCS fuel reserves are marginal, it is desirable to conserve available supplies for use in the flight regimes below Mach 5, where additional control authority may be required for trim and maneuvers. Aerovariation combinations may critically reduce surface effectiveness, forcing a dependence on the RCS for stability in that region. To save fuel for use in the critical region, the NO Y JET position could be used to eliminate commands to the yaw jets. Roll maneuver initiation and rate damping are then provided by ailerons and, below Mach 4.2, by the rudder.

In early entry, aileron deflections produce an adverse yawing moment overpowering the initial rolling moment, inducing a sideslip angle. As β increases, the vehicle rolls in the opposite direction. Taking advantage of this powerful adverse yaw and subsequent roll opposite the initial command, the FCS initially commands aileron deflections that generate a sideslip opposite to the desired roll direction. Since the ailerons are not augmented by the yaw jets in the NO Y JET mode, the initial aileron deflections are larger than would be obtained with the nominal gain schedules (0.8 of the roll rate command versus the nominal 0.1 times the roll rate command). In the roll maneuver sequence in the NO Y JET mode, the pilot desires a right roll, for example. For a right RHC command the initial aileron deflection is left; the vehicle initially rolls left. However, as left β increases, the roll direction will reverse to the right. Then left β will decrease to near zero (assuming no Y c.g. offset) as the normal function of β damping is assumed by the ailerons. The NO Y JET mode is intended only for use down to approximately Mach 5, where the nominal FCS configuration would be reselected. The aileron/rudder provide roll/yaw rate feedback stabilization in the same manner as the nominal system although β damping is not as tight without the RCS. At low dynamic pressure, damping is poor.

17.6 BFS VARIATIONS FROM THE PRIMARY CONTROL SYSTEM

The entry BFS control system will not support the AUTO mode, only the CSS mode (with downmoding capability) in pitch, roll, and yaw channels. There is no automatic nosewheel steering. Inputs to the ADI pitch and roll error needles are discontinued after TAEM.

The BFS provides for pre-engage tracking of the primary FCS. It also provides for takeover-engage initialization with acceptable transients. The D DAP combines the GRTLs and AERO JET functions into a single software unit. Unified AERO_PRL and JET SELECT software are provided in backup.

17.6.1 BFS Variations From the Primary JSL

The BFS combines the JSL for transition, GRTLs, and descent. Another major difference is that BFS is single-fault tolerant to jet failures, whereas the primary avionics software system (PASS) is generally two-fault tolerant. BFS chooses individual jets within a jet group in a different priority order than does PASS. The BFS order is based on maximizing a commanded moment or minimizing crosscoupling.

17.7 WRAP DAP

The WRAP DAP was established as the nominal flight control mode starting with STS-86 in 1997. The Aerojet mode was tested on STS-79 through 83. STS-84 performed the first WRAP DAP maneuver which consisted of the first roll reversal at $M = 21.6$ and dynamic pressure = 40 psf.

For nominal entry, the WRAP DAP is enabled at MM 304 when the Aerojet DAP becomes active. The baseline Aerojet DAP is active until $q\text{-bar} = 10$ when the WRAP DAP becomes active if it is still enabled. Performing an ITEM 45 on SPEC 51 disables the WRAP DAP or returns to the baseline DAP. For a TAL, the WRAP DAP is not automatically enabled and will not activate unless the crew enables it with the item entry. Normally, the WRAP DAP is only enabled for a TAL if required.

The WRAP DAP mode of the Aerojet DAP provides more lateral directional control authority than the baseline DAP. The WRAP DAP is often perceived to have less control margin than that baseline due to wider attitude deadbands, but this is not the case. During its development, the aileron trim limit analysis was revisited with the latest uncertainties. This allowed the WRAP DAP aileron trim limit to be increased from 3° in the baseline to 5° . (The baseline DAP was not changed despite the new analysis findings.) The aileron trim integrator is also four times faster than the baseline. The WRAP DAP has the same GALR as No-Yaw-Jet, which is seven times that of the baseline for $M > 4$. This decreases RCS propellant usage by increasing use of the ailerons and is the reason the WRAP DAP is nominally preferred over the baseline.

The WRAP DAP will not track guidance commands as close as the baseline due to the aileron control reversal inherent in Orbiter control design for $M > 4$. The early entry shuttle flight control system uses an adverse aileron scheme to control the Roll axis. This scheme is used in all modes of the Aerojet DAP but is especially apparent to the crew in the WRAP DAP. For more information about the adverse aileron scheme, reference section 17.8.2.2.1 (Adverse Aileron Scheme) below.

The WRAP DAP also reduces the likelihood of firing RCS jets by lowering the gain on the bank error signal and increasing jet deadbands. Thus, the WRAP DAP complies with guidance command but with a longer response time than the baseline. The WRAP DAP is a compromise between crisp tracking of guidance commands (baseline) and RCS prop savings (no-yaw jet). It is important to understand this relaxed command tracking does not correspond to decreased control authority.

Table 17-2 summarizes the differences between WRAP DAP and baseline for high Mach:

Table 17-2. Comparison between WRAP and Baseline DAPs (high Mach).

| Parameter | WRAP | Baseline |
|-----------------------------------|----------|-----------|
| Small signal response | Aileron | RCS jets |
| Large signal response | RCS jets | RCS jets |
| RCS jet deadbands | 1.35° | 0.35° |
| Roll error gain to Aileron (GALR) | 0.7 | 0.1 |
| Aileron trim limit | 5° | 3° |
| Aileron trim rate limit | 4.2°/min | 1.05°/min |

Even though the WRAP DAP has become the nominal entry DAP mode, during its implementation it was decided to not change the No-Yet-Jet Quantity Flight Rule. An AFT RCS propellant quantity of 10% was retained to protect for possible control problem cases.

Since the WRAP DAP provides increased lateral control, it should be used for offset c.g., control problems, and loss of 2+ jets cases. For example, if during a TAL abort the aileron trim reaches a steady ~1.5°, enabling WRAP DAP will provide increased aileron capability. In addition, the high GALR makes the DAP effective for low aft propellant quantity cases. The WRAP DAP is also less sensitive to density shears and asymmetric boundary layer transitions than the baseline.

The only known case which the baseline DAP consistently out performs WRAP DAP is for a stuck rudder/speed brake at M>4. The high GALR term of the WRAP DAP causes jaw jet firings to be delayed which could be detrimental in a stuck rudder case. The baseline DAP must be selected to allow for earlier jet control firings.

17.8 TRIM SWITCHES

Trimming attempts to create a set of control vectors that will counteract conditions of lateral c.g. offset, asymmetric aerodynamics or a bent airframe. In order to use the orbiter trim switches, the following switches on the corresponding CDR or PLT Trim Panel must be properly configured. The CDR switches are located on the left end of panel F3 and the PLT switches are on the right end.

PANEL ON/OFF - This switch provides power to all trim switches except the RHC beep trim which receives power through the controller.

RHC/PANEL ENABLE/INHIBIT - This switch enables or inhibits all trim switches including the RHC beep trim. The switch is a 2 contact switch. In the event of a switch dilemma, trim switches will be enabled. The CDR switch is powered by CNTL AB1 and BC1. The PLT switch is powered by CNTL BC2 and CA2.

17.8.1 RHC Beep Trim

The simplest trim switch available is RHC Beep trim which relieves longitudinal stick forces by commanding a constant rotation rate about the Orbiter body pitch or roll. When the RHC is out of detent, a rate is being commanded. The beep trim lets the crew command an equivalent rate of a stick deflection past detent by digitally biasing the RHC output with the beep trim (it trims out forces that the crew must otherwise hold). RHC beep trim is only available when flying CSS. Since the beep trim only creates a rate command bias, it does not directly affect aerosurface positions.

The RHC Beep trim is a two-contact "hat" shaped knob on the RHC which is deflected in the direction of desired trim. The amount of commanded rotation is proportional to the length of time the switch is actuated. When the switch is depressed, 0.6 deg/s^2 acceleration is commanded. The maximum pitch trim rate allowed is 1.5 deg/s which is reached after ~ 2.5 seconds of trim deflection. This limit is an I-loaded parameter: DETR_LIM (V97U0643C). The maximum roll trim rate allowed is 2.5 deg/s which is also an I-loaded value: DATLIM (V97U0615C). The RHC pitch trim command (DETM_RHC), however, is sampled at 6.25 Hz which can result in missed pilot beep trim inputs as often as half the time.

Since the Aerojet DAP pitch channel is relatively free of interaction with roll and yaw, pitch beep trim can effectively be used to trim out any pitch rate gyro bias that could pass through the DAP selection filtering process. This enables the use of pitch RHC beep trim to be used for derotation upon landing. Derotating with the trim switch standardizes derotation, reduces main gear induced spring oscillations (bounces) and peak loading, and allows for faster derotation initiation speeds to reduce the time spent in a 2 point stance. (During the 2 pt stance, lateral control is not good and the vehicle is susceptible to crosswind gusts and disturbances). For more information about derotation, see the Controllers System Brief: 13.2.2.4.3 Beep Trim Derotation Logic.

Beep trim could be an effective way to trim for a pitch RGA bias and maybe for a roll bias as well. Along with the sampling rate, the operational problem with beep trim is that there is no way to monitor the exact value of the integrators that compute the beep rate command. Therefore, once a rate is trimmed into the system, there is no way to be sure the command has later been removed with subsequent commands with the Beep trim. To ensure a trim command has been removed, the crew can transition from CSS to AUTO and then back to CSS. This causes the integrators to reinitialize to zero.

17.8.2 Panel Trim Switches

The panel trim switches are 6 momentary switches (3 for CDR and 3 for PLT) which increase their respective trim with each deflection. The CDR switches are located on L2 (CNTL AB1 and BC1) and the PLT switches are on C3 (CNTL BC2 and CA2). Each seat also has a manual body flap control switch which is also a momentary switch located on the same panels. These switches are not limited like beep trim, and therefore can be used to move the aerosurfaces through their entire range of motion.

If both crew stations generate conflicting trim commands in an axis (e.g., if the commander generates a positive pitch trim and the pilot generates a negative pitch trim simultaneously), the resultant trim command in that axis is zero.

17.8.2.1 Pitch Panel Trim

The pitch panel trim directly provides command input to the elevon command. In doing so, the panel trim effectively bypasses the pitch channel's inner loop damping function. As such, there is no known effective use of the pitch panel trim switch.

In order to use the pitch panel trim, the body flap must be in manual or the body flap will cause the elevons to move right back to their scheduled position when the panel trim switch is released.

The elevator pitch trim limit is $+18^\circ$ and -33° which covers the entire range of the elevons. This limit is enforced by the pitch channel software.

17.8.2.2 Roll/Aileron Panel Trim

The roll panel trim command goes directly to the aileron trim integrator and physically repositions the ailerons since the panel trim gain is high enough to override all automatic aileron trim (or speed up the trim process as required). This however is a continuous task. The crew cannot turn off AUTO trim - they can only overpower it with the panel trim. The Aileron trim integrator has a limit of 3° in the Aerojet Baseline mode and 5° in the WRAP DAP mode. This limit will be reached after ~5 seconds of trim panel deflection.

Aileron trim logic is driven by the same yaw error signal that fires yaw jets. Thus, right jet firing adds more left aileron causing the aileron to drive down towards saturation. As aileron trim increases, the requirement for jets decreases but the vehicle sideslip (beta) increases.

To gain an understanding of aileron trim, consider the following discussion adapted from the workbook: *Entry Flight Training for Commander and Pilot*. Before Mach 5, when the rudder becomes effective, the only lateral trim methods available are beta, aileron and jets. During this time if aileron trim becomes saturated, nothing can be done to decrease jet activity. If the crew trims the ailerons opposite of the firing jet they will only change the vehicle beta. In this situation, the crew might as well drive the aileron trim towards zero since a trim reversal is imminent. As a general rule, aileron trim should be away from a firing jet as long as the action decreases jet activity. If trimming away increases jet activity, the ailerons should be trimmed in the opposite direction.

17.8.2.2.1 Adverse Aileron Scheme

The early entry shuttle flight control system uses an adverse aileron scheme to control the Roll axis. This scheme is used in all modes of the Aerojet DAP but is especially apparent to the crew in the WRAP DAP. In the early entry regime of flight, the roll axis is dominated by the

side-slip angle, beta. When the ailerons on the right of the vehicle are moved up and the ailerons on the left are moved down, as would be done on an airplane to cause a roll to the right. However, the end result of the shuttle's response during the early part of an entry, is quite the opposite. The increased drag on the left of the vehicle (caused by dropping the left elevons) will force the vehicle to yaw to the left, creating a sideslip angle (beta).

As expected, the orientation of the ailerons, does induce a roll moment to the right, however, as mentioned, the roll axis is dominated by the sideslip angle. The left Yaw, exposes the right side of the vehicle to the air stream, which in-turn generates a roll to the left. This roll moment is the dominant force in the roll axis during the early entry flight control system. These effects from the sideslip angle are therefore used in the early aerojet DAP to govern the roll axis. WRAP DAP uses both the ailerons and the yaw jets to generate the desired beta angle. The No-Yaw-Jet DAP mode depends solely on the ailerons to generate the desired beta angle, until the rudder becomes active at Mach 5. The baseline DAP uses the RCS jets more heavily (earlier) than WRAP DAP and uses the elevons less aggressively (due to lower gains). The WRAP DAP will also allow the Aileron Trim integrator to trim the system out quicker and to a greater total magnitude than the baseline Aerojet DAP. This explanation illustrates that the roll axis is very much coupled with the yaw axis.

The Aileron trim integrator is used during the early entry flight control system to trim out beta, such that the desired rotation about the roll axis (or lack there of) is maintained by offsetting beta as needed. During the early entry flight regime, adjustments to the aileron trim integrator are made as a function of the commands coming from the yaw axis.

As mach numbers begin to climb down, the shuttle's flight control system starts to transition into that of a more conventional airplane. The nose of the vehicle comes down as the angle of attack is reduced, exposing the vertical stabilizer to the air-stream and making the rudder effective at M=5. The adverse aileron scheme is phased out, and the ailerons are used to control the roll axis more directly.

17.8.2.2.2 Aileron Trim Gains

The gains in the aileron auto trim rate for CSS and Auto begin to diverge at a Mach of 1.26 (Figure 1.10). By Mach 1.25, the gain going to the aileron auto trim in CSS mode is reduced to zero. Simultaneously, the gain in the AUTO mode is increased from 0.1 to 0.25. Reducing the gain to zero while in CSS mode, keeps the AUTO flight control system from adjusting the aileron trim while the crew is flying the vehicle (and Mach<1.25). The reasoning behind this design was to take the AUTO system out of the loop and handover control to the crew.

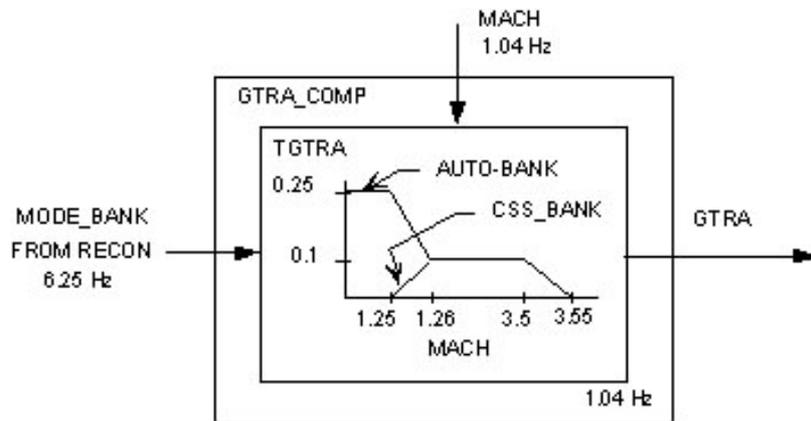


Figure 7.10 Aileron trim integrator roll error gain.

According to the original designers of the flight control system, the astronauts wanted the aileron trim gains to be different in AUTO and CSS. Remember that the Shuttle was the first fly-by-wire aircraft and pilots were not accustomed to aircraft automatically adjusting the vehicle trim. Therefore, as the Shuttle approached the landing site where the astronauts were expected to take over manual control, they wanted the flight control system to be inhibited from automatically adjusting the aileron trim - hence a zero gain on the trim input.

17.8.2.2.1 Aileron Trim Bias

If CSS is selected at the wrong time, the aileron trim will be latched and can possibly create a roll bias.

The aileron trim integrator (ATRIM) produces a limited aileron trim angle command by integrating the sum of the auto aileron trim rate and the derived panel roll rim rate. When a roll command is generated, there is initially a roll rate error, which the flight control system then responds to. The roll rate error feeds into the preliminary aileron command, which is scaled by the gain (Figure 7.10) to generate an auto trim rate. The auto trim rate is then fed to the aileron trim integrator, which creates an aileron trim angle command.

If CSS is selected below Mach 1.25 shortly after a roll command is executed in AUTO, the gain on the aileron AUTO trim rate jumps down to zero, latching the aileron trim integrator at its last value. Since the aileron trim integrator was in the process of compensating for the roll command that had been initiated in AUTO, selecting CSS will latch the aileron trim integrator in a transient state possibly creating a roll bias. This bias occurred during the STS-13 HAC when the commander took CSS after the roll had been started in AUTO.

There are a few techniques that can be used to compensate for this incurred roll bias. The simplest is to reselect the AUTO and allow the integrator to settle back to zero. The length of time required for the integrator to clear out has not been tested, but is believed to be between 10 and 15 seconds.

The second method available is to enable the Trim Switches and compensate using the Roll Panel Trim Switch. Recall that the output of the Roll Panel Trim Switch is summed with that of the AUTO aileron trim and then sent to the integrator. The complications associated with this method are; (1) Added crew workload. The crew must enable the trim switches, and then trim out the bias using the Roll Panel Trim switch while flying the vehicle. And (2), if the Roll Panel Trim switch is moved in the wrong direction, the added bias can further saturate the RHC's ability to control Roll.

The last option available is to use the RHC "Beep" trim. The RHC "Beep" trim can be used to enter a constant bias command from the RHC. This would also involve enabling the trim switches, and correcting for the bias while flying the vehicle. Again there is the risk of applying the trim in the wrong direction, further saturating the RHC. The effectivity of the RHC beep trim integrator is very small and may not be sufficient to correct for a significant amount of aileron trim integrator command.

17.8.2.3 Yaw Panel Trim

Unlike the roll panel trim which goes directly to the trim integrator, the yaw (rudder) panel trim provides a bias to the Ny accelerometer input in the yaw flight control channel. This bias is limited to 0.1g's. Therefore, it is more continually used to null a yaw accelerometer bias than effectively provide yaw trim. The ENTRY TRAJ and VERTICAL SIT displays show both Ny and Ny trim but there is no way on board to distinguish whether the Ny reading is real or a bias. Thus, ground support is required.

The yaw panel trim switch could also be used to offset a lateral AA bias of a maximum of 0.1 g. The yaw panel trim could be used in the case of a bent airframe to cause an acceleration bias and counter-act the effects. If this is not enough, the rudder could also be used to increase lateral acceleration.

The roll/yaw channel limits the Ny acceleration to ± 0.1 g (± 3.2 fps) and hence, the yaw panel trim is also limited. This limit is why the panel trim switch can only offset a lateral AA bias of 0.1 g.

17.8.2.4 Body Flap Manual Command

The body flap manual command can be used to truly affect the position of the elevons. Moving the body flap up creates a nose-up pitching moment which causes the elevons to automatically go down. When the body flap is returned to AUTO, it will chase the elevons back toward their scheduled position. To use this trim switch, the body flap must be in manual control. Since the body flap manual command switch is not technically a trim switch, it is not affected by the F3 Trim Panel power and inhibit/enable switches.

The body flap switches are on panels L2 and C3 and can be used when the body flap control mode is manual. To place the BF under manual control, either of the BODY FLAP PBIs can be depressed. The AUTO will extinguish and both Manual (MAN) lights will illuminate. To go back to automatic software control, either of the BODY FLAP PBIs can be depressed. The BF

control mode will then change from manual to auto, the MAN lights will extinguish and both AUTO PBIs will illuminate. If both crew stations generate conflicting UP/DOWN commands, a body flap UP command will be output to flight control since UP has priority.

References:

Aerojet DAP Roll/Yaw Channel drawing 1.11.2.

Aerojet DAP Pitch Channel drawing 1.11.1.

Controllers Workbook, CONT 21002

Entry Flight Training for Commander and Pilot. August 1985

Hochstein, A. *A&L Class Overview*. April 1999

McWhorter, Larry B. *Space Shuttle Entry Autopilot*. NASA Technical Memorandum. June 1992.

SECTION 18 CREW OPTICAL ALIGNMENT SIGHT

18.1 GENERAL

The crew optical alignment sight (COAS) (fig. 18-1) has two principal functions. It provides optical alignment capability for rendezvous and docking, and it provides redundancy to the two star trackers for aligning the inertial measurement units (IMU's). As noted in the star tracker brief, the star tracker software will not accept stars for IMU alignments if the stars are more than 1.4° from where the general purpose computer (GPC) thinks it should be. Should the IMU's become misaligned by more than this limit, the COAS serves as the alternate method of star tracking. Like the trackers, the COAS is used to establish a line-of-sight (LOS) vector to a star, thus enabling determination of the IMU orientation with respect to the stars. It can be used as the method of coarse alignment with the star trackers completing the job with a fine alignment or it can be used stand alone in the case of a failure of both star trackers.

An orbit calibration is required before the device can be relied upon as a backup to the trackers. No calibration is required for its use as a rendezvous/docking aid.

18.2 HARDWARE DESCRIPTION

The COAS is a cylindrically shaped device about 8 inches long, with a combining glass at one end, a box-like attachment at the other end (containing a rheostat for light intensity control), and a mounting plate attached to the central portion of the cylinder (fig. 18-2).

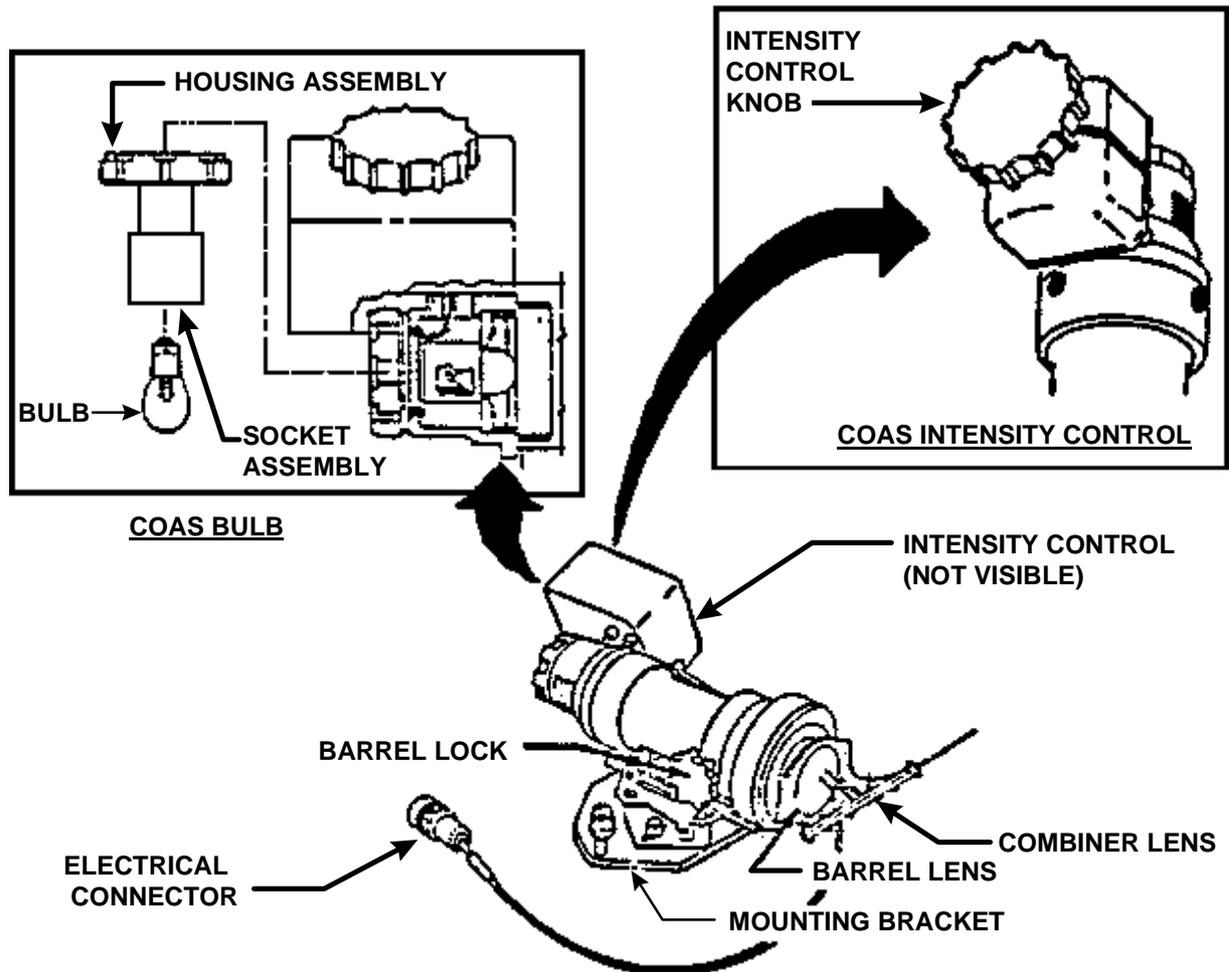


Figure 18-1. - Crew Optical Alignment Sight.

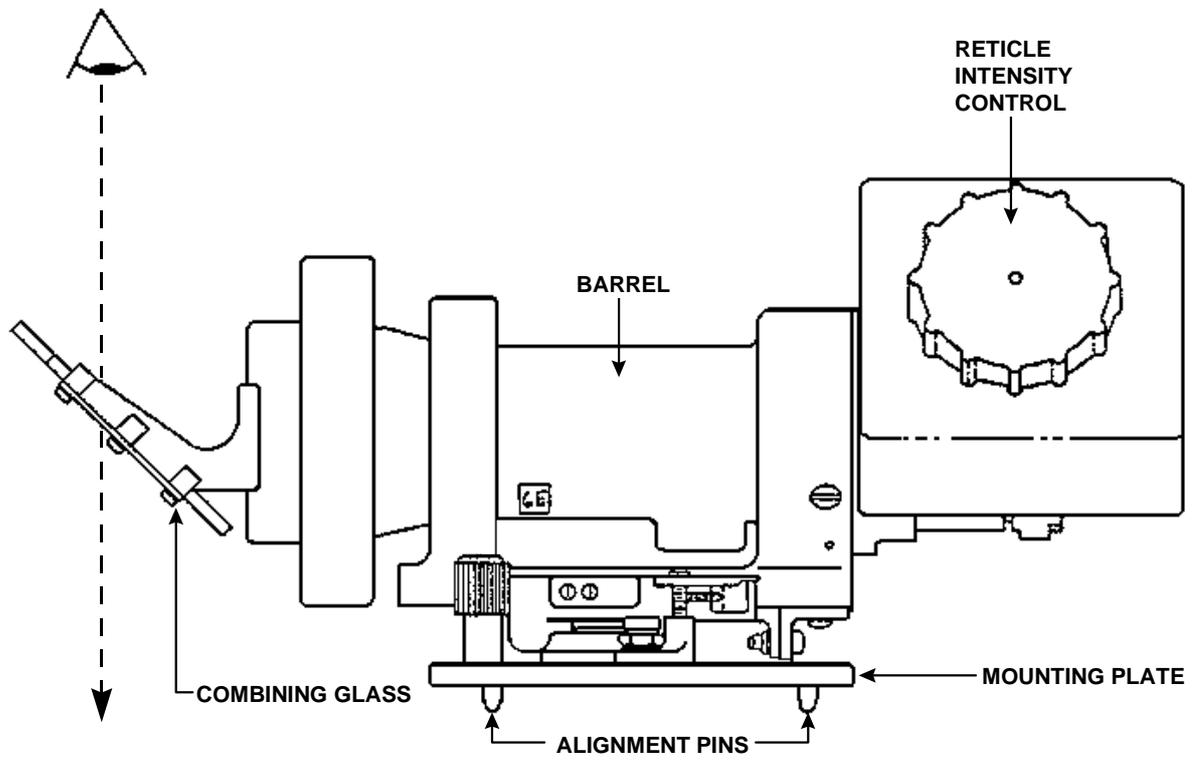


Figure 18-2.- COAS side view.

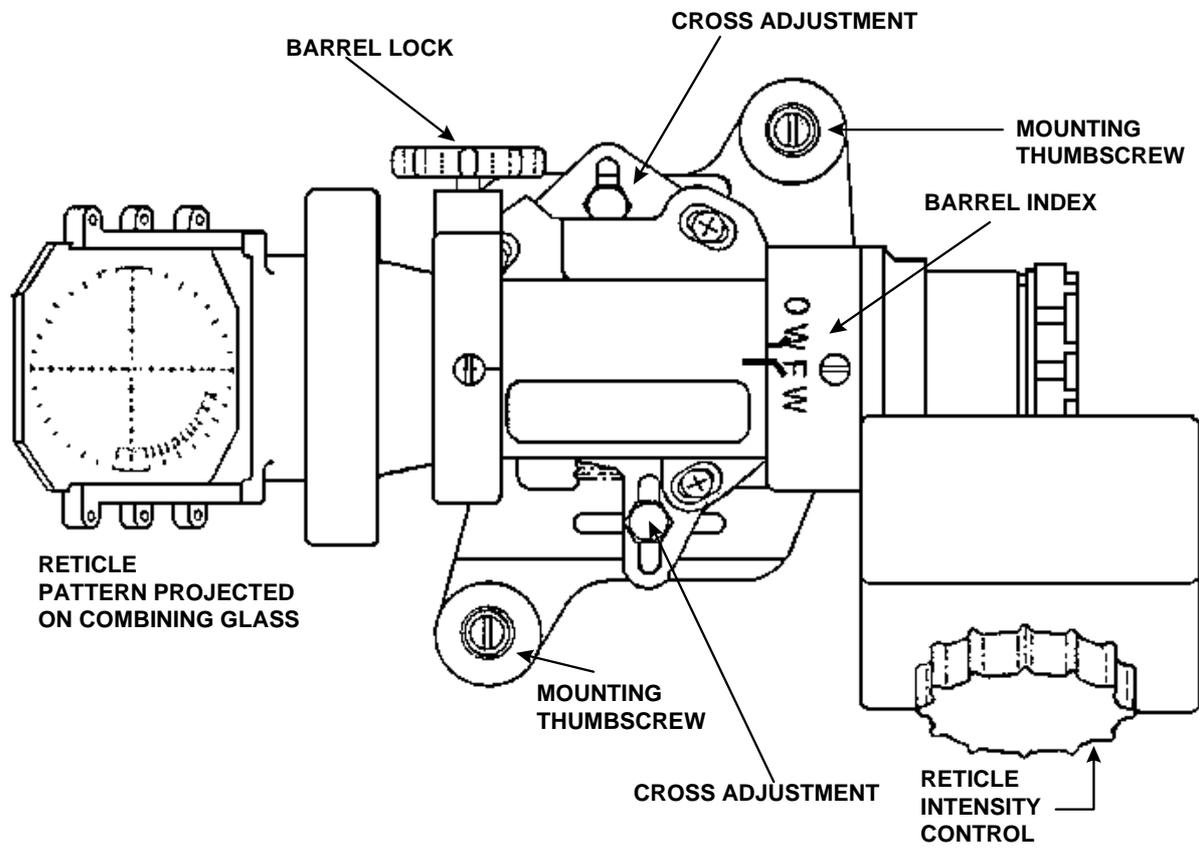


Figure 18-3.- COAS top view.

The COAS flown on the Shuttle is an Apollo modified unit. Modifications included changing the reticle dimmer and mounting such that repeated mounting and demounting cycles would have minimal effect on sighting accuracy/ repeatability. Alignment pins are located on the bottom of the mount. Phase self-centering alignment pins maintain the boresight aiming point after many mount and demount cycles. The LOS, as shown, is looking through the combining glass. This is used to project the reticle pattern onto the field of view (FOV) of the star or target. The knurled knob drives the rheostat, which controls the brightness of the reticle pattern.

Pictured on the combining glass is the reticle pattern as seen when looking through the combining glass at a star or target (fig. 18-3). Also shown are the two mounting thumbscrews used to lock the COAS into position.

The cross adjustments, which mount the COAS to the mounting plate, are preset prior to ground calibration and should not be moved.

In order that the COAS be properly aligned, it must be oriented for the position in which it is mounted. The barrel index allows selection of either forward window (FW) or overhead window (OW) orientation. This orientation is a rotation of the reticle glass and lower portion of the COAS relative to the upper portion and mounting plate. The rotation aligns the COAS boresight axis with the Shuttle body axis associated with the mounting position. The barrel lock fixes the barrel at the correct orientation.

18.3 MOUNTING

The COAS has two mounting positions: one above the commander (CDR) +X FW window (W3) and one in front of the starboard OW (W7) above the aft flight station (fig. 18-4 for COAS viewing directions). The +X mount is boresighted parallel to the +X axis and the -Z mount parallel to the -Z axis of the Shuttle.

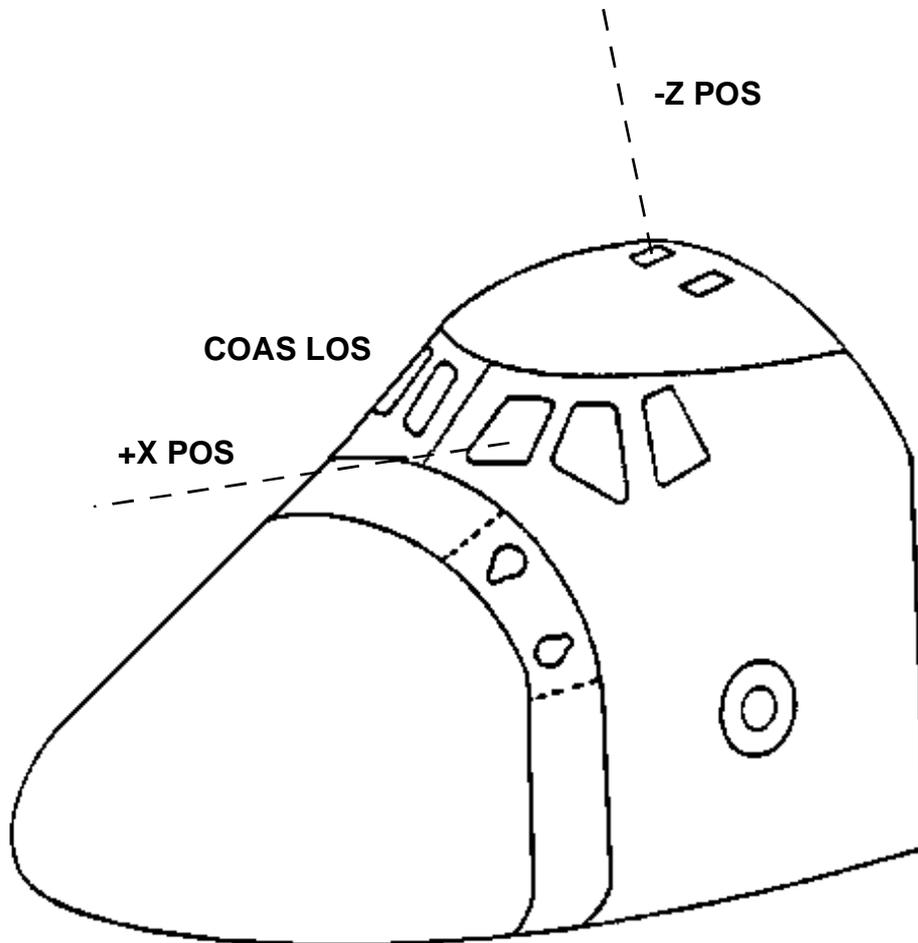


Figure 18-4.- COAS viewing directions.

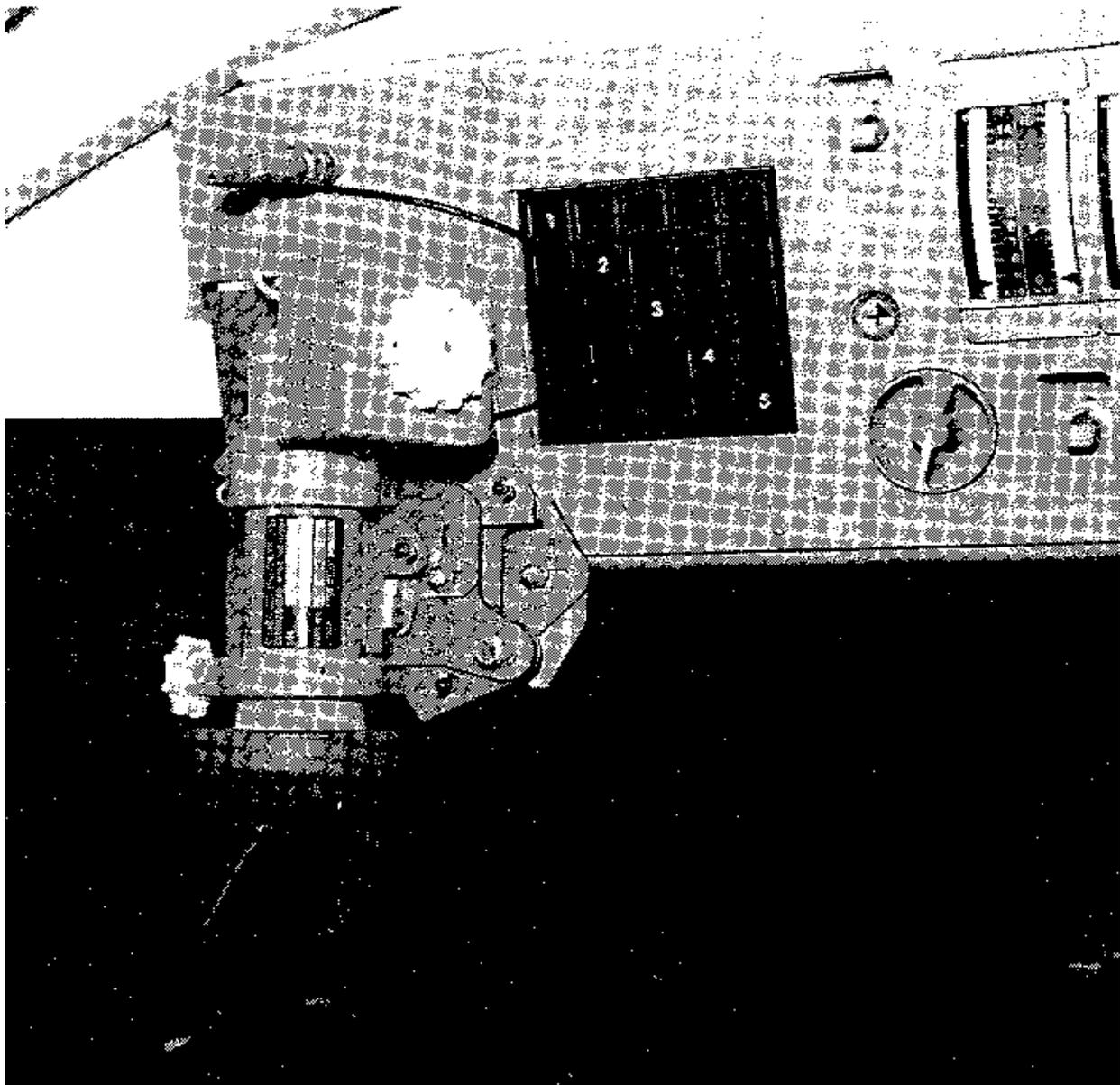


Figure 18-5.- COAS mounted in forward location.

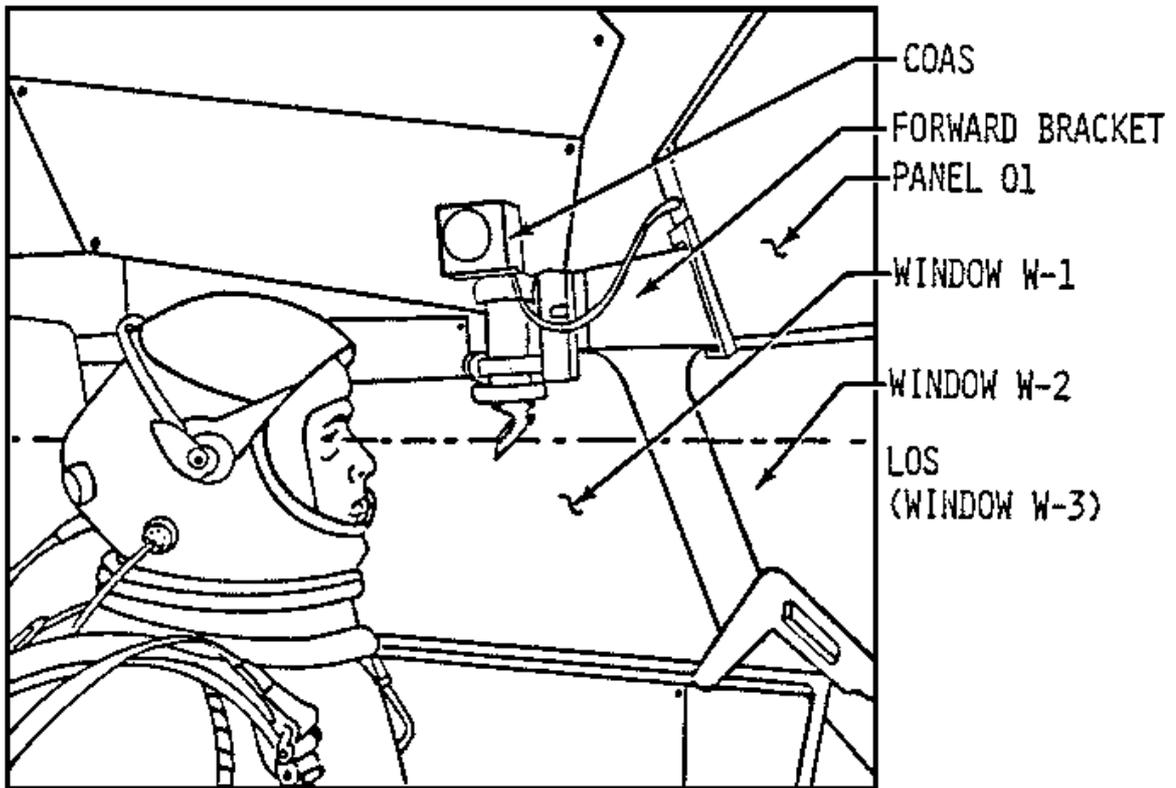


Figure 18-6.- Using COAS in forward position.

18.3.1 Forward Mounting

When taking sightings along the +X axis (figs. 18-5, 18-6), the COAS is mounted on panel 01 and sighted through window W3. With the COAS mounted on panel 01, power to illuminate the reticle is provided from the COAS POWER switch on panel 01. Reticle brightness is controlled by the intensity control on the COAS. A mounting bracket is attached to the COAS prior to mounting it in this position. This bracket is required to lower the unit so that the view through the FW is unobstructed.

18.3.2 Aft Mounting

A COAS mount near the forward edge of the starboard OW (W7) is the aft COAS location for the star sighting or rendezvous. Sightings from this position are made in the -Z direction of the Shuttle body coordinate system (figs. 18-7 and 18-8). COAS power at the aft mount is controlled by a COAS POWER switch on panel 019. The mounting bracket used at the +X position is not required here and should not be attached.



Figure 18-7.- Using COAS in aft position.

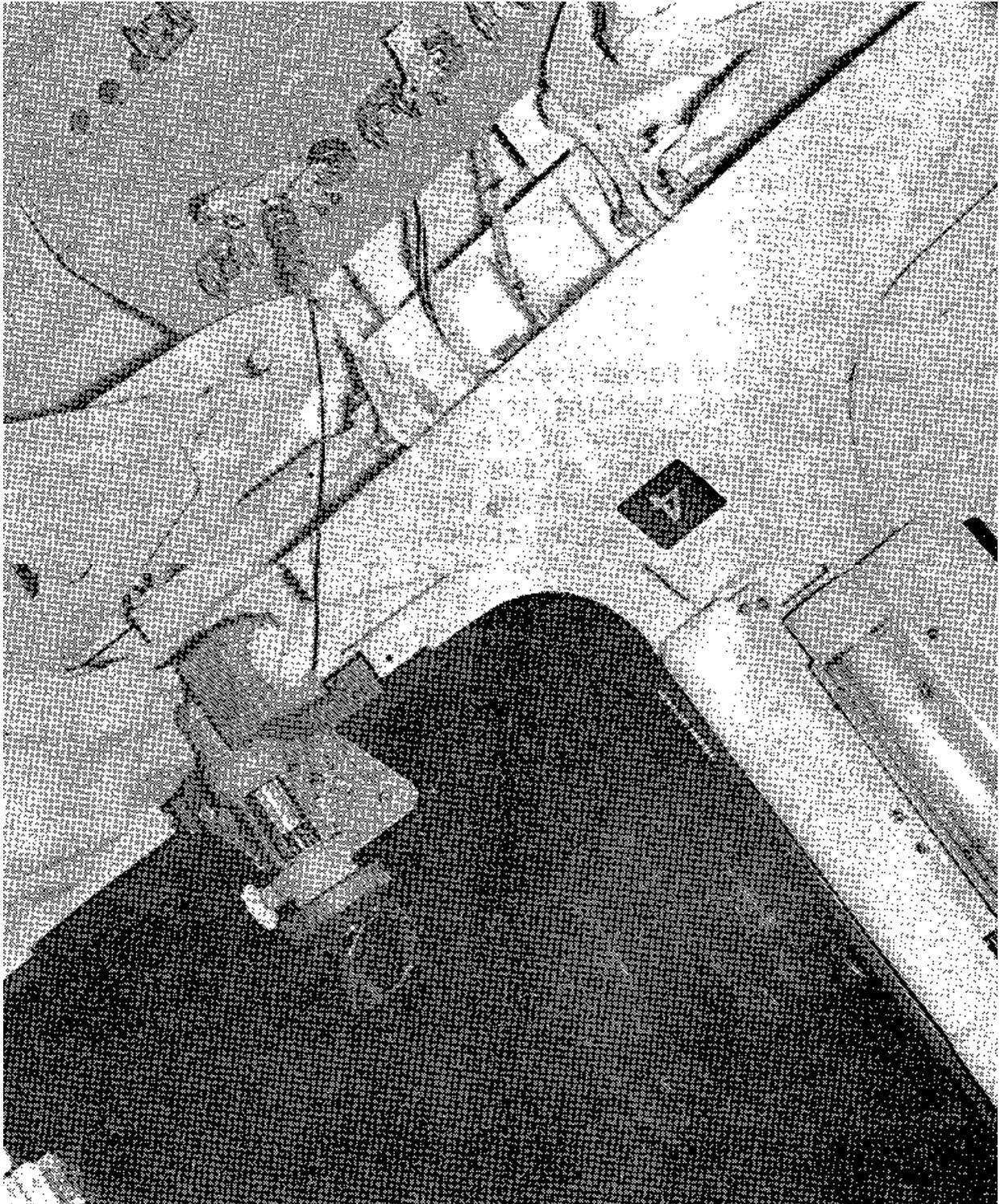


Figure 18-8.- COAS mounted in aft location.

18.4 APERTURE STOP AND FILTER

An aperture stop that reduces the FOV is provided to assist in "sighting on" a star. A filter that facilitates daylight sightings is also provided. The aperture stop and filter both clip on the COAS lens mounting bracket; therefore, they cannot be used simultaneously (figs. 18-9 and 18-10).

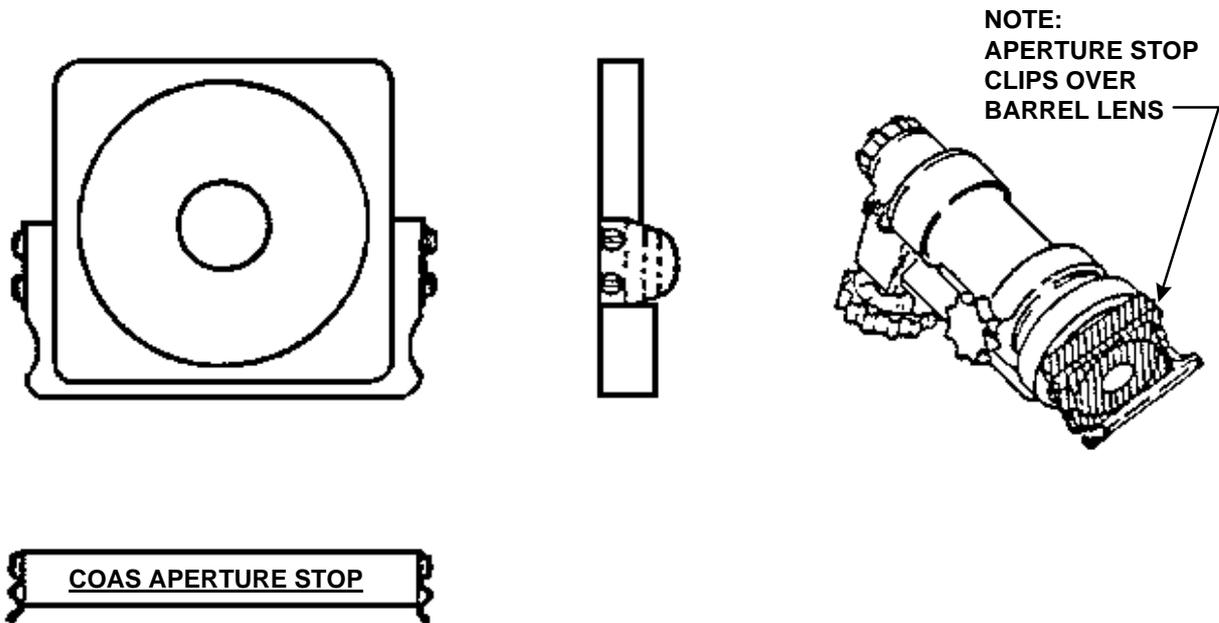


Figure 18-9.- COAS aperture stop installation.

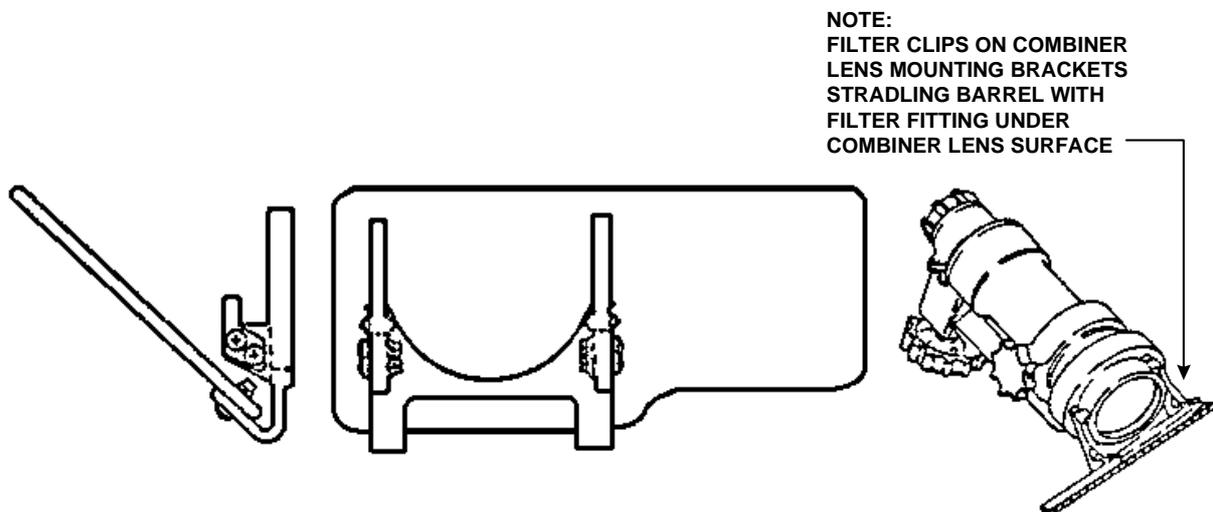


Figure 18-10.- COAS filter installation.

18.5 STOWAGE

Each Orbiter has two COAS's assigned to it, a prime unit and a backup. The prime COAS is always flown unless it has been damaged. For each flight, the prime unit is packaged at Johnson Space Center (JSC) (bldg 36) in a modular locker in a kit/container along with the mounting bracket, filters, spare bulbs, etc. After inspection, the locker is then shipped to Kennedy Space Center (KSC) and installed in middeck locker location MF57M several days prior to launch. After the flight, the locker is returned to JSC (figs. 18-11 to 18-14).

In flight, when not in use, the COAS is stored in its flight container.

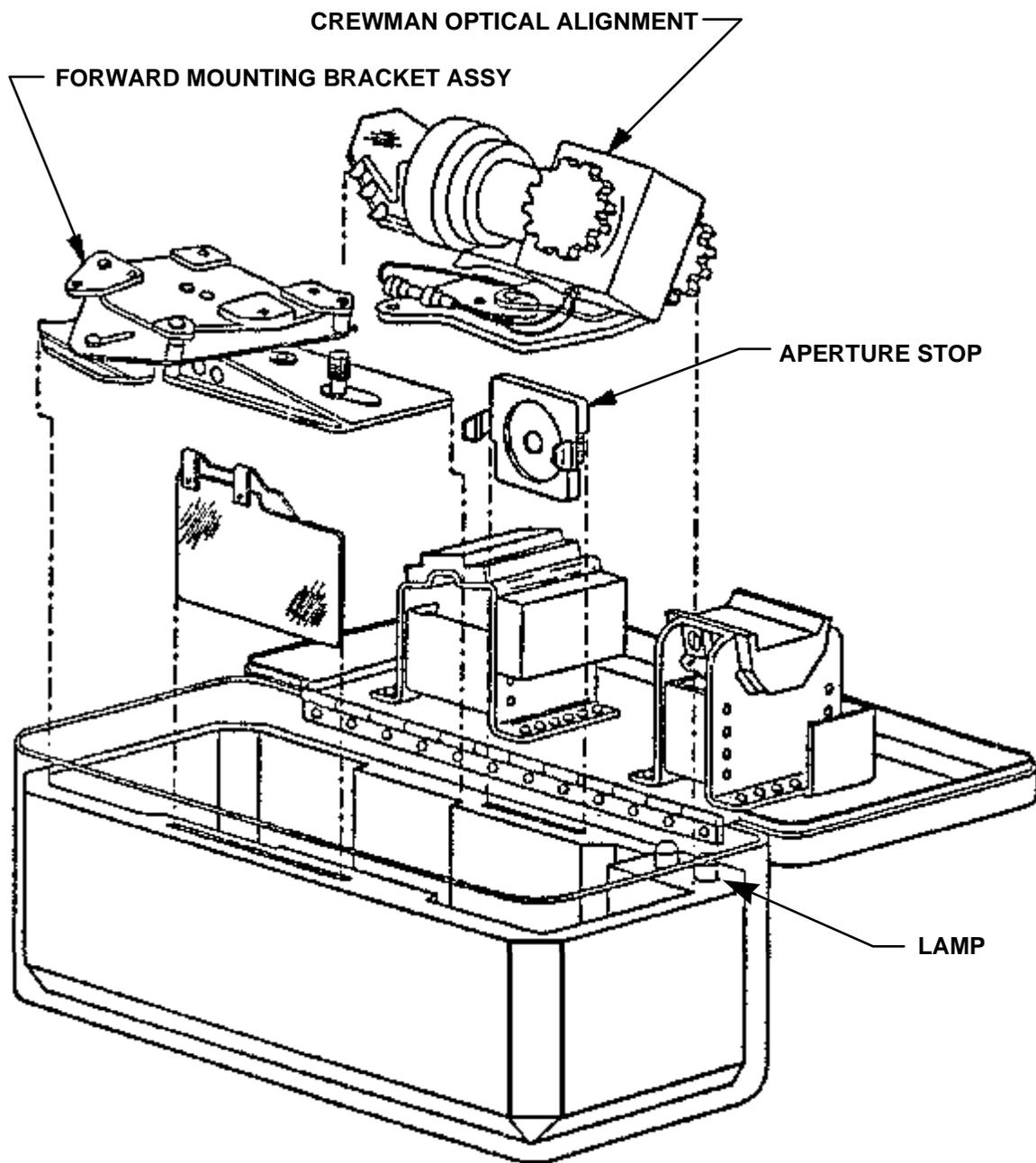


Figure 18-11.- In-flight handling container.

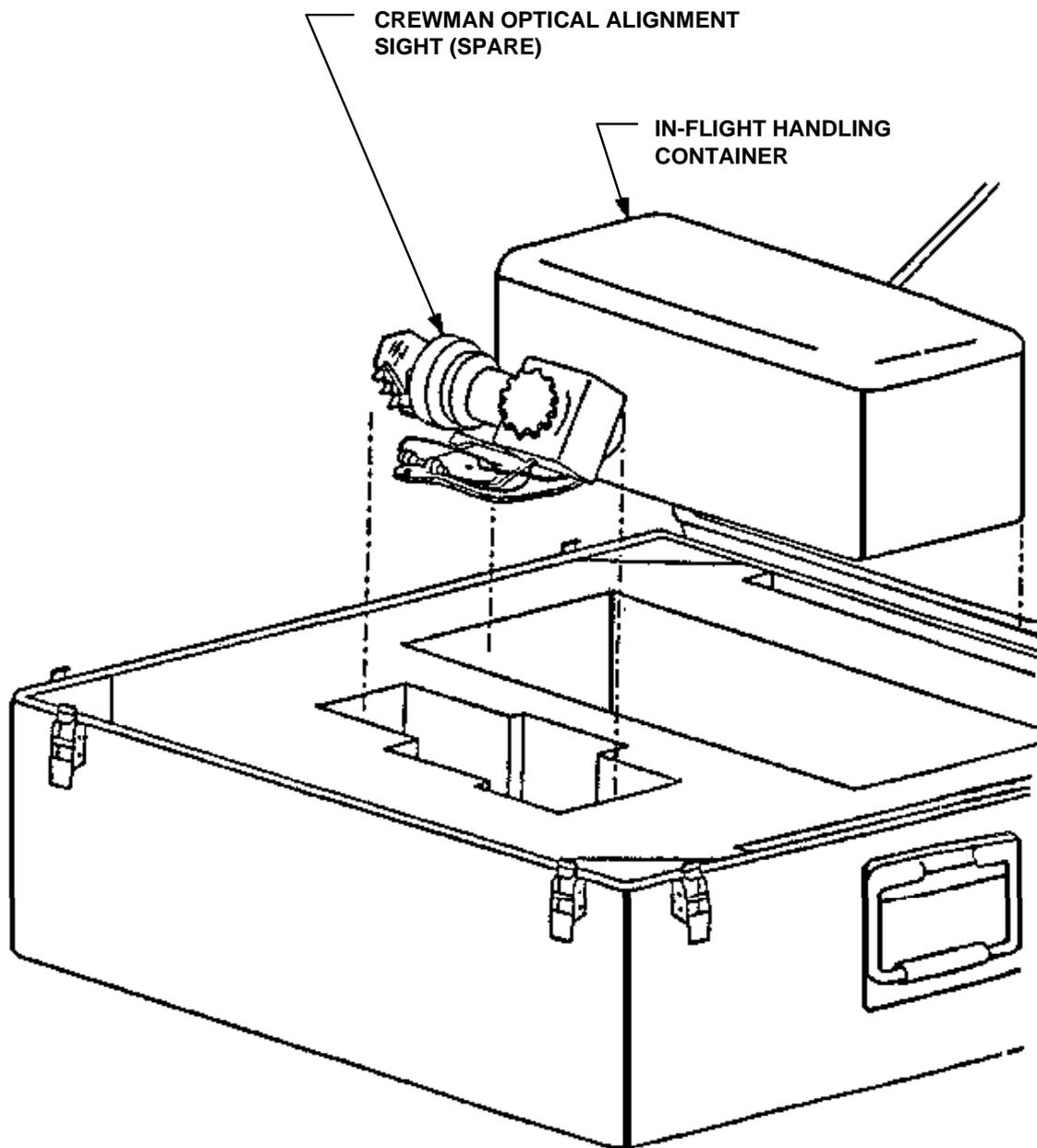


Figure 18-12.- COAS reusable container (shipment).

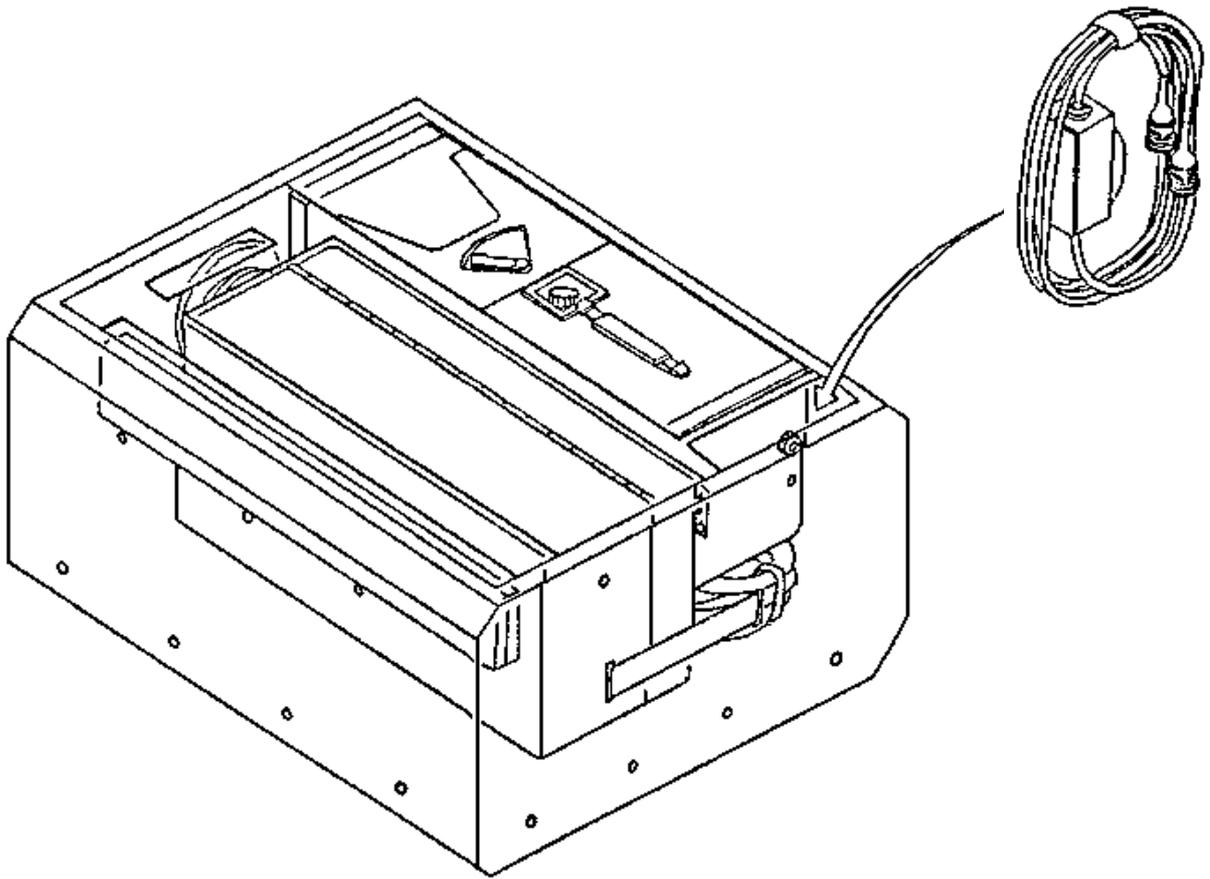


Figure 18-13.- Locker MF57M.

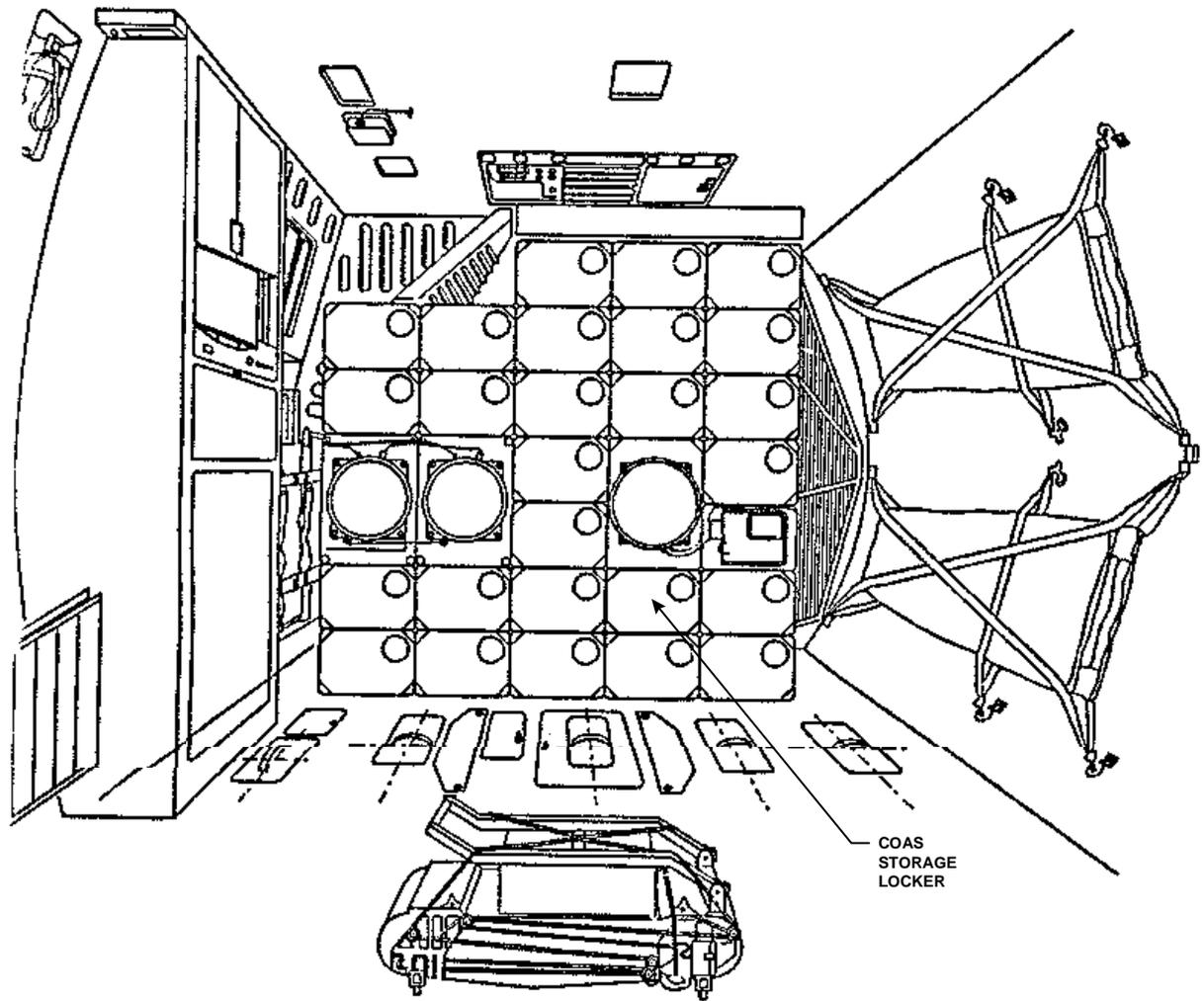


Figure 18-14.- COAS stowage locker.

18.6 STRUCTURAL ALIGNMENT (PRELAUNCH)

The Orbiter interface is via a flat plate of aluminum with thread (female) fasteners and indexing holes. The screws on the COAS mounting plate retract when not in use. Each Orbiter has one mounting bracket that allows the unit to be positioned at the forward (+X) station; the bracket is not necessary at the aft station (-Z). Prior to the maiden flight of each vehicle, the prime COAS, the backup, and the mounting bracket are adjusted at both the forward (+X) and aft (-Z) stations so that they are aligned properly to the vehicle coordinate system. The COAS LOS at both the forward and aft positions is then I-loaded into the flight software. The adjustments to the hardware (COAS and bracket) are intended to be permanent. The software LOS I-load(s) is procedurally fine-tuned by the calibration performed on each flight.

It should be noted that there have been recorded cases of unauthorized adjustment (tampering) on the flight COAS units (i.e., alteration of the forward mounting bracket on OV-099 (prior to 51-L) and adjustments on both the mounting bracket and the prime COAS on OV-104, post-61-B check). These tamperings were deliberate as the adjustment screws on the bracket are set with lockwashers and locknuts.

18.7 COORDINATE SYSTEM

The -Z star tracker is mounted such that the +V axis lies in approximately the same direction as the -Y Shuttle body coordinate axis, and the +H axis lies in approximately the same direction as the +X Shuttle body coordinate axis. These coordinates correspond to those displayed on specialist function (SPEC) 22 for -Z COAS use (fig. 18-15).

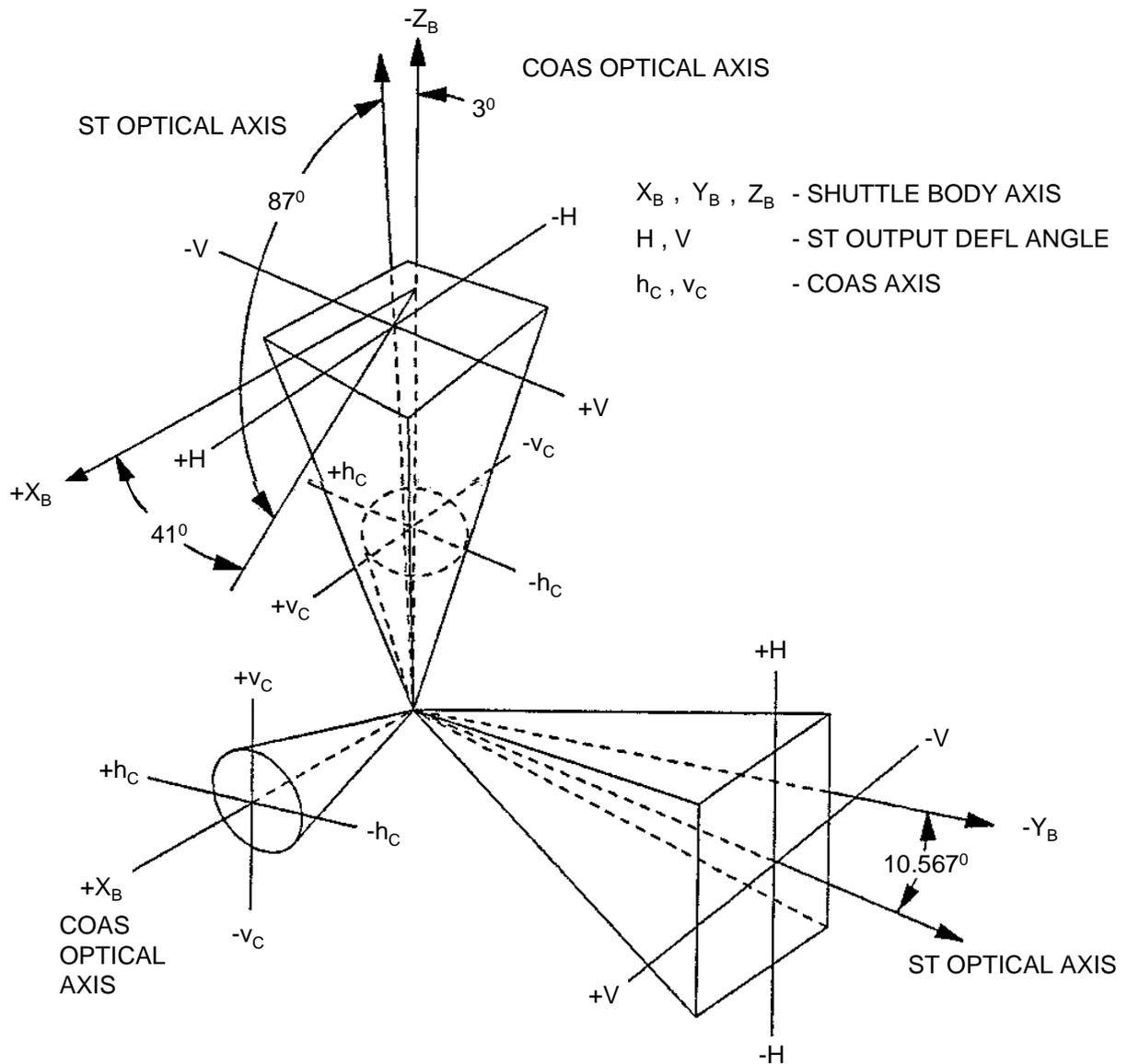


Figure 18-15.- Coordinate system.

18.8 RETICLE PATTERN

When the COAS is in use, the Orbiter is slowly maneuvered so that the star or target moves through the center of the cross on the reticle (fig. 18-16). The desired position of the star is exactly at the intersection of two perpendicular dashed lines. Once this is achieved, the attitude reference pushbutton (ATT REF pb) is depressed to indicate to the GPC that the star is centered in the reticle.

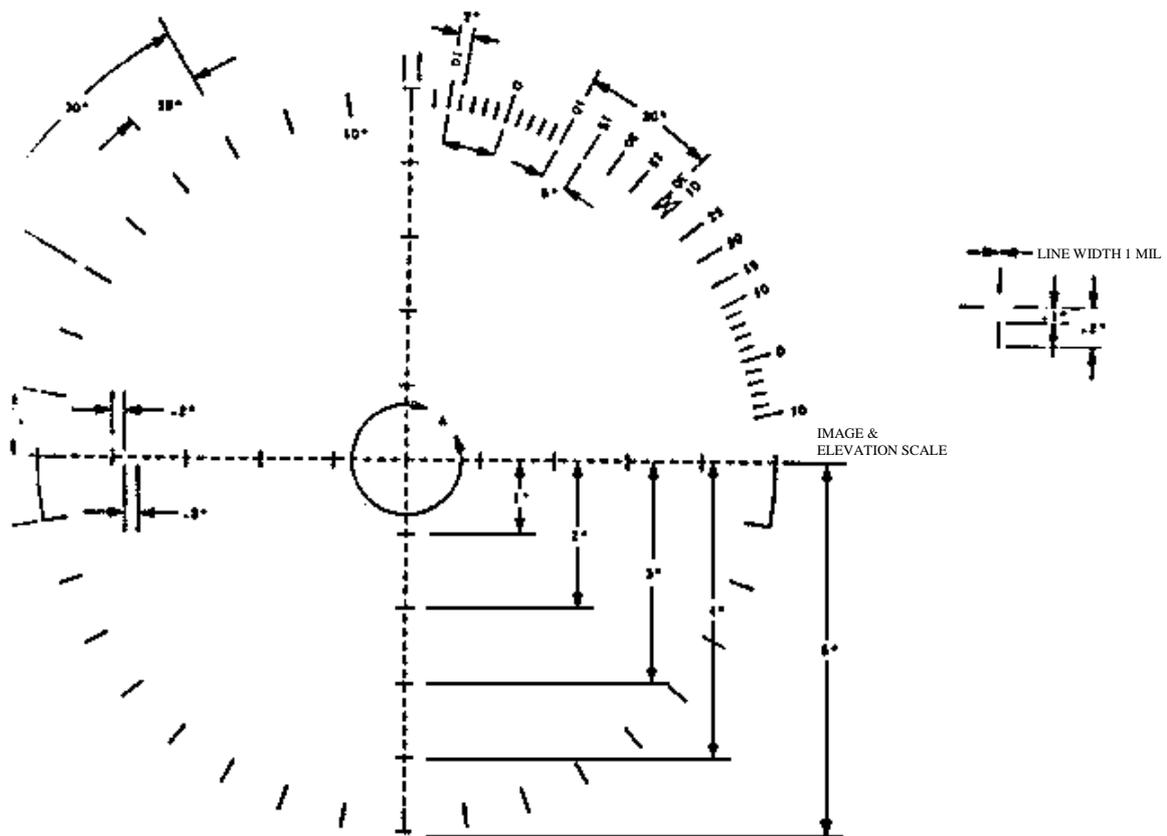


Figure 18-16.- COAS reticle pattern.

18.9 COAS POWER

Power for both the forward and aft COAS stations is supplied from circuit breaker CB 127 (COAS) powered by AC30A, located on panel L4. From this circuit breaker, power is routed to panels 01 and 019 (figs. 18-17 to 18-20). In the event of a COAS power failure, the end of the COAS barrel opposite the reticle can be unscrewed and a flashlight shone through the barrel to illuminate the reticle. The +X or -Z power can also be achieved via panel F1, AC utility. In

order to use the COAS AC utility power, the AC power transfer cable must be available. IFM will then have to instruct the crew on using the pin connector kit to make the power cable compatible with the utility power inlet.

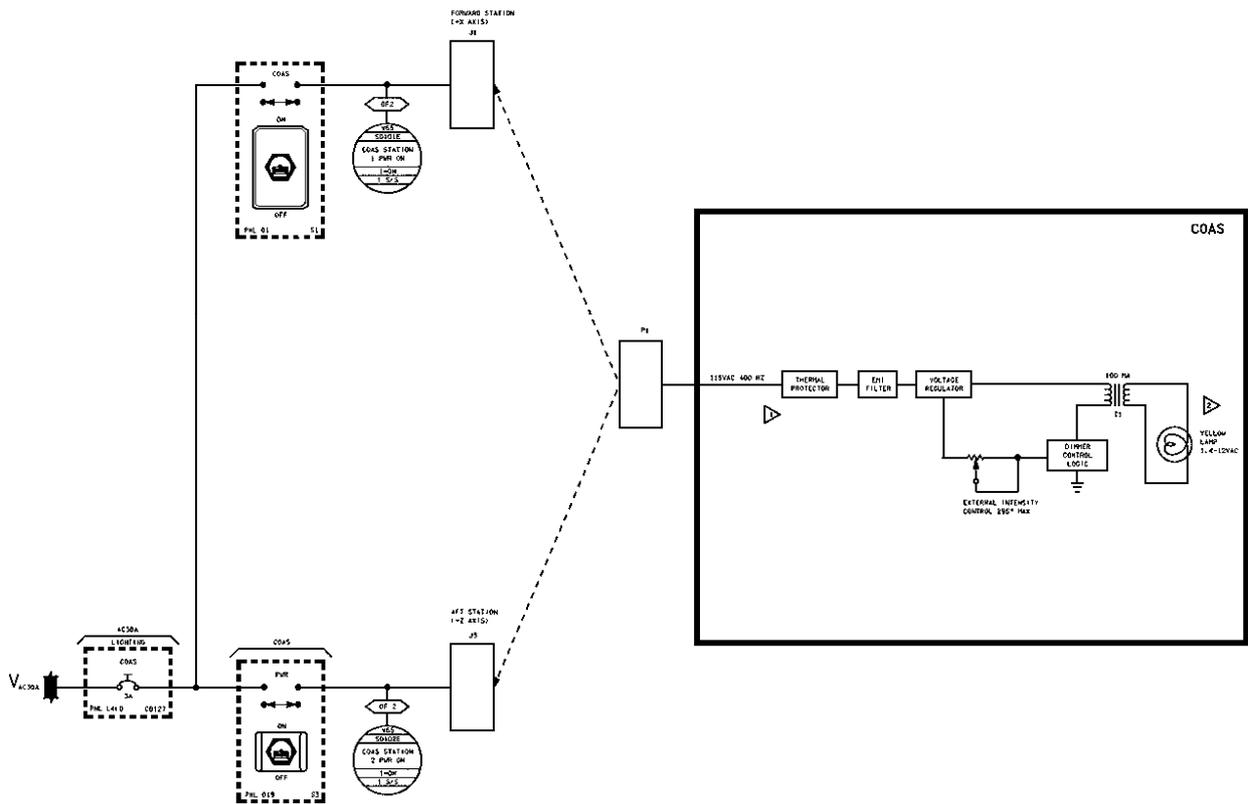


Figure 18-17.- COAS power distribution.

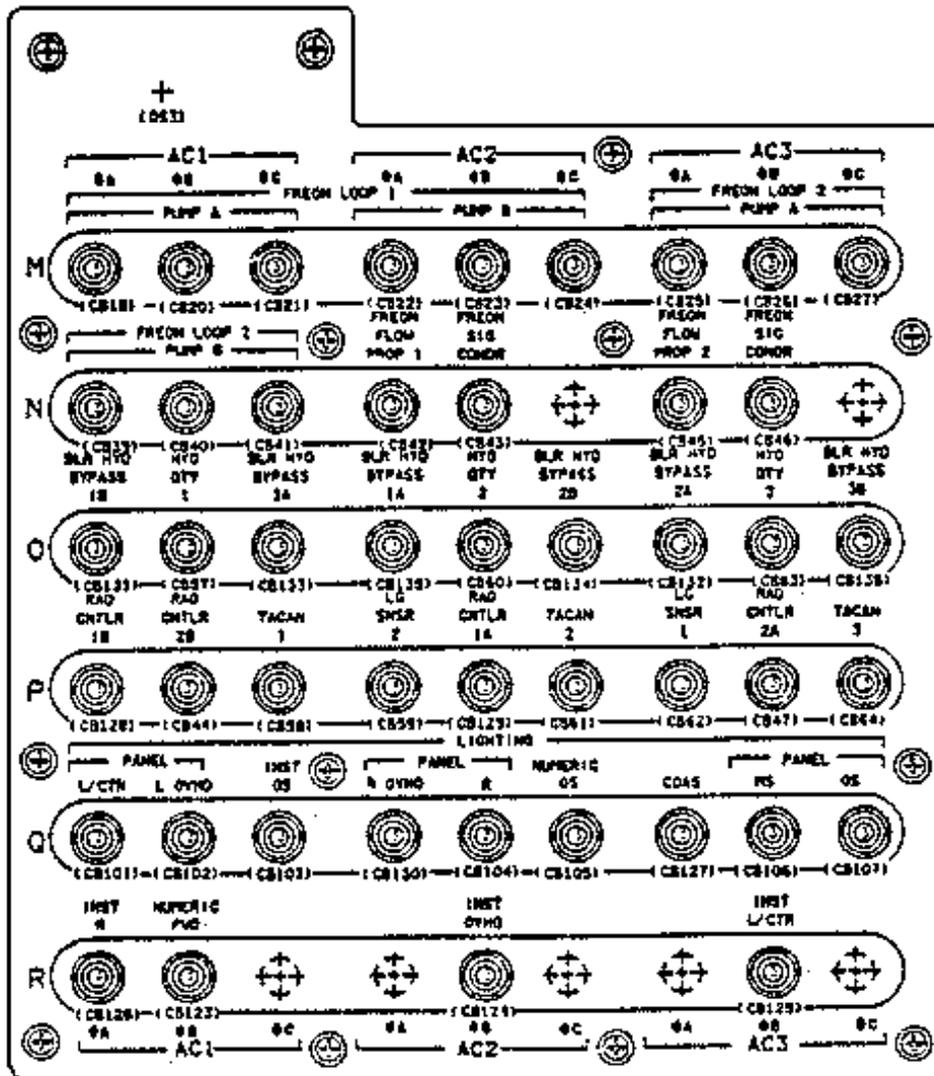


Figure 18-18.- CB 127 on panel L4 (right portion of panel).

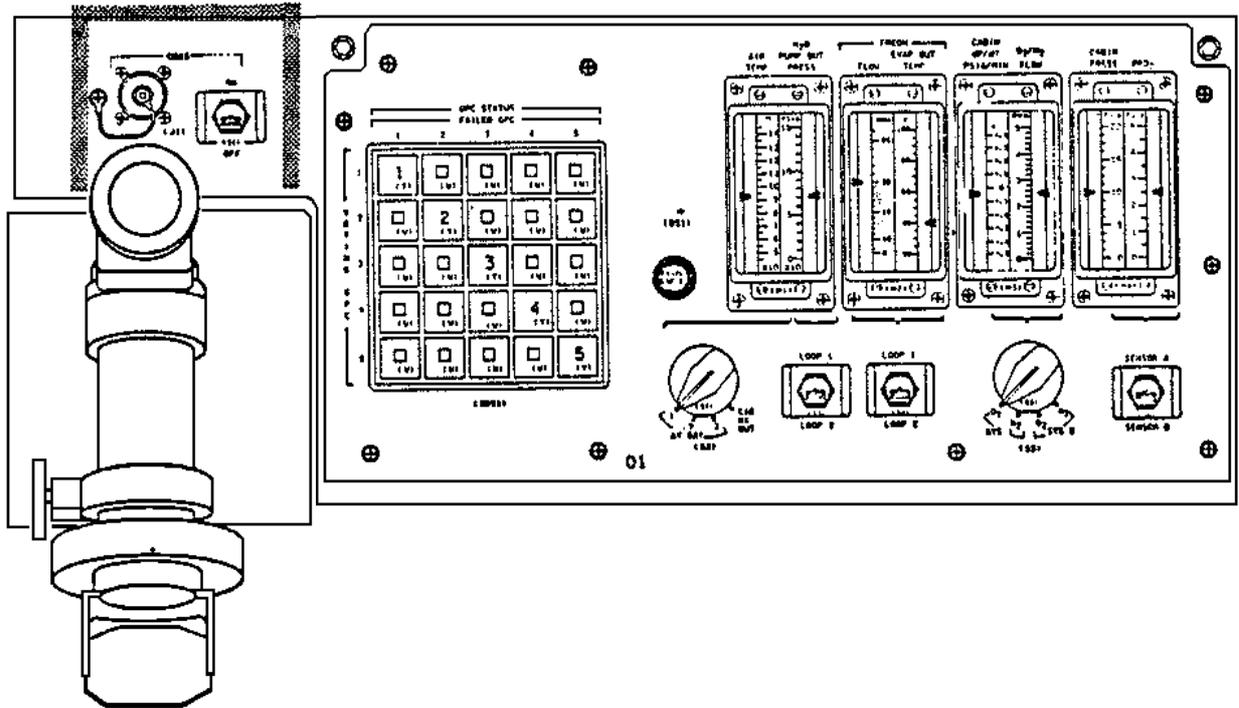


Figure 18-19.- COAS power switch on panel 01.

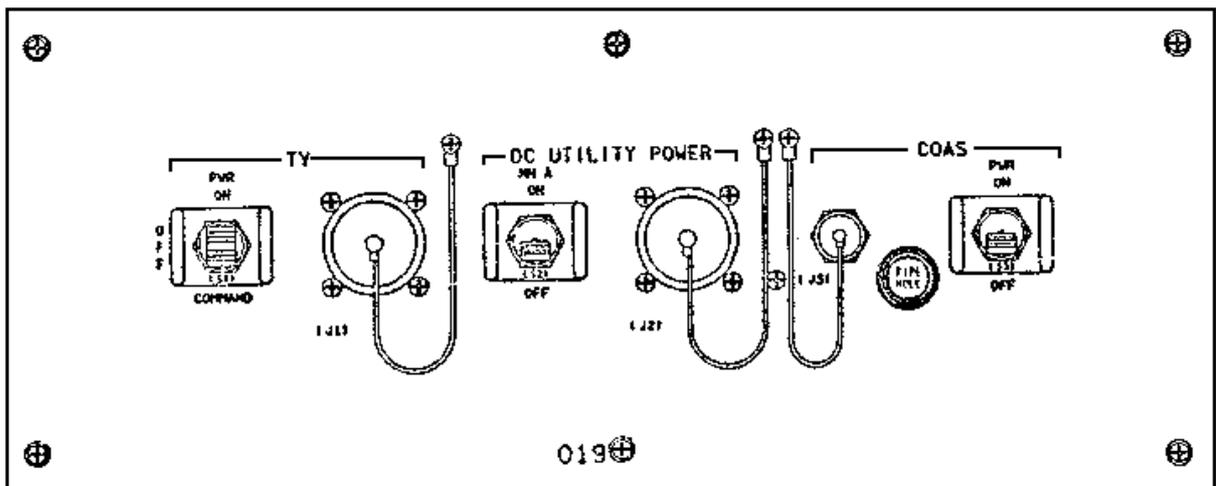


Figure 18-20.- COAS power switch on panel 019.

18.10 ATT REF PUSHBUTTON

In order to establish an LOS vector to a COAS-sighted star, the crewmember needs some method of informing the GPC software when he has a star correctly sighted. Located on panels A6 and F6 (figs. 18-21 and 18-22) is an ATT REF pb. One of its functions is to send a discrete to the GPC telling it to snapshot an LOS vector at the current vehicle attitude. Each ATT REF pb is within easy reach of a crewmember sighting through a COAS in the +X or-Z position. Each is positioned such that it can be depressed with the left hand while the crewmember sights through the COAS as he maneuvers the Shuttle using the rotational hand controller (RHC) with the right hand. The ATT REF pb on panel F8 can also be used through coordination with another crewmember. A COAS mode must be selected on SPEC 22 for the ATT REF pb to have this function; otherwise, its function is attitude direction indicator (ADI) reference definition. While in a COAS mode, the ATT REF pb does not control the ADI. (It should be noted that for use of the L (R, A) ATT REF pb, respectively, multiplexer/demultiplexers (MDM's) FF 1 (2, 3) must be available). The ATT REF pb must be depressed for >320 msec to be recognized (DR 37519-R).

The action of sighting on a star and depressing the ATT REF pb is often referred to as "marking on a star".

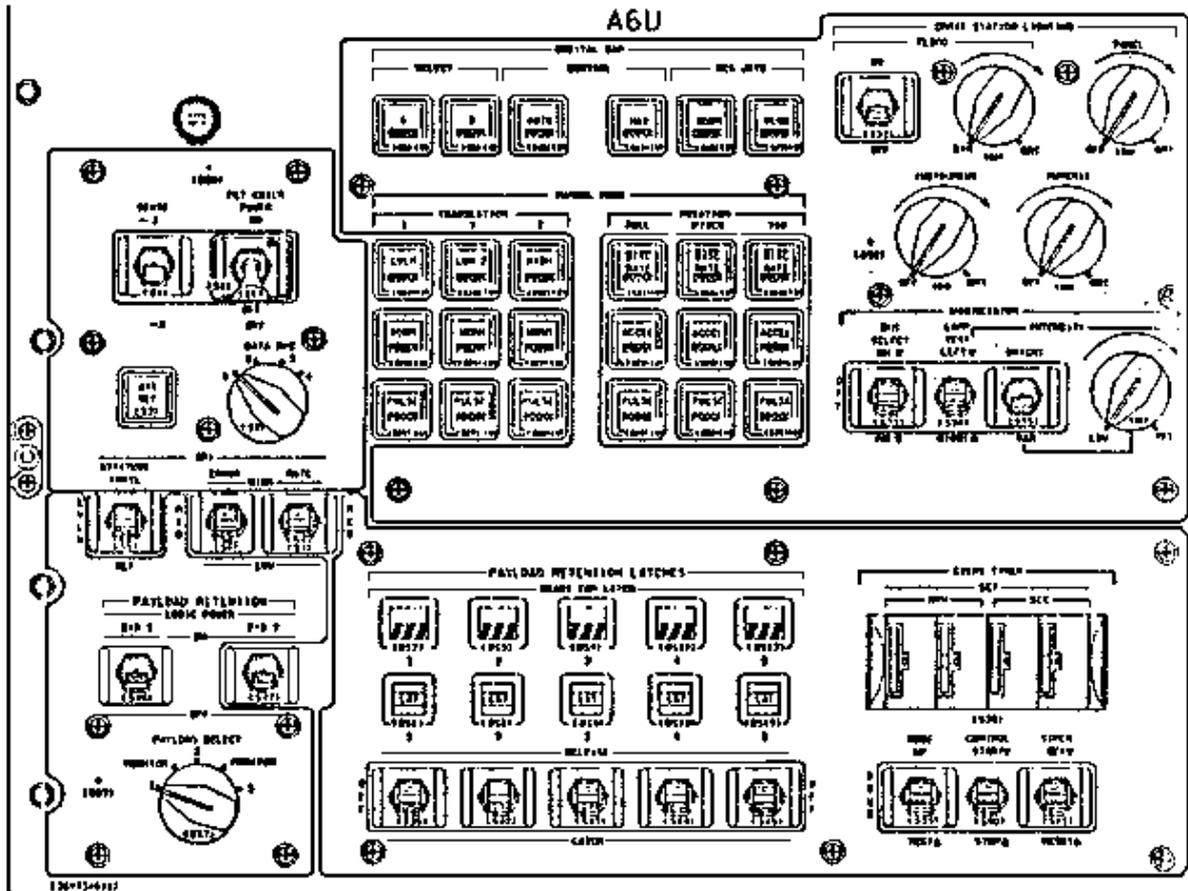


Figure 18-21.- Aft ATT REF pb.

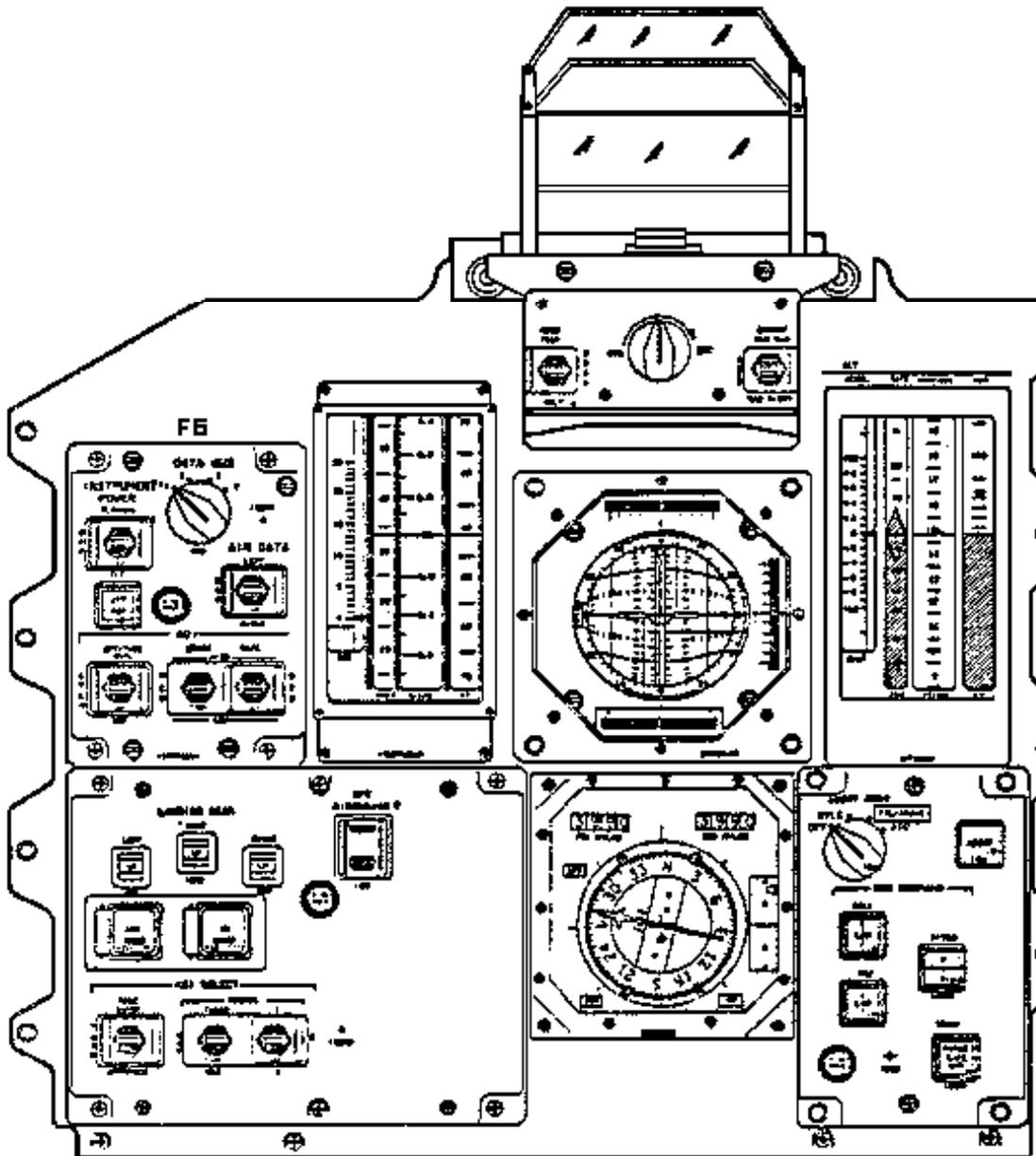


Figure 18-22.- Forward ATT REF pb.

18.11 STAR TRACKER/COAS CONTROL DISPLAY (COAS OPERATIONS) - OPS 2 AND 301

COAS controls are located in the lower right-hand portion of SPEC 22 (fig. 18-23). Item 21 (REQD ID) allows input of the star or target on which COAS marks are to be made. The DDEG X and Y output parameters display the equivalent -Z COAS offsets of a star or target being tracked by the -Z axis star tracker. Item 22 selects the COAS SIGHT MODE in which the ADI ATT REF button performs the COAS mark function and is disabled from its ATT REF function. Item 23 (ACCEPT) accepts the last COAS mark and allows processing of the "mark" data, applies to COAS STAR SIGHT MODE only. Item 24 selects the COAS calibration mode (CAL MODE). Execution of item 25 deselects (DES) COAS operation and causes an "*" to appear next to the item number. Items 22, 24, and 25 are mutually exclusive. Items 26 and 27 allow selection of the mounting location (POS) of the COAS (either +X or -Z, respectively). The calibration (DBIAS) outputs the angular difference between the loaded COAS LOS and the measured calibration COAS LOS in navigation (NAV) BASE coordinates. Items 28 and 29 (UPDATE) are used to incorporate a new COAS bias.

Using the COAS SIGHT MODE or CAL MODE will mode the trackers to TERM/IDLE from any other mode of operation. Only one optical system can be selected at any time.

COAS-sighted stars will always be inserted into columns 1 and 2 of the star table. The first star sighted will be inserted into column 2, and the second into column 1. However, stars are inserted into the table only after an item 23 (ACCEPT), not after the "mark". Column 3 of the star table is not used in COAS operations. In the sighting mode, data on any number of stars subsequent to the first will always be placed into column 1. To replace data in column 2, the star table has to be cleared first, or the sighting mode deselected and reselected. COAS calibrations do not use the star table.

Program Notes

a. COAS ACCEPT without mark

Current software makes no check to ensure a COAS mark has been taken as a condition for processing an ACCEPT item selection. Therefore, erroneous data could go into the star table. This might only be apparent to the crew after a second (valid) star is sighted and the ANG DIF shows a significant error from the expected angular separation.

b. 017 User's Guide

Star ID's >60 illegal in OPS 3.

| XXXX / 022/ | S TRK/COAS CNTL | | XX X | DDD/HH:MM:SS DDD/HH:MM:SS | | |
|-------------|-----------------|------------|-------------|------------------------------|-------|-------|
| S TRK CNTL | Y | Z | S TABLE | 1 | 2 | 3 |
| SELF TEST | 1X | 2X | TRK ID | XXX | XXX | XXX |
| STAR TRK | 3X | 4X | Δ MIN | XXX | XXX | XXX |
| TOT TRK | 5X | 6X | ANG DIF | XXX.X | XXX.X | XXX.X |
| BREAK TRK | 7 | 8 | ERR | XX.XX | XX.XX | XX.XX |
| TERM IDLE | 9X | 10X | SEL | 17X | 18X | 19X |
| | | | S TABLE CLR | 20 | | |
| S TRK | -Y | -Z | COAS | | | |
| REQD ID | 11 XXX | 12 XXX | REQD ID | 21 | XXX | |
| TRK ID | XXX | XXX | Δ DEG X | | ± X.X | |
| S PRES | X | X | Y | | ± X.X | |
| Δ ANG | ± X.XX | ± X.XX | SIGHT MODE | 22 | X | |
| THOLD | 13 X | 14 X | ACCEPT | 23 | | |
| SHUTTER | XX | XX | CAL MODE | 24 | X | |
| MAN OP | 15 X | 16 X | DES | 25 | X | |
| STATUS | XXXXXXXXXX | XXXXXXXXXX | POS + X | 26 X | - Z | 27 X |
| | XXXX | XXXX | Δ BIAS | X.XX | X.XX | |
| | | | UPDATE | 28 | 29 | |

Figure 18-23.- S TRK/COAS CNTL display.

18.12 COAS OPERATIONS

18.12.1 Calibration

The COAS mounting locations are calibrated prior to launch so that the GPC knows the boresighted look angle of each position relative to the NAV base. When the Shuttle reaches orbit, the structural forces applied to the vehicle by its environment have changed; therefore, the structural bending of the vehicle may be different than on the ground. Consequently, the GPC may no longer know the correct COAS boresight in orbit. To correct this condition an on-orbit calibration of the COAS is performed. An orbit calibration is required to guarantee the sighting accuracy required for a critical alignment.

On-orbit calibration is accomplished by sighting on a NAV star while the IMU's are accurately aligned. At this time, the GPC knows the vector to that star. The difference between the two

vectors is the error in the COAS look angle, called the “delta bias“ (D “BIAS”). The GPC then uses the bias in future COAS sightings to correct for misalignment. Once the COAS is calibrated, subsequent demountings and remountings should have no effect on calibration. Refer to figure 18-24 for COAS calibration.

In both calibration and star sighting for alignment, the ATT REF pb defines the LOS vector to the star when depressed. In either case, it is possible to depress the ATT REF pb any number of times until there is confidence that an accurate sighting has been performed. The last “mark” is the one which will be used by software when an UPDATE or ACCEPT is executed (fig. 18-25 and 18-26).

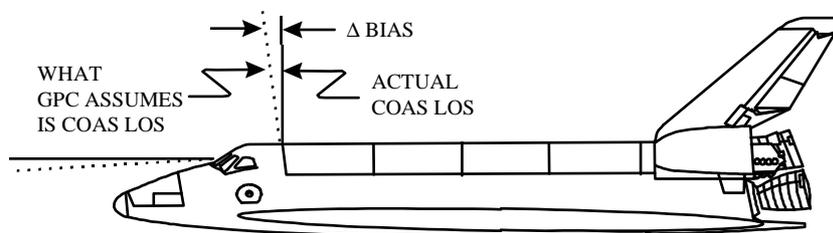


Figure 18-24.- COAS D BIAS rationale (typical for both COAS positions).

If the GPCs are initial program loaded (IPL'd) or an operations sequence (OPS) mode overlay is performed, the on-orbit calibration vector will be overwritten with the preflight I-load. A repeat of the calibration or a relatively simple (but undeveloped and unapproved) general memory Read/Write procedure (GMEM) would be required to reestablish the sighting accuracy.

Current orbiter flight rules state that insufficient data exists to determine if a change in cabin pressure from 14.7 PSIA to 10.2 PSIA or vice versa will result in orbiter structural body bending and a resulting line-of-sight shift large enough to violate the 0.5 degree EI IMU alignment accuracy requirement for entry. If an extended stay is expected at a cabin pressure other than that which the HUD/COAS was calibrated, then a secondary station should be calibrated to provide fail safe redundancy. It is worth noting that during STS-61B the -Z COAS was calibrated at 14.7 psi cabin pressure. Two days later, the cabin pressure was lowered to 10.2 psi to support an EVA and a -Z COAS verification was performed while at 10.2 psi. A total of 10 marks were taken at the lower cabin pressure, with no update, and it was observed that the COAS line of sight had shifted 0.2 degrees toward the -Z axis. So, a very limited set of data does exist, supporting the fact that a change in cabin pressure will result in a LOS shift.

18.12.2 Star Sighting

Use of the COAS for actually aligning IMU's occurs only when the IMU's become so badly misaligned that the star tracker cannot be used or when both star trackers have failed. In either event, a COAS-based IMU alignment is a contingency procedure. The COAS align procedure was exercised on STS-1. Performed after the IMU align and calibration, torquing angles yielded $\pm 0.05^\circ$ error.

18.13 SOFTWARE OPERATIONS

18.13.1 Redundancy Management

The functional redundancy of two trackers and a calibrated COAS is adequate for IMU alignments. No backup to the COAS is provided for rendezvous/ docking. There is no conventional redundancy management (RM: SF or FDI) possible on this device. There is no self-test or built-in test equipment (BITE) function.

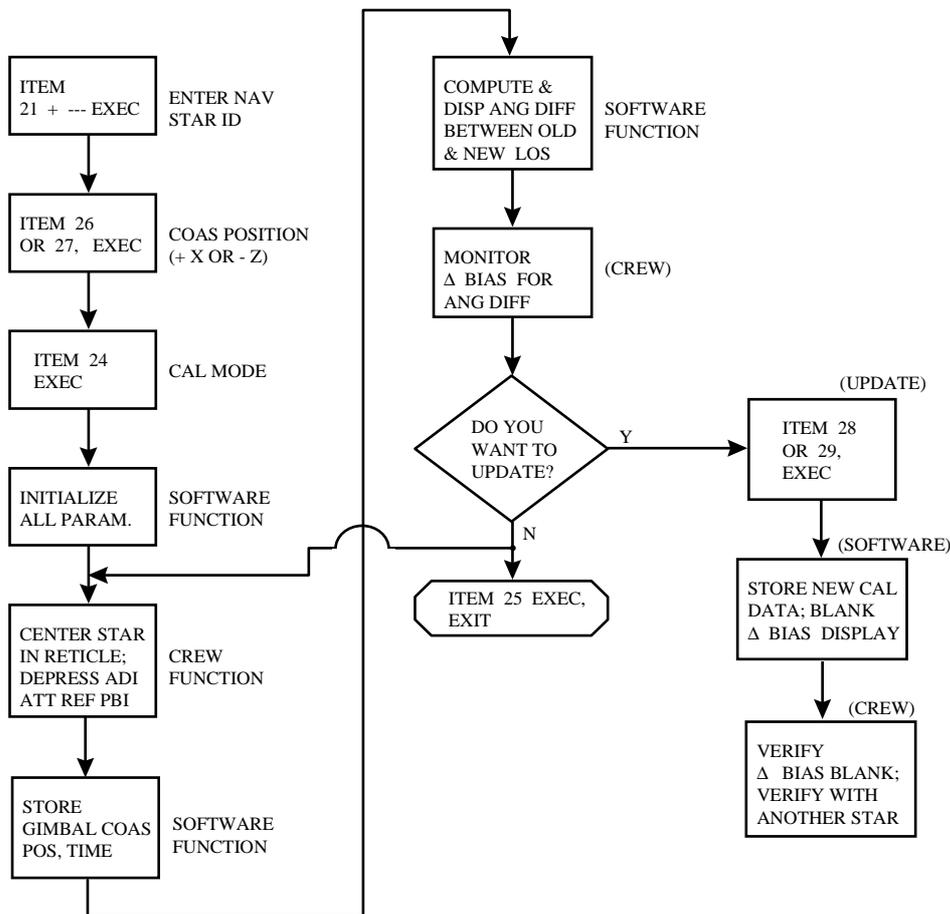


Figure 18-25.- COAS calibration flow diagram.

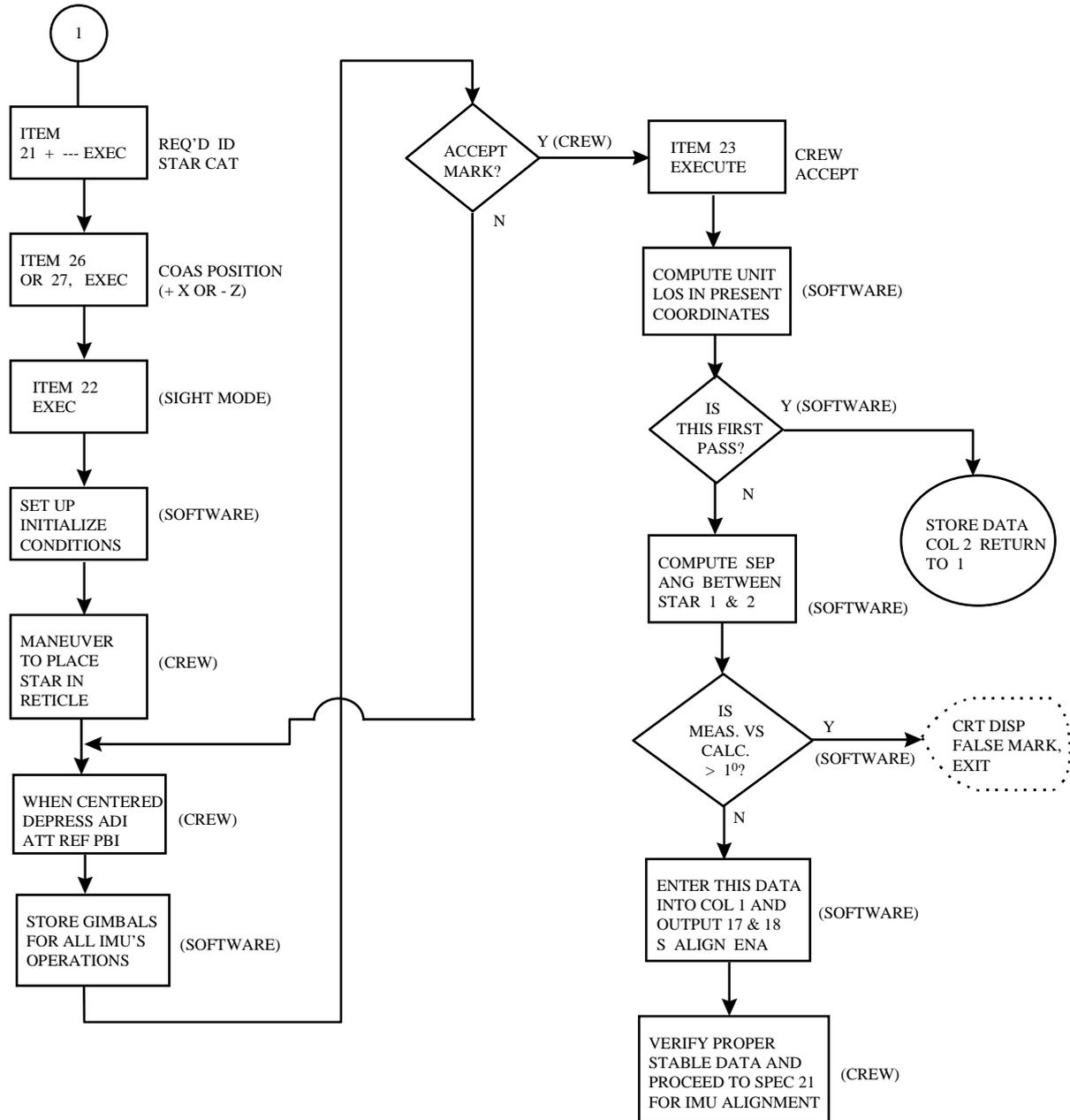


Figure 18-26.- COAS sight mode flow diagram.

18.14 FLIGHT EXPERIENCE

18.14.1 I-Loads - Line of sight

The COAS calibration vector is I-loaded. Flight data has shown that in most cases, this LOS vector would provide sufficient sighting accuracy to support a critical alignment; however, flight data has shown that an orbit calibration is required to guarantee sighting accuracy.

The I-loaded LOS vectors for the HUD/COAS can be observed on SPEC 22 S TRK/COAS CNTL if no update is performed. The +X slot contains LOS unit vector I-loads for the CDR HUD on OV-102, 103, and 105, and the +X COAS on OV-104. The -Z slot contains unit LOS vector I-loads for the -Z COAS on all vehicles (OV-102, 103, 104, and 105). OV-104 does not have HUD I-loads in the +X slot due to a lack of in-flight HUD calibration data. When more in-flight HUD data is obtained on OV-104 the I-loads may be modified to reflect the HUD LOS vector. The vehicle specific I-loads are built based on a history of in-flight calibration data of the HUD/COAS LOS vector.

The I-loads representing the HUD/COAS LOS vectors are unit vectors. The I-load parameters stored in the flight software for the +X and -Z LOS unit vectors are as follows:

| <u>MSID</u> | <u>FSSR Name</u> | <u>Nomenclature</u> |
|-------------|------------------|-------------------------------|
| V99U5613C | I_COAS (2,1) | SLOT 1 (+X), X AXIS COMPONENT |
| V99U5614C | I_COAS (2,2) | SLOT 1 (+X), Y AXIS COMPONENT |
| V99U5615C | I_COAS (2,3) | SLOT 1 (+X), Z AXIS COMPONENT |
| V99U5610C | I_COAS (1,1) | SLOT 2 (-Z), X AXIS COMPONENT |
| V99U5611C | I_COAS (1,2) | SLOT 2 (-Z), Y AXIS COMPONENT |
| V99U5612C | I_COAS (1,3) | SLOT 2 (-Z), Z AXIS COMPONENT |

Although the +X and -Z positions are labeled as such, they may be thought of as LOS vectors 1 and 2, where according to current flight rule priority, slot 1 would be used for the CDR HUD/+X COAS, and slot 2 used for the PLT HUD/-Z COAS. If the I-loads are accurate and a good calibration is performed, the delta bias values to the +X position for a CDR HUD calibration/verification should be small (on the order of 0.12 degrees or less). However, a calibration of the PLT HUD using the -Z slot containing the -Z COAS I-loads would result in a large but consistent delta bias (on the order of 90.0 degrees).

18.14.2 Flight Experience - LOS Calibration

Prior to STS-1, the COAS calibration plan was to perform a calibration in both the +X and -Z locations for several flights of each vehicle and by the resulting data, choose for each vehicle (mounting location and cabin pressure) an accurate I-load that would eliminate the need for repeated in-flight calibration. This plan was based on the assumption that vehicle-specific calibration data would be consistent flight to flight, a single I-load value could be defined based on their averages, and this would satisfy all subsequent requirements for calibration. However,

actual flight data has demonstrated a progressive movement or shifting of calibration marks from flight to flight (figs. 18-27 to 18-31).

This phenomenon is thought to be based on structural flexing/deformation of the crew cabin. The resulting precession of data points makes it impossible to define an exact, all encompassing I-load value. Because of this concern, the procedural in-flight calibration of the COAS (in the -Z location at a minimum) is performed. The bottom line, based on veteran crew inputs, is that there is no problem in calling for a calibration each flight. Most feel it is advisable to learn the use the COAS equipment in case an alignment is necessary.

Numerous unsatisfactory +X COAS calibrations have been observed on numerous missions, including STS-51J, STS-26, STS-29, STS-34, STS-33, STS-32, STS-36, STS-31, STS-41, STS-39, STS-40, STS-48, STS-44, STS-45, and STS-46. The reasons for the unsatisfactory calibrations ranged from not using the proper DAP configuration, poor lighting conditions (moonlight near FOV or too many lights on in the flight deck), poor training/piloting technique, forward window haze or obstructions in the +X COAS FOV, large unexplained flight to flight shifts in the +X COAS LOS, large unexplained +X COAS LOS shifts for multiple calibrations on the same flight, and scattered marks during the same calibration near the 0.12 degree limit.

To provide a reliable back-up IMU alignment device to the star trackers, DTO 0785 "HUD as Back-up to COAS" was performed on several flights (STS-30, STS-41, STS-38, and STS-40). Crew comments recorded during DTO 0785 included that the HUD was easier to use than the +X COAS. The conclusion of the four DTO 0785 flights was that the HUD is an acceptable backup to the COAS. It was noted that the HUD may be superior to the COAS in several respects:

- 1) The HUD is not routinely moved between stations and stowage eliminating a major source of possible misalignment.
- 2) This crew found the HUD to be easier to use than the COAS (green HUD pattern favored over the orange COAS reticle pattern).
- 3) HUD symbology on the test pattern is crisper and clearer than the COAS reticle markings.
- 4) The HUD brightness has a finer intensity control than the COAS reticle dimmer.

After the DTO 0785 results confirmed the HUD to be an adequate backup to the star trackers, the Shuttle Program determined that due to inconsistent +X COAS calibration results, the HUD would be the primary calibration device over the +X COAS. The HUD (CDR and PLT) has demonstrated excellent repeatability as a calibration device from flight to flight and for the same flight (multiple calibrations during a single flight). Current flight Rules state the priority of the calibration devices (HUD and COAS) as a backup to the star trackers. A HUD Calibration was nominally performed instead of +X COAS Cal starting with STS-54 (first time), STS-55, and STS-56.

For additional systems information regarding the HUD, refer to the GNC Systems Briefs, Section 9.0, Heads Up Display. The HUD/COAS calibration procedure is contained in the Generic Flight Data File (FDF), Orbit Operations (ORB OPS) book, GNC tab.

18.14.3 Flight experience - IMU REFSMMATs

Nonorthogonality of the reference stable member matrices (REFSMMATs) causes the small angle dot product computations in the calibration software to be in error. The error becomes significant when the separation angle (D BIAS) is less than 0.1° . The calibration limit 0.12° . The uncorrected REFSMMATs contribute $\sim 0.05^\circ$ and the base-level noise is 0.03 to 0.05° . With uncorrected REFSMMATs, the crewman is hard pressed to meet the calibration criteria. Additionally, feedback is poor since a good mark will appear marginal.

To alleviate this concern, the ground procedurally orthogonalizes the REFSMMATs prior to the calibration. Offline programs exist to perform the computations. The new REFSMMATs are uplinked and incorporated with an alignment prior to the COAS calibration period. New OPS 101 software (OI-12?) for continuous IMU alignment will fix the problem.

Be sure to uplink the normalized IMU REFSMMATS prior to any HUD/COAS calibration.

18.15 SYSTEM FAILURE MODES

All likely failures identified can be compensated, reasonably and easily. No time-critical responses are required to safe the system. The use of the head up display (HUD) is being developed as a potential backup alignment technique to work around a complete loss of the COAS (no backup for rendezvous/docking aid). The remote chance of totally losing the COAS justifies this lack of redundancy.

18.15.1 COAS

a. Failure mode: Incorrect mounting or unable to mount

Cause(s): Improper handling, adjustment mechanism damaged or broken, physical damage to mounting surface

Effect(s):

- (1) Subsystem - correlation between COAS and vehicle axis lost.
- (2) Interfaces - erroneous optical alignment reference with respect to NAV base.
- (3) Mission - difficulty increased in rendezvous and loss of redundancy for IMU alignments.
- (4) Crew/vehicle - possible inability to align IMU and resultant vehicle loss.

Correcting actions: If necessary, stabilize adjustment mechanism. If possible, fasten COAS to mount by normal means or using tape. Recalibrate in flight to establish new LOS vector. Check mark repeatability for the new configuration.

b. Failure mode: Loss of reticle illumination

Causes(s): Bulb fails to light, dimmer fails (open or jammed); bulb failure, vibration or shock, wear or contamination

Effect(s):

- (1) Subsystem - loss of reticle illumination or control thereof.
- (2) Interfaces - reticle brightness cannot be adjusted to the required level for the function being performed.
- (3) Mission - depending on failure (illumination full off, full on, or anywhere in between), sighting on stars or rendezvous operations could be impaired or lost completely.
- (4) Crew/vehicle - possible inability to align IMU and resultant vehicle loss.

Corrective actions: Replace bulb; attempt to use as is; remove assembly cap and bulb, illuminate reticle with flashlight, angle flashlight to vary reticle brightness. Even reflected light from within the cabin has been used to illuminate the reticle.

18.15.2 Mounting/Bracket Assembly

- a. Failure mode: Inability to mount (forward bracket). The mounting bracket provides the means of mounting the COAS to the Orbiter. Improper handling, impact of another object, or damage to the mounting screws could prevent proper mounting. The alternate station would serve as the backup for aligns.
- b. Failure mode: Bracket misaligned (forward only). Can be caused by shock, vibration, or impact. Results in erroneous reference for aligning. Use the aft station and recalibrate to reestablish proper alignment.
- c. Failure mode: Mounting pad damaged (aft only). This is the mounting plate physically attached to the Orbiter. This results in an inability to mount the COAS in the aft station. The forward station should be used. The COAS may be used in the aft if it can be held or taped in place adequately and recalibrated.

18.15.3 Miscellaneous Concerns

- a. Window contamination also may be considered a failure mode. There is a potential for the forward port and/or aft overhead windows to be clouded as a result of solid rocket booster (SRB) separation. It is, however, very unlikely that both stations would be unusable.
- b. Combiner lens damaged. Likelihood of failure is remote as the lens is stored when not in use. No present workaround.

18.16 OPERATIONAL LIMITS

18.16.1 Shuttle Operational Data Book

(4.5.1.2.5) Crewman Optical Alignment Sight

- a. Reticle image - a 10° circle, vertical and horizontal cross-hairs with 1° marks
- b. Elevation scale -10 to 31.5°
- c. COAS to NAV base alignment accuracy
 - 1. Installation to vehicle axis ±0.6 deg/axis, maximum
 - 2. On-orbit calibration
 - (a) Ground-aligned IMU as reference (calibration at 02:23:00 MET) 174 arc sec/axis, 1 σ
 - (b) Star tracker-aligned IMU as reference (1 σ star tracker) 137 arc sec/axis, 1 σ
- d. COAS instrument error (parallax) 52 arc sec/axis, 1 σ
- e. COAS reinstallation after on-orbit calibration (no recalibration) 20 arc sec/axis, 1 σ

18.16.2 FLT EXPERIENCE - ACCURACY

- a. Sighting repeatability (after reinstallation) _12 deg (cal criteria)
- b. COAS LOS movement, flight to flight
 - ±0.45° (max)
 - ±0.15° (1 σ)

| Error Source | 1 Sigma Error magnitude, (arc sec/axis) |
|--|---|
| Installation measurement error (IMU bench installation) | 14 |
| Line replacement unit reinstallation errors (line and on-orbit) - no measurement | |
| IMU | 20 |
| COAS | 20 |
| Navigation base thermal bending (IMU-to-navigation base) | < 15 |
| Structural deformation (overhead window) | |
| Cabin pressure variation | 50 |
| Temperature variation | 4 |
| Glass bending | - |
| Vehicle dynamics | 75 |
| Crew vision | 21 |
| IMU stable member errors (stable member with respect to case) | |
| Pitch to outer roll gimbal (non-orthogonality) | 30 |
| Case to outer roll gimbal misalignment | 28 |
| Resolver errors | 44 |
| COAS instrument error (parallax) | 52 |
| COAS on-orbit calibration error | |
| Ground-alignment IMU as reference (function of time) | 174 |
| On-orbit star tracker aligned IMU (1 sigma star tracker) | 137 |
| RSS 1 sigma/axis/star measurement (<u>budget total</u>) | |
| Ground-aligned IMU (STS-1 timeline) | 215 |
| Star tracker alignment IMU (1 sigma star tracker) | 187 |

Figure 18-27.- Error Budget for COAS/IMU Realignment (Ref. SODB Paragraph 4.5.1.2.3)

(a) Aft Station

| Star pair separation (SPS), deg | 1-Sigma realignment uncertainties, arc sec/axis | |
|---------------------------------------|--|----------------------------------|
| | Ground-aligned IMU | ST aligned IMU (SPS = 90 deg) |
| 30 | 372 | 324 |
| 50 | 261 | 227 |
| 70 | 224 | 195 |
| 90 | 215 | 187 |
| 110 | 224 | 195 |
| 130 | 261 | 227 |
| 150 | 372 | 324 |

(b) Forward Station

| Star pair separation (SPS), deg | 1-Sigma realignment uncertainties, arc sec/axis | |
|---------------------------------------|--|----------------------------------|
| | Ground-aligned IMU | ST aligned IMU (SPS = 90 deg) |
| 30 | 369 | 321 |
| 50 | 257 | 222 |
| 70 | 219 | 189 |
| 90 | 210 | 181 |
| 110 | 219 | 189 |
| 130 | 257 | 222 |
| 150 | 369 | 321 |

Figure 18-28.- On-orbit COAS/IMU Realignment Uncertainties as a function of Star Pair Separation for All Random Errors (Ref. SODB Paragraph 4.5.1.2.3)

SECTION 19 EXTERNAL TANK SEPARATION

19.1 Introduction

External tank (ET) separation is an important part of the ascent profile. It occurs approximately 8.5 minutes after liftoff, shortly following main engine cutoff (MECO). Although the entire ET separation process normally requires less than 30 seconds, a complex interaction of hardware and software is required for its successful completion.

The ET supplies LH2 and LO2 propellants to the three SSMEs. It also serves as the structural backbone for the Orbiter and the SRBs during ascent. At 154 feet long and 27.6 feet in diameter, it is the largest component of the space shuttle vehicle. Empty, it weighs approximately 69,000 lb. When fully loaded with propellant at launch, it weighs approximately 1,660,000 lb.

External tank separation is normally an automatic process. However, the crew has the capability to manually override the automatic External Tank Separation Sequence or to manually initiate it any time in OPS 104, 105, 106, or 601. A contingency separation sequence called "fast separation" can be initiated manually by the crew in OPS 102, 103, or 601. After separation, the tank pitches away from the Orbiter, tumbles, and breaks up upon reentry. Any pieces of the tank that survive entry are intended to fall within a designated ocean impact area.

This systems brief discusses the hardware and software (PASS, Flight Software Version OI-26, 27) involved in the ET separation process. Attention focuses primarily on nominal separation, although abort and off-nominal cases are addressed. It also describes crew interfaces with the separation process and explains how manual separation is accomplished.

19.2 Crew Interfaces

The crew can interface with the ET separation process via several switches and SPEC item entries. These interfaces allow the crew to control the activities of the automatic process.

19.2.1 ET Separation AUTO/MAN Switch and SEP Push-button

These switches are illustrated in SSSH drawing 10.23 sheet 1 of 4. They provide a means for the crew to manually override the automatic separation process and initiate physical separation of the Orbiter and external tank. Separation is normally accomplished without crew intervention with the mode switch in AUTO. If there is reason to delay ET separation or override the automatic separation sequence, the crew can place the mode switch in the MAN position. This inhibits

physical separation until the ET SEP push-button is depressed, the Spec 51 override is entered, or the switch is returned to AUTO.

Each switch position has three contacts. Two of three voting logic determines the position of the switch. Default logic is provided to resolve dilemmas. If failures cause the switch to indicate both the AUTO and MAN position simultaneously or cause no position to be indicated, a dilemma is declared. In OPS 1, switch RM defaults to the MAN switch position. An "ET SEP MAN" message is annunciated to inform the crew. The override display, SPEC 51, is available to the crew and allows manual override of the switch in the event of switch failures.

In OPS 6 (RTLS), switch RM defaults to the AUTO position and an "ET SEP AUTO" message is annunciated. Note that for a TAL case, if ET SEP MAN is annunciated to the crew in OPS 1, and a manual ET SEP is not performed prior to entering OPS 3, there is no way to get off the tank. As of January 1997, a CR is being considered to default to Auto ET SEP for a ET SEP MAN case on a TAL. The earliest this could be implemented is OI-28 or 29. For an ET Sep Inhibit (ET SEP INH) caused by a 17" disconnect valve failure or vehicle rates out of limits, ET SEP defaults to Auto for both the TAL and RTLS cases.

19.2.2 Override Display - Spec 51, item 38 and item 39 (BFS 28, 29)

The PASS override display SPEC 51, is illustrated in fig. 19-1. Item entries 38(28) and 39(29) are legal in MM102, 103, 104, 105, 106, and 601. Item 38(28) provides the capability to override a manual ET separation and reestablish AUTO. It is used when switch failures cause a manual default to be selected. Item 39(29) is used when automatic separation is inhibited and the ET SEP AUTO/MAN switch or SEP push-button are inoperative. This Spec option initiates physical separation of the Orbiter and ET.

19.2.3 ET Umbilical Door Switches

These switches are located on panel R2 and shown in SSSH drawing 15.2, sheet 2 of 3. They are used by the crew after ET separation to stow the centerline latches, close the ET umbilical doors, and latch the doors. Umbilical door hardware is discussed in greater detail in a later section but is the **responsibility of the MMACS flight controllers.**

19.2.4 Override Display - Spec 51, item 40 (BFS 30)

This item entry is illustrated on the PASS override display, figure 19-1. It serves as a backup to the ET umbilical door switch. Item 40 is only legal in MM104, 105, and 106.

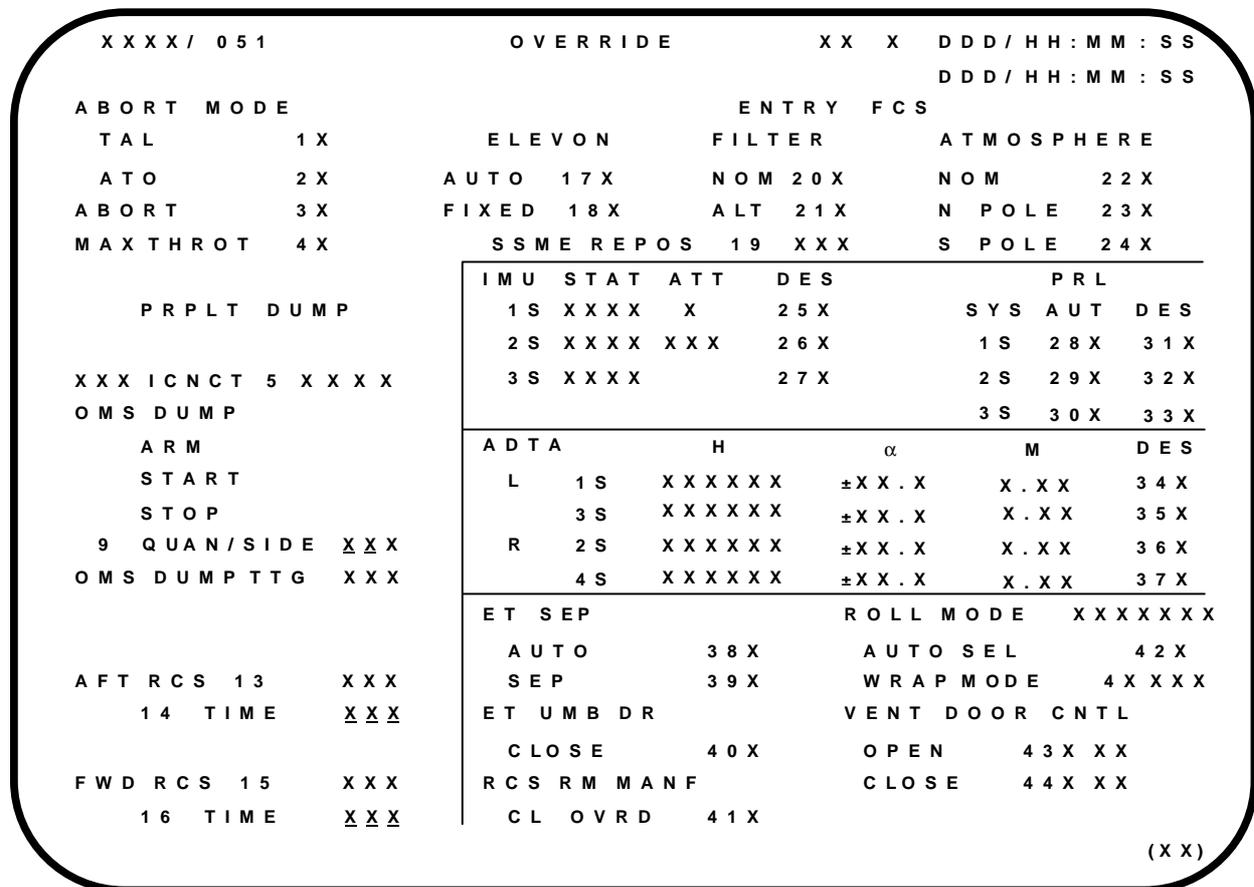


Figure 19-1. Override display

19.3 ET Sep Hardware

To understand the steps executed by the software during ET separation, it is helpful to understand the hardware that is involved in the process. A brief description of this hardware follows. An indication of which flight control discipline has primary responsibility is also listed.

19.3.1 Pyro Initiator Controller (**EGIL Responsibility**)

Pyro initiator controllers (PICs) are explosive devices used throughout the separation process to sever nut/bolt assemblies mating the Orbiter and ET. They have a standard design that is used

throughout the Orbiter vehicle. PICs remain unarmed until they receive an arm command, which consists of a 20-32 V DC signal. This voltage is actually input power for a DC/DC converter. The DC/DC converter charges a capacitor bank which stores sufficient energy to fire a NASA standard initiator (NSI). Once armed, a FIRE command causes the capacitors to discharge to the NSI, creating an explosion.

19.3.2 Forward and Aft ET Structural Attach Points (**MMACS Responsibility**)

There are three Orbiter/ET attach points, one forward and two aft. The forward attach point is supported from the LH2 tank forward ring frame. The aft structure is mounted on the ET aft major ring frame and the LH2 tank longerons. Refer to SSSH drawing 15.3, sheet 1 of 3. Numerous pinned and spherical joints exist within the support structures that allow for multidirectional motion, minimizing bending induced in the support structures as a result of thermal and structural loads.

19.3.2.1 Forward structural attach point (**MMACS Responsibility**)

The forward attach point is designed with rotational freedom about a Y-axis reference line so that changes in ET length resulting from thermal effects will not induce loads into the Orbiter. ET attach hardware consists of two struts joined to a yoke fitting. A bolt runs through the apex of the yoke and is secured inside the Orbiter with a large frangible hex nut. Refer to SSSH drawing 15.3, sheet 1 of 3. This nut is equipped with two pyros, referred to as the structural separation PICs, which explode and shear the nut to allow ET structural separation. Either of the two structural separation PICs is capable of fracturing the nut. A containment vessel is designed to capture the exploded nut-fragments. The strain energy of the bolt, which is released when the nut is exploded, causes the bolt to spring out of its yoke and travel away from the Orbiter. This motion is aided by Belleville springs, which push the freed bolt outside the Orbiter along a guide rod. A device called a hole plugger fills the vacated area. No closeout door is required to cover the forward attach point after separation.

19.3.2.2 Aft structural attach points (**MMACS Responsibility**)

The two aft ET structural support assemblies are slightly different from one another. The starboard (right) aft support is a tripod, making it fixed, while the port (left) structure is a bipod that permits lateral pivotal motion. This design allows unrestrained relative lateral distortion between the aft ET and the Orbiter. A rectangular crossbeam connects the two support structures.

The ET/Orbiter structural interface attachment hardware is the same at both aft attach points. It is very similar to the hardware in the forward assembly. A 2.5-inch diameter tension bolt extends

from the apex of the bipod (tripod) assembly on the ET and engages a frangible nut housed in the Orbiter. Each nut is equipped with two structural separation PICs. ET/Orbiter physical separation is accomplished by firing these PICs which shear the nut. Each bolt has a retracting spring on the ET, which retracts the bolt into the ET strut after nut fragmentation. Orbiter A “blade valve” fills the cavity vacated as the bolt exits. Fragments of the exploded nut are intended to be confined within a blast container on the Orbiter.

19.3.2.3 ET Umbilical Attach Plate

The Attach Plate itself is a MMACS Responsibility, but the disconnect valves located there are a Booster/MPS Responsibility

The ET/Orbiter fluid and electrical interfaces are located at two aft umbilical plate assemblies. These plates are adjacent to, but separated from, the two aft structural attach points. Orbiter/ET interfaces located on the umbilical plates include a 2-inch tank pressurization line, a 17-inch feedline, a 4-inch tank recirculation line (LH2 side only), and an electrical monoball disconnect. In the special case of the OV-102 vehicle, separation cameras are also installed on the umbilical plate and provide coverage of the ET/Orbiter separation process. Refer to SSSH drawing 15.3, sheet 2 of 3.

Each Orbiter umbilical plate is mated with its counterpart on the ET via three frangible nut/bolt combinations. Each nut is equipped with two pyrotechnic detonators. During the ET/Orbiter separation process these pyros, referred to as the ET umbilical unlatch PICs, are fired. This shears the nuts, allowing the mated umbilical plates to separate.

Each umbilical plate is equipped with three hydraulic retractors. These retractors pull the plate approximately 2.5 inches into the Orbiter following release of the three frangible nut/bolt combinations. A solenoid valve incorporated in the retractor device controls fluid flow to the retractor. The solenoid valve is enabled by ET umbilical retract PICs.

The use of the word PIC in the name of the ET umbilical retract PICs is somewhat misleading. This is not an explosive device because it does not contain an NSI. The electrical charge generated by the PIC is used instead to supply electrical power to the solenoid valve. When the solenoid valves are powered, they allow hydraulic fluid to flow to the hydraulic retractors. Only two of the three hydraulic actuators are required to retract the umbilical plates successfully.

Retracting the umbilical plate serves three functions. Primarily, the retract motion disconnects the Orbiter/ET electrical umbilical in the first one-half inch of travel. This releases fluids trapped between the Orbiter and ET LO2/LH2 17 inch disconnect valves. A secondary function of the retractor motion is to provide a redundant means for closing the 17 inch disconnect valves (the primary actuation force is pneumatic). As the umbilical plate retracts, these valves are forced closed via mechanical linkages. This function is especially important because it is the primary,

and only, means of closing the 17-inch disconnect valves during a fast ET separation (to be discussed later). In addition, each Orbiter umbilical plate has three X-Y plane stabilizing bungees that hold the plate in position after separation from the ET umbilical plate.

19.3.3 ET Umbilical Doors (MMACS Responsibility)

After the ET separates from the Orbiter, the two aft umbilical openings are exposed. Two doors with thermal protective tiles cover the exposed umbilical areas - shielding them from entry heating. These doors are approximately 50 inches square and are attached to the Orbiter by two hinges. Each door is driven open/closed by two, three-phase AC motors. The doors also have two different types of latches: centerline latches hold the doors open during ascent, and uplock latches hold the doors closed during orbit and entry. Refer to SSSH drawing 15.2, sheet 1 of 3.

The doors are held in their ascent stow position by two centerline latches that engage both the left and right doors. Each centerline latch engages a fitting on the outer edge of the doors. After ET separation, the latches are disengaged and allow the doors to close. This is accomplished by rotating the latches 58° (a minimum rotation of 33.75° is required to free the doors). After rotating, the two centerline latches retract into their stow positions, the umbilical doors are closed and then held in place by uplock latches. There are three uplock latches located inside each umbilical cavity. The latches engage three uplock rollers located on the inside of each door. The latches are designed to prevent the doors from vibrating or reopening during entry. When the door is within 2 inches of being closed, the three uplock rollers contact the three uplock latches. Ready-to-latch limit switches indicate that the uplock rollers have engaged the latches. The signals enable the uplock latches to drive closed, compressing the aerothermal seals around each door.

Nominally, the umbilical doors are closed manually by the crew following ET separation. This is accomplished via switches on panel R2. During an RTLS or TAL abort, the doors close automatically. Automatic door closure reduces crew operations during time-critical abort situations. For the RTLS abort, the doors close following ET separation in MM601. For a TAL abort, the doors are closed in MM304. Note that the software which closes the umbilical doors during a TAL is open loop. No check is made to verify that ET separation has occurred prior to closing the umbilical doors. The automatic close capability for TAL cases was a new function added to the vent door sequence in Flight Software Release OI-20 (CR89354).

19.3.4 Main Propulsion System Umbilical Lines (Booster/MPS Responsibility)

The Orbiter and ET umbilical connections provide paths of transfer for O₂ and H₂ between the ET and Orbiter. The umbilical lines include 2-inch GO₂ and GH₂ pressurization lines, a 4-inch LH₂ recirculation line, and 17-inch LO₂ and LH₂ feedlines (figure 19-2). The 2-inch lines at each attach point are used to maintain the ullage pressure in the LO₂ and LH₂ tanks. This is achieved by routing gaseous propellants to the tanks from the SSMEs. The 4-inch recirculation line on the LH₂ side is used to circulate LH₂ through the SSMEs during prelaunch. This provides thermal conditioning for the main engines before start. A 4-inch line is not installed on the LO₂ system. During prelaunch, LO₂ used for thermal conditioning of the SSMEs is expelled overboard. LH₂ flows to the main engines during ascent through the port side 17-inch feedline and LO₂ flows through the starboard line. At ET separation, the Orbiter umbilical lines are separated from the ET umbilical lines using special disconnect hardware.

19.3.4.1 17-Inch Feedline Disconnect (Booster/MPS Responsibility)

The 17-inch LO₂/LH₂ feedline disconnects are shown in figure 19-3. Each feedline disconnect consists of a pair of two-position, pneumatically actuated, flapper valves (one on each side of the Orbiter/ET interface). A feedline disconnect latch locks the flappers in the open position during ascent, allowing propellant to flow from the ET to the Orbiter. Post-MECO, the latch is unlocked to ready the flapper valves for closure. The feedline disconnect latches were not part of the original Orbiter design but were added beginning with STS-26. They prevent accidental valve closure during main engine operation. If a valve failed closed during this time, it would starve the main engines of propellant and result in catastrophic shutdown.

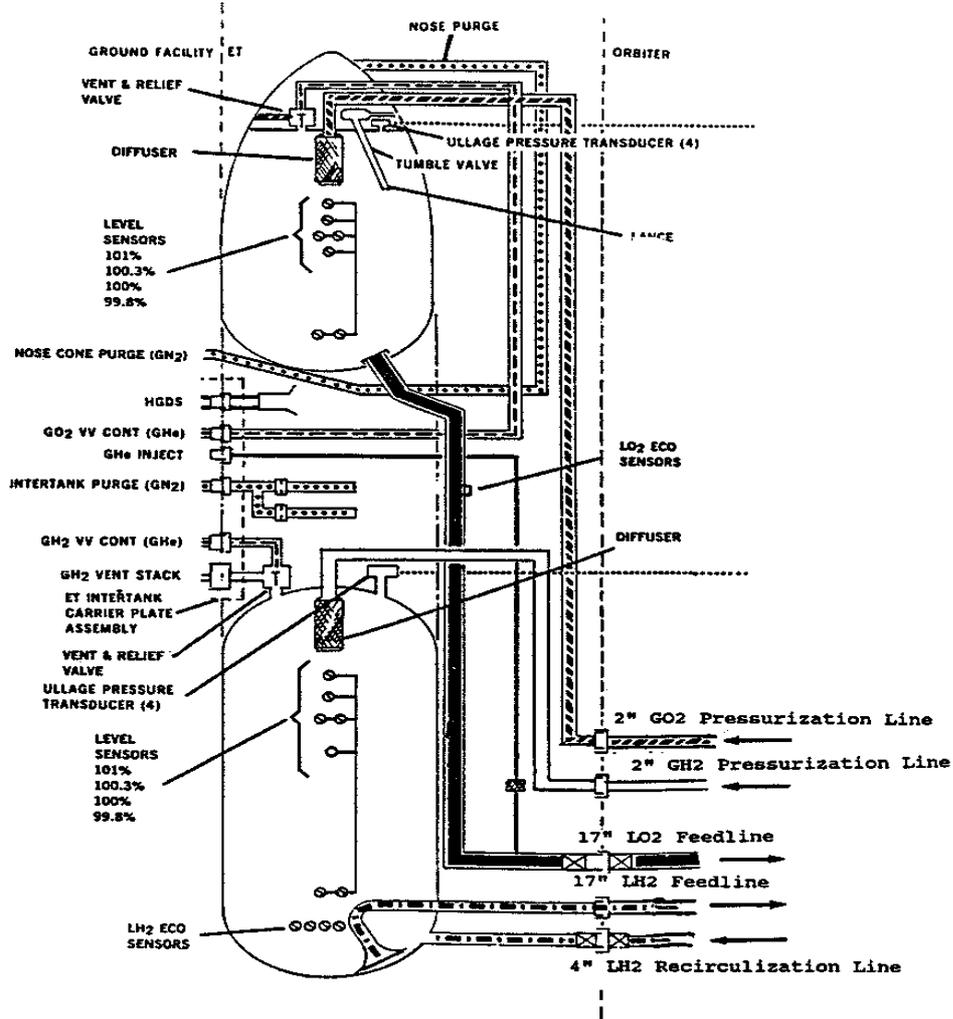


Figure 19-2. External Tank Interconnect Lines

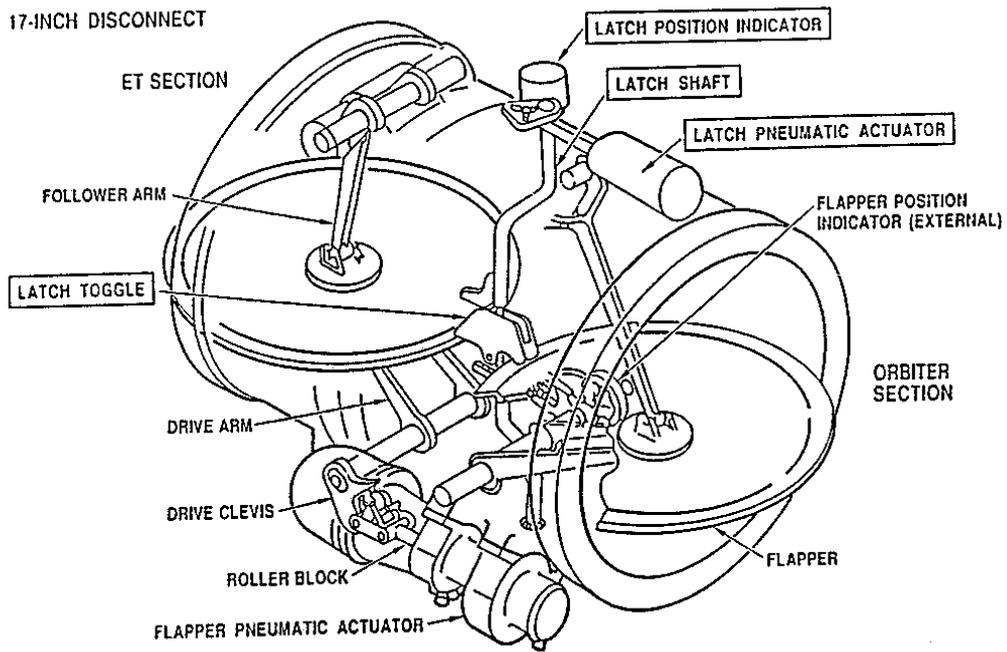


Figure 19-3. 17" LO2/LH2 Feedline Disconnect

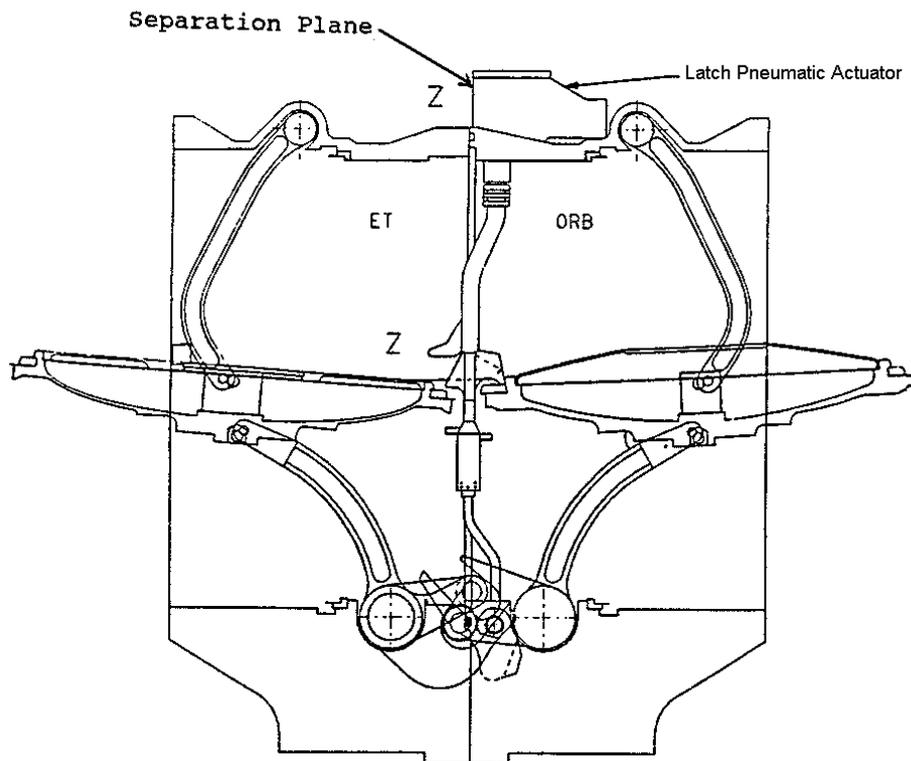


Figure 19-4. 17" LO2/LH2 Feedline Disconnect

There are two flapper valve closed position indicators for each 17-inch disconnect. These redundant position indicators are used by software to determine the position of the valves. The LH2 line closed position indicator telemetry is carried through MDMs FA2 and FA4 while the LO2 feedbacks travel through FA2 and FA3. The MDMs that carry this telemetry are significant because commfaults can affect the ET separation process. OI-22 and subsequent software qualifies the indicators for LO2 and LH2, such that any premature closed indicator or commfault

will not be used in the ET SEP Sequence logic. Only one indicator per LO2/LH2 is required by the ET SEP Sequencer.

A secondary means of closing the 17-inch feedline disconnect flapper valves is provided through mechanical linkages. When the umbilical plate is retracted into the Orbiter by the hydraulic retractors, a mechanical linkage system forces the valves closed. The mechanical system is especially important because it is the only means of closing the valves during "fast separation" ET separations.

The 17-inch valves disconnect during the separation process. The ET half of the disconnect departs with the ET and the Orbiter half remains with the Orbiter. The separation plane is shown in figure 19-4.

19.3.5 MPS Valves (Booster/MPS Responsibility)

Refer to figures 19-5a and 19-5b throughout this discussion.

19.3.5.1 Prevalves.

The six MPS prevalves terminate flow of propellants to the main engines and help to safe the system. The SSME OPS software commands the prevalves to close at MECO.

19.3.5.2 MPS LO2/LH2 Feedline Relief Isolation Valves.

The MPS LO2/LH2 feedline relief isolation valves prevent propellant from escaping through the 17-inch feedline relief valves during SSME operation. The isolation valves are opened post MECO and allows gasses trapped within the 17-inch line to reach the relief valve.

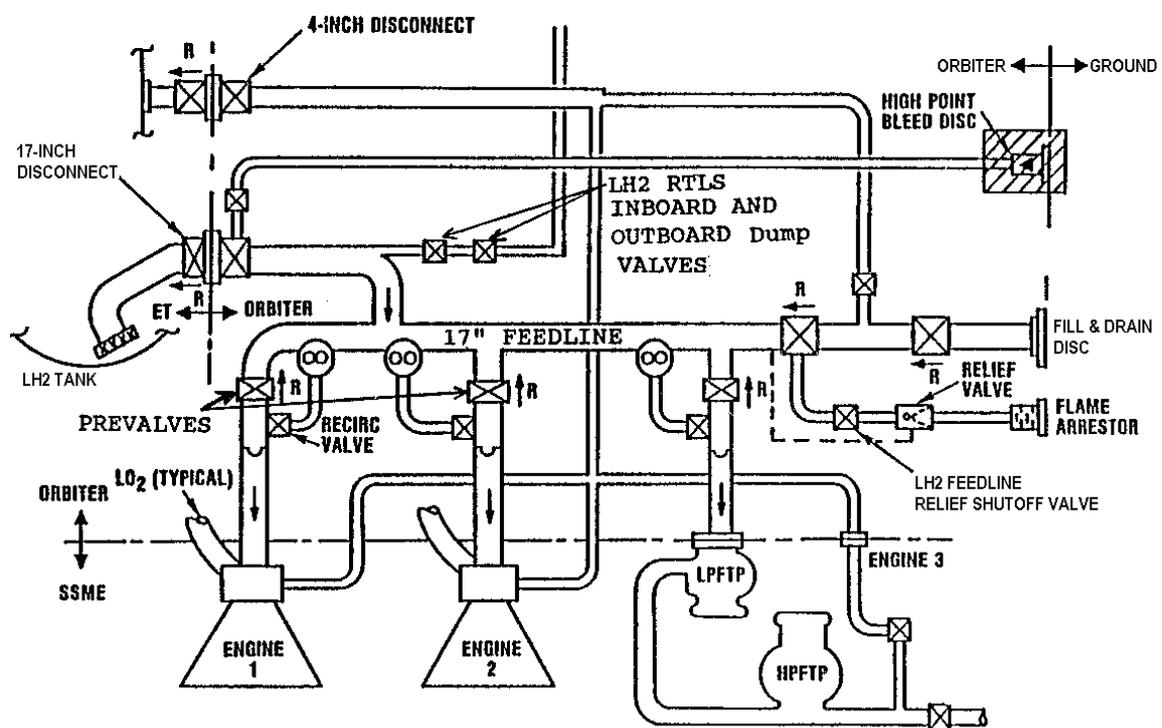


Figure 19-5a. MPS LH2 System

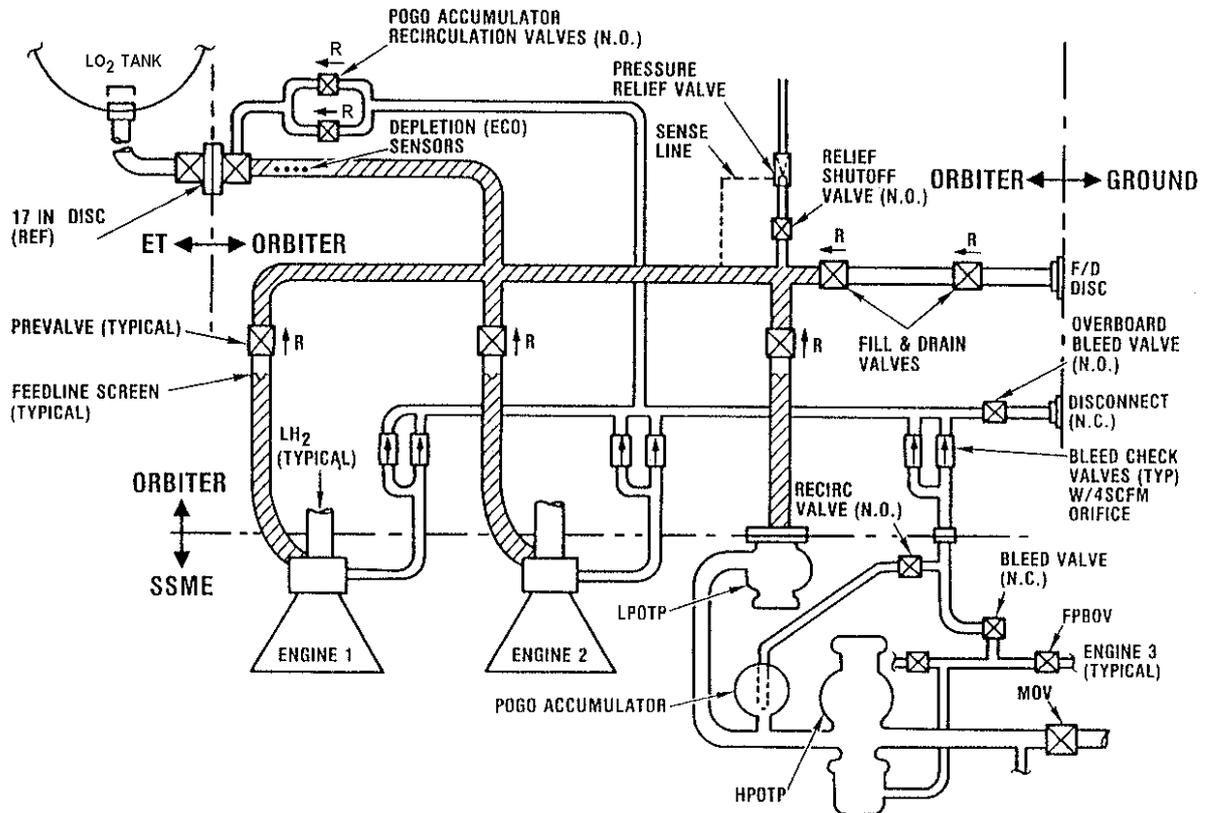


Figure 19-5b. MPS LO2 system

19.3.5.3 MPS LO2/LH2 Feedline Relief Valves.

The feedline relief valves allow propellants trapped in the 17 inch feedlines to escape when pressure builds up to the relief setting. It prevents the line from over-pressurizing and rupturing.

19.3.5.4 LH2 Backup Dump Valves - Inboard and Outboard (aka RTLS Dump Valves).

The LH2 backup dump valves provide the primary means of relieving pressure in the 17-inch feedlines post MECO. During a nominal ET separation, these valves are opened to provide pressure relief by exposing the feedline to the vacuum of space. In case of RTLS and TAL aborts, the 17-inch line is also purged with helium to assist in forcing LH2 out the dump valves. This is necessary to ensure that the LH2 is cleared from the system despite the higher ambient pressure experienced during these aborts.

19.3.6 ET Tumble Valve

The ET tumble valve was part of the original Orbiter design but is no longer in use. **The following is retained for historical purposes.** It is located at the top of the ET and was designed to allow a small amount of GO₂ to escape from the tank following separation. The escaping GO₂ caused the tank to begin a slow end-over-end tumble. The tumbling action was intended to ensure that the ET would accurately land in a predetermined footprint during reentry. However, the tumble valve was found to be of minimal value in maintaining the ET footprint and had several risks associated with it. If the valve failed open during ascent, it could result in a catastrophic explosion. Therefore, the tumble valve hardware was disabled prior to each flight. External tanks currently in production and use do not include a tumble valve. However, the ET Separation Sequence software still contains tumble valve commands.

19.4 ET Separation Software

ET separation is a complex process achieved through the interaction of numerous software modules. These include the ET Separation Sequencer, the transition digital autopilot (Trans DAP), the GN&C Moding, Sequencing, and Control (MSC) sequencer, the Master Events Controller (MEC) SOP, and the SSME OPS module. Most of the separation activities are accomplished by just two of these modules, the ET Separation Sequencer and the Trans DAP. An overview of the software interfaces during ET separation follows.

The MSC module schedules execution of the ET Separation Sequencer at the correct time in the ascent profile. Nominally this occurs when MECO is confirmed. The SSME OPS module sets the MECO confirmed flag and other discrettes that are required by the MSC to achieve proper scheduling. The SSME OPS also provides flags to the ET Separation Sequencer, which indicate the mode of separation that is required during contingency abort cases. Physical separation of the Orbiter and ET is accomplished by the ET Separation Sequencer; it is responsible for preparing the Orbiter for separation and then executing the physical separation. Three modes of separation can be accommodated: automatic (nominal, RTL, or TAL), manual, and fast ET separation. Each of these are discussed in detail later. Once separation from the ET has been achieved, the Trans DAP performs an automatic -Z translation away from the tank. It also monitors vehicle rates and determines when the translation should be terminated. The automatic separation process can be manually overridden by the crew at any time. Figure 19-6 illustrates the interaction between the modules.

Each software module involved in the separation process is discussed in detail in the following sections.

19.4.1 SSME OPS Sequence

The SSME OPS sequencer serves several important functions in the ET separation process. First, it sets the MECO confirmed flag. This flag is set when indications from main engine sensors confirm that the engines have shut down. The MECO confirmed flag represents the beginning of the ET separation processes and is used by both the MSC sequencer and ascent DAP software. It indicates to the MSC sequencer that it is time to schedule execution of the ET separation sequence module. The ascent DAP uses this flag as an indication to terminate operation and hand Orbiter control over to the Trans DAP.

Another function of the SSME OPS module is to monitor manual ET separation discrettes during MM102, MM602, and pre-MECO in MM103. Manual inputs during these periods affect the mode of separation executed by the ET separation sequencer. However, the sequencer is not normally active during these early phases of flight and cannot monitor for manual discrettes. A fast separation is performed if the crew initiates manual ET separation in MM102 or MM601. It is also performed in MM103 if the crew initiates separation after two main engines have failed. The SSME OPS sends the appropriate indications to the ET separation sequencer if a fast separation is requested. It also sets the appropriate flags and delays to provide the proper SSME shutdown sequence for these cases. A fast separation can be performed in the BFS in OPS 102 only.

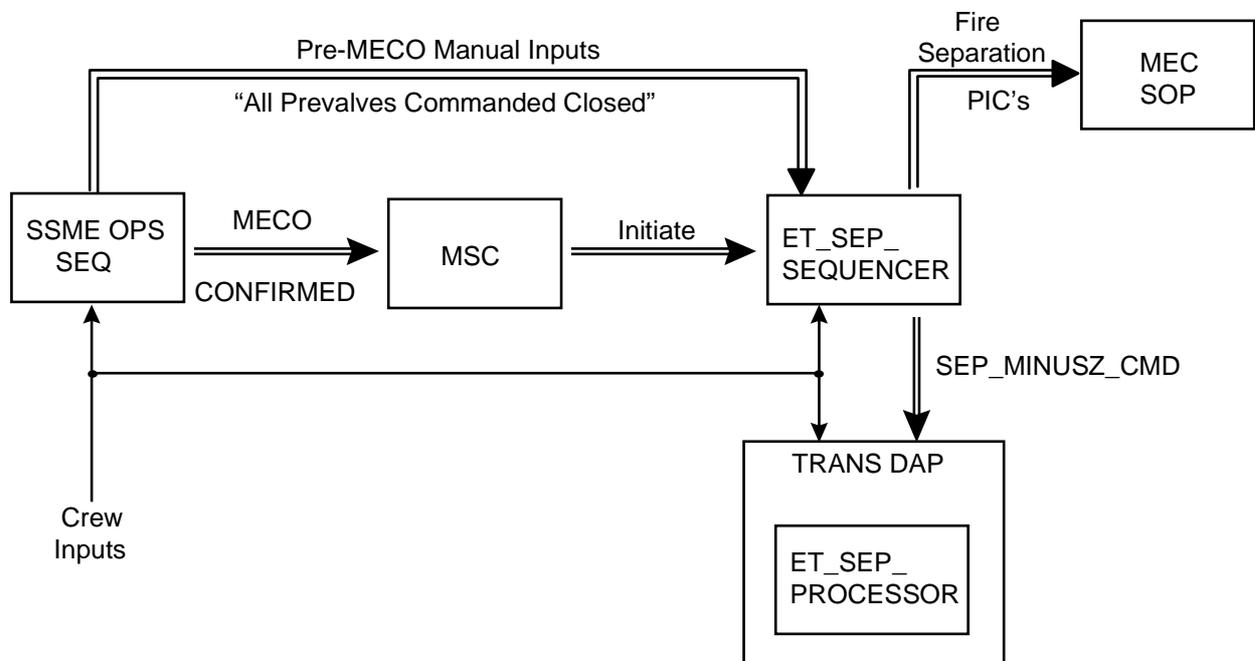


Figure 19-6.- ET separation software interfaces

The final function of the SSME OPS sequencer relating to ET separation is to initiate timers for closing the LO2 and LH2 prevalves following MECO. The timers allow the engines to be shutdown safely by ensuring that the prevalves are closed at the proper time. When these timers expire, the SSME OPS sequencer commands the prevalves closed and sets the "All Prevalves Commanded Closed" discrete. This discrete is required by the ET separation sequencer to perform automatic ET separation correctly.

19.4.2 ET Separation Sequencer

The ET separation sequencer is responsible for achieving physical separation of the Orbiter and ET. It performs this function for nominal separations as well as for all abort cases. The activities it performs during nominal, ATO, AOA, and TAL aborts are essentially the same. RTLS abort activities differ significantly. This module may be active during ascent first stage (MM102), ascent second stage (MM103), OMS 1 insertion (MM104), OMS 2 insertion (MM105), insertion coast (MM106), or during an RTLS (MM601). The sequencer then operates cyclically at 6.25 Hz until the ET structural separation fire commands are issued on nominal, ATO, AOA, and TAL missions or until the umbilical doors are closed and latched on an RTLS.

Numerous functions must be performed to prepare the Orbiter for ET structural separation. All of these functions are accomplished by the ET separation sequencer. It determines if the separation is to be manually inhibited, or the mode of separation if Auto. It closes the feedline disconnect valves, gimbals the SSME nozzles to the proper position, deadfaces the ET/Orbiter interface, and unlatches and retracts the umbilical plates. The sequencer arms the structural separation PICs, performs limit tests on roll, pitch, and yaw body rates, angle of attack, sideslip angle, and tests for feedline disconnect valve closure before continuing with an automatic separation. If no inhibit conditions exist, the sequencer fires the structural separation PICs and issues commands to the Trans DAP to begin the -Z translation away from the tank.

The following discussion provides a step-by-step description of the sequencer as it performs each of these functions. It focuses primarily on activities during a nominal ET/Orbiter physical separation. Some discussion is also provided regarding sequencer activity during manual, fast, and RTLS separations. It may be helpful to refer to information provided previously in section 19.3 (ET Sep Hardware) to understand the activities of the software.

19.4.2.1 Nominal, TAL, ATO, AOA, Automatic ET Orbiter Physical Separation

Physical separation of the Orbiter and ET normally occurs automatically in MM103 and requires no special initialization from the crew. The sequencer proceeds through each of the steps described below to accomplish ET separation. The entire process normally requires approximately 18 seconds. Table 19-I provides a summary timeline of the ET separation process.

It should be noted that the numbers assigned to each of the following steps do not correspond to the step numbers given in the FSSR or DDS. The intention of this discussion is to describe events in the order in which they occur during a nominal separation. The actual software accomplishes this task through fairly complex control logic. Numbers assigned to the steps in the software do not necessary indicate the order in which the steps are executed. Therefore, the steps have been renumbered to sequential order here to reduce the confusion caused by the numbering system used in the onboard software its documentation.

Step 1 - Open Feedline Relief Isolation Valve

The MPS LH2/LO2 feedline relief isolation valves are commanded open. This allows propellants trapped in the 17-inch feedline to reach the feedline relief valves. These valves relieve pressure within the feedline when pressure builds up to the relief valve pressure limit. Pressure can build up in the line when the main engine prevalues and feedline disconnect valves are closed later in the sequence.

Step 2 - Unlock Umbilical Door Centerline Latches

The umbilical door centerline latches are unlocked. This allows for subsequent closure of the umbilical doors.

TABLE 19-I. - ET SEPARATION SUMMARY TIMELINE

| Step | Time (sec) | Sequence of events for nominal separation | Software module |
|------|------------|--|-------------------------------|
| | 0 | MECO command | Guidance |
| | 0.960 | MECO confirmed | SSME ops |
| 1 | 1.079 | ET separation sequence initiated | MSC |
| 2 | | Feedline relief isolation valve opened | ET Sep. Seq. |
| | | Unlock umbilical door centerline latches | ET Sep. Seq. |
| | 4.710 | Terminate ascent DAP, initiate Trans DAP | Trans DAP |
| 3 | 4.196 | "All Prevalves Commanded Closed" discrete set true | SSME ops |
| 4 | | Unlock 17-inch feedline disconnect latches | ET Sep. Seq. |
| | | Arm ET tumble system (if present) | ET Sep. Seq. |
| 5 | 7.317 | Allow prevalves to close | ET Sep. Seq. |
| 6 | | Separation camera ON | ET Sep. Seq. |
| 6 | | Fire Tumble valve system (if present) | ET Sep. Seq. |
| 7 | | Monitor 17-inch feedline latch positions | ET Sep. Seq. |
| 8 | | Close 17-inch feedline disconnect valves | ET Sep. Seq. |
| 9 | | Position main engines | ET Sep. Seq. |
| 10 | | 9.717 | Arm ET umbilical unlatch PICs |
| 11 | 11.317 | Allow ET umbilical unlatch PICs to charge | ET Sep. Seq. |
| 12 | | Deadface electrical connections | ET Sep. Seq. |
| 13 | | Fire ET umbilical unlatch PICs | ET Sep. Seq. |
| 14 | | Fire ET umbilical retract PICs | ET Sep. Seq. |
| 15 | | Open LH2 backup (RTL) dump valves | ET Sep. Seq. |
| 16 | 16.917 | Allow umbilical plate to retract | ET Sep. Seq. |
| 17 | | Arm the structural separation PICs | ET Sep. Seq. |
| 18 | 18.517 | Check automatic separation inhibits | ET Sep. Seq. |
| 19 | | Check for manual overrides | ET Sep. Seq. |
| 20 | | Issue "SEP MINUS Z" command to Trans DAP | ET Sep. Seq. |
| 21 | 18.677 | Orbiter/ET physical separation | ET Sep. Seq. |
| | | Issue "ET SEP CMD" flag | ET Sep. Seq. |
| 22 | | Zero MECs output buffers | ET Sep. Seq. |
| 23 | | (ET Separation Sequencer terminated) | |
| | | Thrusters ON (Four forward and six aft) | Trans DAP |
| | | MPS red and amber status lights turned off | DDS |
| | 25.000 | "SEP COMPLETE" flag | Trans DAP |
| | | MM104 transition | Trans DAP |
| 24 | (varies) | ET Umbilical Door Closure | Crew Action |
| 25 | 240:00 | Close LH2 backup (RTL) dump valves | MPS Dump Seq. |

Step 3 - Monitor for "All Prevalves Commanded Closed"

The ET Sep Sequencer then waits for the "All Prevalves Commanded Closed" discrete to be set by the SSME OPS sequence. At MECO, the SSME OPS sequence initiates timers for closing the LO2 and LH2 prevalves. These times allow for a safe shutdown of the main engines by ensuring that the prevalves are closed at the proper time. When the timers have expired, the prevalves are commanded closed and SSME OPS sets the "All Prevalves Commanded Closed" discrete.

Step 4 - Unlock 17-inch Feedline Disconnect Latches

Once the ET Sep Sequencer receives the "All Prevalves Commanded Closed" discrete, it commands the 17-inch LO2 and LH2 feedline disconnect latches to unlock. This frees the 17 inch feedline disconnect flapper valves in preparation for their closing. The sequencer also sends a command to arm the ET tumble system.

Step 5 - Allow Prevalves to Close

The sequencer pauses 1 second after receiving the "All Prevalves Commanded Closed" discrete to allow time for the prevalves to close and the feedline disconnect latches to unlock. This ensures that the disconnect valves are closed under conditions of no propellant flow.

Step 6 - Enable Separation Camera On/Tumble Valve

The separation camera located on the Orbiter umbilical plate is commanded ON (OV-102 only) and the ET tumble valve enabled command is set (but the tumble valve is no longer flown).

Step 7 - Monitor 17-inch Feedline Latch Positions

The sequence now monitors the 17-inch feedline disconnect latch indications to determine if the latches have closed. A voting scheme is used to determine the latch position.

Each feedline disconnect, LO2 and LH2, has two redundant indicators for the latch locked and unlocked positions; i.e., locked A and B and unlocked A and B. The sequencer looks at each discrete and its associated commfault. It determines if the locked discrettes are OFF and the unlocked discrettes are ON. If any of the indications are commfaulted, the sequencer assumes that the commfaulted system is indicating the locked position. When any three of the four discrettes indicate that the latches are unlocked, the sequencer assumes that the 17 inch feedline disconnect flapper valves are unlatched. The three out of four voting allows the ET Sep Sequence to proceed with one indication failed.

If failures or commfaults cause the sequencer to receive less than three unlocked indications, a manual separation is required. A secondary means of closing the 17 inch feedline disconnect flapper valves is provided during umbilical plate retract, as described in step 14.

Step 8 - Close 17-inch Feedline Disconnect valves

Having confirmed that the 17-inch feedline disconnect valves are unlatched (LH2 and LO2), the sequencer commands these valves to close.

Step 9 - Position Main Engines

The sequencer sends a flag to the MPS TVC CMD SOP to position the SSME nozzles in the MPS dump positions. In the event of a TAL or RTLS abort the main engines are commanded to their entry stow positions. The K-loaded values for the MPS dump and the entry stow positions are currently the same.

Step 10 - Arm ET Umbilical Unlatch PICs

The sequencer waits for 2.3 seconds to expire after the LO2 and LH2 feedline disconnect valves were commanded closed. It then sets the "ET Umbilical Unlatch PIC Arm" Flag. The PIC arm flag signals the MEC to arm the ET umbilical unlatch PICs.

Step 11 - Allow ET Umbilical Unlatch PICs to Charge

The sequencer waits for 1.5 seconds to elapse after it issued the "ET Umbilical Unlatch PIC Arm" command. This pause allows the PICs to become fully charged. The pause also allows a total of 3.8 seconds to elapse after the feedline disconnect valves were commanded closed (3.8 sec = 2.3 sec (step 10) + 1.5 sec (step 11)). The 3.8-second time period is a spec requirement to allow time for the feedline disconnect valves to become fully closed.

Step 12 - Deadface Electrical Connections

The sequencer deadfaces the electrical connectors between the Orbiter and ET. This step involves resetting the ET developmental flight instrumentation (DFI) power and terminating the main propulsion system signal conditioner power.

Step 13 - Fire ET Umbilical Unlatch PICs

The ET umbilical unlatch PICs are fired. Detonation of these PICs severs the three nut/bolt assemblies mating the Orbiter and ET umbilical plates.

Step 14 - Fire ET Umbilical Retract PICs

The ET umbilical retract PICs are fired (energized). Recall that these PICs are not explosive devices. They power solenoid valves that allow hydraulic fluid to flow to the three hydraulic retractors on each umbilical plate. The hydraulic retractors draw the umbilical plates 2.5 inches into the Orbiter. The retraction disconnects the Orbiter/ET electrical umbilical in the first one-half inch of travel. The retract motion also closes the 17-inch feedline disconnect valves if they failed to close when commanded in step 8. The retract motion also releases fluids trapped between the Orbiter and the ET 17-inch disconnect valves. Only two hydraulic actuators are required to retract the umbilical plates successfully.

Step 15 - Open LH2 RTLS Inboard and Outboard (Backup) Dump Valves

The LH2 RTLS inboard and outboard (backup) dump valves are opened to relieve the pressure buildup caused by trapped propellants in the feedline. The dump valves are the primary means of relieving pressure in this line. In step 1 above, the MPS LH2/LO2 feedline relief isolation valve was opened to provide a backup system to the dump valves. If the dump valves do not open, pressure in the LH2 manifold builds to the relief valve setting and the relief valve opens. This prevents the LH2 feedline from rupturing.

Step 16 - Allow Umbilical Plate to Retract

The sequencer then waits 5.5 seconds from when it commanded the ET umbilical retract PICs to fire (step 14). This delay allows time for the LO2 and LH2 umbilical plates to fully retract.

Step 17 - Arm the Structural Separation PICs

The sequencer issues a command to the master events controller (MEC) to arm the ET/Orbiter structural separation PICs. The sequencer pauses 1.5 seconds to allow time for the PICs to become fully charged.

Step 18 - Check Automatic Separation Inhibits

The sequencer checks for automatic separation inhibits, which includes testing the position of the LO2/LH2 17-inch feedline disconnect valves. These valves were commanded closed in step 8 (they should have also been closed mechanically when the umbilical plates retracted in step 14). Each 17 inch disconnect flapper valve is equipped with two position feedbacks. At least one feedback must indicate the closed position, and not be commfaulted, for each flapper valve in order to proceed with automatic separation. The sequencer also checks that roll, pitch, and yaw body rates are less than 0.7°/sec. These rates were established to prevent recontact between the ET and Orbiter. If any of the above tests are not satisfied, the separation is inhibited. An "ET SEP INH" message and a class 3 alert light and tone are annunciated to notify the crew.

Separation cameras are also turned off to conserve film. The sequencer does not proceed until the out-of-tolerance parameter comes back within tolerance or the crew selects to continue separation by manually overriding the inhibit. During an RTLS abort, any inhibits are automatically overridden and ET SEP occurs automatically at MECO plus 13.2 seconds. During a TAL abort, only the disconnect failed open inhibit is overridden automatically (15 second SEP delay). (See also section 19.2.1: ET Separation AUTO/MAN Switch and SEP Push-button) Vehicle rates must be damped manually or coincidentally for separation to proceed.

Step 19 - Check for Manual Overrides

If all automatic separation criteria are satisfied, the sequencer makes a final check of the ET SEP AUTO/MAN switch position. If the switch is in AUTO, it proceeds with the automatic separation. All inhibits and manual override checks are bypassed following this step.

Step 20 - Issue "SEP MINUS Z" Command to Trans DAP

The Sequencer issues the "SEP MINUS Z" command to the Trans DAP. This discrete commands the Trans DAP to begin the Orbiter -Z translation which separates it from the ET. The "SEP MINUS Z" command is issued before actual Orbiter/ET physical separation in order to allow 160 ms for the Trans DAP to configure for the -Z RCS burn.

Step 21 - Orbiter/ET Physical Separation

The sequencer commands the separation camera on. This command is needed in case the camera was turned off earlier because of automatic separation inhibits. It then commands the Orbiter/ET structural separation PICs to fire. This command causes the MEC SOP to fire the separation PICs, separating the Orbiter from the ET. The sequencer also issues a discrete, the "ET SEP CMD" flag, to other software modules signifying that ET separation has occurred. This causes the MPS dedicated display driver sequencer to turn off the MPS red status lights that were turned on at MECO. It also causes the GPCs to initiate steering commands for ET separation. In a TAL abort, it causes the vent door control software to command all vent doors to the closed position to prevent ingestion of propellant during the propellant dump. Finally, a timer is started that prevents the ET umbilical door centerline latches from stowing for 2 seconds after physical separation. This delay prevents any movement of the umbilical doors or latches until the ET is clear of the Orbiter.

Step 22 - Zero MECs Output Buffers

The sequencer commands the MEC SOP to zero the MECs' critical command output buffers and reset all of the non-critical fire 3 commands.

Step 23 - ET Separation Sequencer Terminated

The ET Separation Sequencer is terminated once the ET structural separation fire commands are issued.

Step 24 - ET Umbilical Door Closure

Umbilical door closure normally is accomplished manually by the crew via switches on panel R2 (excluding RTLS and TAL aborts). The doors are closed manually to allow the crew to monitor the door closure process and take appropriate action if a problem is observed. When the crew manually closes the doors, the software described in this step is bypassed.

In case of an AOA, time does not permit the crew to close the doors manually. The ET Sep Sequencer provides the option to close the umbilical doors automatically. The crew commands automatic door closure in OPS 1 by executing an item 20 on the GNC 51 override spec (PASS, refer to Flight Rule A10.1.13-1). When the ET separation sequencer receives this command it executes the following steps.

First, the ET door centerline latches are commanded to stow, assuming the 2-second timer that was started at structural separation has expired. The sequence allows 6 seconds for the latches to release and then commands the left and right ET doors to close. The sequencer waits another 48 seconds to allow time for complete door closure (single-motor time). The left and right uplock latches are then commanded to latch. Twelve seconds are allowed for these latches to close (single-motor time).

There is some risk involved in selecting the automatic umbilical door closure sequence described above. The sequencer issues commands based on nominal motor drive time and does not check or stop for anomalous operation. The potential exists to drive the doors against the centerline latches (if they do not stow properly), damaging the drive mechanism and preventing proper ET door closure. This risk is considered acceptable during an AOA.

In the case of an RTLS or TAL abort, umbilical door closure is accomplished automatically without any commanding required by the crew (or spec item entries). The doors are closed by the Separation Sequencer during an RTLS abort using a sequence similar to the one described above. During a TAL abort, the umbilical doors are closed in MM304 by software contained in the vent door sequencer.

Step 25 - Close LH2 RTLS Inboard and Outboard (Backup) Dump Valves

For OI-22 and subsequent, the backup dump valves are closed by the MPS dump sequence at MECO +240 seconds.

The following is retained for historical purposes:

Prior to OI-22, the ET Sep Sequencer waited for the expiration of an 80-second timer before closing the LH2 RTLS inboard and outboard (backup) dump valves. These valves were opened in step 15 to relieve feedline pressure. The original Orbiter software set the value of this timer to only 23 seconds. It was updated with the OI8C flight software version as a result of CR 89399. The additional time increased system reliability against overpressurization. The CR also revised sequencer logic so that the MEC master reset would be done before expiration of the timer instead of after. This ensures that the MEC is reset before MM304 transition in case of a TAL abort.

19.4.2.2 Manual Separation

The crew has the capability to manually inhibit ET separation at any point or to manually initiate the separation in the presence of automatic inhibits. This capability is provided by the ET SEP AUTO/MAN switch and the SEP push-button or item entries to SPEC 51. This was described earlier in the crew interface section. In the following discussion, it is assumed that manual separation has been declared and one of the following conditions is true:

1. It is MM103 and "MECO Confirmed" has been set
2. It is MM104, 105, or 106

If manual separation is initiated under other than these conditions, a fast separation or RTLS separation is executed.

Before the initiation of manual separation, the ET Separation Sequencer proceeds with automatic separation as described above. Throughout the automatic separation process, the sequencer continuously monitors for manual inputs. When the ET SEP switch is placed in the MAN position, the sequencer commands the ET separation cameras OFF and waits for the SEP push-button. It does not proceed further in the sequence until the push-button is engaged. The cameras are commanded off to conserve film in the event that the automatic sequence had reached a point where the cameras were turned on.

When the SEP push-button is engaged, the sequencer resumes the ET separation process where it was interrupted when the ET SEP switch was taken to MAN. The sequence then proceeds as it would for an automatic separation except for one important difference. The check for automatic separation inhibits is bypassed (step 18 above). The sequence proceeds directly to step 19 and issues the "SEP MINUS Z" command to the Trans DAP. All other sequencer activities proceed as described above for the automatic separation.

19.4.3 Transition Digital Autopilot

The Trans DAP is one of six digital autopilots that provide flight control throughout the numerous flight regimes of the shuttle. The primary responsibility of the Trans DAP is to provide control logic during insertion and deorbit phase of flight. For a detailed description of the Trans DAP see the Transition Digital Autopilot Systems Brief in the GNC's Systems Briefs Book.

The first duty of the Trans DAP when it assumes Orbiter control following MECO is to perform ET separation. The Trans DAP is active during a NOM, TAL, AOA, and ATO ET separation. It serves three major functions in the separation process. First, it maintains Orbiter rate and attitude deadbands throughout the various phases of the separation. Second, it executes the Orbiter -Z translation, assuming no automatic inhibits or crew overrides. Third, it monitors vehicle rates during separation and issues the "SEP COMPLETE" flag, signifying completion of the separation process.

The separation process can be divided into two distinct phase, the mated coast phase and the -Z translation phase. The -Z translation represents the only automatic reaction control system (RCS) translation of the mission. Each phase of separation is described in detail below.

19.4.3.1 Mated Coast

The Trans DAP assumes Orbiter control from the ascent DAP following issuance of the "MECO Confirmed" flag by the SSME OPS sequencer. When the ascent DAP receives the "MECO Confirmed" flag, it initiates an attitude hold. The Trans DAP maintains this attitude hold when it assumes Orbiter control 3.75 seconds later. This begins the mated coast phase. Attitude and rate deadbands during this period are +/-10.0 degrees and +/-0.5 degree/sec, respectively.

While the Trans DAP maintains the Orbiter within the specified deadbands, the separation sequencer performs the functions described in the previous section. The separation sequencer nominally requires approximately 18 seconds to complete its functions, assuming no automatic inhibits or crew overrides. Throughout the mated coast period, the Trans DAP monitors for the "SEP_MINUSZ_CMD" from the separation sequencer. This flag is the signal from the sequencer that it has completed preparations for ET/Orbiter physical separation. Upon receiving this flag, the Trans DAP initiates the -Z translation.

19.4.3.2 -Z Translation

When the Trans DAP receives the "SEP_MINUSZ_CMD" from the separation sequencer, two events occur. First, the Trans DAP takes a snapshot of the Orbiter attitude and maintains that attitude throughout the -Z translation. Second, commands are issued to the RCS jets to begin firing for the -Z translation. The jet commands come from a dedicated module within the Trans DAP called the ET_SEP_PROCESSOR. This module commands firing of 10 -Z translational jets, four forward and six aft. The jets begin to fire at the same time the ET separation sequencer commands the ET structural separation PICs to detonate. These PICs shear the nut/bolt assemblies mating the Orbiter and ET. A minimum number of thrusters is required in each thruster group to ensure that Orbiter/ET recontact does not occur during the translation. Flight Rule A8.1.2-5 provides guidelines for the minimum number of RCS jets required.

At initiation of the -Z translation, attitude control is inhibited for 2.96 seconds. This is done to prevent attitude control jet activity from reducing Orbiter translation rates during the critical period immediately following physical separation. The attitude hold is implemented by temporarily severing communication between attitude control modules in the Trans DAP and the RCS jets. This allows attitude control modules to continue running during the 2.96-second inhibit and prevents the need for system reinitialization at resumption of attitude control. Attitude control is resumed by re-enabling communication between control module and thrusters.

At the end of the attitude control inhibit period, the Trans DAP maintains attitude deadbands of $+3.0^\circ$ and rate deadbands of $+0.50/\text{sec}$. Throughout the translation, the DAP does not fire any jets in opposition to the -Z jets. Trans DAP jet select logic prevents up-firing jets from being commanded while a -Z translation is commanded. To correct errors in pitch, down-firing -Z translation jets are turned off.

Typically, the Trans DAP turns off two forward down-firing jet immediately following the 2.96-second attitude control inhibit to keep the Orbiter within its pitch rate deadband. Analysis shows that firing the 10 -Z translation jets for 2.96 seconds causes a positive pitch rate of approximately 0.5 deg/sec . The rate results from the unbalanced moment arms of the forward and aft jets.

As the -Z translation proceeds, the ET_SEP_PROCESSOR monitors Orbiter translation rates. It compares the Orbiter translation rate it receives from the ascent/entry attitude processor to an I-loaded value when the -Z translation rate reaches this value, the ET_SEP_PROCESSOR issues the "SEP_COMPLETE" flag. The current value of this I-load is 4 ft/sec for a NOM, AOA, or ATO separation and 11 ft/sec for a TAL abort separation.

The "SEP COMPLETE" flag signifies termination of the ET_SEP_PROCESSOR and notifies other Trans DAP software modules that separation is complete. The Trans DAP then automatically modes to MM104 and configures itself for the coming OMS burns. Automatic

transition to MM104 is the crews indication that the separation process is complete. Post-ET separation attitude deadbands are +3.5 degrees and rate deadbands are +0.5°/sec in all axes.

19.4.3.3 Manual -Z translation

The crew normally allows the -Z translation to proceed automatically. However, they have the capability to assume manual control of the translation at any time. This may become necessary in certain failure scenarios or to execute a detailed test objective (DTO). Manual control is activated by moving the translational hand controller (THC) out of detent. It is not reversible. Taking manual control causes the ET_SEP_PROCESSOR to terminate its -Z translation jet commands. However, it continues to monitor vehicle rates. It is the crew's responsibility to see that the I-loaded -Z translation rate is achieved. When the Orbiter reaches this rate, the ET_SEP_PROCESSOR issues the "SEP_COMPLETE" flag and the Trans DAP automatically modes to MM104.

Taking the rotational hand controller (RHC) out of detent or moding the DAP to MAN via the DAP PBI panel, before or during the -Z translation, does not prevent the -Z burn from occurring automatically.

If the crew does not achieve the I-loaded -Z translation rate, the ET_PROCESSOR does not issue the "SEP_COMPLETE" flag. There are two consequences of this. First, the Trans DAP does not automatically transition to MM104. The crew has to manually execute the transition. Second, the impact is seen in the deadband maintained by the Trans DAP. Attitude deadbands of 3° remain in effect instead of the 3.5 degree deadbands normally in effect following ET separation. In such cases, the "SEP_COMPLETE" flag is issued when an OMS engine is commanded to fire or OPS 1 is exited.

19.4.3.4 Fast Sep

A fast separation is a contingency separation procedure that greatly reduces the time between ET SEP CMD and structural separation. A fast separation is performed if the ET SEP PB is depressed during MM102 or 601, or in MM103 with 2 SSMEs failed.

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| | |
|--|-----------|
| GLOBAL POSITIONING SYSTEM..... | 4 |
| 20.1 INTRODUCTION..... | 4 |
| 20.2 GPS SYSTEMS OVERVIEW..... | 4 |
| 20.3 MAGR-S3S | 9 |
| 20.3.1 <i>MAGR-S3S Schematics</i> | 10 |
| 20.3.2 <i>Radiation Protection</i> | 10 |
| 20.3.2.1 Results and Conclusions from the Proton Testing..... | 11 |
| 20.3.3 <i>Battery Operations</i> | 11 |
| 20.3.4 <i>GPS Encryption</i> | 12 |
| 20.3.4.1 Zeroize Switch..... | 12 |
| 20.3.4.2 MAGR-S3S Memory..... | 13 |
| 20.3.4.3 Pre-Launch Encrypt Key Loading Process..... | 14 |
| 20.3.5 <i>Power Input</i> | 14 |
| 20.3.6 <i>GPS Operations</i> | 14 |
| 20.3.6.1 GPS Operational Modes..... | 15 |
| 20.3.6.2 GPS Channel Operations / Peripheral Channel..... | 16 |
| 20.3.6.3 GPS Tracking States and C/No..... | 17 |
| 20.4 SHUTTLE GPS IMPLEMENTATION..... | 18 |
| 20.4.1 <i>Orbiter Signal Environment</i> | 18 |
| 20.4.2 <i>Shuttle Operations</i> | 19 |
| 20.4.3 <i>GPS Redundancy Management / QA Checks</i> | 20 |
| 20.4.3.1 QA Override | 21 |
| 20.4.3.2 QA 1 - FOM | 22 |
| 20.4.3.3 QA 2 - GPS to Nav Compare | 22 |
| 20.4.3.4 QA 3 - GPS to Last Cycle Compare..... | 22 |
| 20.4.3.5 QA 4 - GPS to GPS Compare..... | 22 |
| 20.4.3.6 QA Fault Summary Messages | 23 |
| 20.4.3.7 GPS QA and GAX I-Loads | 23 |
| 20.4.4 <i>GPS SOP</i> | 25 |
| 20.4.4.1 GPS Data Processing..... | 25 |
| 20.4.4.2 GPS Serial Data Messages..... | 26 |
| 20.4.5 <i>Shuttle GPS Configuration / Crew Interface</i> | 33 |
| 20.4.5.1 Single String GPS Configuration | 33 |
| 20.4.5.1.1 Single String GPS Switch Configuration | 33 |
| 20.4.5.1.2 Single-String GPS Installation | 34 |
| 20.4.5.2 Single String GPS in PASS | 35 |
| 20.4.5.2.1 PASS Horizontal Situation Display (SPEC 50)..... | 35 |
| 20.4.5.2.2 PASS GPS Status Display (SPEC 55) | 37 |
| 20.4.5.3 Single String GPS in BFS | 48 |
| 20.4.5.3.1 BFS Horizontal Situation Display (SPEC 50) | 48 |
| 20.4.5.3.3 BFS GPS Status Display (SPEC 55)..... | 49 |
| 20.4.5.4 Three String GPS Configuration | 53 |
| 20.4.5.4.1 Three String GPS Switch Configuration..... | 53 |
| 20.4.5.4.2 Three String GPS Displays | 58 |
| 20.4.6 <i>MCC Monitoring of GPS Data</i> | 61 |
| 20.4.6.1 Single String Displays | 61 |
| 20.4.6.2 Three String Displays | 65 |
| A: GLOSSARY OF GPS TERMS..... | 69 |
| B: IN FLIGHT ANOMALY LOG | 72 |
| C: NOTES OF INTEREST | 73 |
| <u>GPS PRT USER NOTES:</u> | 73 |
| <u>MAGRS-3S DRS (LINK-007)</u> | 76 |
| <u>PASS USER NOTES</u> | 78 |
| D: REFERENCES | 87 |

| | |
|--|----|
| FIGURE 1 MAJOR SEGMENTS OF THE NAVSTAR GLOBAL POSITIONING SYSTEM..... | 5 |
| FIGURE 2 GPS POSITION SOLUTION COMPUTATION | 7 |
| FIGURE 3 MAGR-S3S CABLE CONNECTIONS (FRONT OF BOX)..... | 10 |
| FIGURE 4 MAGR-S3S BATTERY LOCATION (BACK OF LRU)..... | 11 |
| FIGURE 5 GPS SATELLITE MASK ANGLE | 19 |
| FIGURE 18 GPS SELECTION FILTER CANDIDATE LOGIC | 21 |
| FIGURE 6 GPS CONFIGURATION (SINGLE-STRING FLIGHTS) | 34 |
| FIGURE 7 HORIZONTAL SITUATION DISPLAY (SPEC 50) FOR THREE TACAN & ONE GPS..... | 35 |
| FIGURE 8 PASS GPS STATUS DISPLAY (SPEC 55)..... | 37 |
| FIGURE 9 GPS AUTO/INHIBIT/FORCE FUNCTIONS FOR TRIPLE-STRING GPS | 46 |
| FIGURE 10 BFS HORIZONTAL SITUATION (SPEC 50)..... | 49 |
| FIGURE 11 BFS GPS STATUS DISPLAY (SPEC 55)..... | 50 |
| FIGURE 12 THREE-STRING GPS HARDWARE LOCATIONS..... | 54 |
| FIGURE 13 THREE-STRING GPS CONFIGURATION PANEL LAYOUT | 55 |
| FIGURE 14 THREE-STRING GPS WIRING CONFIGURATION..... | 56 |
| FIGURE 15 THREE-STRING GPS DATA FLOW AND POWER MATRIX..... | 57 |
| FIGURE 16 HORIZONTAL SITUATION DISPLAY (SPEC 50) 3 GPS – NO TACAN | 58 |
| FIGURE 17 BFS HORIZONTAL SITUATION DISPLAY (SPEC 50) FOR THREE-STRING GPS | 59 |
| FIGURE 19 PASS GPS STATUS DISPLAY (OLD)..... | 61 |
| FIGURE 20 PASS GPS 2 STATUS DISPLAY (NEW) | 61 |
| FIGURE 21 GPS TO PASS NAVIGATION COMPARE DISPLAY (OLD)..... | 62 |
| FIGURE 22 GPS TO PASS NAVIGATION COMPARE DISPLAY (NEW) | 62 |
| FIGURE 23 GPS STATUS MINI DISPLAY | 62 |
| FIGURE 24 GPS QUICK DISPLAY..... | 63 |
| FIGURE 25 GPS SATELLITE DESELECT | 63 |
| FIGURE 26 PPS STATUS DISPLAY | 64 |
| FIGURE 27 BFS GPS STATUS DISPLAY..... | 64 |
| FIGURE 28 BFS GPS COMPARISON DISPLAY | 65 |
| FIGURE 29 GPS STATUS DISPLAY - THREE STRING | 65 |
| FIGURE 30 BFS GPS STATUS DISPLAY – THREE STRING..... | 66 |
| FIGURE 31 SENSORS THREE-STRING DISPLAY | 66 |
| FIGURE 32 BFS SENSORS THREE-STRING DISPLAY..... | 67 |
| | |
| TABLE 1 GPS CODE COMPARISON..... | 6 |
| TABLE 2 GPS FIGURE OF MERIT (FOM)..... | 8 |
| TABLE 3 TIME FIGURE OF MERIT | 9 |
| TABLE 4 DEFINITION OF PPS_STATUS_WORD - MSID V74M4459P (1459P), (3459P) FOR GPS 1, (2), (3) | 12 |
| TABLE 5 MAGR-S3S I/O MEMORY | 13 |
| TABLE 6 MAGR-S3S NAV MEMORY | 14 |
| TABLE 12 GPC TO GPS DATA DESCRIPTION | 26 |
| TABLE 13 GPS TO GPC DATA ONE (I) CONTENTS..... | 26 |
| TABLE 14 NAV MODE WORD RCVR (I)..... | 27 |
| TABLE 15 GPS STATUS FOM GDOP RCVR (I)..... | 29 |
| TABLE 16 CHANNEL STATUS (I) WORD 1 | 29 |
| TABLE 17 CHANNEL STATUS (I) WORD 2 | 29 |
| TABLE 18 CHANNEL STATUS (I) WORD 3 | 29 |
| TABLE 19 CHANNEL STATUS (I) WORD 4 | 30 |
| TABLE 20 CHANNEL STATUS (I) WORD 5 | 30 |
| TABLE 21 CHANNEL STATUS (I) WORD 6 | 31 |
| TABLE 22 CHANNEL STATUS (I) WORD 7 | 31 |
| TABLE 23 CHANNEL STATUS (I) WORD 8 | 31 |
| TABLE 24 CHANNEL STATUS (I) WORD 9..... | 31 |
| TABLE 25 GPC TO GPS DATA ONE MESSAGE DESCRIPTION | 32 |
| TABLE 26 GPC TO GPS DATA TWO MESSAGE | 32 |
| TABLE 27 GPS CONTROL WORD..... | 33 |

| | |
|--|----|
| TABLE 7 GPS GAX MESSAGES..... | 47 |
| TABLE 8 BFS GPS INDICATION..... | 50 |
| TABLE 9 VALID BFS GPS STATUS OUTPUTS (STAT)..... | 50 |
| TABLE 10 VALID BFS GPS RECEIVER HARDWARE MODES | 51 |
| TABLE 11 BFS GPS FIGURE OF MERIT (FOM)..... | 52 |

GLOBAL POSITIONING SYSTEM

20.1 INTRODUCTION

To replace the TACAN navigation aid function, NASA decided to use the DoD's NAVSTAR Global Positioning System (GPS) to provide the necessary entry navigation function currently provided by TACANs; it was originally planned that all vehicles would have TACANs replaced by GPS, however because of the planned retirement of the Shuttle fleet in 2010, only OV-105 will be equipped with three GPS receivers (known as three-string). OV-103 and OV-104 will continue to retirement using three TACAN receivers, and both are also equipped with a single-string GPS receiver.

Space Shuttle GPS implementation became more complicated following the Program decision to use GPS for much more than just a TACAN replacement. Unlike TACAN stations, which are fixed to a point on the earth, the GPS satellite constellation provides worldwide coverage, and as a result can provide navigation information for all phases of flight. This can minimize or even eliminate the need for ground tracking for what are today routine state vector updates to the Orbiter. However, due to funding and Shuttle design restrictions, it was decided to limit the Phase A and Phase B GPS implementations to the Space Shuttle orbit and entry phases of flight. The ascent phase of flight was considered, but it was decided against due to limitations in the AP101S GPCs.

For the purpose and scope of this document based on the current GPS implementation, assume the following:

GPS is to be used as a TACAN replacement for entry on OV-105 only;

GPS is available for On-Orbit use, but not normally used to update navigation;

Even though GPS performance has been shown to be better than TACAN, it is not considered a Precision Landing System (continue to use MLS);

GPS can help to reduce and/or eliminate the need for state vector updates during Orbit phase, however it is unlikely that it will be used for this during normal operations;

As currently implemented in the Shuttle Navigation architecture, GPS is not considered adequate for use as a Relative Nav sensor for Rendezvous operations (continue to use Star Tracker, COAS, and Rendezvous Radar).

20.2 GPS SYSTEMS OVERVIEW

(Section 20.2 contains GPS theory of operation, with a Glossary of Terms located in Appendix A. Space Shuttle GPS implementation begins in Section 20.4).

The NAVSTAR Global Positioning System is a space-based radio positioning navigation system that provides three-dimensional position, velocity and time information to suitably equipped users anywhere on or near the surface of the earth. The GPS constellation consists of 24+ NAVSTAR satellites operating in 10,980 NM orbits in six planes, an inclination of 55 degrees with respect to the equator, and at orbital velocities of 14,500 km/hr. As of 12/06 the GPS Satellite Constellation consisted of 31 operational satellites. The NAVSTAR satellites are in 12 hr orbits. Because of the orbital parameters of the satellites, the visible constellation over any point on the earth repeats roughly every 12 hours, however there is some slow drift due to the motion of the earth about the sun (approximately 4 minutes per day). The Department of Defense (DoD) finances NAVSTAR satellites and their support equipment, but their navigation signals are available free of charge to anyone. The NAVSTAR satellite constellation control center facility is located at Shriver Air Force Base in Colorado Springs, Colo.

MAJOR SEGMENTS OF THE GPS SYSTEM

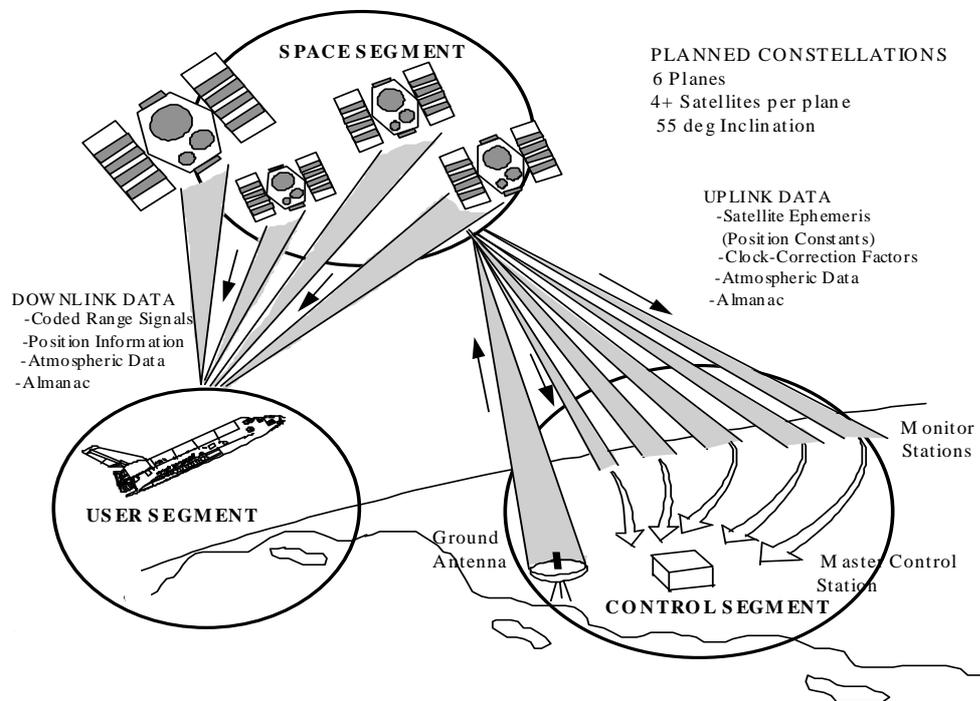


Figure 1 Major Segments of the NAVSTAR Global Positioning System

The Navstar Global Positioning System can be broken into three major pieces or segments:

The Space Segment, which consists of Navstar satellites plus active on-orbit spares

The User Segment, which consists of military and civilian receivers

The Control Segment, which consists of five unmanned monitor stations, a master control station, and a set of ground antennas installed in widely separated locations around the globe.

GPS navigation utilizes a technique called pseudo-ranging, where the pseudorange is the time shift required to align (correlate) a replica of the GPS code generated in the receiver with the received GPS code; this time shift can be scaled into distance by the speed of light. The time shift is the difference between the time of the signal reception (measured in the receiver time frame) and the time of emission (measured in the satellite time frame); if the receiver and satellite clocks are perfect this is the true range from satellite to receiver antenna. This range is called the pseudorange because the clocks are never perfect and it therefore is not the true range. During calculation of the receiver position, the clock bias is solved for allowing an accurate position solution to be computed despite having inaccurate measurements of the range to the satellites being tracked (this does however require the introduction of another variable into the equations and therefore necessitate four simultaneous pseudorange measurements to solve for the three-dimensional position).

GPS availability around the globe has approached 100% with a 24+ satellite constellation, and provides continuous worldwide coverage from +/- 75° latitude (with a Position Dilution of Precision (PDOP) ≤ 6). It is expected to be 98% reliable, have a continuous fix rate, unlimited system capacity, and no potential for ambiguity.

GPS has two main signals that it sends out for reception, PPS for Precise Positioning System and SPS for Standard Positioning System. PPS is encrypted and limited to a specific user set as defined by the DoD, while SPS is available to anyone. The Shuttle fleet utilizes PPS for navigation but is also certified to use SPS.

P-Code, or Precision code, is a very long sequence of pseudorandom binary modulations on the GPS carrier at a rate of 10.23 MHz; the P-code sequence does not repeat itself for about 267 days, however it is reset once a week so the entire sequence is never broadcast continuously. Each one-week segment of the P-code is unique to one GPS satellite. Encrypted P-code carrier signals may be accessed if the receiver contains encrypted PPS capability and if the receiver is "keyed". If the receiver is keyed, then the P-code accessed internal to the GPS receiver becomes Y-code (see Table 1 for code comparison accuracies).

Table 1 GPS Code Comparison

| GPS Accuracy (3σ) | |
|-------------------|---|
| PPS | Horiz - 21m Vert - 29m Time - 200ns |
| SPS | Horiz - 100m Vert - 140m Time - 340ns |

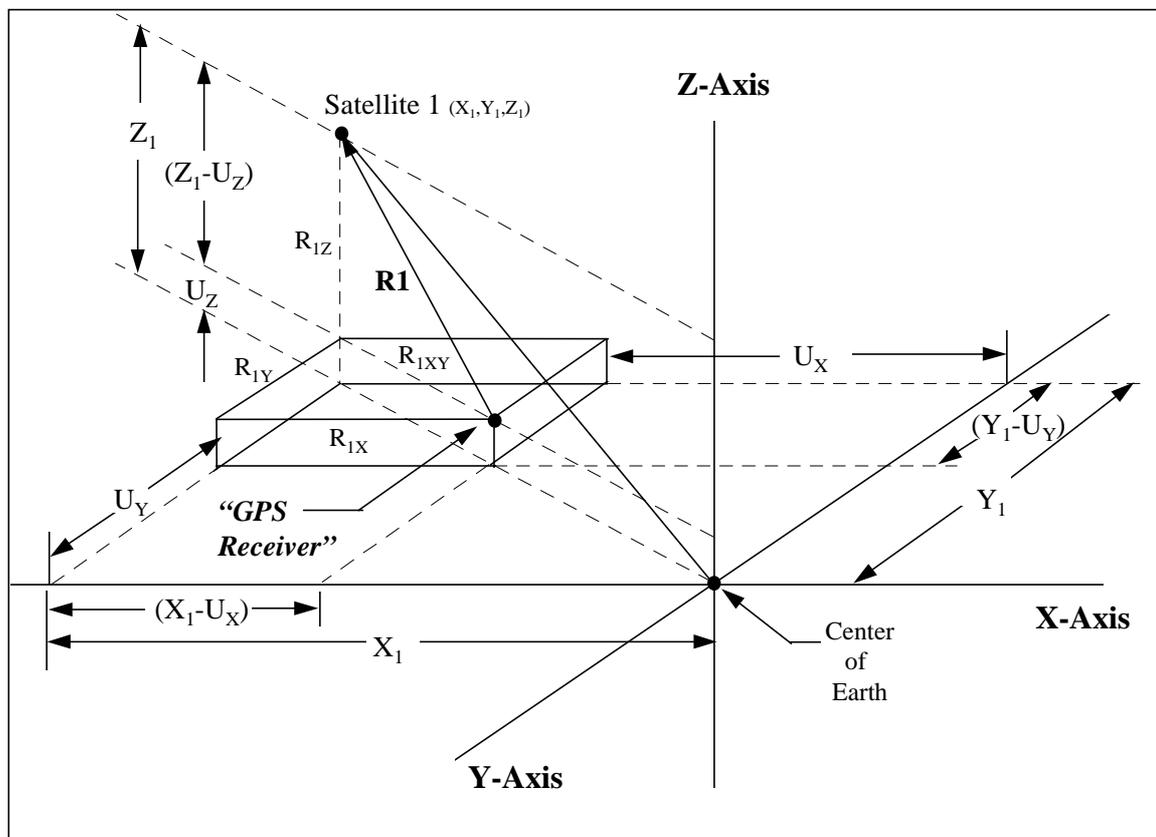
C/A code, or Coarse/Acquisition code, is a sequence of 1023 pseudorandom binary biphase modulations on the GPS carrier broadcast at a rate of 1.023 MHz, thus having a code repetition period of one millisecond (as opposed to the 267 day P-code). C/A code is available to all users who have a GPS receiver, including commercial, private, and military users. C/A code however, is subject to Selective Availability, which may be turned on by the DoD at any time. If Selective Availability (SA) is turned on, the DoD may introduce small errors into the C/A code that result in small user position and/or velocity navigation errors. Currently (as of 12/06), selective availability is disabled indefinitely, however the DoD reserves the right to activate it at any time. When SA is off, SPS accuracy is quite good and in fact the most accurate GPS receivers in the world are civilian survey-grade receivers which track SPS code and PPS carrier signal (without the need for encryption keys).

The GPS satellite constellation sends out carrier signals on two frequency bands L1 and L2, at 1575.42 MHz and 1227.6 MHz respectively. The two L-band frequencies provide spherical ranging information (3D position and 3D velocity, with a time tag). The GPS satellites also have an S-band uplink operating at 2227.5 MHz, and a downlink operating at 1783.74 MHz. The uplink data contains satellite ephemeris (position) constants, as well as clock correction factors, atmospheric data, and almanac data; all of which are essential to maintain accurate position, velocity and clock information aboard the satellites. The S-band uplink and downlink are not used by GPS users, only the DOD control segment. The L1 band contains the C/A-code and P-code data along with some system data. The L2 band contains either C/A-code plus system data, or P-code plus system data, or P-code only (no system data). Typically L2 operates using P-code plus system data.

To calculate a navigation state vector, a GPS PPS receiver acquires the GPS downlinked code signals on frequencies L1 and/or L2 and computes the user's position coordinates. Since there are four unknowns in the position equations (three position and clock bias), the receiver must acquire at least four satellites. The basic equations used to compute the pseudorange are shown below (See Figure 2 for a geometric representation of GPS Receiver position).

Calculation of GPS Receiver Position Coordinates

$$\begin{aligned} (X_1 - U_X)^2 + (Y_1 - U_Y)^2 + (Z_1 - U_Z)^2 &= (R_1 - C_B)^2 \\ (X_2 - U_X)^2 + (Y_2 - U_Y)^2 + (Z_2 - U_Z)^2 &= (R_2 - C_B)^2 \\ (X_3 - U_X)^2 + (Y_3 - U_Y)^2 + (Z_3 - U_Z)^2 &= (R_3 - C_B)^2 \\ (X_4 - U_X)^2 + (Y_4 - U_Y)^2 + (Z_4 - U_Z)^2 &= (R_4 - C_B)^2 \end{aligned}$$



$$(R_1)^2 = (R_{1x})^2 + (R_{1y})^2 + (R_{1z})^2$$

$$(R_{1x}) = (X_1 - U_x)$$

$$(R_{1y}) = (Y_1 - U_y)$$

$$(R_{1z}) = (Z_1 - U_z)$$

$$(R_1)^2 = (X_1 - U_x)^2 + (Y_1 - U_y)^2 + (Z_1 - U_z)^2$$

Figure 2 GPS Position Solution Computation

In the above equations, X_1, Y_1, Z_1 represents the known position of satellite 1; X_2, Y_2, Z_2 represents the known position of satellite 2; X_3, Y_3, Z_3 represents the known position of satellite 3; X_4, Y_4, Z_4 represents the known position of satellite 4; and U_x, U_y, U_z represents the unknown position of the user. The range to satellite 1 (R_1) is equal to the speed of light (c) times the time it took for the carrier signal to get to the user (ΔT_1). The clock bias (C_B) is the difference in time between the GPS satellite constellation's extremely accurate atomic clocks, and the clock in the user's GPS receiver (typically less accurate). When at least four satellites have been acquired, it is easy to see that the series of four equations with four unknowns may be solved to determine the user's position (U_x, U_y, U_z) and clock bias (C_B). Most modern receivers can track all satellites in view (sometimes a dozen or more) and this will result in an overdetermined solution to the above equations which greatly reduces the effects of measurement errors; the MAGR/S-3S on the Shuttle cannot perform navigation with more than four satellites so it does not benefit from this.

To calculate velocity, a GPS Receiver uses the equations of motion to calculate the Doppler shift. Again, since there are four unknowns in the velocity equations, the receiver must lock on to at least four satellites. The first three satellites are required to compute each component of velocity (U_x, U_y , and U_z), and the fourth satellite is required to calculate the clock drift C_B . The basic equations used to compute the user velocity are shown below:

Calculation of GPS Receiver Velocity Components

$$\begin{aligned} (X_1 - U_X) (\underline{X}_1 - \underline{U}_X) + (Y_1 - U_Y) (\underline{Y}_1 - \underline{U}_Y) + (Z_1 - U_Z) (\underline{Z}_1 - \underline{U}_Z) &= R_1(\underline{R}_1 + \underline{C}_B) \\ (X_2 - U_X) (\underline{X}_2 - \underline{U}_X) + (Y_2 - U_Y) (\underline{Y}_2 - \underline{U}_Y) + (Z_2 - U_Z) (\underline{Z}_2 - \underline{U}_Z) &= R_2(\underline{R}_2 + \underline{C}_B) \\ (X_3 - U_X) (\underline{X}_3 - \underline{U}_X) + (Y_3 - U_Y) (\underline{Y}_3 - \underline{U}_Y) + (Z_3 - U_Z) (\underline{Z}_3 - \underline{U}_Z) &= R_3(\underline{R}_3 + \underline{C}_B) \\ (X_4 - U_X) (\underline{X}_4 - \underline{U}_X) + (Y_4 - U_Y) (\underline{Y}_4 - \underline{U}_Y) + (Z_4 - U_Z) (\underline{Z}_4 - \underline{U}_Z) &= R_4(\underline{R}_4 + \underline{C}_B) \end{aligned}$$

As in the position equations from above, X_1, Y_1, Z_1 represents the known position of satellite 1; X_2, Y_2, Z_2 represents the known position of satellite 2; and so on. The range to satellite 1 (R_1) is equal to the speed of light (c) times the time it took for the carrier signal to get to the user (ΔT_1). The clock bias (C_B) is the time drift between the GPS satellite constellation's extremely accurate atomic clocks, and the clock in the user's GPS receiver (typically this drift is almost entirely due to the receiver's drift).

The satellite velocity components for satellite 1 are $\underline{X}_1, \underline{Y}_1, \underline{Z}_1$; the velocity components for satellite 2 are $\underline{X}_2, \underline{Y}_2, \underline{Z}_2$; and so on. The frequency shift is defined in the equations as \underline{C}_B . The range rates (\underline{R}) are defined as the change in frequency ($\Delta\phi$) divided by the frequency received (ϕ) times the speed of light (c). When at least four satellites have been acquired, and the position is known from solving the previous set of equations, it is easy to see that the series of four equations with four unknowns may be solved to determine the user's velocity ($\underline{U}_X, \underline{U}_Y, \underline{U}_Z$) and frequency shift (\underline{C}_B).

Once the GPS derived position, velocity, and time components of the state vector are computed, the state vector is output from the receiver in WGS-84 (Worldwide Geodetic System, 1984) coordinates. The Shuttle GPC's then take the WGS-84 state information from the GPS, and convert it to the M-50 (Mean of 1950) coordinate frame for use by the onboard navigation software.

Each GPS data stream is composed of 1500 bits of data, which takes approximately 12.5 minutes before the entire message repeats. Each data stream contains telemetry words, handover words (HOW), clock correction factor, ephemeris constants, and almanac/health status. For PPS use, a "HOW" (Hand Over Word) is employed in the telemetry that matches an address of the PPS signal, in turn allowing use of the PPS signal for high precision calculation of the user position, velocity, and time. Thus, a keyed PPS receiver could take as long as 12.5 minutes to begin using PPS signals for state determination. Prior to downloading the HOW, it will operate in SPS mode despite being keyed.

In addition to computing a state vector, the GPS receivers also generate a number of other parameters that reflect the quality of the state vector calculations. These parameters can then allow the user to weigh the state vector data based on the amount of error present in the GPS state vector calculation. GPS parameters that provide insight into the quality of the data include the Figure of Merit (FOM), and the Geometric Dilution of Precision (GDOP).

The FOM, also called QA1 (for Quality Assessment 1) in the Shuttle implementation, is a measure of total 3D-position error that can be expected in the GPS derived state vector. The FOM is calculated internally by the MAGR and it combines the expected geometric errors (GDOP) and the estimated MAGR Kalman filter errors. The FOM is a scaled integer, ranging from 1 (smallest error) to 9 (largest error). Table 1 contains values of FOM as output from the GPS receivers (the FOM is generally a very conservative estimate and actual error is usually much lower).

Table 2 GPS Figure of Merit (FOM)

| FOM (integer) | Corresponding Error (m) | Feet |
|---------------|-------------------------|---------------|
| 1 | < 25 | < 75 |
| 2 | 25 to 50 | 75 to 150 |
| 3 | 50 to 75 | 150 to 225 |
| 4 | 75 to 100 | 225 to 300 |
| 5 | 100 to 200 | 300 to 600 |
| 6 | 200 to 500 | 600 to 1500 |
| 7 | 500 to 1000 | 1500 to 3000 |
| 8 | 1000 to 5000 | 3000 to 15000 |

| | | |
|---|--------|---------|
| 9 | > 5000 | > 15000 |
|---|--------|---------|

The Geometric Dilution of Precision (GDOP) is a measure of the purely geometrical contribution to the uncertainty in a dynamic position fix for the three position coordinates plus clock offset in the navigation solution. GDOP is a scaled integer ranging from 1 to 15, where a lower value indicates the most favorable satellite geometry and 15 the least. GDOP is a function of the Line of Sight (LOS) or look angles to the satellites in view. The closer the satellites are to 90 degrees separation, the lower the GDOP, and the smaller the errors introduced due to geometry. A high GDOP indicates poor satellite geometry and as a result errors can be introduced into the navigation solution. A good GDOP means a favorable geometric separation of satellites (ideally at least 4 satellites separated by at least 60 degrees).

NOTE: Due to line of sight restrictions, the lowest GDOP attainable is ~1.5, which results in the integer 2 on crew and ground displays.

Table 3 Time Figure of Merit

| TFOM | Estimated Time Error | Equivalent Pseudo-Range Bias |
|------|----------------------------|------------------------------|
| 1 | < 1 nanosecond | < 0.3 meters |
| 2 | 1 to 10 nanoseconds | 0.3 – 3 meters |
| 3 | 10 to 100 nanoseconds | 3 – 30 meters |
| 4 | 100 to 1000 nanoseconds | 30 – 300 meters |
| 5 | 1 to 10 microseconds | 300 – 3000 meters |
| 6 | 10 to 100 microseconds | 3 – 30 kilometers |
| 7 | 100 to 1000 microseconds | 30 – 300 kilometers |
| 8 | 1 to 10 milliseconds | 300 – 3000 kilometers |
| 9 | > 10 milliseconds or fault | > 3000 kilometers |

NOTE: Based on Flight Data, FOM and TFOM are highly correlated; TFOM is not expected to be below 3 in the MAGR-S3S

20.3 MAGR-S3S

The specs on the Collins Avionics manufactured Miniature Airborne GPS Receiver- 3 String for Shuttle (MAGR-S3S) are as follows:

Size: 3.21"W x 6.78"H x 12.0"D

Weight: 13 lb.

Location: GPS1 (Avionics Bay 1): GPS2 (Avionics Bay 3B): GPS3 (Avionics Bay 3A)

User Interface – Multiple Serial Data ports. MDM Serial data interface for use with shuttle’s flight critical MDMs. 1553 serial data port for lab testing via PGSC laptop interface.

Operational Startups are broken into 3 categories:

Cold Start: If the receiver has lost primary and battery power or never been powered on, then the receiver uses the default Almanac stored on EEPROM. The receiver does not contain a current almanac, ephemerides, or encrypt key. Time to download an updated almanac, to lock 4 satellites and have a nav solution will be 20-30 minutes.

NOTE: A loss of NVRAM (Non-Volatile RAM) data due to loss of primary and battery power will require a Cold Start of the receiver regardless of the length of the power cycle.

Warm Start: The MAGR has a current almanac, but no current ephemerides. It may or may not contain current encrypt key information. Any power off period of longer than 6 hours will require the receiver to

download the ephemerides again and encrypt key and will be considered a Warm start. The time to lock 4 satellites and have a navigation solution is on the order of 2-12 minutes.

Hot Start: If the receiver has been power cycled for less than 6 hours and not lost NVRAM data, then it will be considered a Hot start using the most recent ephemerides and almanac. The time to 4 lock and have a navigation solution is less than 5 minutes.

NOTE: The Hot Start feature has been inhibited in Link -006 due to some long start ups seen during flight tests. This feature may be re-introduced in subsequent releases of the firmware.

20.3.1 MAGR-S3S Schematics

RADIO RECEIVER R-2512A/U EXTERNAL CONNECTORS

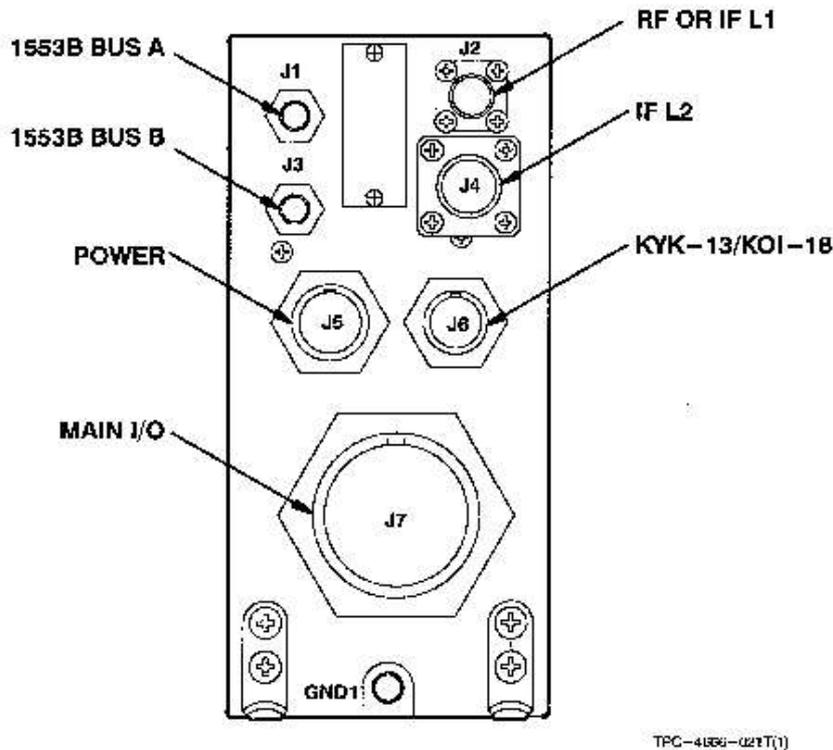


Figure 3 MAGR-S3S Cable Connections (Front of Box)

20.3.2 Radiation Protection

The receiver has been designed such that no radiation induced hard failures (permanent failures which require LRU repair) would occur within one year of continuous operation in a space environment. The minimum time between failures (MTBF), with the failure defined as a receiver reset due to a Single Event Upset (SEU), shall be at least 720 hours. One would expect one SEU per 30 days per GPS or one per 10-day flight for the three-string GPS configuration.

20.3.2.1 Results and Conclusions from the Proton Testing

The estimate of the overall MTBF for the unit due to interaction with protons is

MTBF for nominal environment = 1 every 479 days.
MTBF for solar flare = 1 every 1 day.

During proton testing the loss of use of the encryption key occurred several times. Those words are chosen carefully because it is not clear that the key itself was ever corrupted. In some cases, system memory appears to have been altered such that the unit believed the key to be invalid. In other cases, the only recovery from a failure was to erase the key. In any event, a conservative estimate of key loss due to protons is

MTBF for nominal environment = 1 every 589 days
MTBF for solar flare = 1 every 1.24 days.

20.3.3 Battery Operations

The MAGR-S3S is equipped with an internal battery supply (See Figure 4) so that while the receiver is unpowered, the non-volatile memory (NVRAM) maintains its integrity. If battery power and primary power are lost, the NVRAM is lost and the MAGR-S3S defaults to EEPROM which is set at the factory and cannot be written to (EEPROM is read-only). Battery removal/battery failure will result in MAGR loss of encryption key (PPS service), almanac (uses default almanac on EEPROM), all satellite ephemerides, and the MAGR will lose its current time and position and assume a default position (between Pad A and B), and it will lose the elapsed time indication. The MAGR-S3S will need to perform a cold start in this scenario.

The battery power is also used during zeroize operations. The MAGR-S3S can be zeroized while unpowered (28 VDC Primary Power source removed), but NVRAM will not clear until primary power is applied and the receiver is able to reinitialize in SPS Mode. A zeroize shorts out the zeroize circuit via battery power, and is not known by the receiver until primary power is applied. If the zeroize occurs while primary power is off, (zeroize switch is thrown with battery power), the zeroize will occur when primary power is reapplied.

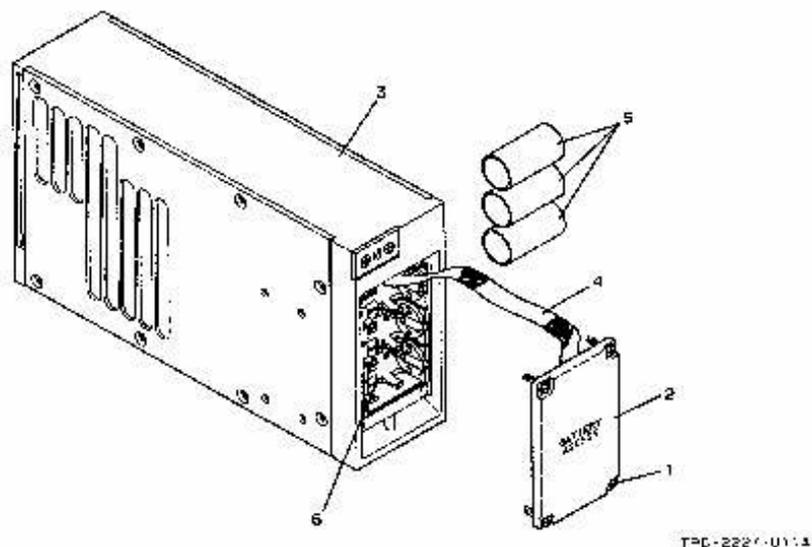


Figure 4 MAGR-S3S Battery Location (Back of LRU)

20.3.4 GPS Encryption

The Miniature Airborne GPS Receiver - Shuttle (MAGR-S3S) utilizes encrypted GPS satellite signals provided by the Department of Defense (DoD). In order to use the encrypted signals, the DoD requires an encrypt key to enable the GPS receiver to decode the encrypted signals. The MAGR-S3S uses a Group Unique Key (GUK), sometimes referred to as a yearly encrypt key. The device used to load the encrypt key is called the KYK-13 (pronounced "kick-13") key loader. Proper loading of the encrypt key during prelaunch operations allows the MAGR-S3S to receive satellite signals using Y-code tracking in the PPS submode. Without the encrypt key, the MAGR-S3S would only be able to track satellites in the SPS mode using C/A and/or P-code. A number of parameters are downlisted that enable the ground to monitor encrypt key operations and status. The encrypt key parameters are contained in one downlist word per GPS, called the PPS_STATUS_WORD and are documented on the following page. Also, if no key is loaded, all satellites will be tracked on L1 in C/A code only (although the peripheral channel may attempt to track on other codes and frequencies).

20.3.4.1 Zeroize Switch

Also, for security of the encryption key, the MAGR-S3S is equipped with a Zeroize switch which can be used to clear non-volatile memory of only the encryption key information. The zeroize switch should be used any time the orbiter performs an abort and lands in a foreign country or in the United States at a non-DoD secure facility. Nominal landings at KSC, Edwards, or Northrup do not require the receiver to be zeroized. Landings at any site other than KSC, Edwards, or Northrup requires the zeroize function to be performed for all three GPS receivers. Zeroize can be performed with the GPS powered off, but will not occur until primary power is applied. The Zeroize circuit is shorted by battery power when the switch is toggled. Once the MAGR-S3S has been zeroized it will automatically reinitialize and operate only in SPS mode. The MAGR-S3S cannot mode to PPS tracking once it has been zeroized, because it will no longer have an encryption key. A zeroize of the MAGR will result in a loss of encryption key information, but the satellite ephemerides and almanac will be retained if valid.

Table 4 Definition of PPS_STATUS_WORD - MSID V74M4459P (1459P), (3459P) for GPS 1, (2), (3)

| BIT PARAMETER | BIT # | BIT VALUE | DEFINITION |
|--|-------|-----------------------|---|
| Daily Key in Use (Verified) V74X4467B, (1467B), (3476B) | 0 | 1 Nominal PPS Mode | Key for the current day has been obtained from and verified against the Satellite Vehicle (SV) subframe data. GUK user mode and daily key has been obtained from satellite. |
| | | 0 SPS Mode | The receiver does not contain a verified key for the current day. Could be due to the fact the receiver has been zeroized or hasn't downloaded the daily key yet. (Possibly due to power off) |
| Daily Key in Use (Unverified) V74X4468B, (1468B), (3468B) | 1 | 1 SPS Mode | The receiver is waiting for subframe data and does not contain a key for the current day. Receiver still needs to download key for the day. |
| | | 0 Nominal PPS Mode | Key for the current day has been obtained and verified or the GUK issue number is invalid or expired |
| Daily Key in Use (Incorrect) V74X4469B, (1469B), (3469B) | 2 | 1 | The issue number is invalid. The receiver does not contain a key for the current day. (Could contain an expired key.) |
| | | 0 Nominal PPS Mode | Set when a new GUK is entered |
| Key Failed Parity V74X4470B, (1470B), (3470B) | 3 | 1 SPS Mode | If any key entered contains a parity error |
| | | 0 | If new keys are entered that pass the parity test |
| GUV User | 4 | 1 | Receiver using GUV's. Key is intended for Military (non- |

| | | | |
|--|------|-----------------------------|---|
| V74X4471B, (1471B), (3471B) | | | shuttle) users. |
| 2 Hour Alert V74X4472B, (1472B), (3472B) | 5 | 1 | Indicating that there are 2 hours or less before expiration of the current key, if the subsequent key is not available. Receiver needs to download the future key before the rollover to the next GPS day occurs (which is Midnight GMT). |
| Insufficient Key V74X4473B, (1473B), (3473B) | 6 | 1 | Indicating that insufficient consecutive keys have been loaded to accomplish the mission duration. N/A for Shuttle operations, will always be zero. |
| Erase Failure No MSIDs on Downlist | 7 | 1 | Indicating that the receiver failed to erase or zeroize a portion of memory. Troubleshooting: If fail to zeroize, cycle zeroize switch, holding for one second in the zeroize position. If No Joy then power cycle receiver. |
| Receiver Contains Keys V74X4474B, (1474B), (3474B) | 8 | 1 | Key entry successful |
| Mission Duration V74W4475B, (1475B), (3475B) | 9-15 | 0-44 Nominally 1 Or 2 | Representing the number of consecutive days for which the receiver has keys (from the current day) BIT 9 is the MSB and bit 15 is the LSB. From GMT 0:00 - 22:00 hr. will have one key, One future key will be loaded between 22:00- 23:59 hr. |

NOTE: Shuttle operations use Group Unique Key (GUK) and Group Unique Variable (GUV), shuttle does not use Crypto Key Weekly (CKW) operations.

20.3.4.2 MAGR-S3S Memory

The MAGR-S3S uses non-volatile memory to store the ephemerides for all SVs. The MAGR-S3S is capable of storing up to 32 ephemerides in NVRAM. The ephemeris data is updated at 1Hz from the NAVSTAR satellites. If the satellite downlink message indicates the SVs ephemeris has expired, the ephemerides will be deleted from the non-volatile memory (NVRAM). This memory can retain information when primary power is lost, but if battery power fails, the NVRAM is no longer protected. If battery power is lost, the MAGR-S3S must acquire fresh ephemeris data from each satellite. The non-volatile memory also retains the encryption key, current time and position.

During prelaunch operations, if available, a future key will be loaded into the non-volatile memory as well as the current key. Keys cannot be loaded in flight for security reasons.

Table 5 MAGR-S3S I/O Memory

| Memory Location | Size | Type | Description | SRU Name |
|-----------------|------|--------|-----------------|---------------|
| 0:0000 - 0:7FFF | 32k | EEPROM | Processor | I/O Processor |
| 0:8000 - 0:9FFF | 8k | RAM | Processor | I/O Processor |
| 0:A000 - 0:EBFF | 19k | none | | |
| 0:EC00 - 0:FFFF | 1k | RAM | Security Module | NAV Processor |
| 0:F000 - 0:F3FF | 1k | RAM | Timer Interrupt | I/O Processor |
| 0:F400 - 0:F7FF | 1k | RAM | Interprocessor | I/O Processor |
| 0:F800 - 0:FFFF | 2k | none | | |
| 1:0000 - 1:7FFF | 32k | EEPROM | Processor | I/O Processor |
| 1:8000 - 1:FFFF | 32k | EEPROM | Processor | I/O Processor |
| 2:0000 - 2:7FFF | 32k | EEPROM | Processor | I/O Processor |
| 2:8000 - 2:9FFF | 8k | NVRAM | Common Bus | NAV Processor |
| 2:A000 - 2:FFFF | 24k | RAM | Processor | I/O Processor |
| 3:0000 - 3:FFFF | 64k | None | | |
| 4:0000 - 4:1FFF | 8k | RAM | ARINC/INST/HVQK | I/O Processor |
| 4:2000 - 4:3FFF | 8k | RAM | 1553 | I/O Processor |
| 4:4000 - 4:9FFF | 24k | none | | |

| | | | | |
|-----------------|-----|-------|------------|---------------|
| 4:A000 – 4:FFFF | 24k | NVRAM | Common Bus | NAV Processor |
|-----------------|-----|-------|------------|---------------|

Table 6 MAGR-S3S NAV Memory

| Memory Location | Size | Type | Description | SRU Name |
|-----------------|------|--------|-----------------------|------------------|
| 0:0000 - 0:7FFF | 32k | EEPROM | Processor | NAV Processor |
| 0:8000 - 0:DFFF | 24k | RAM | Processor | NAV Processor |
| 0:E000 – 0:E7FF | 2k | RAM | Correlators | Signal Processor |
| 0:E800 – 0:EBFF | 1k | none | | |
| 0:EC00 – 0:FFFF | 1k | RAM | Security Module | NAV Processor |
| 0:F000 – 0:F3FF | 1k | RAM | Timer Interrupt | NAV Processor |
| 0:F400 – 0:F7FF | 1k | RAM | Interprocessor | NAV Processor |
| 0:F800 – 0:FFFF | 2k | RAM | Low Power Time Source | NAV Processor |
| 1:0000 – 1:7FFF | 32k | EEPROM | Processor | NAV Processor |
| 1:8000 – 1:FFFF | 32k | EEPROM | Processor | NAV Processor |
| 2:0000 – 2:7FFF | 32k | EEPROM | Processor | NAV Processor |
| 2:8000 – 2:9FFF | 8k | NVRAM | Common Bus | NAV Processor |
| 2:A000 – 2:DFFF | 16k | None | | |
| 2:E000 – 2:FFFF | 8k | RAM | Processor | NAV Processor |
| 3:0000 – 3:FFFF | 64k | none | | |
| 4:0000 – 4:9FFF | 40k | none | | |
| 4:A000 – 4:FFFF | 24k | NVRAM | Common Bus | NAV Processor |

20.3.4.3 Pre-Launch Encrypt Key Loading Process

Prelaunch, for each flight, a new GUK encrypt key will be loaded into each MAGR-S3S. The key and PPS operations will have already been checked out prior to GNC manning in the prelaunch time frame. Prelaunch encrypt key loading operations begin with zeroizing the receiver, which will delete any current or expired KYK-13 information stored in NVRAM. The next step is to load the future key (if available) to ensure it was loaded into volatile memory before the current key is loaded. The future key is loaded first because the MAGR-S3S only shows the status of the current key loaded. So by loading the future key first it is possible to confirm that the future key was properly loaded, and that the receiver will know it is a GUK user, and that the future key passes the parity check. The future key will not get satellite encrypt verification because it isn't the current daily key. MAGR-S3S is capable of storing a maximum of 2 yearly keys in memory (one current & one future). The future key may not always be available (not yet released by the DoD), so it may not always be stored in memory.

20.3.5 Power Input

The GPS subsystem converts the +28 VDC primary power to internal operating voltages as required. The current MAGR-S3S design uses 40-Watts of power at the receiver and 2.2-Watts for each of the pre-amps. (Each receiver utilizes an upper and lower antenna pre-amp). The total power used for each GPS subsystem is therefore 45 watts per string.

20.3.6 GPS Operations

At powerup, the MAGR-S3S detects the presence of Serial Input/Output interface, and will begin initializing the routines that allow it to communicate with the GPCs. So long as the NVRAM has not been cleared during the power off period, the MAGR-S3S will determine the length of time that it has been powered off, and execute its power up sequence accordingly. If the receiver has been off for more than 30 seconds, it

will perform the power on self test before resuming the previous (to power outage) operational mode. Otherwise, the receiver will skip the self test and resume operations. In the event that the NVRAM has been cleared, the receiver will execute the Long Term Power Outage processing after transitioning from OFF to NAV mode.

20.3.6.1 GPS Operational Modes

Technically speaking, the MAGR has four operational modes: OFF, TEST, INIT, and NAV. OFF is the unpowered state of the MAGR, during which the MAGR is doing nothing. The NVRAM and the encryption key are maintained by the power supplied by the MAGR's internal battery. The OFF mode is controlled by the MAGR power switch which is located on panel A13 on single string vehicles and panel O7 on three-string vehicles. The other three modes are software modes that are commanded by the crew using the GPS Status Display, SPEC 55.

TEST - TEST mode can be commanded by the crew using the proper item entries on the GPS Status Display, SPEC 55. Once commanded to TEST mode, the receiver waits for 6 seconds, during which time another operational mode (INIT or NAV) may be commanded. Once the 6 second wait is up, the receiver runs the GPS R/P Built-In-Test (BIT) for fault isolation and identification. The BIT takes about 2 minutes to complete, during which all data output from the MAGR to GPCs is suspended and crew displays show zeros. At BIT completion, the MAGR will resume outputting data, which for the TEST mode includes the BIT Fault Log Words. The result of the test (found in the fifth bit of Word 1) is displayed to the crew as an up or down arrow on SPEC 55 indicating a pass or fail. The arrow will remain until the crew commands another GPS mode. The MAGR will not exit the TEST mode without being commanded to do so by the crew. It should be noted that the commanded self test, BIT, is different from the Power-On Self Test.

INIT - INIT mode may be commanded by the crew by entering the appropriate item entries on SPEC 55. This mode allows the MAGR to attain a stable timing and frequency standard that is required to communicate to the GPCs. Data is then sent from the GPCs to the GPS to give the GPS an initial state vector as well as the lever arm corrections. After initialization is complete, the announced reset and filter restart flags should be set to off. The GPS will remain in INIT mode until it is commanded to either the TEST or NAV modes. If the receiver is not under GPC control when it is powered up, it will transition directly to NAV mode after initialization is complete. The INIT process should take approximately 15 seconds to complete.

NAV - NAV mode may be commanded by the crew via the item entries on SPEC 55. In NAV mode, the GPS is tracking satellites and providing its state vector information to the GPCs. There are three submodes to the NAV mode: aided (INS), unaided (PVA), and prelaunch standby (STA).

The receiver uses the prelaunch standby mode when it is commanded to NAV in OPS 9 prior to launch, in OPS 1 before MM102, and after transitioning to OPS 9 after landing. In this submode, the GPCs give the receiver the correct position vector, a velocity vector of zero, and attitude aiding data. This submode is also known as the stationary submode.

In all other phases of flight, the receiver will be in either the INS or the PVA submode. The PVA submode is used when a valid aiding state is not available to the MAGR. When valid aiding data is available, the receiver operates in the INS submode. Determination of valid aiding data relies on several factors. The rate that aiding data is supplied must be between 4 and 100 Hz. The aiding data must also have vertical position error as estimate by the filter of less than 1,000,000 meters. Finally, the aiding state must have a time stamp with a value that is no greater than 2 seconds in the past or 0.005 seconds in the future (this is known as Age of Aiding, or AoA). The MAGR runs on time supplied by the GPS constellation which is compared to the timetag associated with the aiding state. The aiding state timetag is supplied by the GPCs, which usually get their time from the MTUs. If the GPCs have to downmode to their internal time, there is a much greater potential for timing problems with this aiding state.

In the INS mode, the MAGR has two navigation routines, called Background nav (running at 1Hz) and Foreground nav (running at 6.25 Hz). Foreground nav propagates the latest position, velocity, and IMU misalignment errors to the time of the most recent aiding state from the IMUs. In background nav, a Kalman filter estimates ECEF (Earth-Centered, Earth-Fixed) position and velocity errors (6 states), IMU misalignment (3 states), baro altimeter bias (1 state) and GPS receiver clock bias and drift (2 states). The

measurement inputs to the filter are satellite pseudorange, deltarange, and baro altimeter measurements (if available). The position and velocity error vectors from the Kalman filter are used to correct the foreground navigation solution that is sent to the GPC (this is the GPS state vector) and back to background navigation. Foreground navigation velocity is used to aid the code tracking loops during acquisition and state 3 tracking to reduce the size of the code phase search window and decrease acquisition time. In state 5 tracking, delta range (carrier phase) measurements are used to aid the code tracking loops (see 20.3.6.3 for a description of tracking states).

If IMU aiding is lost or invalid, the receiver will operate in PVA filter mode; in this mode the IMU misalignment states in the filter are replaced by three ECEF acceleration error states. Because IMU aiding is not present, the foreground navigation solution must be propagated between measurements using a mathematical model rather than accumulated IMU velocity. In all MM except 901, 101, 305, 602, and 603, ballistic propagation is used to propagate the foreground solution at 10 Hz. In MM 304, 305, 602, and 603, great circle propagation is used.

In OI-30 in the event of an MTU failure, it is necessary for the ground to uplink a compensation to adjust the GPS correction factor for time associated with aiding data. In OI-32, a workaround has been added which will automate this time adjustment and update the age of aiding every 60 seconds (when announced GPS filter restarts occur). This adjustment will drive the AoA to a target value of 240 ms. One issue with this automatic compensation is that it must be enabled prior to the GPS moding to PVA after MTU failure. At the nominal GPC clock drift rate, the crew could have less than an hour to enable automatic compensation (even less time if on the BFS). If the automatic compensation is not enabled prior to aiding loss, a ground uplink will be required to recover aided operation. (For more information on this issue, see agenda/minutes from Ascent/Entry Flight Techniques #224, GPS Operation with MTU Failures)

When the receiver loses track of a satellite, it will attempt to reacquire that satellite for up to two minutes (state 7 will be indicated when the receiver is attempting to re-acquire a lost satellite). If it is still unsuccessful, it will replace it with any alternate satellite that has been acquired on the peripheral channel. An alternate satellite is a healthy, visible satellite that has not been assigned to a Nav channel, has not been operator deselected and has an ephemeris in the receiver. The receiver will attempt to acquire this alternate satellite for ten seconds; if it is unsuccessful it will then attempt acquisition using other code and frequency combinations. The receiver will only look for satellites on the visible list. In the event that a satellite has anti-spoofing enabled or the MAGR is unauthorized or unkeyed for that satellite, the MAGR will use C/A-code only mode tracking. In this scheme the receiver will reduce the priority of P-code reacquisition attempts if the track history was C/A-code only. In addition, the receiver will also inhibit ionospheric delay measurements. It will also monitor the navigation message and receiver authorization status to determine whether to leave C/A-only mode track.

Upon transition to NAV mode, the receiver builds the clock-ephemeris database by retrieving the ephemerides of the satellites on the initial visible list from the NVRAM. This database is maintained and updated as information becomes available from the constellation. An ephemeris is deleted from the clock-ephemeris database once the satellite drops off the visible list. Satellite ephemerides are stored in the NVRAM for all available satellites so long as there are no more than 32 satellites in the constellation.

20.3.6.2 GPS Channel Operations / Peripheral Channel

The MAGR-S3S has five hardware channels for tracking satellites; four channels are assigned as "measurement" or "primary" channels and are used to track the signals of satellites being used for navigation. Any of the five hardware channels can be used for either primary or peripheral tasks.

The fifth channel is called the peripheral channel and it is used for the following tasks (in order of priority):

- 1) Acquisition of alternate satellites in case of loss of lock of one of the primary channels. If this happens and the primary channel cannot reacquire the satellite within 10 seconds, the primary and peripheral channels will switch and the former peripheral channel will be used for primary tracking. In this case an increase in GDOP could occur due to an alternate satellite being used. Satellites in the alternate list must be healthy, not a primary navigation satellite, and the receiver must have an almanac or ephemeris for it.

The satellites in the alternate list are prioritized from highest to lowest elevation angle with respect to the local horizontal, as well as data source (ephemeris or almanac), and track history.

2) Special data collection

3) Channel calibration (P(Y) code only); every 20 minutes the peripheral channel will acquire each navigation satellite using the same code and frequency as the primary channel to calibrate the hardware channels and measure any differential channel delay, which for the MAGR-S3S should be less than 1 meter.

4) If a change in the selected primary navigation satellites is to take place (based on GDOP), the peripheral channel will first attempt to acquire and track the new satellite; once acquired it will switch hardware channels with that primary channel. For a new navigation satellite, C/No of the new satellite must be greater than 32 dB-Hz, or the C/No of the old satellite must be below 35 dB-Hz; this is to avoid state 3 tracking.

5) Collection of ephemerides for each visible satellite for which the receiver does not have a valid ephemeris.

6) Alternate satellite health bit checks.

7) Ionospheric error estimation (if keyed). Every 30 seconds the receiver will attempt to track each satellite being tracked by the primary channels for 2 to 5 seconds on the frequency not being tracked by the primary channel. If the receiver is unkeyed, it will not perform any ionospheric corrections.

8) Determination of what code and frequency combinations can be tracked is done for all visible satellites for which the receiver has an ephemeris. This is known as expansion activity, and is usually seen under difficult tracking conditions (during plasma or while on the pad).

20.3.6.3 GPS Tracking States and C/No

When the receiver tracks satellites, a measure of the signal strength is called the C/No ratio, which is the carrier plus noise to noise ratio, and it is measured in decibel Hertz (dB-Hz). The higher the C/No, the easier the signal is to track and a high C/No typically has less measurement errors than a low C/No. In general, C/No is a function of satellite elevation angle with respect to the antenna gain pattern and local horizontal, RF noise, multipath, skin or bending effects, antenna masking, and other errors such as ionospheric scintillation. Typical C/No values for normal tracking (state 5 - see below) range from approximately 26-50 dB-Hz. State 5 tracking with a C/No greater than 34-35 dB-Hz will likely yield good tracking performance and good pseudorange/deltarange measurement accuracy.

The state of a channel indicates the type of activity being conducted on that channel; the channel track states are described here in order of preference (first being best tracking performance):

State 5: Code and carrier frequency lock on a satellite (pseudorange and deltarange). C/No is typically above 26.5 dB-Hz, although it can be lower. State 5 should be seen most of the time after ET separation, on orbit, and during entry when not in plasma.

State 3: Only code phase being tracked (pseudorange). Receiver velocity will be less accurate; state 3 tracking is typically seen with lower C/No ratios (26.5-14 dB-Hz). May occur with satellites which are at a low elevation angle with respect to the antennas, or if the receiver thinks it is being jammed.

State 7: A channel that was tracking a satellite (state 5 or 3) but has lost lock and is trying to reacquire will indicate state 7 tracking. The MAGR/S-3S will attempt to reacquire for 10 seconds on the same code/frequency before selecting a satellite in the alternate list, assuming the alternate has been acquired by the peripheral channel.

State 2: The channel is attempting to establish lock on P (or Y) code without first tracking C/A code; this occurs if the receiver has a very accurate position, velocity, and time solution, most likely seen after a short power cycle.

State 1: The channel is attempting to establish lock on C/A code, and will later attempt to transition to P or Y code.

State 0: The channel is idle; short periods are acceptable (up to 6 seconds) but longer periods are not.

20.4 SHUTTLE GPS IMPLEMENTATION

20.4.1 Orbiter Signal Environment

The orbiter signal environment does not always meet the receiver design assumptions, and the Satellite Selection algorithms have been modified to account for this. There are several factors which increase the possibilities of the receiver to experience extended GDOP chimneys:

Antenna Gain – Greater than 70° from antenna boresight, the antenna gain drops off dramatically

Vehicle Masking – Occurs in the wing plane and in the direction of the tail, causing masking and multipath of GPS signals

Plasma Effects – During entry, the plasma build up will significantly hinder the lower antenna's ability to acquire GPS signals. The upper antenna has reduced cone angle, but still has satellite tracking capability. Plasma typically occurs between 300-200 kft. of altitude. The lower GPS antenna is typically totally obscured between 230 – 200 kft.

Dynamic Attitudes of the Orbiter – there are a wide range of vehicle attitudes (typically on orbit) which will carry the antennas out of the local horizontal attitude and hinder signal acquisition. (MAGR-S3S antenna select logic assumes an LVLH attitude)

External Structure Masking – During ascent, the external tank is a large structure that blocks the lower antennas. Structures such as the International Space Station can also hinder signal acquisition due to their large area obscuring the antennas.

Multipath- In the proximity of the wing plane, both antennas experience multipath problems due to signal reflections off of external structures. Known multipath sources include the Space Station during docked periods, large payloads outside the payload bay, and the launch tower.

The MAGR Satellite Selection logic selects satellites for availability by deeming satellites below the radius of the earth plus 80 km in the 0° LVLH frame as unavailable. Above 80 Km the visible SV mask angle is calculated dependent on the Earth tangent depression angle. (See Figure 5) Any satellite below the LVLH 0° (or Biased LVLH negative elevation angle for altitudes above 80 km) is not considered available. Although the receiver may track these satellites, they are not used to compute the GPS Navigation State.

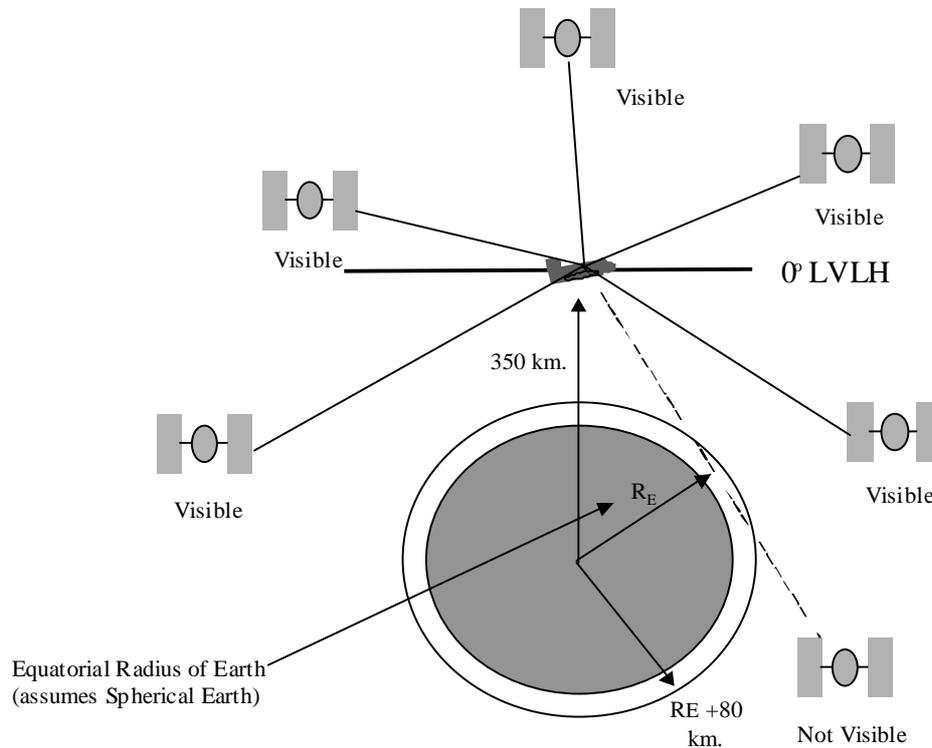


Figure 5 GPS Satellite Mask Angle

20.4.2 Shuttle Operations

MAGR-S3S Operational capabilities include:

Normal Startup – The GPS is capable of performing a nominal start up given an initial estimate of Position, Velocity & Time (PVT). The Initialization uncertainties are 100 km / 150 m/sec / 60 sec (P/V/T).

Satellite Selection – The receiver can autonomously select a set of satellites for tracking. These satellites are chosen such that they yield the optimal navigation performance based on geometry, signal strength, and track history.

Satellite Incorporation – The GPS Navigation filter is able to incorporate data from available satellites, even when less than four are available (an LVLH attitude is assumed).

Filter restart – The GPS Navigation filter can be restarted at anytime by a manual or autonomous command. The filter restart does not unnecessarily reset the GPS satellite tracking loops, but opens the covariance settings in the internal Kalman Filter.

Satellite Monitoring – The receiver monitors the health status of all available satellites being tracked. It will not utilize measurements or data from an unhealthy satellite, or manually deselected satellites.

MAGR-S3S Operational Modes:

Off – External Power to GPS is removed or the power control discrete is inactive. While powered off, the internal batteries maintain the non-volatile memory (NVRAM) and all the information contained within. The receiver is capable of accepting the PPS key upload while in OFF Mode. (KYK-13 Key Loader)

INIT – INIT Mode allows the MAGR to attain stable operations by obtaining a current Position, Velocity, & Time from the GPC.

NAV – The receiver is in NAV Mode when it is outputting State Vectors.

Prelaunch Standby – Executed only in OPS 9. The receiver autonomously transitions to this navigation submode when INIT is completed in OPS 9.

Aided Navigation – The receiver autonomously modes to the Aided mode when valid aiding data is received. When aiding data is available and satellites are visible, the MAGR-S3S will weigh the data from both sources to maintain estimates.

Unaided Navigation – The receiver autonomously transitions to this mode when the GPC designates any OPS phase other than OPS 9 and the aiding data is invalid. The MAGR-S3S is capable of operating in Unaided Mode (PVA), but is not certified for Shuttle Navigation.

S/TEST– The GPS receiver performs full commanded self-test using the GPS R/P Built-In-Test (BIT) capability for fault isolation and identification. After the first 6 seconds in TEST mode, the receiver will mode itself to BIT during which period (about 2 minutes) there is no output from the receiver. All GPS receiver data, including the data header, received by the GPC will be zero. When BIT is complete, the receiver will again output valid header and data (See Appendix B: BIT FAULT LOG WORDS). The result of the self-test (pass or fail) will be in the bit 4 of word 1 in the BIT Fault Log (Collins Fault Log Word 10) and is available on the crew display. Upon completion of the commanded BIT, the receiver will remain in TEST mode and will not automatically transition to INIT or NAV.

GPS Lever Arm Correction – The navigation software uses the antenna lever arm correction when attitude data is available. The correction uses the attitude data for each antenna to translate the GPS solution from antenna phase centers to vehicle center of navigation. This information (Lever Arm Data) is provided to the MAGR-S3S by the GPC.

Center of Navigation – The center of navigation is the vehicle CG during on orbit and the NAV base during ascent and entry. If attitude data is not available, or is invalid, the MAGR-S3S State Vector will be relative to the antennas.

20.4.3 GPS Redundancy Management / QA Checks

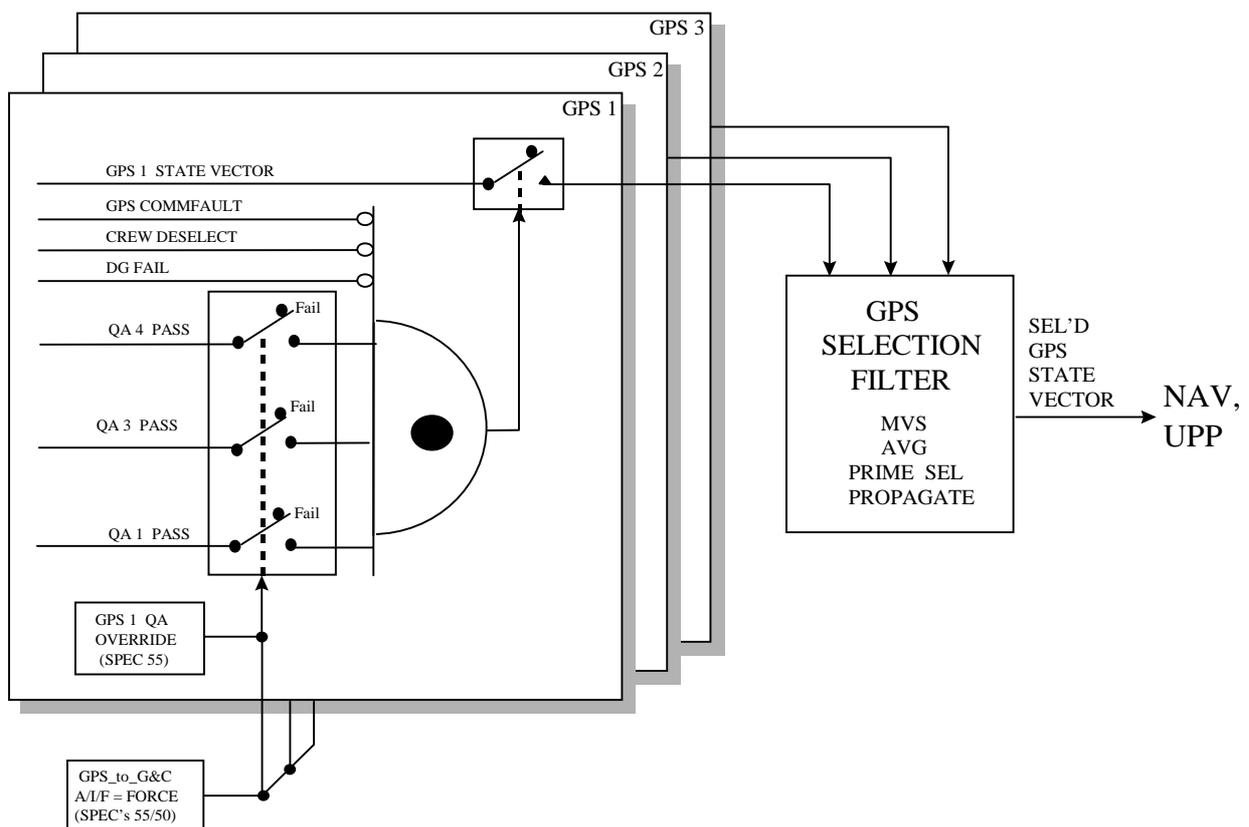


Figure 6 GPS Selection Filter Candidate Logic

Unlike the classical Redundancy Management (RM) schemes applied to other Shuttle navigation sensors, the GPS utilizes a cyclic quality assessment (QA) process, which is more conducive to its performance. Due to the nature of its operation, whereby poor satellite coverage may result in periods of invalid data, it would not be prudent to 'permanently' remove a unit from availability due to these questionable periods. The QA scheme employed uses three specific criteria, QA1, QA3 and QA4 to remove a given GPS from candidacy to the GPS Selection Filter. Violation of any of these QA's will temporarily remove that LRU from the GPS selection filter for the cycles it violates the QA threshold; the same checks are made on the ensuing cycle. The crew has the option to override the QA functions from a specific unit (except for a QA4 dilemma) should it be determined that good GPS data is being kept out of the selection filter due to an erroneous QA signal (see sections 20.4.5.2.2 and 20.4.5.3.2).

With three eligible candidates available, mid-value Select (MVS) logic is used by the GPS selection filter to select a unit for processing; at the two level, the candidate data is averaged; at the one level, the single GPS is prime selected. When no eligible GPS candidates are available, the selection filter will propagate the last valid data for processing. If none of the GPS states are candidates and at least one of the GPS receivers is not deselected or bypassed, the selected GPS state will be the selected state from the previous navigation cycle propagated to the current filter time (up to an I-loaded amount of time). IMU accelerometer data is used to propagate this selected state. As long as the IMU accelerometer data is good, the propagated selected state can still be used to update NAV (assuming GPS was not already being processed into NAV). If IMU accelerometer data is bad, the propagated selected state will be corrupted along with NAV. The GPS selected state will not be usable until a GPS receiver with a valid state is once again a candidate for selection.

If none of the GPS states are candidates and all three GPS receivers are not available (each receiver is either deselected or bypassed), the selected GPS state gets reset to zero and designated as unavailable. GPS data should not be used until a GPS receiver with a valid state is once again a candidate for selection. After an I-loaded period of time, the crew will be alerted via a GAX message and an SM alert; the propagation time will be displayed as "Last Sel Fil Update" on SPEC 55 (see Figure 9).

In the BFS, the selection filter uses a component-by-component mid value select at the three level. At the two level, the BFS will select the receiver with the lower FOM; if both receivers have the same FOM the BFS will select the low-numbered LRU. At the one-level, the BFS will use the available receiver, and it will not have an available state at the zero-level.

In addition to a QA 1, 3 or 4 flag being set, a GPS data good (DG) signal, crew deselect, or commfault will remove a GPS unit from candidacy within the GPS selection filter. Forcing GPS data into UPP (Force to G&C) is a continuous function (PASS only), whereas forcing GPS into nav is a one-shot transfer in the PASS and continuous in the BFS. During entry on the PASS, GPS data sent to UPP is subject to metering in order to limit state vector updates sent to guidance. Metering is also applied when switching the UPP state vector source from GPS to NAV. An item entry on SPEC 55 allows the crew to override the metering function and could force a large guidance transient.

20.4.3.1 QA Override

The QA Override function is available on SPEC 55, via an ITEM 29, 30 and 31. Commanding QA Override makes a GPS available for the selection filter even though it may have failed one or more of the QA checks. Note that invoking QA Override does not override crew deselections or a GPS data good failure. Invoking QA Override does not clear a GPS Dilemma.

Should a GPS dilemma occur (QA4), a crew response is required (deselect or powerdown). Even though a GPS DG signal normally removes a unit from candidacy in the GPS selection filter, the DG flag cannot be trusted to break the dilemma, as periodic DG dropouts may occur nominally, based on satellite constellation conditions and the GPS Kalman filter states. This restriction is meant to prevent a dilemma from being resolved by an inadvertent downmoding of the good unit.

20.4.3.2 QA 1 - FOM

The Figure of Merit (FOM) or QA1 is not an orbiter flight software check of the GPS, but rather an internal GPS hardware performance estimate of one sigma position errors expected from the associated GPS navigation solution. FOM, also called $P1\sigma$ (pronounced P-one-sigma) in three string GPS, is displayed on SPEC 55 and shipped directly from the GPS to the GPC for use by the GPC in determining if that GPS solution will be used as a SF candidate to update navigation.

The MAGR-S3S FOM calculation is heavily based on satellite geometry (GDOP). It is not uncommon to see a high GDOP causing a high FOM several seconds later. High FOM due to a high GDOP will correct itself when a better geometry of satellites moves into view. A high FOM due to reasons other than a high GDOP should be looked at more closely, since it indicates the GPS has off nominal performance not directly attributable to poor satellite geometry.

The QA1 check is available in both the PASS and BFS, and also in the single and three string GPS configurations. One difference to note between the PASS and BFS, is that the PASS uses a less than function "<" for QA1, whereas the BFS uses a less than or equal "<=" to function to determine QA1 failures. This results in different I-loaded thresholds for QA1.

20.4.3.3 QA 2 - GPS to Nav Compare

QA2 is the onboard navigation comparison to GPS. The orbiter flight software check compares the GPS 1, 2, and 3 solutions with the on-board navigation state. If the GPS 1, 2, or 3 comparisons exceeds any of the QA2 thresholds, a QA2 fail is declared for position, or velocity, or both.

The QA2 limits are I-loaded and time dependent (ramp up to a max threshold over time since the last GPS update was performed). If any of the QA2 position or velocity components exceeds the QA2 thresholds, the whole GPS state fails QA2. Nominally, QA2 failures will be driven by error growth in the onboard propagated navigation state.

The QA2 check does not factor in to the selection filter, so QA2 fail will not prevent a selected GPS state from updating navigation. Once the selected state is sent to NAV, navigation performs its own comparison, called the Nav Edit Ratio function. The Nav Edit Ratio function is sometimes called the "secondary QA2 check". If the secondary QA2 check is failed (RATIO > 1), the LRU with the high edit ratio may be used by forcing GPS to NAV.

The QA2 check is available in the BFS single and three string GPS configuration, however there is no onboard indication of QA2 data output on BFS SPEC 55. The nav edit ratio check function (on SPEC 50) can be observed to determine the difference in GPS state and onboard navigation.

20.4.3.4 QA 3 - GPS to Last Cycle Compare

QA3 compares a given GPS to its previous nav cycle (propagated to the current time using IMU data) to see if any large jumps in velocity or position were detected. The receiver must have a valid nav solution on the current and previous nav cycles, otherwise the GPS fails QA3. QA3 should catch any rapid spikes in position or velocity from one cycle to the next.

The PASS flight software provides crew insight to QA3 on SPEC 55 in single string and triple string configurations. The position and velocity ratios (difference/threshold) are displayed along with a status indicator.

BFS does not support QA3.

20.4.3.5 QA 4 - GPS to GPS Compare

QA4 compares one GPS solution to the other GPS solutions. If a common pair of position or velocity differences exceeds the QA4 threshold, the common receiver fails the LRU to LRU check. A QA4 dilemma requires crew intervention to resolve.

It is possible to have a QA4 Dilemma at the 3-level and at the 2-level. A QA4 dilemma will be present at the 3 level, if all three LRU to LRU difference checks exceed the QA4 threshold. To resolve a 3-level dilemma, the GNC operator should look at the other QA checks to observe how each of the LRUs are violating the QA1, QA2, and QA3 checks. It is recommended to perform a QA Override on the LRU with the smallest FOM to allow NAV to process GPS data. Another option is to deselect the LRU with the largest FOM, to remove the worst GPS from the 3-level dilemma. However, if the deselect option is selected, a 2-level dilemma may be present. If the 2-level dilemma occurs, follow the steps located below for a 2-level dilemma.

A QA4 miscompare occurs at the two level when both selected GPS LRUs disagree by more than the QA4 threshold. For a QA4 miscompare at the 2-level, it is recommended to deselect the LRU with the largest FOM. If both LRUs have the same FOM, then deselect the LRU that disagrees most with nav (QA2).

Note that QA4 does not function while on-orbit with only a single GPS powered.

The PASS flight software provides crew insight to QA4 via position and velocity ratio (difference/threshold) on SPEC 55 along with status indicators.

BFS supports a GPS to GPS compare (not called QA4). BFS does not have QA 4 dilemma logic like the PASS, or crew interface to QA4 output on BFS SPEC 55.

20.4.3.6 QA Fault Summary Messages

RM DMLA GPS - Will be annunciated for a QA4 dilemma at the 2 or 3 LRU level

G55 GPS QA 1, 2, 3 - Will be annunciated for a GPS LRU 1, 2 or 3 fail of QA 1, 3, or 4 for greater than an I-loaded number of cycles.

20.4.3.7 GPS QA and GAX I-Loads

GPS has a considerable number of I-loads related to QA checks and GNC Annunciation (GAX) messages for crew alerts. Many of the I-loads are dependent on the OPS mode and the flight software OI being flown. The following tables contain the PASS I-loads for OI-30. Keep in mind that these parameters are merely I-loads and may change.

NOTE: A number of I-loads have been increased to their maximum values in order to avoid nuisance alarms, specifically during crew sleep periods.

| OI-30 PASS GPS I-LOADS (FOR ALL OPS) | |
|--------------------------------------|---|
| DOWNLIST SCALE FACTORS | |
| DELR_SF | 0.0016457884 (ft) |
| DELV_SF | 0.1 (fps) |
| DELR_UVW_POS_SF | 10.0 (ft) |
| DELV_UVW_VEL_SF | 0.1 (fps) |
| QA2_RTOL_SF | 10.0 (ft) |
| QA2_VTOL_SF | 0.1 (fps) |
| QA2_DELR_RATIO_SF | 0.01 (fps) |
| QA2_DELV_RATIO_SF | 0.01 (fps) |
| QA3_DELR_RATIO_SF | 0.01 (fps) |
| QA3_DELV_RATIO_SF | 0.01 (fps) |
| QA4_DELR_RATIO_SF | 0.01 (fps) |
| QA4_DELV_RATIO_SF | 0.01 (fps) |
| DT_QA2_SF | 1.0 (sec) |
| GPS CONFIGURATION DATA | |
| GPS_INSTALLED | ON, ON, ON (3 receivers) OFF, ON, OFF (1 receiver) |
| GPS_LOCKOUT | 3 (1 or 3 receivers) |

| GPS GAX MESSAGE COUNTERS | |
|---------------------------------|---------------------------------|
| GPS_FAIL_FOM_GAX_MSG_CNT_MIN | 32,760 nav cycles (~34 hours) * |
| GPS_QA3_GAX_MSG_CNT_MIN | 32,760 nav cycles (~34 hours) * |
| GPS_QA4_GAX_MSG_CNT_MIN | 32,760 nav cycles (~34 hours) * |
| GPS_QA4_DILEMMA_GAX_MSG_CNT_MIN | 32,760 nav cycles (~34 hours) * |
| GPS DATA QUALITY ASSESSMENT | |
| GPS_QA_TIME_THRESH | 1.90 sec |
| GPS STATE PROPOGATION | |
| GPS_IMU_VEL_THRESH | 0.05 fps |

| OI-30 PASS GPS I-LOADS (FOR OPS 2 and 8) | |
|--|-----------------------------------|
| GPS GAX MESSAGES/ALERTS | |
| GPS_ALERT_DT | 1,728,000.0 (sec) (~20 days) |
| GPS_GAX_OFF_ALERT_RMNR_DT | 1,728,000.0 (sec) (~20 days) |
| N_RATIOS_ABOVE_LIMIT_MIN | 32,760 nav cycles (~34 hours) |
| GPS QA TOLERANCES | |
| MAX_FOM_ACCEPT_PFS | 2 |
| GPS_QA2_RTOL_UVW_A(1-3) | 500.0, 500.0, 500.0 (ft) |
| GPS_QA2_RTOL_UVW_B(1-3) | 0.13889, 1.89815, 0.09259 (fps) |
| GPS_QA2_RTOL_UVW_C(1-3) | 0.0, 0.0, 0.0 (ft/s2) |
| GPS_QA2_RTOL_MAX_UVW(1-3) | 2000.0, 21000.0, 1500.0 (ft) |
| GPS_QA2_VTOL_UVW_A(1-3) | 2.0, 1.0, 1.0 (fps) |
| GPS_QA2_VTOL_UVW_B(1-3) | 0.00203, 0.00009, 0.00009 (ft/s2) |
| GPS_QA2_VTOL_UVW_C(1-3) | 0.0, 0.0, 0.0 (ft/s3) |
| GPS_QA2_VTOL_MAX_UVW(1-3) | 24.0, 2.0, 2.0 (fps) |
| GPS_QA3_RTOL | 200.0 (ft) |
| GPS_QA3_VTOL | 0.6 (fps) |
| GPS_QA4_RTOL | 500.0 (ft) |
| GPS_QA4_VTOL | 5.0 (fps) |
| GPS UPDATE RATE | |
| T_GPS_CYCLE | 5400.0 (sec) (~ 90 minutes) |

| OI-30 PASS GPS I-LOADS (FOR OPS 1 and 3) | |
|--|--------------------------------------|
| GPS GAX MESSAGES/ALERTS | |
| GPS_ALERT_DT | 300.0 (sec) (5 min) |
| GPS_GAX_OFF_ALERT_RMNR_DT | 1,728,000.0 (sec) (~20 days) |
| N_RATIOS_ABOVE_LIMIT_MIN | 5 nav cycles (~19.2 sec) * |
| GPS QA TOLERANCES | |
| MAX_FOM_ACCEPT_PFS | 5 |
| GPS_QA2_RTOL_UVW_A(1-3) | 500.0, 500.0, 500.0 (ft) |
| GPS_QA2_RTOL_UVW_B(1-3) | 1.8056, 8.1944, 3.75 (fps) |
| GPS_QA2_RTOL_UVW_C(1-3) | 0.0, 0.0, 0.0 (ft/s2) |
| GPS_QA2_RTOL_MAX_UVW(1-3) | 7000.0, 30000.0, 14000.0 (ft) |
| GPS_QA2_VTOL_UVW_A(1-3) | 8.0, 8.0, 8.0 (fps) |
| GPS_QA2_VTOL_UVW_B(1-3) | 0.006111, 0.003333, 0.003333 (ft/s2) |
| GPS_QA2_VTOL_UVW_C(1-3) | 0.0, 0.0, 0.0 (ft/s3) |
| GPS_QA2_VTOL_MAX_UVW(1-3) | 30.0, 20.0, 20.0 (fps) |

| | |
|------------------------------|-------------------------|
| GPS_QA3_RTOL GPS_QA3_VTOL | 500.0 (ft) 5.0 (fps) |
| GPS_QA4_RTOL GPS_QA4_VTOL | 500.0 (ft) 5.0 (fps) |
| GPS UPDATE RATE | |
| T_GPS_CYCLE | 40.0 (sec) |

* Note the nav cycle rate is increased at MLS processing from every 3.84 seconds to 1.94 seconds. The numbers in brackets assume a 3.84 second nav cycle.

20.4.4 GPS SOP

One of the functions of the GPS SOP is to interpret the data coming from the GPS receivers. When a set of GPS data is received by the GPCs, the GPS_TO_GPC module is run to properly format the data that is coming in. In the 3 string configuration, the data from different receivers is not required to be time homogenous, but the data set from a particular receiver must be time homogenous. In MM304, 305, 602, and 603 the GPS SOP collects barometric aiding data from the air data system to aid the GPS receivers. Similarly, radar altimeter data is collected in MM305 and 603 for aiding the GPS receivers. Commands such as the GPS operational mode and the satellite deselected are collected by the GPS SOP and sent to the GPS receivers. Reference states are also collected by the GPS SOP for the purpose of aiding the GPS receivers. Time uplinks that are to keep the GPS receivers in sync with the GPCs are done through the GPS SOP.

Upon all OPS transition, except when moding from OPS 1 to 6 the GPS mode indicator is blanked and reset to the current operating state as reported in the NAV_MODE_WORD by each GPS receiver. The GPS filter restart and init request flags are also set to zero to indicate the no request has been made to perform either of these functions. Before the first pass through the GPS SOP, the time reference conversion and number of LRUs installed are initialized to preserve these parameters across all OPS transitions.

20.4.4.1 GPS Data Processing

The GPS receivers communicate to the GPCs through the use of a serial digital MDM input and output data channel. During nominal flight operations, two blocks of data are sent to the GPS receivers from the GPCs at a rate of 6.25 Hz. The GPS receiver sends one block of data to the GPCs at a rate of 1.04 Hz. These data blocks or messages consist of 32 halfwords of hexadecimal data. Data being sent to the GPS receivers from the GPCs is formatted by the GPC_TO_GPS portion of the GPS_SOP. The GPC_TO_GPS data processing code reads the crew commanded operational state (GPS_MODE_CMD_SOP) from the GPS_SOP to know what data message format to send to each receiver. The GPS_TO_GPC data processing code deciphers the data message coming from each receiver by reading the NAV_MODE_WORD to see the operational mode of that receiver.

The first message that is sent by the GPCs to the receivers (GPC_TO_GPS_DATA_ONE) is of the Message 1 format, which contains some of the receiver aiding data, including the position and velocity aiding data from the orbiter's state vector, and its time tag. The satellites that have been manually deselected by the crew are also included in Message 1. Message details are contained in the next section of this document. The GPS_CONTROL_WORD is also found just after the header in Message 1. It contains the operational mode command from the GPCs to the receiver and some of the aiding data valid flags and filter restart commands. As previously stated, the second data message word (GPC_TO_GPS_DATA_TWO) that is sent from the GPCs to the receivers and the message that the receivers send to the GPCs (GPS_TO_GPC_DATA_ONE) are determined by the crew commanded operational mode and the current operational mode of each receiver.

In TEST mode, the second data message that is sent from the GPCs to the GPS has all of its components set to zero. Since the header for this message contains all zeros, the receiver will ignore the remainder of this message. The receiver will send data to the GPC in the Message 4 format. This format includes the test BIT_FAULT_LOG_WORD providing the results of the self test as well as the GPS NAV_MODE_WORD.

In INIT mode, the second data message that is sent to the GPS receiver is in the Message 2 format. Message 2 contains the lever arm corrections for the GPS antenna placement, as well as antenna bore sight direction in the orbiter's body coordinate system. This information is used by the GPS receivers to determine which satellites are within the field of view of the GPS antennas. This data is only sent to the receiver during the INIT mode because it should not change throughout the flight. During the INIT mode the GPS receiver sends data to the GPCs in the Message 5 format. This message has very little data in it, just the GPS NAV_MODE_WORD and the almanac reference week, as well as the PPS status word.

In NAV mode, the second data message sent by the GPC is Message 3. Message 3 provides the orbiter attitude to the GPS receivers for use in determining which satellites should be on the visible list. The BARO and radar altimeter aiding data and time tags which are used in OPS 305 are also provided in Message 3. The GPS receiver sends Message 6 to the GPCs, containing the receiver's solution for position and velocity with the timetag for that solution. Message 6 also contains the FOM and GDOP for that receiver, as well as the channel status word. The channel status word contains information about which satellites are being tracked on what channel, as well as the signal to noise ratio and tracking code (P-Code, Y-Code, or C/A Code). The GPS NAV_MODE_WORD is also contained in Message 6 as it is in Messages 4 and 5.

The NAV_MODE_WORD is the first part of any data message that is sent from a receiver to the GPCs, after the header, whether it is Message 4, 5, or 6. The NAV_MODE_WORD informs the GPCs what mode the receiver is in and whether or not the receiver has the init data and almanac. By knowing what mode receiver is in, the GPCs know what data message to expect from the GPS receiver. The receiver also informs the GPCs whether it is in PPS or SPS operation and whether it is in the PVA or INS nav submodes. Other flags regarding resets, lever arm corrections, and UTC are also included.

GPS receiver input and output processing is configurable by both the crew and I-loaded values. During OPS 1, 2, 3, 6, and 8, the GPS_IO_CNTL flag, which controls the I/O configuration of the GPS system, is set to 2. This is the flight configuration for the GPS I/O indicating that the GPS to GPC input data will be shipped at a rate of 1.04 Hz, and the GPC to GPS output will occur at a rate of 6.25 Hz. The GPS_IO_CNTL flag is displayed as Item 10 on SPEC 55, though it is not possible for the crew to change this flag in any OPS mode other than OPS 9. In OPS 9 an I/O configuration of 0, 1, or 2 may be selected. Upon entering OPS 9, GPS_IO_CNTL is set to 0 indicating that GPS receiver I/O has been disabled. If this flag is set to a value of 1, the only I/O that will occur is 6.25 Hz non-commfaultable GPS to GPC input. GPC to GPS output follows GPS to GPC input when they occur on the same cycle. During OPS 9 when GPC to GPS outputs are disabled (GPS_IO_CNTL = 1 or 0) the ground may command the GPS receivers using the LDB/TCS operators or some other GPS receiver test stimuli.

20.4.4.2 GPS Serial Data Messages

This section specifies the general format of the GPS MDM serial I/O. The index (*I*) in the following tables represents the number of the GPS subsystem string, 1, 2 or 3.

Table 7 GPC to GPS Data Description

| I/O Data Blocks | Number of Halfwords |
|---------------------------------|---------------------|
| GPC_TO_GPS_DATA_ONE(<i>I</i>) | 32 |
| GPC_TO_GPS_DATA_TWO(<i>I</i>) | 32 |
| GPS_TO_GPC_DATA_ONE(<i>I</i>) | 32 |

Table 8 GPS to GPC data One (I) Contents

| R/P MODE | GPS_TO_GPC_DATA_ONE Contents | Halfword Number | Units | Scale Factor |
|----------|---|-----------------|-------|--------------|
| TEST | HEADER_4 (DDDD) | 0 | | |
| | NAV_MODE_WORD_RCVR(<i>I</i>) | 1 to 2 | | |
| | BIT_FAULT_LOG_WORD_RCVR (<i>I</i> , 1 to 18) | 3 to 20 | | |
| | SET_OPERATING_TIME | 21 to 22 | Hours | |
| | CPCI_ID | 23 to 24 | | |

| | | | | |
|------------------|------------------------------|----------|--------|------------------|
| | SPARE | 25 to 31 | | |
| INIT | HEADER_5 (EEEE) | 0 | | |
| | NAV_MODE_WORD_RCVR (I) | 1 to 2 | | |
| | T_GPS_WEEK_RCVR (I) | 3 | Weeks | 1 |
| | T_GPS_SEC_RCVR (I) | 4 to 6 | Sec | 10 ⁻⁸ |
| | PPS_STATUS_WORD_RCVR (I) | 7 | | |
| | ALMANAC_REF_WEEK_RCVR (I) | 8 | | |
| | SPARE | 9 to 31 | | |
| NAV | HEADER (FFFF) | 0 | | |
| | NAV_MODE_WORD_RCVR (I) | 1 to 2 | | |
| | T_GPS_WEEK_RCVR (I) | 3 | Weeks | 1 |
| | T_GPS_SEC_RCVR (I) | 4 to 6 | Sec | 10 ⁻⁸ |
| | R_GPS_ECEF_RCVR (I) | 7 to 12 | ft | 1/16 |
| | X Position | | | |
| | Y Position | | | |
| | Z Position | | | |
| | V_GPS_ECEF_RCVR (I) | 13 to 18 | ft/sec | 1/512 |
| | X Velocity | | | |
| | Y Velocity | | | |
| | Z Velocity | | | |
| | GPS_STATUS_FOM_GDOP_RCVR (I) | 19 | | |
| | PPS_STATUS_WORD_RCVR (I) | 20 | | |
| | CHAN_STATUS (I) | 21 to 29 | | |
| TFOM_RCVR (I) | 30 | | | |
| MISALIGNMENT RSS | 31 | arc sec | 1 | |

Table 9 NAV Mode Word RCVR (I)

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-----------------------------------|------------|---------------------|---------------------------|
| Nav data validity flag | 0 | 1 | Valid |
| Actual GPS R/P mode | 1-2 | <u>1</u> <u>2</u> | Test Mode |
| | | 0 0 | INITIALIZATION mode |
| | | 0 1 | NAVIGATION mode |
| | | 1 0 | NAVIGATION mode |
| | | 1 1 | Not Used |
| BIT in progress flag | 3 | 0 | BIT in progress |
| Prelaunch standby mode flag | 4 | 1 | Standby mode ON |
| Initialization data request | 5 | 1 | Need initialization data |
| Almanac downloaded | 6 | 1 | Download complete |
| PPS or SPS mode flag | 7 | 1 | PPS Mode |
| | | 0 | SPS Mode |
| UTC valid flag | 8 | 1 | UTC Valid |
| | | 0 | UTC Not Valid |
| Lever arm correction flag | 9 | 1 | Correction used |
| Spare | 10-12 | - | - |
| GPS filter Mode | 13-14 | <u>13</u> <u>14</u> | Unaided Mode (PVA) |
| | | 0 0 | Aided Mode (INS) |
| | | 0 1 | Not Used |
| | | 1 0 | Not Used |
| | | 1 1 | Not Used |
| Spare | 15 | - | - |
| Announced reset flag | 16 | 1 | Reset Complete |
| Initialization states flag | 17 | 1 | Initialization states set |
| IMU reconfiguration complete flag | 18 | 1 | Reconfiguration complete |
| Filter restart complete flag | 19 | 1 | Filter restart complete |
| R/P autonomous reset flag | 20 | 1 | R/P autonomously reset |
| Spare | 21-29 | - | - |

| GPS Mode Command Check | 30-31 | <u>30</u> | <u>31</u> | |
|------------------------|-------|-----------|-----------|---------------------|
| | | 1 | 1 | Test Mode |
| | | 1 | 0 | Initialization Mode |
| | | 0 | 1 | Navigation Mode |
| | | 0 | 0 | Not Used |

Table 10 GPS Status FOM GDOP RCVR (I)

| Bit Parameter | Bit Number | Bit Value | Meaning |
|--|------------|-----------|---|
| Spare | 0-1 | - | Not used |
| Incorporation less than 4 measurements | 2 | 1 | R/P incorporating less than four measurements in solution |
| R/P Failure | 3 | 1 | R/P Failed |
| Spare | 4 | - | Not Used |
| Battery Low | 5 | 1 | R/P battery voltage low |
| Spare | 6 | - | Not Used |
| GDOP | 7-10 | 0-15 | Integer indication of GDOP |
| Figure of merit | 11-14 | 0-9 | Integer indication of estimated position error |
| Spare | 15 | - | Not used |

Table 11 Channel Status (I) WORD 1

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-----------------------------------|------------|-----------|---------------------------------------|
| Channel 1 Fault | 0 | 1 | Channel Fault |
| Channel 1 Antenna | 1 | 0 1 | Lower antenna Upper antenna |
| Channel 1 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 1 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 1 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 1 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 1 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 12 Channel Status (I) WORD 2

| Bit Parameter | Bit Number | Bit Value | Meaning |
|---------------------------------------|------------|---|--|
| Channel 1 Carrier to Noise Ratio, C/N | 0-5 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 1 Code Tracking Status | 6-7 | <u>6</u> <u>7</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |
| Channel 2 Carrier to Noise Ratio, C/N | 8-13 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 2 Code Tracking Status | 14-15 | <u>14</u> <u>15</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |

Table 13 Channel Status (I) WORD 3

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-------------------|------------|-----------|--------------------------------|
| Channel 2 Fault | 0 | 1 | Channel Fault |
| Channel 2 Antenna | 1 | 0 1 | Lower antenna Upper antenna |

| | | | |
|-----------------------------------|-------|--------|---------------------------------------|
| Channel 2 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 2 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 2 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 2 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 2 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 14 Channel Status (I) WORD 4

| Bit Parameter | Bit Number | Bit Value | Meaning |
|---------------------------------------|------------|---|--|
| Channel 3 Carrier to Noise Ratio, C/N | 0-5 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 3 Code Tracking Status | 6-7 | <u>6</u> <u>7</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |
| Channel 4 Carrier to Noise Ratio, C/N | 8-13 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 4 Code Tracking Status | 14-15 | <u>14</u> <u>15</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |

Table 15 Channel Status (I) WORD 5

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-----------------------------------|------------|-----------|---------------------------------------|
| Channel 3 Fault | 0 | 1 | Channel Fault |
| Channel 3 Antenna | 1 | 0 1 | Lower antenna Upper antenna |
| Channel 3 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 3 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 3 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 3 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 3 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 16 Channel Status (I) WORD 6

| Bit Parameter | Bit Number | Bit Value | Meaning |
|---------------------------------------|------------|---|--|
| Channel 5 Carrier to Noise Ratio, C/N | 0-5 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 5 Code Tracking Status | 6-7 | <u>6</u> <u>7</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |
| Channel 6 Carrier to Noise Ratio, C/N | 8-13 | 0-63 | Carrier-to-noise ratio with LSB represents 1 dB Hz |
| Channel 6 Code Tracking Status | 14-15 | <u>14</u> <u>15</u> 0 0 1 0 0 1 1 1 | Code Tracking P Code Tracking C/A Code Tracking Y Code Tracking Not Used |

Table 17 Channel Status (I) WORD 7

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-----------------------------------|------------|-----------|---------------------------------------|
| Channel 4 Fault | 0 | 1 | Channel Fault |
| Channel 4 Antenna | 1 | 0 1 | Lower antenna Upper antenna |
| Channel 4 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 4 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 4 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 4 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 4 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 18 Channel Status (I) WORD 8

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-----------------------------------|------------|-----------|---------------------------------------|
| Channel 5 Fault | 0 | 1 | Channel Fault |
| Channel 5 Antenna | 1 | 0 1 | Lower antenna Upper antenna |
| Channel 5 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 5 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 5 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 5 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 5 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 19 Channel Status (I) WORD 9

| Bit Parameter | Bit Number | Bit Value | Meaning |
|-------------------|------------|-----------|--------------------------------|
| Channel 6 Fault | 0 | 1 | Channel Fault |
| Channel 6 Antenna | 1 | 0 1 | Lower antenna Upper antenna |

| | | | |
|-----------------------------------|-------|--------|---------------------------------------|
| Channel 6 Frequency | 2 | 0 1 | L1 frequency L2 frequency |
| Channel 6 Ephemeris | 3 | 1 | Ephemeris downloaded |
| Channel 6 State | 4-6 | 0-7 | Indication of channel operating state |
| Channel 6 Hardware Channel Number | 7-9 | 0-7 | Receiver physical channel |
| Channel 6 Satellite Number | 10-15 | 0-63 | PRN number of satellite being tracked |

Table 20 GPC to GPS Data One Message Description

| R/P Mode | GPC_TO_GPS_DATA_ONE Content | Halfword Number | Units | Scale Factor |
|-----------------------|-----------------------------|-----------------|--------|------------------|
| TEST/ INIT/ NAV | HEADER_1 (AAAA) | 0 | | |
| | GPS_CONTROL_WORD | 1 to 2 | | |
| | T_REF_WEEK_OUT | 3 | weeks | 1 |
| | T_REF_SEC_OUT | 4 to 6 | sec | 10 ⁻⁸ |
| | R_REF_OUT | 7 to 12 | ft | 1/16 |
| | X,Y, Z POSITION COMPONENTS | | | |
| | V_REF_OUT | 13 to 18 | ft/sec | 1/512 |
| | X,Y, Z VELOCITY COMPONENTS | | | |
| | SPARE | 19 to 22 | | |
| | CREW_SAT_HEALTH_STATUS | | | |
| | SIG_R_COV | 27 to 28 | ft | 1 |
| | SIG_V_COV | 29 | ft/sec | .01 |
| SPARE | 30 to 31 | | | |

Table 21 GPC to GPS Data Two Message

| R/P Mode | GPC_TO_GPS_DATA_TWO content | Halfword content | Units | Scale factor |
|-------------------------------|-----------------------------|------------------|-------|--------------------|
| INIT | HEADER_3 (CCCC) | 0 | | |
| | GPS_UPPER_ANTENNA_NB | 1 to 3 | | |
| | GPS_LOWER_ANTENNA_NB | 4 to 6 | | |
| | GPS_UPPER_ANTENNA_CG | 7 to 9 | | |
| | GPS_LOWER_ANTENNA_CG | 10 to 12 | | |
| | GPS_UPPER_ANTENNA_BORESIGHT | 13 to 15 | | |
| | GPS_LOWER_ANTENNA_BORESIGHT | 16 to 18 | | |
| | SPARES | 19 to 32 | | |
| | NAV | HEADER_2 (BBBB) | 0 | |
| BODY ROLL ATTITUDE (I) | | 1 | rad | PI/2 ¹⁵ |
| BODY PITCH ATTITUDE (I) | | 2 | rad | PI/2 ¹⁵ |
| BODY YAW ATTITUDE (I) | | 3 | rad | PI/2 ¹⁵ |
| SPARE | | 4 to 6 | | |
| BARO AIDING TIME TAG WEEK | | 7 | week | 1 |
| BARO AIDING TIME TAG | | 8 to 10 | sec | 10 ⁻⁸ |
| RADAR ALTIMETER TIME TAG WEEK | | 11 | week | 1 |
| RADAR ALTIMETER TIME TAG | | 12 to 14 | sec | 10 ⁻⁸ |
| BARO ALTIMETER | | 15 to 16 | ft | 1 |
| RADAR ALTIMETER | | 17 | ft | 1/2 |
| SPARES | 18 to 32 | | | |

Table 22 GPS Control Word

| Bit Parameter | Bit Number | Bit Value | | Meaning |
|---|------------|--------------------|----|---|
| GPS Mode Command | 0-1 | 0 | 1 | TEST Mode INITIALIZATION Mode NAVIGATION Mode Not Used |
| | | 0 | 0 | |
| | | 0 | 1 | |
| | | 1 | 0 | |
| | | 1 | 1 | |
| Flag indicating position and velocity available for announced reset | 2 | 1 | | Data available |
| Flag indicating baro data good | 3 | 1 | | Data good |
| Flag indicating radar data good | 4 | 1 | | Data good |
| Command GPS R/P to differential mode | 5 | 1 | | Differential mode command |
| Flag indicating valid position and velocity data | 6 | 1 | | Position and velocity data valid |
| Spare | 7 | | | |
| Flag indicating position and velocity available for GPS initialization | 8 | 1 | | Initial state available |
| Flag indicating position and velocity available for receiver state reset due to IMU reconfiguration | 9 | 1 | | Reset state available |
| Lever arm corrections available | 10 | 1 | | Lever arm corrections available |
| Spares | 11-14 | | | |
| Filter restart flag | 15 | | | Filter restart flag |
| Flag indicating valid attitude data | 16 | 1 = valid attitude | | |
| GN&C GPC Major Mode | 17 to 27 | | | |
| Spares | 28-29 | | | |
| GPS Mode Command | 30 to 31 | 30 | 31 | TEST Mode INITIALIZATION Mode NAVIGATION Mode Not Used |
| | | 1 | 1 | |
| | | 1 | 0 | |
| | | 0 | 1 | |
| | | 0 | 0 | |

20.4.5 Shuttle GPS Configuration / Crew Interface

With the first flight of OI-27 (STS-96, reference FSW CR 92097B and CR 92067D) the GPS software was able to support mixed fleet operations of a single string or three GPS receivers (following removal of TACANs). A mixed fleet operation of both single-string GPS (with TACANs) or triple-string GPS will continue until the end of the Shuttle program, with OV-103 and 104 being single-string and OV-105 three-string. The driver behind mixed fleet operations is the Orbiter Maintenance Down Period (OMDP) schedule when TACANs, wiring, and antennas are removed, and GPS receivers, wiring, and antennas were installed in OV-105. The other Orbiters will not be upgraded to three-string GPS because of the anticipated end of the Shuttle program in 2010.

20.4.5.1 Single String GPS Configuration

20.4.5.1.1 Single String GPS Switch Configuration

The single string configuration for OV-103 and OV-104 is not the same due to panel and wiring differences between the vehicles, but both vehicles do use Flight Critical MDM FF2 for communication

between the MAGRS-3S and GPC (See Figure 7). The single string GPS switch configurations are shown in SSSH Drawing 9.18. On OV-103, each preamp can be powered via MNA or MNC, however nominally only MNC is used to power the preamps because it is also used to power the MAGR itself. The MNA preamp circuit breakers are nominally opened on OV-103 and the preamp power switches remain in MNC. For OV-104, only MNB is used to power both MAGR and preamps so no special configuration is required. A control bus and its associated main bus power supply with corresponding RPC's, supplies primary power to the GPS Receiver. In addition, an encryption switch has also been installed, which will be moved from the "NORMAL" position to the "ZEROIZE" position for non-approved landing sites (for security protection). When placed in the Zeroize position, the GPS will default to SPS mode, and cannot be returned to the PPS mode until a new key is loaded by ground equipment. The encrypt switch is powered via internal GPS receiver battery power.

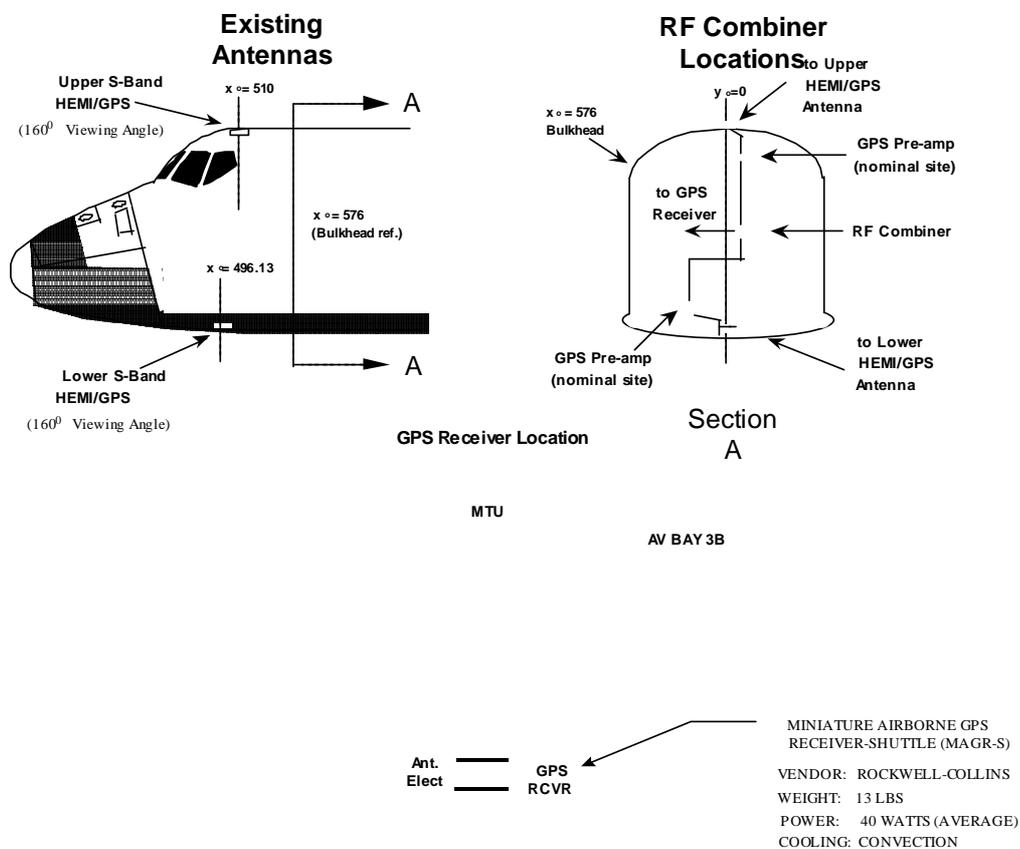


Figure 7 GPS Configuration (Single-String Flights)

20.4.5.1.2 Single-String GPS Installation

The installation locations of the GPS receivers and antennas are shown in Figure 7; the antennas installed on single-string vehicles as well as the antennas for GPS 2 on OV-105 are co-located with S-band Hemi antennas.

20.4.5.2 Single String GPS in PASS

20.4.5.2.1 PASS Horizontal Situation Display (SPEC 50)

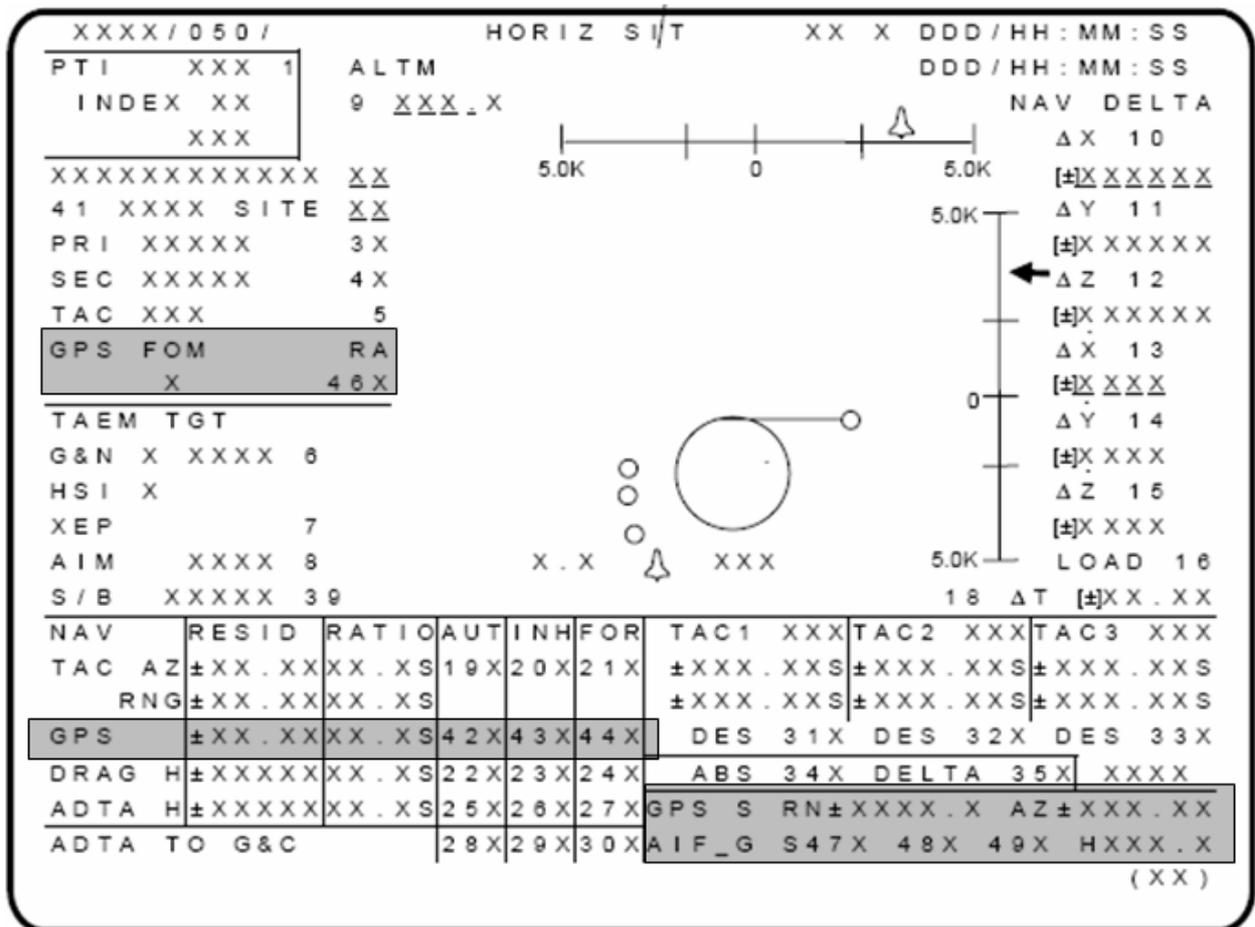


Figure 8 Horizontal Situation Display (SPEC 50) for Three TACAN & One GPS

In OI-30 the PASS HORIZONTAL SITUATION Display (SPEC 50) supports single string GPS operations in OPS 3, and the GPS STATUS Display (SPEC 55) provides a crew interface, health and status monitoring, and control items for GPS.

On SPEC 50 (see Figure 8, shaded area only), in addition to all of the TACAN data and item entries that were retained, an abbreviated set of GPS parameters is available, namely, the commands and data relevant to GPS incorporation into onboard navigation and guidance. The GPS FOM is displayed below the TACAN selection section. Next to this, item 46 allows radar altimeter data to be used to aid the GPS receiver. The GPS range and azimuth data (at the bottom-right of the screen) display the distance to the selected TACAN station, and can be used to compare the TACAN measurements to GPS to aid in identifying TACAN problems.

RESID and RATIO are indications of GPS data quality (lower left); RESID is the difference in nautical miles between the GPS and navigational software position vectors. RATIO is the ratio of the position or velocity residual to the maximum allowable position or velocity residual. A 'P' for position or a 'V' for velocity will be driven in the first space of the RATIO data field to indicate which sort of ratio is being displayed (the 'P' or 'V' will be double overbright if a propagated state is being used because no GPS

state has passed the residual edit test on the current NAV cycle). Both fields will be blank in OPS 1 or OPS 6, or if no GPS receivers are installed.

If $RATIO < 1$ (indicating good agreement between GPS and navigation), the status field next to it will be blank; if $RATIO > 1$, a '↓' will be driven in that status field; if $RATIO > 1$ and the crew force GPS data to NAV anyway (by executing item 44), an '↑' will be driven in the status field.

Items 42, 43, and 44 set the GPS to Navigation Auto/Inhibit command and the GPS to Navigation Force command. Executing item 42 ('AUTO') allows the navigation filter state(s) to be automatically reset/updated with the selected GPS state (when available) every I-Load specified time interval (for OI-30, every 42 seconds in Ops 3). Executing item 43 ('INHIBIT') inhibits the resetting of navigation filter state(s) with the selected GPS state. Items 42 and 43 are mutually exclusive.

Executing item 44 ('FORCE') forces the navigation filter state to be automatically reset/updated with the selected GPS state on the next navigation cycle, the FORCE command taking precedence over the setting of the GPS to Navigation Auto/Inhibit command on that cycle. While the FORCE command is being executed, the setting of the GPS to Navigation Auto/Inhibit command remains as set by the crew. The 'FORCE' command is removed by navigation after the navigation reset is executed, so that the previous setting of the GPS to Navigation Auto/Inhibit command again controls the resetting of the navigation state with the selected GPS state on subsequent navigation cycles. Note that the 'FORCE' is a one-time force. An '*' next to the item number represents the selected setting. Note that only one setting will be indicated on a given navigation cycle. The indication is FORCE if the GPS to Navigation Force Command was selected. If the GPS to Navigation Force Command was not selected, the indication is either AUTO or INHIBIT, depending on the selection of the GPS to Navigation Auto/Inhibit Command.

The initial setting of the GPS to Navigation Auto/Inhibit command is 'INHIBIT'. It will also be set to 'INHIBIT' at OPS transitions. The initial setting of the GPS to Navigation Force command is 'OFF' and it will be set to 'OFF' at OPS transitions. The GPS to Navigation commands can also be set on SPEC 50 with the setting there also shown on the STATUS display. If any of these items are executed in OPS 1 or OPS 6, an 'ILLEGAL ENTRY' message will result. In OPS 3, execution of these items is controlled by the value of the GPS_LOCKOUT I-load parameter. If GPS_LOCKOUT is 0, 1, or 2, execution of these items will produce an 'ILLEGAL ENTRY' message in OPS 3 as well; if GPS_LOCKOUT is 3 or more, execution of these items in OPS 3 will be permitted.

Items 47, 48, and 49 set the GPS to Guidance and Control Auto/Inhibit/Force command. An '*' next to the item number set by the application software represents the selected setting. The three item entries are mutually exclusive. Selecting item 47 ('AUTO') commands the application software to automatically reset/update the user parameter (UPP) state vector with the selected GPS state every navigation cycle (when a selected state is available), rather than resetting the UPP state with the selected navigation filter state. Item 48 ('INHIBIT') inhibits the application software from using the selected GPS state to reset the UPP state. Item 49 sets the command to 'FORCE'. In this setting, the command performs two functions in the application software: (1) it forces the GPS receiver states from all installed receiver states, that are determined by the receiver to have good data (receiver data good is on) and are not deselected by the crew (item 26, 27, or 28 (DES RCVR) on SPEC 55), to be processed by the GPS state selection filter, forcing a selection of a GPS state; and (2) it forces the use of this selected state in resetting the UPP state for guidance. Unlike the GPS to Navigation Force command, which is a one-time force, the GPS to guidance and control 'FORCE' condition remains active until either item 47 or 48 is selected.

If item 47 is executed and no selected GPS state vector is available, the application software will command a down arrow in the status field before the item 47 label. The down arrow will be removed once a selected state becomes available. If item 49 is executed and no selected GPS state is available and all installed receivers are either deselected or indicate that their state data are not good, the application software will also command a down arrow in the status field before the item 47 label. The down arrow will be removed once the state of a receiver which has not been deselected is indicated as good by the receiver, since the GPS to G&C FORCE setting will force a selected state to be made available on that navigation cycle.

In preland, if TACAN is not in FORCE, item 47 ('AUTO') has been selected, MLS data good, and a selected GPS state is available, the application software will command a down arrow in the parameter status indicator field before the item 47 label, indicating to the crew that the processing of MSBLS

measurements by the application software and using the MSBLS derived navigation state for guidance has precedence over the "automatic" use of the selected GPS state in providing the state for guidance. If item 49 ('FORCE') has been selected, however, no down arrow will appear in the parameter status indicator field before the item 47 label and the following application software conditions apply:

- (1) the application software will use the selected GPS state (when available) for guidance;
- (2) MSBLS measurements will not be processed by the application software and if TACAN is in AUTO, TACAN measurements will be processed.

The GPS to Guidance and Control command will initially be set to 'INHIBIT'. It will also be set to 'INHIBIT' at OPS transitions. The GPS to G&C command can also be set on SPEC 55 with the setting there also shown on the HORIZ SIT display.

If any of these items are executed in OPS 1 or OPS 6, an 'ILLEGAL ENTRY' message will result. In OPS 3, execution of these items is controlled by the value of the GPS_LOCKOUT I-load parameter. If GPS_LOCKOUT is 0, 1, or 2, execution of these items will produce an 'ILLEGAL ENTRY' message in OPS 3 as well; if GPS_LOCKOUT is 3 or more, execution of these items in OPS 3 will be permitted.

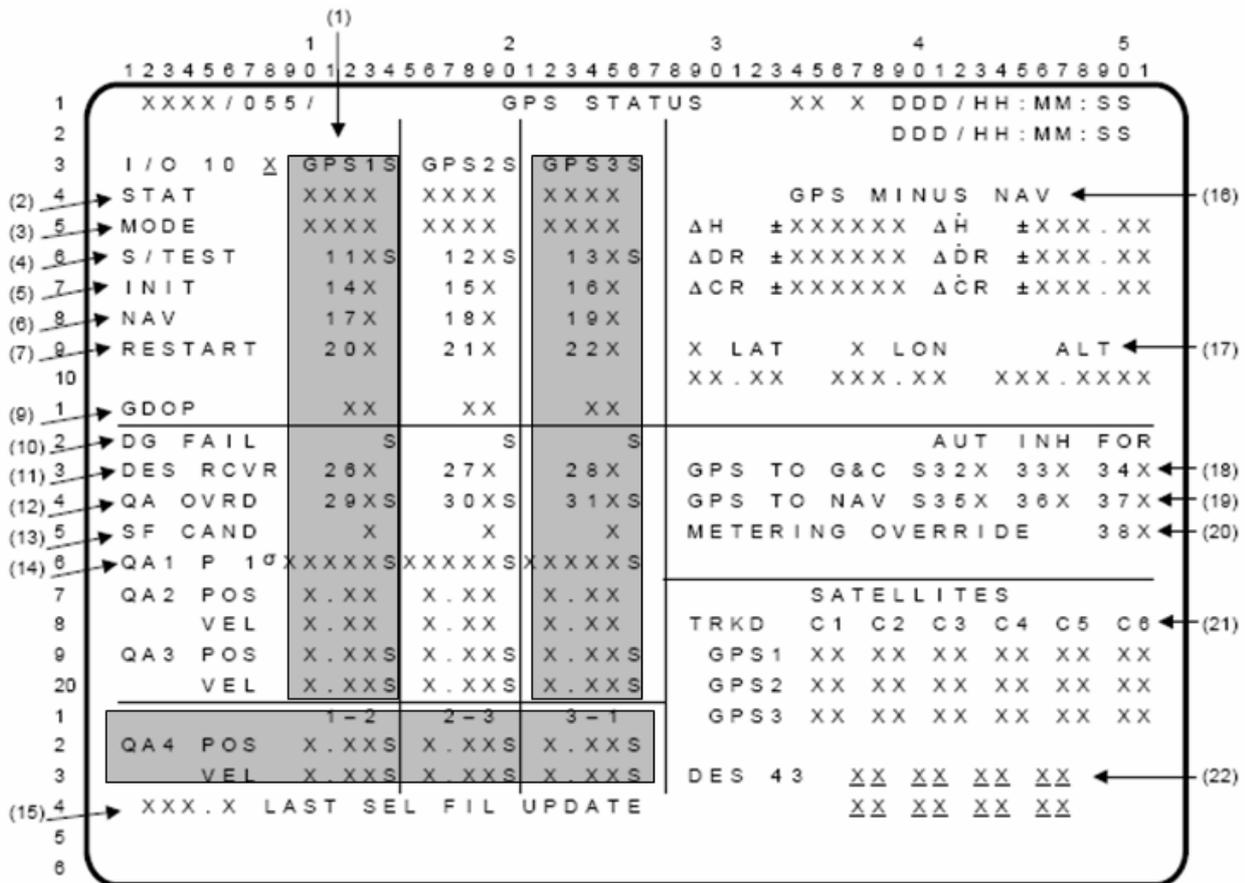


Figure 9 PASS GPS STATUS Display (SPEC 55)

20.4.5.2.2 PASS GPS Status Display (SPEC 55)

The Global Positioning System STATUS display (SPEC 55) provides the crew with the capability to monitor and control GPS operations. It is available in GNC OPS 1, 2, 3, 6, 8, and 9. All information associated with receivers which are not installed shall be blanked with the exception of 'NI' (Not Installed) being displayed in the appropriate receiver status fields. Execution of item entries associated with the

uninstalled receivers shall generate an 'ILLEGAL ENTRY' message. NOTE: Items 1 thru 9 are not used and execution of these items will result in an 'ILLEGAL ENTRY' message.

On SPEC 55 (see Figure 9 , the shaded area is for three string configuration only), during single-string operations, only the data and item entries applicable to GPS #2 will be activated; any commands addressing non-installed units will be regarded as illegal entries. In addition to the data associated with GPS's #1 and #3 being blanked, all data related to QA4 is also de-activated, as this quality assessment requires at least two units for relative comparison; all other QA data is still applicable to a single GPS.

SPEC 55 is available in all OPS Modes (OPS 1/6, 2, 3, 8, and 9), but is primarily intended to be an on-orbit display.

(1) GPS1, GPS2, GPS3. Item 10 provides the capability to control GPS receiver I/O in GNC OPS 9. 'Item 10+X EXEC' (where X=0, 1, or 2) controls the execution of one of the following three GPS receiver I/O options:

0 - Disabled GPS receiver I/O, indicated by a '0' beside item 10. When in GNC OPS 9, all GPS cyclical processing functions shall be suspended.

1 - 6.25 Hz non-commfaultable GPS to GPC input only, indicated by a '1' beside item 10.

2 - Flight GPS I/O of 1.04 Hz GPS to GPC input and 6.25 Hz GPC to GPS output indicated by a '2' beside item 10.

GPS I/O shall initially be disabled in GNC OPS 9 and a '0' shall be driven beside item 10. If no GPS receivers are installed, then GPS receiver I/O is disabled in all OPS and attempts to execute item 10 shall cause an 'ILLEGAL ENTRY' message. Otherwise, on the first execution of 'ITEM 10+1 EXEC', a '1' is driven beside item 10; on the second item 10 execution, the only values can be 'item 10+0 EXEC' or 'item 10+2 EXEC'. An 'item 10+2 EXEC' will enable flight I/O and drive a '2' beside the item. When in I/O state 1 or 2 and 'item 10+0 EXEC' will return GPS I/O to the disabled state and display a '0' beside the item. The legal sequence may be repeated indefinitely in GNC OPS 9.

The GPS1, GPS2, and GPS3 column headers identify the item entries and display parameters associated with a GPS receiver. For the single string GPS configuration, an '*' will be displayed next to the GPS column header(s) in the parameter status field if the receiver's state is selected. When three GPS receivers are installed, a 'Blank' will be displayed in the column header of every GPS eligible for selection.

The corresponding fields for uninstalled receivers will remain blank.

(2) STAT. This field displays the status of each GPS receiver.

| Display | Message Description |
|---------|--|
| NI | Receiver is not installed |
| M | Communication fault |
| BIT | Receiver Built-In-Test in progress |
| RPF | Receiver processor failure |
| BATT | GPS battery low |
| DGF | GPS data good failure |
| (blank) | Nominal operation – No GPS related anomaly |

If a receiver processor failure occurs, a CLASS 3 alarm will be annunciated by GAX for all operational sequences, except for OPS 9. 'RPF' and 'M' will be displayed double overbright. STAT is active regardless of whether the corresponding receiver is installed.

(3) MODE. This field displays the current operating mode of the receiver.

| Display | Message Description |
|---------|--|
| INIT | Initialization mode |
| TEST | Self-Test mode |
| INS | Inertial Navigation System submode |
| PVA | Position, Velocity, Acceleration submode |
| Site | 3-character vehicle location mnemonic |
| (blank) | Receiver is not in an operating mode |

If the receiver is reporting prelaunch standby or stationary mode, a three-character mnemonic representing the last crew-selected IMU calibration site or landing site is displayed in the mode field (prior to launch the IMU calibration site, as selected on SPEC 104, is displayed; post liftoff, the first three characters of the last selected runway on SPEC 50 are displayed). When the receiver is reporting prelaunch standby or stationary mode, the site location will supersede 'INS' or 'PVA' but not 'INIT' or 'TEST'. The receiver will normally be in prelaunch standby nav submode only in MM901 and MM101.

The latest valid receiver mode is displayed. If the receiver is not installed, the displayed mode, which is initialized to blank at an OPS transition, will remain blank. If the receiver is powered off or failed at an OPS transition, the displayed mode will remain blank until the receiver enters a valid mode. Once the receiver is in a valid mode, the displayed mode will not be blank for that OPS, since the latest valid mode remains on the display, even if the receiver is powered off or if it fails. If the mode field is blank, the STAT parameter (2) should be referenced for further information. During a communications fault the receiver MODE indication will remain in the operating mode last displayed.

(4) S/TEST. Items 11, 12, and 13 provide the capability to command a self-test for the associated GPS receiver. An '*' shall be displayed while self-test is being commanded. 'BIT' will be driven in the receiver status field while the self-test is in progress. 'BIT' will be removed from the receiver status field on completion of the self-test. The receiver will remain in S/TEST upon completion of S/TEST until commanded to INIT or NAV mode, at which time the '*' will be removed. Reentering an item to command an S/TEST while the receiver is already in S/TEST will not result in a restart of the S/TEST and will result in an 'ILLEGAL ENTRY' message. A new S/TEST can be commanded only when a receiver is in INIT or NAV mode. The items of (4), (5), and (6) for an individual receiver are mutually exclusive.

A status indicator provides feedback on the progress and completion of the commanded self test. The status indicator for each receiver is initially an up arrow and will be blanked while a self-test is in progress. Upon completion of the self-test, an up arrow will be shown if the self-test completed successfully or a down arrow will be shown if the receiver failed the self-test. The up or down arrow will remain on the display until the receiver enters either INIT or NAV mode.

For uninstalled receivers, executing item 11, 12, or 13 will result in an 'ILLEGAL ENTRY' message.

(5) INIT. Items 14, 15, and 16 provide the capability to command the initialization of a GPS receiver. An '*' shall be displayed while initialization operations are in progress and while the receiver remains in the INIT mode. The receiver gives no indication when it completes the initialization mode. Commanding INIT mode while the receiver is in the INIT mode will result in an 'ILLEGAL ENTRY' message. The receiver will

remain in the INIT mode until commanded to S/TEST or NAV mode, at which time the '*' will be removed. The items of (4), (5), and (6) or an individual receiver are mutually exclusive.

For uninstalled receivers, executing item 14, 15, or 16 will result in an 'ILLEGAL ENTRY' message.

(6) NAV. Items 17, 18, and 19 provide the capability to mode the GPS receiver to the navigation mode. An '*' shall be displayed once the receiver is commanded to the navigation mode. The receiver will remain in the NAV mode until commanded to S/TEST or INIT mode, at which time the '*' will be removed. Commanding NAV mode while the receiver is in NAV mode will result in an 'ILLEGAL ENTRY' message. If a receiver is commanded to NAV before a commanded S/TEST or INIT has been completed, the receiver will transition to the NAV mode when it completes the S/TEST or INIT mode. The items of (4), (5), and (6) for an individual receiver are mutually exclusive.

For uninstalled receivers, executing item 17, 18, or 19 will result in an 'ILLEGAL ENTRY' message.

(7) RESTART. Items 20, 21, and 22 provide the capability to command each GPS receiver to restart its navigation filter. This option is available only when the receiver is in the INS NAV submode, indicated by 'INS' being displayed in the MODE field. An 'ILLEGAL ENTRY' message will be generated if the RESTART command is entered and the receiver is not in the INS NAV submode. An '*' shall be displayed in the associated item field while the filter restart operations are in progress for that receiver and the '*' will be removed upon receiver completion of the restart. If the receiver is in the S/TEST or INIT modes, commanding a receiver filter restart will result in an 'ILLEGAL ENTRY' message. For uninstalled receivers, executing item 20, 21, or 22 will result in an 'ILLEGAL ENTRY' message.

(9) GDOP. This field displays the geometric dilution of precision (GDOP) computed in each GPS receiver. GDOP is a function of GPS satellite geometry. Displayed values of GDOP range from 1 to 15, with a smaller number indicating a more favorable satellite geometry. The GDOP data will be blank: (1) in GN&C OPS 9, OPS 1, MM601 when it is determined by the GPS SOP to be static; and (2) in GN&C OPS 2, OPS 3, OPS 8, MM602 and MM603 when GPS data is determined by the GPS RCVR State Proc to be invalid.

The corresponding fields for uninstalled receivers will remain blank.

(10) DG FAIL. This field indicates whether the receiver data used by the GPS receiver state processing principal function are valid. If the receiver is in the navigation mode and the receiver data are invalid, or the receiver state time tag is more than 3.84 seconds older than the current navigation filter state time tag, a down arrow is displayed in double overbright; otherwise the status field is blanked. The status field is also blanked if the receiver is in the TEST or INIT mode.

The corresponding fields for uninstalled receivers will remain blank. All fields are blanked in OPS 9.

(11) DES RCVR. Items 26, 27, and 28 provide the capability for each receiver to be deselected or reselected (toggle) for use by the GPS selection filter. Deselection is indicated by an "*" adjacent to the DES item number. A blank indicates that the associated receiver has not been deselected.

Execution of item entries 26, 27, or 28 in GN&C OPS 1, OPS 9, or MM601 will result in an 'ILLEGAL ENTRY' message. GPS receivers can be deselected or reselected from HORIZ SIT DISP in OPS 3 and MM602 and 603 (3 GPS receivers installed). In OPS 2 a receiver can be deselected or reselected from REL NAV DISP. Deselection or reselection will carry over OPS transitions.

(12) QA OVERRIDE. Items 29, 30, and 31 are used to force a given receiver's state to be a candidate for GPS state vector selection even though that state may have failed any of three quality assessments (indicated by a down arrow next to one or more of the QA1, QA3, and QA4 checks described in (14)). An '*' is displayed if the override has been selected. QA data continues to be calculated for a GPS even if it

has been selected for a QA OVERRIDE. Note that a QA OVERRIDE does not override a receiver–data–good failure on a GPS (see (10)) or a crew deselection of a GPS (see (11)). In either case, an attempted QA OVERRIDE of that GPS will drive a down arrow in the associated parameter status field, indicating that the QA OVERRIDE command is not being honored. The override applies continuously until it is removed by again executing the item, at which time the parameter field is blanked.

Note that items 29, 30, and 31 will not override a QA4 dilemma and force a given receiver’s state to be a candidate for GPS state vector selection.

For uninstalled receivers, executing item 29, 30, or 31 will result in an ‘ILLEGAL ENTRY’ message.

These parameters are blanked in OPS 9, OPS 1, and MM601.

(13) SELECTION FILTER CANDIDATE. An ‘*’ in the status field indicates that the state for the given receiver is a candidate for GPS state selection. A down arrow indicates that the state for the given receiver is not a candidate for GPS state selection. In order to be a candidate for selection the receiver cannot have been deselected (see (11)); must have valid data as indicated by the receiver (see (10)); must have passed QA1, QA3, and QA4 quality assessments (see (14)) unless the QA override for that receiver (see (12)) has been set; or it must have been forced (see (18)).

The field will also remain blank in OPS9, OPS 1, and MM601.

The corresponding field for uninstalled receivers will remain blank.

(14) QA1, QA2, QA3 and QA4

QA1 P1 This field displays the Figure of Merit (FOM) or estimate of position error for each receiver. The FOM is related to the error as follows:

| FOM integer | Position error (P), one sigma (1σ) [feet] |
|-------------|--|
| 0 | (blank) |
| 1 | 75 |
| 2 | 175 |
| 3 | 250 |
| 4 | 325 |
| 5 | 650 |
| 6 | 1650 |
| 7 | 3275 |
| 8 | 16400 |
| 9 | 99999 |

The FOM data from the GPS Receiver State Processing Principal Function will be blank when the receiver is not in NAV mode (i.e., in TEST or INIT mode) or when the GPS NAV data used by the GPS Receiver State Processing Principal Function are not valid. They are blanked in OPS 9, OPS 1, and MM601 when determined by the GPS SOP to be static. If the FOM exceeds a pre–mission determined I–Load tolerance, a down arrow shall be driven in the appropriate parameter status indicator field next to the displayed FOM value and displayed double overbright. If the receiver state estimate fails the QA1 FOM quality assessment check, it is excluded as a candidate from the GPS state vector selection process, unless forced to be a candidate for selection by the crew ((13) and (18)), in which case the down arrow remains.

The corresponding fields for uninstalled receivers will remain blank.

QA2 POS & VEL This selection displays the maximum position (POS) and maximum velocity (VEL) residual ratios. The ratios give the crew an indication as to how well the GPS and NAV states agree only and are not used to determine candidacy for GPS state vector selection. The position and velocity residual ratios are computed by differencing the navigation state and the individual GPS state in UVW coordinates, and dividing each of the UVW difference components by a corresponding UVW I-Load derived tolerance.

The calculation of UVW position and velocity QA2 residuals shall be in a coordinate system that is GNC Major Mode dependent. In Modes 304, 305, 602, and 603, the UVW coordinate system shall be an earth relative coordinate system. For all other major modes where these residuals are computed, the UVW coordinate system shall be an inertial coordinate system. The actual QA2 calculations use all six components of position and velocity, but only the maximum position and velocity residual ratio is displayed.

The data are static when the receiver is in the TEST or INIT mode and the display field will be blank in OPS 9, OPS 1, and MM601.

The corresponding fields for uninstalled receivers will be blank for all OPS.

QA3 POS & VEL This section displays the QA3 position (POS) and velocity (VEL) residual ratios. These ratios are computed as follows: (1) The position and velocity differences between the current GPS receiver orbiter state estimate which has been propagated to the current navigation filter time and that receiver's valid state estimate from the previous navigation cycle which has been propagated to the current navigation filter time is computed; (2) the ratios are then computed by dividing the position and velocity differences by pre-mission determined I-Load tolerances. The QA3 test gives the crew an indication of the stability of the receiver position and velocity estimates. The GPS receiver state processing QA3 software tests the vector magnitude of the position and velocity difference against pre-mission determined I-Loads. If the QA3 test for either a position or velocity ratio fails (ratio > 1) that receiver's state is then excluded as a candidate from the GPS state vector selection process, unless forced to be a candidate for selection by the crew ((12) and (18)). A down arrow shall then be displayed next to the associated ratio value for a QA3 test failure. The down arrow is displayed if the failure condition exists, regardless of the state of the QA override discrete or the GPS to G&C FORCE discrete. The down arrow is removed when the failure condition no longer exists. If the QA3 blanking flag for a receiver is ON, the QA3 position and velocity ratios and their corresponding status indicators for that receiver are blanked.

The data are static when the receiver is in the TEST or INIT mode and the display field will be blank in OPS 9, OPS 1, and MM601.

The corresponding fields for uninstalled receivers will be blank for all OPS.

QA4 POS & VEL A GPS receiver is available for QA4 processing (receiver to receiver residual ratio calculations) if the receiver data is determined to be valid. Receiver data is valid if its DG FAIL status is showing a blank (10) and the receiver is selected (11). However, QA4 failure/dilemma detection will only be performed on receiver which passes QA1 check or has its QA Override set ON. Display of QA4 Ratios and Status Indicators: This section displays the Quality Assessment 4 (QA4) position (POS) and velocity (VEL) residual ratios for comparison of GPS receiver to GPS receiver. Column 1 displays the differences between GPS receiver 1 and 2; column 2 displays the differences between 2 and 3; and column 3 displays the differences between 3 and 1. Data is displayed whenever the receiver data used by the GPS receiver state processing principal function is valid (DG FAIL blank, (10)) for both GPS receivers in the associated pair. If data for either GPS receiver in the pair of receivers being compared is invalid, then the QA4 data and the associated status indicators shall be blanked for that pair of receivers.

When receiver state data is valid for only one GPS receiver, the QA4 data and the associated status indicators shall be blanked for all pairs of receivers.

The down arrow, is removed when the limit exceedance condition no longer exists for each GPS receiver pair combination or when at least one receiver in a pair combination is deselected, or has failed its QA1 check and its QA Override reset to OFF.

The data is static when the receiver is in the TEST or INIT mode. The display fields shall be blank in GN&C OPS 1, OPS 9, and MM 601.

This section is applicable only when three GPS receivers are installed. For zero or one GPS receiver installed, all QA4 fields shall be blanked.

QA4 Fail: At the three GPS receivers available for QA4 level, if any QA4 receiver to receiver comparisons are failed, and it can be determined which receiver is inconsistent with the remaining two receivers, then that receiver fails QA4 and is excluded as a candidate from the GPS state vector selection process, unless forced to be a candidate by the crew ((12) and (18)). A down arrow shall be displayed to the right of the difference to indicate the two GPS receiver pairs which exceeded the position or velocity difference limit. For example, if GPS receiver 2 were to have failed, then a down arrow should appear for pair 1 - 2 data (column 1) and pair 2 - 3 (column 2). The identification of GPS receiver 2 as common to the two columns displaying a down arrow would show that GPS receiver 2 has failed the QA4 test. GPS receiver 2 would also show a down arrow next to the GPS column header (1). If the number of GPS receivers available for QA4 changes on a later execution cycle to be less than 3, then the QA4 fail indicator is set to OFF (to indicate no failure).

QA4, 3 GPS in Dilemma: At the three GPS receivers available for QA4 level, if all three QA4 receiver to receiver pair comparisons fail, then all three GPS receivers are declared to be in dilemma. When all three GPS receivers are in dilemma, '?' will appear by the GPS column header for all three GPS receivers. The dilemma established at the three GPS receiver level will be reassessed at each subsequent execution cycle as long as there are at least two receivers with valid data for QA4 processing. In a rare scenario where at least two receivers transition from data valid to invalid on the same execution cycle, the three GPS level dilemma established on the prior execution cycle will be latched until either:

- 1) If two or more GPS receivers are later available for QA4 processing and conditions no longer exist for a QA4 dilemma on that execution cycle.
- 2) Two GPS receivers are deselected.

For a QA4 dilemma, neither the GPS to GNC FORCE command (18) nor the QA override (12) will override the QA4 dilemma and force a given receiver's state to be a candidate for GPS state vector selection. The crew must resolve the dilemma and then deselect GPS receivers until only a single GPS receiver remains selected.

QA4, 2 GPS in Dilemma: At the two GPS receivers available for QA4 level, a down arrow shall be displayed by the appropriate QA4 ratio if the remaining GPS receiver pair has exceeded the position or velocity difference limit. Both remaining receivers shall be removed from the candidate set for the GPS state vector selection on the current execution cycle and a '?' will appear by the GPS column header of each receiver.

(15) TIME SINCE LAST SELECTION FILTER UPDATE. This field displays the minutes since the last selection of a GPS state by the GPS state vector selection software function. It indicates the time since at least one of the criteria defined in (13) has been satisfied by any receiver. Should the time since the last selection of a GPS state exceed a pre-mission defined value, GAX will announce a class 3 message and an alert tone, and the time will be displayed double overbright. The time will be reset to zero at OPS transition, after an on-orbit IPL, when GPS receiver data updates the selected state, and when all three receivers are not available. The field will be blanked until a selected GPS state becomes available.

The field will also be blank in OPS 9, OPS 1, and MM601.

(16) GPS MINUS NAV. This section displays the components of the position and velocity vector differences between the selected GPS state estimate and the orbiter navigation filter state estimate in UVW coordinates: altitude (ΔH), downrange (ΔDR), and crossrange (ΔCR) for both position and velocity. Units are ft and ft/sec for position and velocity, respectively. The data fields are blanked when no GPS state is available.

The field will also be blank in OPS 9, OPS 1, and MM601.

(17) GEODETIC COORDINATES. This section displays the geodetic coordinates of the selected GPS state: altitude (ALT), latitude (LAT), and longitude (LON). Altitude is displayed in nautical miles. Latitude is displayed in degrees from the equator with an 'N' or 'S' displayed in the field before the LAT title for northern (LAT > 0) or southern (LAT < 0) hemisphere. Longitude is displayed in degrees from the Greenwich meridian with an 'E' or 'W' displayed in the field before the LON title if the orbiter is east (LON > 0) or west (LON < 0) of the Greenwich meridian, respectively. In MM 304, 305, 602, and 603, altitude is displayed with units of kilo-feet. The data fields are blanked when no GPS state is available.

The field will also be blank in OPS 9, OPS 1, and MM601.

(18) GPS TO GUIDANCE AND CONTROL (G&C). Items 32, 33, and 34 set the GPS to Guidance and Control Auto/Inhibit/Force command. An "*" next to the item number set by the application software represents the selected setting. The three item entries are mutually exclusive. Selecting item 32 ('AUTO') commands the application software to automatically reset/update the user parameter (UPP) state vector with the selected GPS state every navigation cycle (when a selected state is available), rather than resetting the UPP state with the selected navigation filter state. Item 33 ('INHIBIT') inhibits the application software from using the selected GPS state to reset the UPP state. Item 34 sets the command to 'FORCE'. In this setting, the command performs two functions in the application software: (1) it forces the GPS receiver states from all installed receiver states, that are determined by the receiver to have good data (receiver data good is ON) and are not deselected by the crew (item 26, 27, or 28 (DES RCVR)), to be processed by the GPS state selection filter, forcing a selection of a GPS state; and (2) it forces the use of this selected state in resetting the UPP state for guidance. Unlike the GPS to Navigation Force command, which is a one-time force, the GPS to guidance and control 'FORCE' condition remains active until either item 32 or 33 is selected.

If item 32 is executed and no selected GPS state vector is available, the application software will command a down arrow in the status field before the item 32 label. The down arrow will be removed once a selected state becomes available. If item 34 is executed and no selected GPS state is available and all installed receivers are either deselected or indicate that their state data are not good (receiver data good is OFF), the application software will command a down arrow in the status field before the item 32 label. The down arrow will be removed once the state of a receiver which has not been deselected is indicated as good by the receiver, since the GPS to G&C FORCE setting will force a selected state to be made available on that navigation cycle. In preland, if TACAN is not in FORCE, item 32 has been selected, MLS data good, and a selected GPS state is available, the application software will command a down arrow in the parameter status indicator field before the item 32 label, indicating to the crew that the processing of MSBLS measurements by the application software and using the MSBLS derived navigation state for guidance has precedence over the "automatic" use of the selected GPS state in providing the state for guidance.

If item 34 has been selected, however, no down arrow will appear in the parameter status indicator field before the item 32 label and the following application software conditions apply:

- (1) The application software will use the selected GPS state (when available) for guidance;
- (2) MSBLS measurements will not be processed by the application software;
- (3) TACAN measurements will be processed if TACAN is in AUTO.

The GPS to Guidance and Control command will initially be set to 'INHIBIT'. It will also be set to 'INHIBIT' at OPS transitions. The GPS to G&C command can also be set on SPEC 50 with the setting there also shown on the GPS STATUS display.

These items will be allowed or not allowed in a given Operational Sequence depending on the value of the GPS_LOCKOUT I-Load parameter.

| GPS_LOCKOUT VALUE | Description and Use |
|-------------------|---|
| 0 | <ul style="list-style-type: none"> No GPS receivers installed |
| 1 | <ul style="list-style-type: none"> GPS receivers installed Capability to use selected GPS state in the onboard system locked out |
| 2 | <ul style="list-style-type: none"> GPS receivers installed Use of selected GPS state in the onboard system allowed in OPS 2 and OPS 8 |
| ≥ 3 | <ul style="list-style-type: none"> GPS receivers installed Use of selected GPS state in the onboard system allowed in OPS 2, OPS 8, and OPS 3 |

| GPS_LOCKOUT VALUE | 0 | 1 | 2 | ≥ 3 |
|---|---|---|-----|-----------------------|
| GPS_TO_NAV_AI (Items 35 and 36) | I | I | 2,8 | 2,3,8, 602, 603 |
| GPS_TO_NAV_FORCE (Item 37) | | | | |
| GPS_TO_GC_AIF (Items 32,33,34 Legality) | | | | |

“I” indicates ILLEGAL ENTRY

A Number indicates the OPS or MM when the Item Entry is available.

NOTE: All above item entries are ILLEGAL in OPS 9, OPS 1 and MM601 regardless of the value of GPS_LOCKOUT

(19) GPS TO NAV COMMAND. Items 35, 36, and 37 set the GPS to Navigation Auto/Inhibit command and the GPS to Navigation Force command. While in GN&C MM 201 or OPS 8, executing item 35 (GPS to Navigation Auto/Inhibit command is set to ‘AUTO’) allows the navigation filter state to be automatically reset/updated with the selected GPS state (when available). Automatic ‘AUTO’ reset/update of the navigation filter state is not permitted in MM 202. Automatic updates occur at an I-Load specified time interval. Also, the automatic GPS updates to navigation will be terminated, even though the GPS to Navigation Auto/Inhibit command is in ‘AUTO’, if the selected GPS residual ratio edit flag is ON, or the time since the selected GPS state was last updated with receiver data is greater than or equal to an I-loaded limit.

While in OPS 3 or OPS 6, executing item 35 (GPS to Navigation Auto/Inhibit command is set to ‘AUTO’ and MLS measurements not available) allows the navigation filter state to be automatically reset/updated with the selected GPS state (when available). Automatic updates occur at one of two I-Load specified time intervals, depending on whether or not the orbiter is above an I-Loaded altitude for changing GPS update rates. Also, the automatic GPS updates to navigation will be terminated, even though the GPS to Navigation Auto/Inhibit command is in ‘AUTO’, if the selected GPS residual ratio edit flag is ON, the orbiter is at or below an I loaded lower limit altitude for automatic GPS updates, or the time since the selected GPS state was last updated with receiver data is greater than or equal to an I-Loaded limit. Executing item 36 (GPS to Navigation Auto/Inhibit command is set to ‘INHIBIT’) inhibits the resetting of the navigation filter state(s) with the selected GPS state. Items 35 and 36 are mutually exclusive.

Executing item 37 (GPS to Navigation Force command is set to ‘FORCE’) forces the navigation filter state to be automatically reset/updated with the selected GPS state (when available) on the next navigation cycle, the FORCE command taking precedence over the setting of the GPS to Navigation Auto/Inhibit command on that cycle. While the FORCE command is being executed, the setting of the GPS to Navigation Auto/Inhibit command remains as set by the crew. The ‘FORCE’ command is removed by navigation after the navigation reset is executed, so that the previous setting of the GPS to Navigation

Auto/Inhibit command again controls the resetting of the navigation state with the selected GPS state on subsequent navigation cycles. Note that the 'FORCE' is a one-time force.

An '*' next to the item number represents the selected setting. Note that only one setting will be indicated on a given navigation cycle. The indication is FORCE if the GPS to Navigation Force Command was selected. If the GPS to Navigation Force Command was not selected, the indication is either AUTO or INHIBIT, depending on the selection of the GPS to Navigation Auto/Inhibit Command. The initial setting of the GPS to Navigation Auto/Inhibit command is 'INHIBIT'. It will also be set to 'INHIBIT' at OPS transitions.

The initial setting of the GPS to Navigation Force command is 'OFF' and it will be set to 'OFF' at OPS transitions. The GPS to Navigation commands can also be set on SPEC 50 with the setting there also shown on the STATUS display.

These items will be allowed or not allowed in a given Operational Sequence depending on the value of the GPS_LOCKOUT I-Load parameter. See tables in (18).

For a graphical description of how GPS is incorporated in to navigation and G&C, see Figure 10 below.

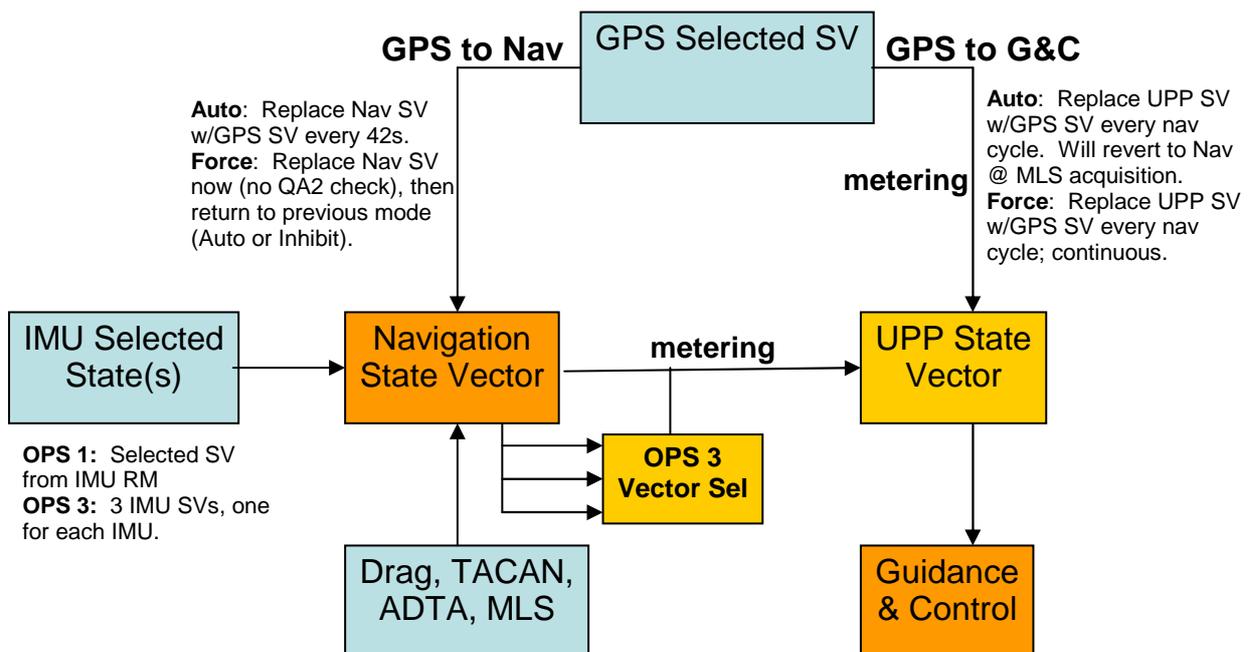


Figure 10 GPS Incorporation Data Flow

(20) METERING OVERRIDE. Item entry 38 (METERING OVERRIDE) shall toggle the discrete controlling the override of navigation principal function metering of the updates to the user parameter reset state. The user parameter reset state is used by the user parameter navigation computation functions and is eventually used by guidance. Depending on the configuration of the navigation system as determined by item entries, either the selected GPS state or the baseline navigation state can be used to update the user parameter reset state. The updates to the user parameter reset state, defined as the difference between the user parameter reset state from the previous navigation cycle propagated to current time and either the baseline navigation state or the selected GPS state as determined by the crew, are metered to be within values defined by permission altitude dependent I-Loads to limit guidance transients. Metering of the updates to the user parameter reset state occurs under the following conditions: continuous metering limit checking when the selected GPS state is being used to update the user parameter reset state; metering until the update to the user parameter state lies within the I-Load tolerance, when switching from using the selected GPS state to using the baseline navigation state to update the user parameter reset state; metering until the update to the user parameter state lies within the I-Load tolerances, when using

the baseline navigation state to update the user parameter reset state and the baseline navigation state has been reset to the selected GPS state. Item 38 is initialized OFF, with no asterisk being displayed (blank), so that metering is being performed under the conditions defined above. When toggled to ON, with an asterisk being displayed, metering is overridden and the updates to the user parameter state are not limited. In this case, the user parameter reset state is either the selected GPS state or the baseline navigation state as determined by prior item entries.

Execution of item 38 in any OPS or major mode other than GN&C OPS 3, and MM 602 and 603 will result in an 'ILLEGAL ENTRY' message being generated.

(21) SATELLITES TRACKED. This section shall display the GPS satellites being tracked by each receiver. Up to a maximum of six receiver channels can be displayed. If the receiver has less than six channels, the data will be blanked for the nonexistent channels. The order of the satellite number displayed reflects a one-to-one correspondence to the receiver channel tracking that satellite. Values from GPS SOP include 'blank' (no satellites being tracked) and 1 to 63. This data will be blank when the receiver is not in the NAV mode or experiencing a communication fault. The corresponding fields for uninstalled receivers will remain blank.

(22) DESELECT SATELLITES. Item entries 43 to 50 allow the crew to deselect satellites, valid satellite ID's being 1 to 63, as candidates for selection by the GPS receivers by building, based on Item Entries, a derived health status word. The '+' delineator deselects satellites and the '-' delineator removes satellites from being deselected as described below. As an example, 'ITEM 43 + 23 EXEC' will indicate to all GPS receivers that satellite ID 23 should not be used as a candidate for selection. If another satellite was previously deselected by item entry 43, an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and no change to deselected satellites occurs. If this satellite, ID 23 in this example, is currently deselected by another Item Entry, one of item entries 44 to 50 in this example, an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and no change to deselect satellites occurs.

'ITEM 43 - 23 EXEC' will indicate to all GPS receivers that satellite 23 should once again be considered as a candidate for selection. If item Entry 43 previously deselected satellite 23, then 'ITEM 43 - 23 EXEC' will result in a blank being displayed, and satellite 23 being available for selection. If satellite 23 is not currently deselected by item Entry 43, then an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and there is no change to deselected satellites.

Entry of satellite number of 0, less than - 63, or greater than + 63 will cause an 'ILLEGAL ENTRY' message to be issued. All satellites are initialized as available for selection. Satellite Deselection information for item entries 43 to 50 shall be maintained in the GNC major function base.

Table 23 GPS GAX Messages

| MML no. | Message Text Major | Message Text Minor | Discussion |
|-----------|--------------------|--------------------|---|
| V94X6021X | RM DLMA | GPS | Message is annunciated for a QA4 dilemma between 2 or 3 GPS's |
| V94X6022X | G55 QA FL GPS | 1 | Message is annunciated whenever there is a failure of QA1, QA3, QA4 for GPS 1 |
| V94X6023X | G55 QA FL GPS | 2 | Message is annunciated whenever there is a failure of QA1, QA3, QA4 for GPS 2 |
| V94X6024X | G55 QA FL GPS | 3 | Message is annunciated whenever there is a failure of QA1, QA3, QA4 for GPS 3 |
| V94X6018X | G55 GPS FAIL | 1 | Message is annunciated whenever GPS 1 has declared a hardware failure (RPF) |
| V94X1049X | G55 GPS FAIL | 2 | Message is annunciated whenever GPS 2 has declared a hardware failure (RPF) |
| V94X6019X | G55 GPS FAIL | 3 | Message is annunciated whenever GPS 3 has declared a hardware failure (RPF) |

| | | | |
|-----------|--------------|-----|---|
| V94X1050X | NAV EDIT | GPS | Message is annunciated whenever nav edit ratio is greater than 1 |
| V94X1048X | NO UPDATE | GPS | Message is annunciated whenever time since last update to GPS SF is greater than I-Load |
| V91X0026X | BCE STRING 1 | GPS | Message is annunciated whenever GPS 1 is commfaulted/ Powered Off |
| V91X0027X | BCE STRING 2 | GPS | Message is annunciated whenever GPS 2 is commfaulted/ Powered Off |
| V91X0028X | BCE STRING 3 | GPS | Message is annunciated whenever GPS 3 is commfaulted/ Powered Off |

20.4.5.3 Single String GPS in BFS

20.4.5.3.1 BFS Horizontal Situation Display (SPEC 50)

In OI-30 the BFS HORIZONTAL SITUATION Display (SPEC 50) supports single string GPS operations in OPS 3, and the GPS STATUS Display (SPEC 55) provides a crew interface, health and status monitoring, and control items for GPS.

On SPEC 50 (see Figure 11 shaded area only), an abbreviated set of GPS parameters is available, namely, the commands and data relevant to GPS incorporation into onboard Nav. The GPS range and azimuth data (at the bottom-right of the screen) display the distance to the selected runway, unlike the PASS where they display bearing and range to the selected TACAN station.

RESID and RATIO are indications of GPS data quality (lower left); RESID is the difference in nautical miles between the GPS and navigational software position vectors. RATIO is the ratio of the position or velocity residual to the maximum allowable position or velocity residual. A down arrow will be displayed beside the ratio parameter if it is greater than or equal to one. Note that it is possible for the GPS velocity residual to be outside of ratio limits and in this case a down arrow might be displayed even though the position ratio is within limits.

Items 42, 43, and 44 set the GPS to Navigation Auto/Inhibit command and the GPS to Navigation Force command. Executing item 42 ('AUTO') allows the navigation filter state(s) to be automatically reset/updated with the selected GPS state (when available) on every navigation filter cycle (as opposed to the PASS which updates navigation at a lower I-loaded rate). Executing item 43 ('INHIBIT') inhibits the resetting of navigation filter state(s) with the selected GPS state. Executing item 44 ('FORCE') forces the navigation filter state to be automatically reset/updated with the selected GPS state, regardless of whether the GPS state passes the ratio edit tests; unlike in the PASS, the BFS GPS force command is a continuous command that updates navigation on every navigation cycle. Because the BFS does not have a GPS to G&C option, the continuous, every-cycle force command is functionally similar to a Force of GPS to G&C in the PASS, in that it essentially eliminates IMU velocity from guidance. These three items are mutually exclusive.

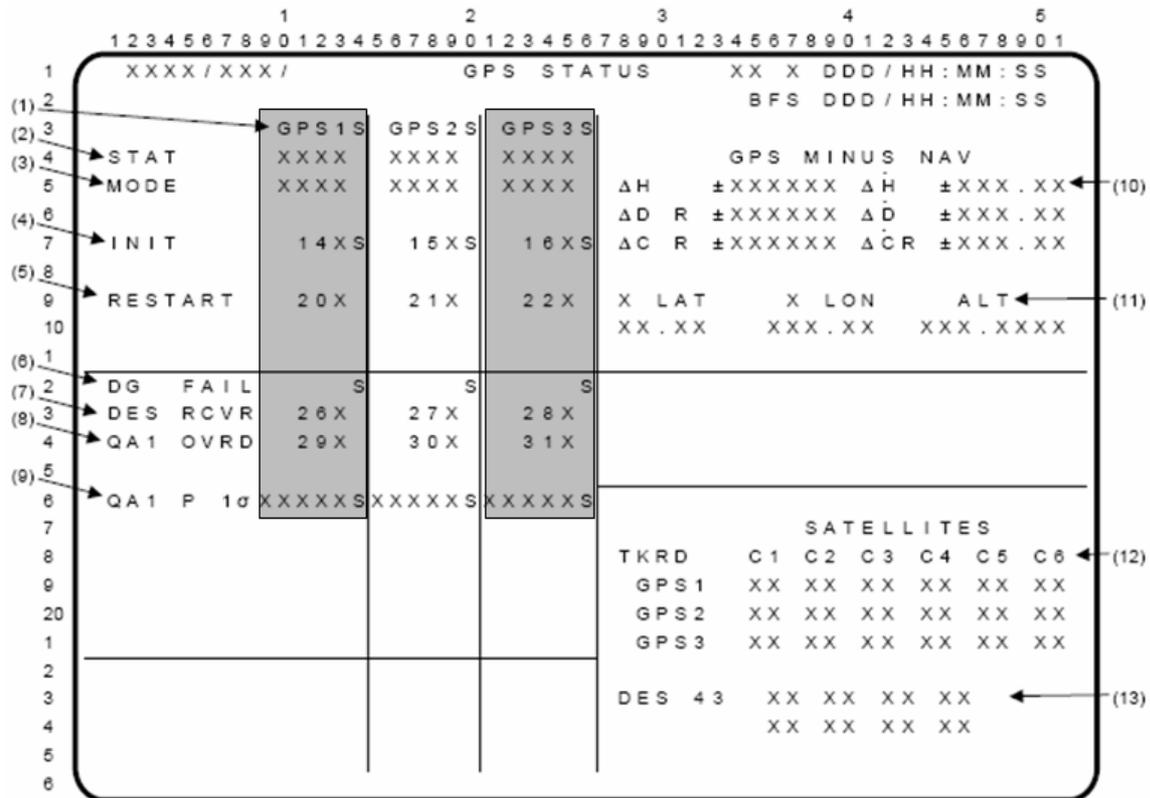


Figure 12 BFS GPS Status Display (SPEC 55)

(1) The GPS parameter status indicator shall display the navigation status of each receiver. The determination of which symbol to display for each receiver shall be made by examining the NAV variables GPS1IND, GPS2IND, and GPS3IND according to the Table 24 .

Table 24 BFS GPS INDICATION

| Display | Message Description |
|---------|---|
| blank | Receiver data available for NAV use |
| ↓ | Receiver data not available for NAV |
| ? | Receiver data available for NAV use but rejected due to a receiver-to-receiver miscompare |

(2) STAT. This field displays the status of each GPS receiver. If the GPS is operating normally, the field is blanked; otherwise, one of the following status codes will be displayed double overbright:

Table 25 VALID BFS GPS STATUS OUTPUTS (STAT)

| Display | Message Description |
|---------|--|
| NI | Receiver is not installed |
| M | Communication Fault |
| RPF | Receiver processor failure |
| BATT | GPS battery low |
| DGF | GPS data good failure |
| (blank) | Nominal operation – No GPS related anomaly |

If a receiver processor failure (RPF) occurs, a CLASS 3 alarm will be annunciated by GAX. 'RPF' will be displayed double overbright. STAT is active regardless of whether the corresponding receiver is installed.

- (3) MODE. This field displays the current operating mode of the receiver.

Table 26 VALID BFS GPS RECEIVER HARDWARE MODES

| Display | Message Description |
|---------|---|
| INIT | Initialization mode |
| TEST | Self-Test mode |
| INS | Inertial Navigation System submode |
| PVA | Position, Velocity, Acceleration navigation submode |
| STA | Prelaunch Standby navigation submode |
| (blank) | Receiver is not in an operating mode |

The MODE field shall display the current operating modes of the GPS receivers. There are three basic operating modes for the receiver: 1) Initialization, 2) Test, and 3) Navigation. The Navigation mode has three submodes: aided (INS), unaided (PVA), and Prelaunch Standby (STA). When any one of the GPS receivers is not in the Initialization mode, Test mode or one of the three recognized Navigation submodes, the MODE field for that GPS receiver shall be blanked.

- (4) INIT. Items 14, 15, and 16 provide the capability to command the initialization of a GPS receiver. An '*' shall be displayed while initialization operations are in progress and while the receiver remains in the INIT mode. To the right of each INIT mode command status field is the command result field, which displays the most-recently commanded initialization attempt. If the initialization was successful, the result field shall be blanked. If the initialization failed, the result field will display a down arrow (↓). Commanding INIT mode while the receiver is in the INIT mode will result in an 'ILLEGAL ENTRY' message. The receiver will remain in the INIT mode until commanded to S/TEST or NAV mode, at which time the '*' will be removed.

For uninstalled receivers or when the BFS is not engaged, executing item 14, 15, or 16 will result in an 'ILLEGAL ENTRY' message.

- (5) RESTART. Items 20, 21, and 22 provide the capability to command each GPS receiver to restart its navigation filter. This option is available only when the receiver is in the INS NAV submode, indicated by 'INS' being displayed in the MODE field. An 'ILLEGAL ENTRY' message will be generated if the RESTART command is entered and the receiver is not in the INS NAV submode. An '*' shall be displayed in the associated item field while the filter restart operations are in progress for that receiver and the '*' will be removed upon receiver completion of the restart. If the receiver is in the S/TEST or INIT modes, commanding a receiver filter restart will result in an 'ILLEGAL ENTRY' message.

For uninstalled receivers, executing item 20, 21, or 22 will result in an 'ILLEGAL ENTRY' message.

- (6) DG FAIL. The DG FAIL status indicator shall display each GPS receiver's assessment of its own navigation state data validity. If the data good flag is OFF for a particular GPS receiver, a down arrow shall be displayed in the status indicator field corresponding to that receiver. If the data good flag is ON for a particular GPS receiver, its corresponding status indicator shall be blanked.

- (7) DES RCVR. Items 26, 27, and 28 provide the capability for each receiver to be deselected or reselected (toggle) for use by the GPS selection filter. Deselection is indicated by an asterisk '*' adjacent to the DES item number. A blank indicates that the associated receiver has not been deselected. If the GPS receiver is manually deselected on either the PASS GPS STATUS DISPLAY or HORIZ SIT DISPLAY (3 GPS receivers installed), it does not need to be manually deselected on the BFS GPS STATUS because these item entries are DK listen.

(8) QA 1 OVERRIDE. Items 29, 30, and 31 are used to force a given receiver's state to be a candidate for GPS state vector selection even though that state may have failed the QA1 threshold test. An "*" is displayed if the override has been selected. QA 1 data continues to be calculated for a GPS receiver even if it has been selected for a QA 1 OVERRIDE.

(9) QA1 P1 σ This field displays the Figure of Merit (FOM) or estimate of position error for each receiver. The FOM is related to the error as follows:

Table 27 BFS GPS FIGURE OF MERIT (FOM)

| FOM integer | Position error (P), one sigma (1 σ) [feet] |
|-------------|--|
| 0 | (blank) |
| 1 | 75 |
| 2 | 175 |
| 3 | 250 |
| 4 | 325 |
| 5 | 650 |
| 6 | 1650 |
| 7 | 3275 |
| 8 | 16400 |
| 9 | 99999 |

The FOM data from the GPS Receiver State Processing Principal Function will be blank when the receiver is not in NAV mode (i.e., in TEST or INIT mode) or when the GPS NAV data used by the GPS Receiver State Processing Principal Function are not valid.

If the FOM exceeds a pre-mission determined I-Load tolerance, a '↓' shall be driven in the appropriate parameter status indicator field next to the displayed FOM value and displayed double overbright. If the receiver state estimate fails the QA1 FOM quality assessment check, it is excluded as a candidate from the GPS state vector selection process.

If a particular GPS receiver's estimate of position error has passed the I-Loaded threshold test for an I-Loaded number of consecutive cycles, the parameter status indicator field for the P1 σ field corresponding to that particular GPS receiver shall be blank.

The corresponding fields for uninstalled receivers will remain blank.

(10) GPS MINUS NAV. This section displays the components of the position and velocity vector differences between the selected GPS state estimate and the orbiter navigation filter state estimate in UVW coordinates: altitude (ΔH), downrange (ΔDR), and crossrange (ΔCR) for both position and velocity. Units are ft and ft/sec for position and velocity, respectively. The data fields are blanked when no GPS state is available.

(11) GEODETIC COORDINATES. In MM304, 305, 602, and 603, altitude is displayed with units of kilo-feet. All other modes will display nautical miles.

This section displays the geodetic coordinates of the selected GPS state: altitude (ALT), latitude (LAT), and longitude (LON). Altitude is displayed in nautical miles. Latitude is displayed in degrees from the equator with an 'N' or 'S' displayed in the field before the LAT title for northern (LAT > 0) or southern (LAT < 0) hemisphere. Longitude is displayed in degrees from the Greenwich meridian with an 'E' or 'W'

displayed in the field before the LON title if the orbiter is east (LON > 0) or west (LON < 0) of the Greenwich meridian, respectively.

The data fields are blanked when no GPS state is available.

(12) SATELLITES TRACKED. This section shall display the GPS satellites being tracked by each receiver. Up to a maximum of six receiver channels can be displayed. If the receiver has less than six channels, the data will be blanked for the nonexistent channels. The order of the satellite number displayed reflects a one-to-one correspondence to the receiver channel tracking that satellite. Values range from 'blank' (no satellites being tracked) to 1 through 63.

The corresponding fields for uninstalled receivers will remain blank.

(13) DESELECT SATELLITES. Item entries 43 to 50 allow the crew to deselect satellites, valid satellite ID's being 1 to 63, as candidates for selection by the GPS receivers by building, based on Item Entries, a derived health status word. The '+' delineator deselects satellites and the '-' delineator removes satellites from being deselected as described below. As an example, 'ITEM 43 + 23 EXEC' will indicate to all GPS receivers that satellite ID 23 should not be used as a candidate for selection. If another satellite was previously deselected by item entry 43, an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and no change to deselected satellites occurs. If this satellite, ID 23 in this example, is currently deselected by another Item Entry, one of item entries 44 to 50 in this example, an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and no change to deselect satellites occurs.

'ITEM 43 - 23 EXEC' will indicate to all GPS receivers that satellite 23 should once again be considered as a candidate for selection. If item Entry 43 previously deselected satellite 23, then 'ITEM 43 - 23 EXEC' will result in a blank being displayed, and satellite 23 being available for selection. If satellite 23 is not currently deselected by item Entry 43, then an 'ILLEGAL ENTRY' message will be generated, the displayed data is unchanged and there is no change to deselected satellites.

Entry of satellite number of 0, less than - 63, or greater than + 63 will cause an 'ILLEGAL ENTRY' message to be issued. All satellites are initialized as available for selection. Satellite Deselection information for item entries 43 to 50 shall be maintained in the GNC major function base.

For the BFS, the satellite deselect option will only be necessary for satellites that are deselected in the PASS in OPS 2, 8, or 9. BFS will DK-listen for satellite deselections in OPS1, 3, and 6.

20.4.5.4 Three String GPS Configuration

20.4.5.4.1 Three String GPS Switch Configuration

As mentioned earlier, OV-105 will be the only three string GPS vehicle. A mixed fleet operation of either single-string GPS (with three TACANs) or triple-string GPS (no TACANs) will continue until the end of the program.

Figure 13 shows the Triple-String GPS hardware locations within the Avionics Bays, as well as their respective Antenna and associated hardware installation.

Flight Vehicle 3-String GPS Installation

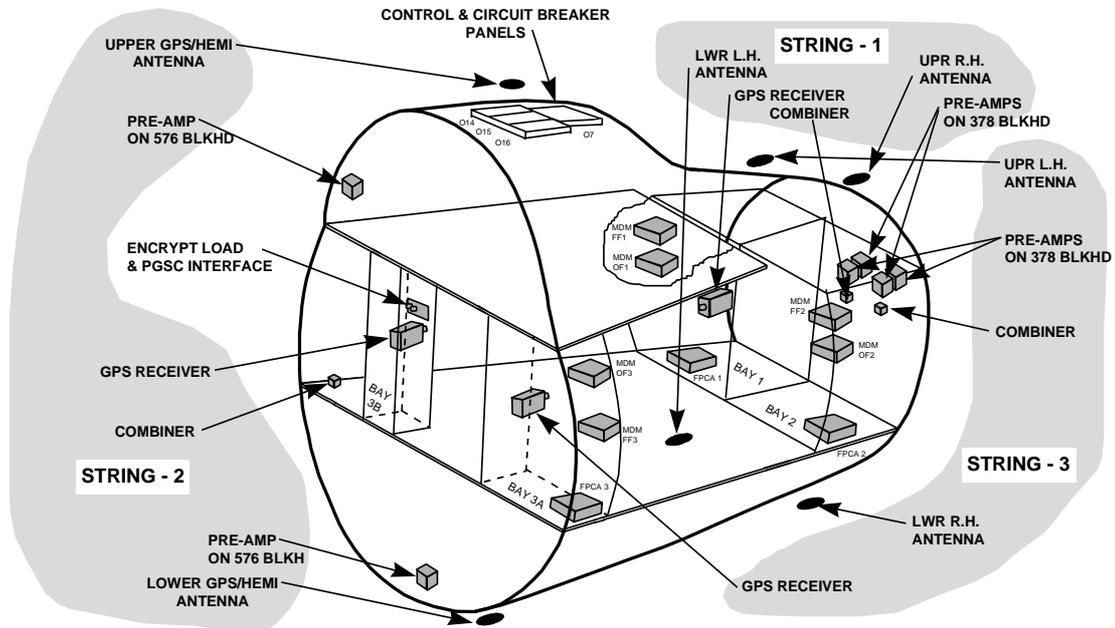


Figure 13 Three-String GPS Hardware Locations

Figure 14 shows the panel layout for the 3-String GPS configuration. The three string location for individual GPS switches will be on Panel O7, in the previous location of the TACAN Rotary switches and Thumbwheels. Note the power source for the Receiver and the Pre-Amps is a single bus power source, and each Receiver has its own guarded Zeroize switch.

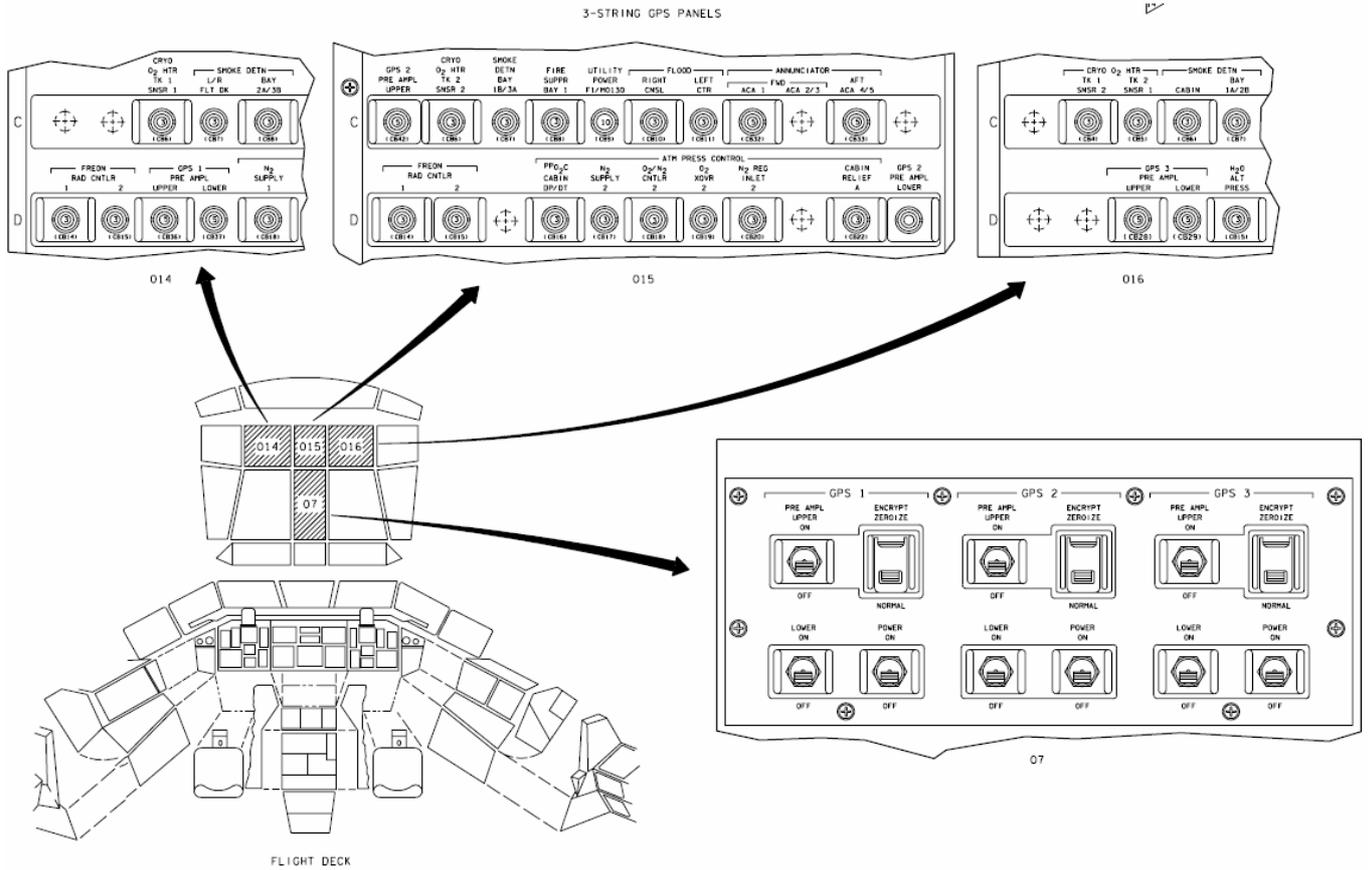


Figure 14 Three-String GPS Configuration Panel Layout

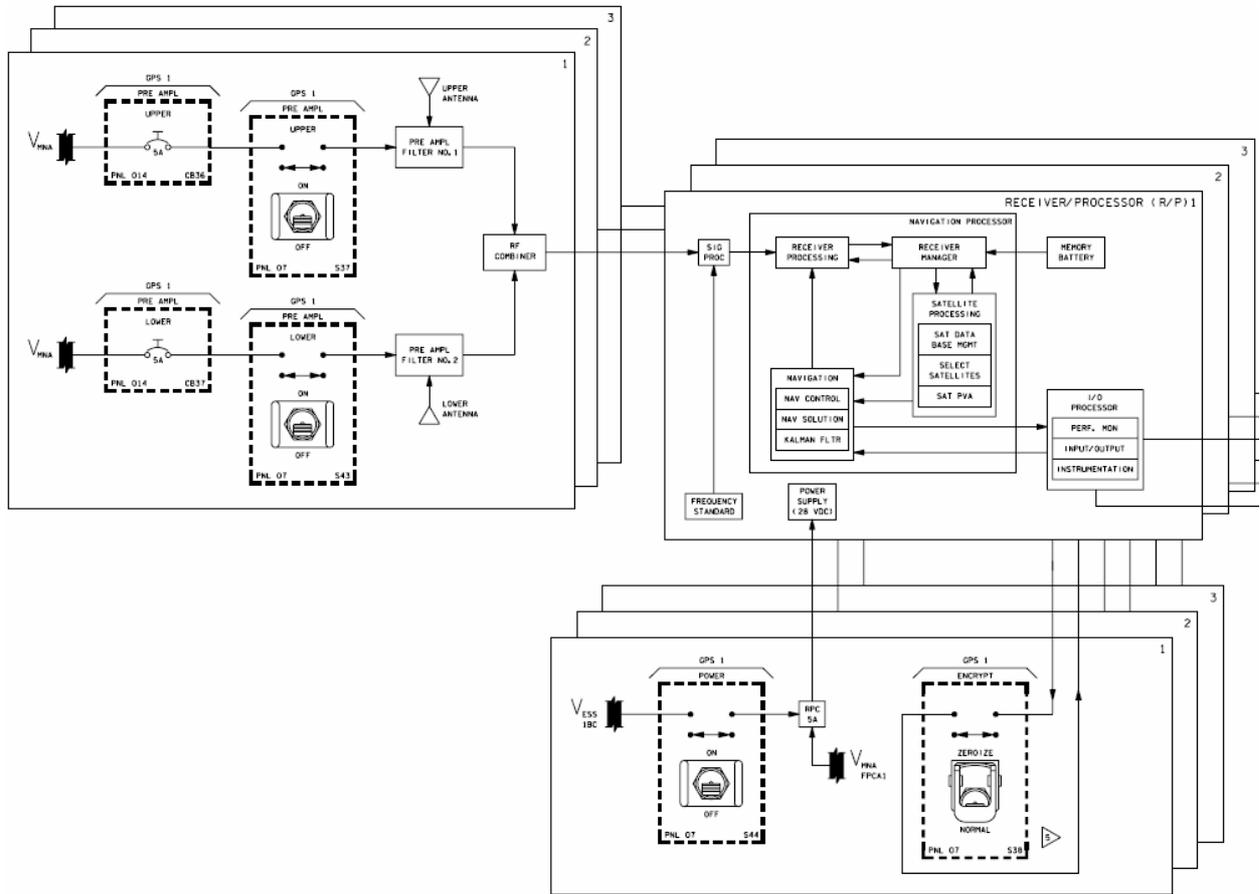


Figure 15 Three-String GPS Wiring Configuration

Figure 16 shows the data flow and power signals associated with the three-string GPS configuration including panel switches, CBs, antenna assignments, Av Bay locations, etc.

3-STRING GPS CONFIGURATION

| | GPS 1 | GPS 2 | GPS 3 |
|------------------|---------------|---------------|---------------|
| POWER | VMNA FPCA1 | VMNB FPCA2 | VMNC FPCA3 |
| CONTROL BUS | ESS 1BC | ESS 2CA | ESS 3AB |
| ON/OFF PNL O7 | SW 44 | SW 46 | SW 48 |
| DATA PATH | MDM FF1 | MDM FF2 | MDM FF3 |
| < 4 SAT's | MDM OF1 | MDM OF2 | MDM OF3 |
| LOCATION | AV BAY 1 | AV BAY 3B | AV BAY 3A |

| | UPPER PRE-AMPL | | | LOWER PRE-AMPL | | |
|------------------|----------------|---------------|--------------|----------------|---------------|--------------|
| GPS | 1 | 2 | 3 | 1 | 2 | 3 |
| POWER | VMNA O14 | VMNB O15 | VMNC O16 | VMNA O14 | VMNB O15 | VMNC O16 |
| CB's (5A) | cb 36 O14 | cb 43 O15 | cb 28 O16 | cb 37 O14 | cb 44 O15 | cb 29 O16 |
| ON/OFF PNL O7 | SW 37 | SW 39 | SW 41 | SW 43 | SW 45 | SW 47 |
| ANTENNA | UPPER RH | UPPER HEMI | UPPER LH | LOWER LH | LOWER HEMI | LOWER RH |

Figure 16 Three-String GPS Data Flow and Power Matrix

20.4.5.4.2 Three String GPS Displays

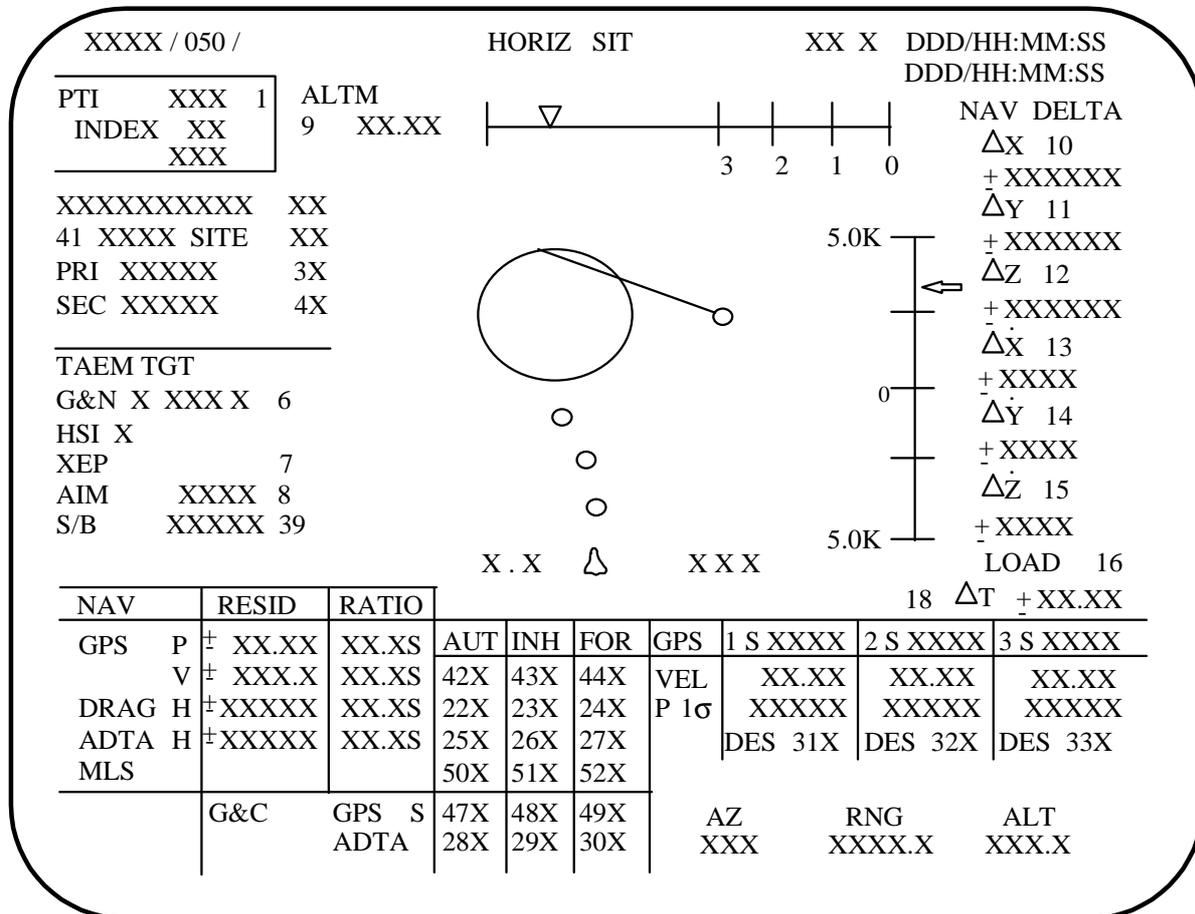


Figure 17 Horizontal Situation Display (SPEC 50) 3 GPS – NO TACAN

SPEC 50 (see Figure 17): With the removal of TACANs from the Horizontal Situation Display, a full complement of GPS data has been installed. Velocity and P1sigma (Statistical error estimate equivalent to FOM) for each LRU has been added, in addition to the azimuth, range and altitude derived from the selected GPS state vector. The GPS-derived azimuth and range is referenced to the runway threshold, as opposed to the TACAN station (similar to the BFS single-string). In addition to the GPS Auto/Inhibit/Force item entries, the residual and ratio values for both the position and velocity (derived from the selected GPS state vector) were added to the Nav section of the display. For consistency, the GPS_to_G&C Auto/Inhibit/Force item entries are located in the G&C section of the display.

A feature that became available in OI27 is the MLS Auto/Inhibit/Force function, which controls incorporation of MLS data into the onboard state vector. Unlike earlier FSW, the crew can now Force MLS data into nav, which may have been erroneously edited out due to a miscompare with a bad nav state. Conversely, the crew can also inhibit MLS data, based on an MCC call regarding a suspect MLS Ground Station, thereby defaulting back to GPS data. The double overbright display of the character string “MLS” is displayed when MLS data is being utilized by the navigation filter and the MLS-derived navigation state is being sent to guidance. The MLS A/I/F feature is only available on three-string GPS equipped vehicles (OV-105 only).

The intent (similar to what had occurred for TACANs) is that if the MLS A/I/F function is in the same state as the GPS_to_G&C or the GPS_to_NAV A/I/F function, valid MLS data flow would inhibit flow of GPS data (to G&C and NAV), thus defaulting nav to be MLS-driven. MLS in auto can be overridden by placing GPS to force; however, MLS in force cannot be overridden. While in auto, if MLS data becomes invalid, it

SHUTTLE OPS
GNC/JSC-12843

ORIG: J. MCGREGOR
07/26/07: FINAL, PCN 1

GPS
SB 20.0

20.4.6 MCC Monitoring of GPS Data

20.4.6.1 Single String Displays

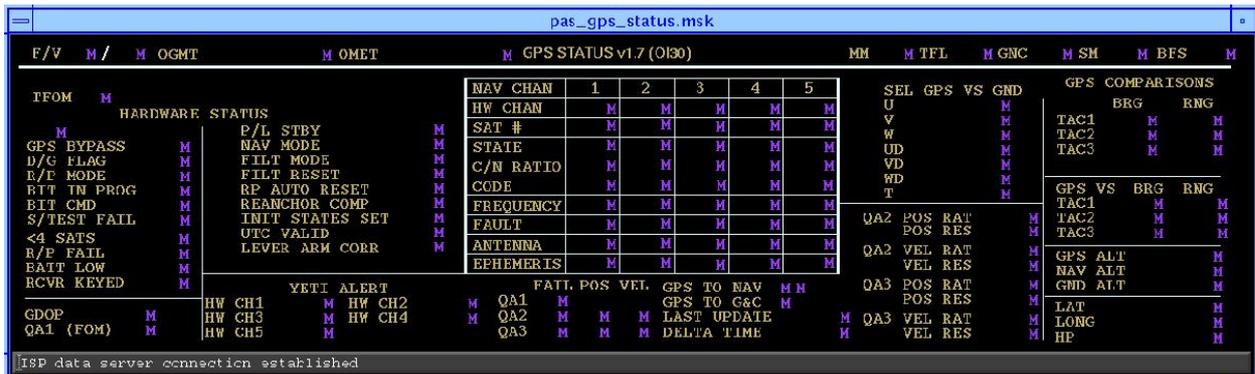


Figure 19 PASS GPS Status Display (Old)

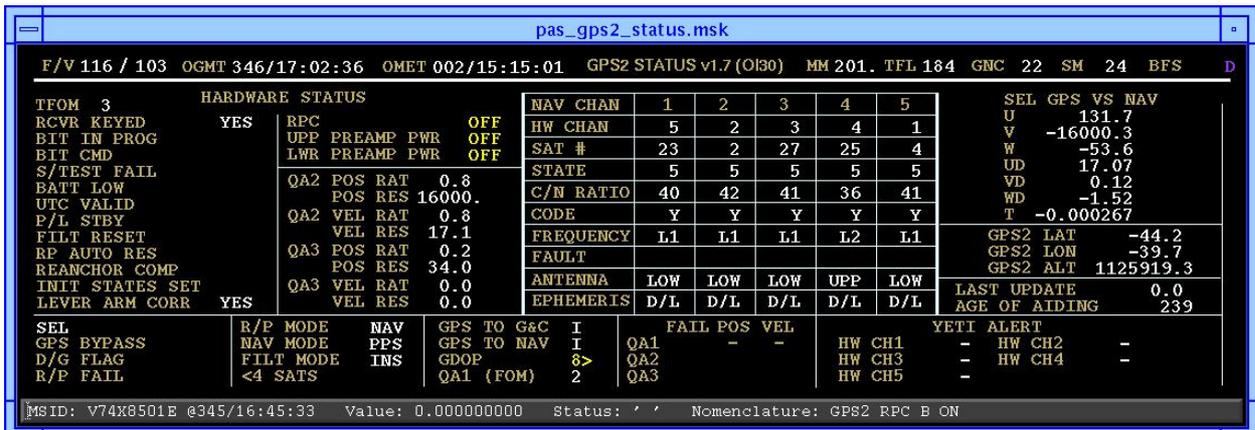


Figure 20 PASS GPS 2 Status Display (New)

For single-string vehicles (OV-103 and OV-104), the GPS displays most heavily used are the PASS GPS Status display (Figure 19 and Figure 20) and the GPS to PASS Comparison display (Figure 21 and Figure 22). The new GPS status display can be used for both single and triple string flights (one display per GPS LRU). The old GPS status display is only for single-string.

The PASS GPS Status displays, used both on-orbit and during ascent and entry, provide a detailed view of GPS status (bites, modes, YETI events, etc) as well as measures of GPS performance (complete channel status data, QA checks, and compares to navigation and TACAN). This display also contains the GPS state vector, expressed in terms of latitude, longitude, and HP (height of perigee – old display) or altitude (new display). Two typos in the old display are the SEL GPS VS GND, which actually displays GPS against navigation, and “DELTA TIME” which is actually the age of aiding. These have been corrected in the new displays. One other item of note is the “Antenna” indication. This is not necessarily the antenna which is tracking the satellite (it is impossible to truly know which antenna is tracking because the signals are combined prior to the connection to the MAGR receiver). This indication is based only on orbiter attitude and antenna boresight alignment with respect to the line-of-sight to the satellite. While this is an assumption, in most cases the indication should be correct; however it is possible that the other or

The GPS Status Mini display (Figure 23) is a small display which shows some of the status items contained in the PASS GPS status display, but without the individual channel data. This is a good display to use for a quick look when a GPS issue is suspected.

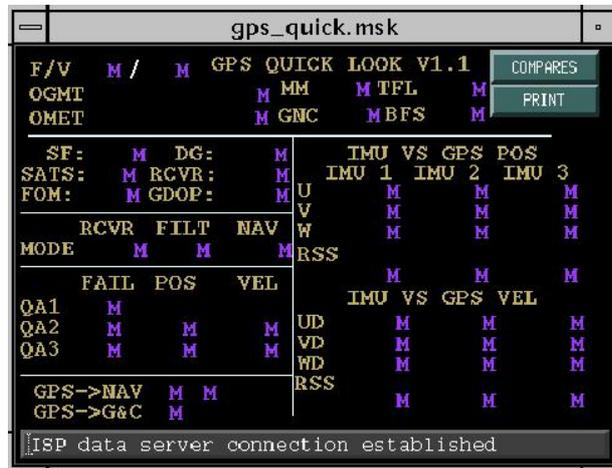


Figure 24 GPS Quick Display

The GPS Quick display (Figure 24) contains most of the same data as the GPS Status Mini display but in a different shape and also contains GPS to IMU compares which are useful during OPS 3.

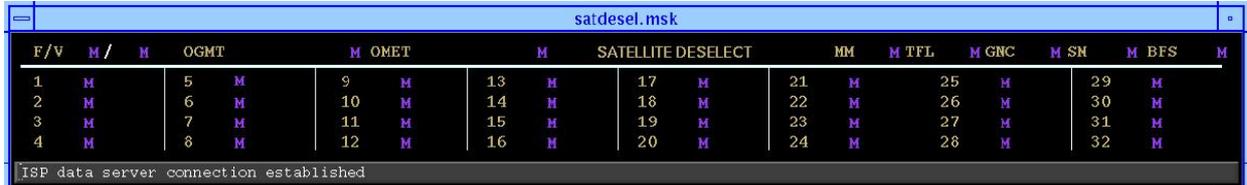


Figure 25 GPS Satellite Deselect

The GPS satellite deselect display (Figure 25) simply shows which (if any) satellites have been deselected by the crew on Spec 55.

| pps.msk | | | | | | | | | | | |
|---------|------|---|---------------------------------|---|------|-------|---------------------|---|-------|--|---|
| F/V | M/ | M | OGMT | M | OMET | M | GPS ENCRYPT KEY-PPS | | | | |
| MM | MTFL | M | GNC | M | SM | M | BFS | M | | | |
| | | | GPS 1 | | | GPS 2 | | | GPS 3 | | |
| | | | PPS DAILY KEY IN USE VERIFIED | | | M | | | M | | M |
| | | | PPS DAILY KEY IN USE UNVERIFIED | | | M | | | M | | M |
| | | | PPS DAILY KEY IN USE INCORRECT | | | M | | | M | | M |
| | | | PPS KEY FAILED PARITY | | | M | | | M | | M |
| | | | PPS-GUV USER | | | M | | | M | | M |
| | | | PPS - 2 HOUR ALERT | | | M | | | M | | M |
| | | | PPS - INSUFFICIENT KEYS | | | M | | | M | | M |
| | | | PPS - RECEIVER CONTAINS KEYS | | | M | | | M | | M |
| | | | PPS - MISSION DURATION FOR KEYS | | | M | | | M | | M |
| | | | PPS OR SPS | | | M | | | M | | M |
| | | | GPS BYPASS | | | M | | | M | | M |
| | | | NAV DG (PM) | | | M | | | M | | M |
| | | | R/P MODE | | | M | | | M | | M |
| | | | QA1 (FOM) | | | M | | | M | | M |
| | | | GDOP | | | M | | | M | | M |

ISP data server connection established

Figure 26 PPS Status Display

The PPS Status Display (Figure 26) is used both for single and three string operations and displays information about the status of the PPS key.

| bfs_gps_status.msk | | | | | | | | | | | | | | | |
|--------------------|----|-----|--------------------|---|------|---|-----|------------------------|-----|-----|-----|-----|------|-----|-----|
| F/V | M/ | 10M | OGMT | M | OMET | M | BFS | GPS STATUS v1.2 (0130) | MM | M | TFL | M | BFS | M | |
| | | | GPS CHANNEL STATUS | | | | | GPS COMPARISONS | | | | | | | |
| HARDWARE STATUS | | | GPS CHANNEL STATUS | | | | | GPS COMPARISONS | | | | | | | |
| | | | CHANNEL | 1 | 2 | 3 | 4 | 5 | SEL | GPS | VS | GND | TAC1 | BRG | RNG |
| M | | | R/P MODE | M | | | | | | | | | M | M | M |
| | | | NAV MODE | M | | | | | | | | | M | M | M |
| | | | FILT MODE | M | | | | | | | | | M | M | M |
| | | | NAV DG | M | | | | | | | | | M | M | M |
| | | | <4 SATS | M | | | | | | | | | M | M | M |
| | | | R/P FAIL | M | | | | | | | | | M | M | M |
| | | | BATT LOW | M | | | | | | | | | M | M | M |
| | | | GDOP | M | | | | | | | | | M | M | M |
| | | | QA1 (FOM) | M | | | | | | | | | M | M | M |
| | | | QA2 POS RAT | M | | | | | | | | | M | M | M |
| | | | POS RES | M | | | | | | | | | M | M | M |
| | | | QA2 VEL RAT | M | | | | | | | | | M | M | M |
| | | | VEL RES | M | | | | | | | | | M | M | M |
| | | | GPS LAT | M | | | | | | | | | M | M | M |
| | | | LONG | M | | | | | | | | | M | M | M |
| | | | ALT | M | | | | | | | | | M | M | M |
| | | | HP | M | | | | | | | | | M | M | M |
| | | | DELTA TIME | M | | | | | | | | | M | M | M |

ISP data server connection established

Figure 27 BFS GPS Status Display

The BFS GPS status display (Figure 27) is similar to the PASS status display (Figure 19) but does not contain as much detail due to the limitations of the downlist in the BFS. Detailed channel status is not available and QA1, 3, and YETI comps are not available on the BFS.

| Selected_gps_bfs_compares.msk | | | | | | | | | | | | |
|-------------------------------|-------|------|------|----------------|----------|------|----------|----------------------|---------------------|-----|-----|--|
| F/V | M/10M | MM | M | BFS | SELECTED | GPS | COMPARES | v1.0 | GPS | REF | | |
| OGMT | M | OMET | M | OI | M | GN | M | SM | M | BF | M | |
| BFS IMU VS SEL GPS COMPARES | | | | SEL GPS TO NAV | | | | BFS TACAN VS SEL GPS | | | | |
| IMU1 | | | IMU2 | | | IMU3 | | | BRG | | RNG | |
| U | M | | M | | M | | M | | 1 | M | M | |
| V | M | | M | | M | | M | | 2 | M | M | |
| W | M | | M | | M | | M | | 3 | M | M | |
| POS RSS | M | | M | | M | | M | | AIR DATA VS GPS ALT | | | |
| UD | M | | M | | M | | M | | SEL | | | |
| VD | M | | M | | M | | M | | 1 2 3 | | | |
| WD | M | | M | | M | | M | | HIP DG | M | M | |
| VEL RSS | M | | M | | M | | M | | R/P | M | M | |
| T | M | | M | | M | | M | | NAV DG | M | M | |
| | | | | | | | | | FOM | M | M | |
| | | | | | | | | | GDOP | M | M | |

ISP data server connection established

Figure 28 BFS GPS Comparison Display

The BFS GPS Comparison Display (Figure 28) is the BFS equivalent of the PASS GPS Comparison Display, minus MLS compares as the BFS does not use MLS. The air data section is also slightly different, as only selected air data (instead of individual ADTA raw data) is compared against GPS.

20.4.6.2 Three String Displays

| gps_status_pass_mod.msk | | | | | | | | | | | | | | | | | | |
|-------------------------|------|------|--------------|------|----------------------|-----------|-----------------------|--------|-----|----------------------|-----|-----------|------|------|------|------|------|---|
| F/V | 103 | OGMT | 346/16:59:18 | OMET | 002/15:11:43 | GPS | SUMMARY STATUS (PASS) | v01.00 | MM | 201 | TFL | 184 | GNC | 22 | SM | 24 | BFS | D |
| GPS 1 CHANNEL STATUS | | | | | GPS 2 CHANNEL STATUS | | | | | GPS 3 CHANNEL STATUS | | | | | | | | |
| CHANNEL | 1 | 2 | 3 | 4 | 5 | CHANNEL | 1 | 2 | 3 | 4 | 5 | CHANNEL | 1 | 2 | 3 | 4 | 5 | |
| HW CHAN | 0G | 0G | 0G | 0G | 0G | HW CHAN | 1 | 2 | 3 | 4 | 5 | HW CHAN | 0G | 0G | 0G | 0G | 0G | |
| SAT # | 0G | 0G | 0G | 0G | 0G | SAT # | 8 | 13 | 24 | 16 | 1 | SAT # | 0G | 0G | 0G | 0G | 0G | |
| STATE | 0G | 0G | 0G | 0G | 0G | STATE | 5 | 5 | 5 | 5 | 5 | STATE | 0G | 0G | 0G | 0G | 0G | |
| C/N RATIO | 0G | 0G | 0G | 0G | 0G | C/N RATIO | 38 | 38 | 40 | 34 | 35 | C/N RATIO | 0G | 0G | 0G | 0G | 0G | |
| CODE | PG | PG | PG | PG | PG | CODE | Y | Y | Y | Y | C/A | CODE | PG | PG | PG | PG | PG | |
| FREQUENCY | L1G | L1G | L1G | L1G | L1G | FREQUENCY | L1 | L1 | L1 | L2 | L1 | FREQUENCY | L1G | L1G | L1G | L1G | L1G | |
| FAULT | G | G | G | G | G | FAULT | L1 | L1 | L1 | L2 | L1 | FAULT | G | G | G | G | G | |
| ANTENNA | LOWG | LOWG | LOWG | LOWG | LOWG | ANTENNA | LOW | LOW | LOW | UPP | UPP | ANTENNA | LOWG | LOWG | LOWG | LOWG | LOWG | |
| EPHEMERIS | G | G | G | G | G | EPHEMERIS | D/L | D/L | D/L | D/L | D/L | EPHEMERIS | LOWG | LOWG | LOWG | LOWG | LOWG | |
| GPS 1 CHANNEL STATUS | | | | | GPS 2 CHANNEL STATUS | | | | | GPS 3 CHANNEL STATUS | | | | | | | | |
| CHANNEL | 1 | 2 | 3 | 4 | 5 | CHANNEL | 1 | 2 | 3 | 4 | 5 | CHANNEL | 1 | 2 | 3 | 4 | 5 | |
| HW CHAN | 0G | 0G | 0G | 0G | 0G | HW CHAN | 1 | 2 | 3 | 4 | 5 | HW CHAN | 0G | 0G | 0G | 0G | 0G | |
| SAT # | 0G | 0G | 0G | 0G | 0G | SAT # | 8 | 13 | 24 | 16 | 1 | SAT # | 0G | 0G | 0G | 0G | 0G | |
| STATE | 0G | 0G | 0G | 0G | 0G | STATE | 5 | 5 | 5 | 5 | 5 | STATE | 0G | 0G | 0G | 0G | 0G | |
| C/N RATIO | 0G | 0G | 0G | 0G | 0G | C/N RATIO | 38 | 38 | 40 | 34 | 35 | C/N RATIO | 0G | 0G | 0G | 0G | 0G | |
| CODE | PG | PG | PG | PG | PG | CODE | Y | Y | Y | Y | C/A | CODE | PG | PG | PG | PG | PG | |
| FREQUENCY | L1G | L1G | L1G | L1G | L1G | FREQUENCY | L1 | L1 | L1 | L2 | L1 | FREQUENCY | L1G | L1G | L1G | L1G | L1G | |
| FAULT | G | G | G | G | G | FAULT | L1 | L1 | L1 | L2 | L1 | FAULT | G | G | G | G | G | |
| ANTENNA | LOWG | LOWG | LOWG | LOWG | LOWG | ANTENNA | LOW | LOW | LOW | UPP | UPP | ANTENNA | LOWG | LOWG | LOWG | LOWG | LOWG | |
| EPHEMERIS | G | G | G | G | G | EPHEMERIS | D/L | D/L | D/L | D/L | D/L | EPHEMERIS | LOWG | LOWG | LOWG | LOWG | LOWG | |

ISP data server connection established

Figure 29 GPS Status Display - Three String

The three-string GPS status display (Figure 29) shows detailed status information for all installed receivers; the data contained in the display is similar to the single-string status display. The size of this

display has necessitated an additional workstation and monitor for the sensors console for three-string GPS operations, which should be installed prior to the first three-string flight.

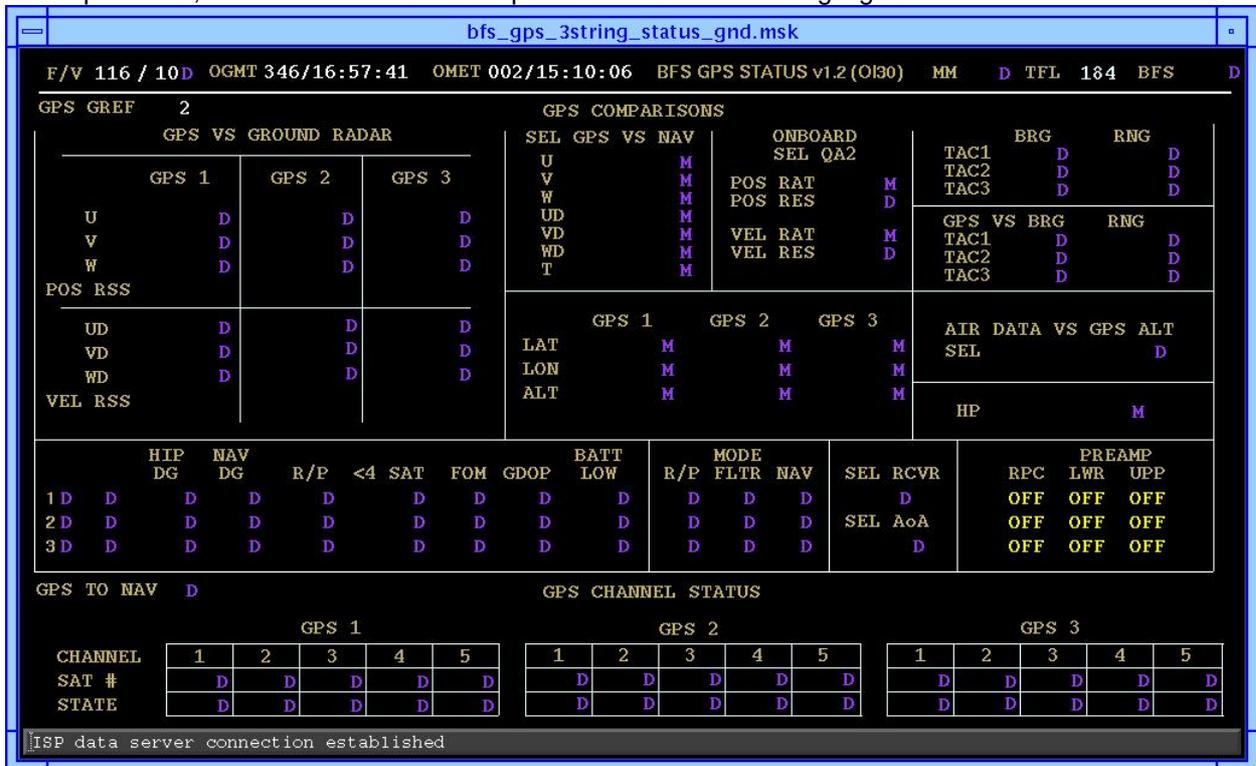


Figure 30 BFS GPS Status Display – Three String

The BFS also has a GPS status display for three string operations, shown in Figure 30.



Figure 31 Sensors Three-String Display

The sensors three-string display (Figure 31) is similar to the sensors TACAN display, except that limited GPS status data and compares are displayed in place of the TACAN data. GPS to ground tracking compares as well as GPS status data are available. For a more complete GPS status during three string operations, the three string GPS status display should be consulted.

| sensors_bfs_mod.msk | | | | | | | | | | | | | | | | | | | | | | | | | |
|---------------------|---|--------------|------------|------------------|-------|---|-------|---------|--------------|--------------|---------|-------------------|---------|---------|--------|-----------|-----------|----------------|---------|-----|----------|-----|-----|------------|--|
| STS-116 | | OV-10D | | GMT 346/17:05:57 | | | MET | | 002/15:18:22 | | | BFS SENSORS v1.04 | | | MM D | | BFS D | | TFL 164 | | | | | | |
| IMU | | IMU BITE | | | 1 | 2 | 3 | TACAN | | BRG | RNG | STA | BRG DG | RNG DG | SW | STATUS | | ROLL PITCH YAW | | | | | | | |
| 1 | D | D | HAINS GOOD | | | D | D | D | 1 | D | D | D | D | D | OFF | D | ATT D D D | | | | | | | | |
| 2 | D | D | MUX FAIL | | | D | D | D | 2 | D | D | DD | D | D | OFF | D | ERR D D D | | | | | | | | |
| 3 | D | D | A/D FAIL | | | D | D | D | 3 | D | D | DD | D | D | OFF | D | | | | | | | | | |
| SEL | | D | | PLATFORM FAIL | | | D | D | D | SEL | | D | | D | | | | | | | | | | | |
| USE | | D | | DIG SUBSYS FL | | | D | D | D | 1/GND | | EDIT | | RESID | | ELEV | | | | | | | | | |
| RPC | | 1 | | DC/DC FAIL | | | D | D | D | 2/GND | | BRG | | D BRG | | A/I/F | | | | | | | | | |
| A | B | 2 | A | DISC PWR FAIL | | | D | D | D | 3/GND | | RNG | | D RNG | | D | | | | | | | | | |
| B | C | 3 | C | RESOLVER FAIL | | | D | D | D | GPS | | HIP DG | NAV DG | R/P | <4 SAT | FOM | GDOP | R/P | FLTR | NAV | BATT | LOW | RPC | VAR DNLIST | |
| 1 | | TRANS WD1 FL | | | D | D | D | 1 | D | D | D | D | D | D | D | D | D | D | D | D | D | D | OFF | 1 | |
| 2 | | TRANS WD2 FL | | | D | D | D | 2 | D | D | D | D | D | D | D | D | D | D | D | D | D | D | OFF | 2 | |
| 3 | | D1/D8 SEQ | | | D | D | D | 3 | D | D | D | D | D | D | D | D | D | D | D | D | D | D | OFF | 3 | |
| A | | D2 SEQUENCER | | | D | D | D | POS RSS | | VEL RSS | | POS RESID FL | | D | | SEL RCVR | | D | | | | | | | |
| B | | D3 SEQUENCER | | | D | D | D | 1/GND | | POS RESID FL | | D | | SEL AoA | | D | | | | | | | | | |
| C | | D4 SEQUENCER | | | D | D | D | 2/GND | | VEL RESID FL | | D | | D | | | | | | | | | | | |
| 1 | | D5 SEQUENCER | | | D | D | D | 3/GND | | NAV AIF | | D | | | | | | | | | | | | | |
| 2 | | CB/SW | | | RPC | | ADTA | | D 1 | D | D 2 | D | MACH | | D | ALPHA NAV | | D | | | | | | | |
| 3 | | 1 OP | | | 2 OP | | D 3 | | D | D 4 | D | EAS | | 0 | ADS | | D | | | | | | | | |
| 4 | | 2 OP | | | 3 OFF | | A OFF | | C OFF | | DG | | USE | | D | TAS | | D | | | | | | | |
| 1 | | 3 OFF | | | 4 OFF | | B OFF | | C OFF | | A/I/F | | D | | VREL | | D | | D | | | | | | |
| 2 | | 4 OFF | | | SW | | RPC | | AVG PAL | | AVG PAU | | H RESD | | D | | ALT NAV | | D | | QBAR NAV | | | | |
| 3 | | 1 OFF | | | 2 OFF | | A OFF | | B OFF | | H A/I/F | | D | | GND | | D | | ADS | | D | | | | |
| 4 | | 2 OFF | | | 3 OFF | | A OFF | | C OFF | | H EDIT | | D | | RW | | D | | H DOT | | D | | | | |
| 1 | | 3 OFF | | | 4 OFF | | RA | | ALT | | RW | | NAV ALT | | D | | D | | D | | D | | | | |
| 2 | | 4 OFF | | | 1 D | | 2 D | | D | | D | | D | | D | | D | | D | | D | | | | |
| 3 | | 1 D | | | 2 D | | 3 D | | D | | D | | D | | D | | D | | D | | D | | | | |
| 4 | | 2 D | | | 3 D | | 4 D | | D | | D | | D | | D | | D | | D | | D | | | | |

Figure 32 BFS Sensors Three-String Display

The BFS three-string sensors display (Figure 32) is a replacement for the TACAN BFS sensors display. In this display, GPS data has been added to TACAN information so the display can be used on any vehicle. No GPS to ground compare information is available with the BFS.

APPENDICES

A: GLOSSARY OF GPS TERMS

B: IN FLIGHT ANOMALY LOG

C: NOTES OF INTEREST

D: REFERENCES

A: GLOSSARY OF GPS TERMS

Almanac – A coarse table location of all NAVSTAR satellites at a given time. Current almanac of all GPS satellites can be acquired from any satellite.

Bandwidth - A measure of width of the spectrum of a signal (frequency domain representation of a signal) expressed in Hertz.

Binary Pulse Code Modulation - Pulse modulation using a string (code) of binary numbers. This coding is usually represented by ones and zeros with definite meanings assigned to them, such as changes in phase or direction of a wave.

C/A-Code or “Coarse Acquisition Code”- is an L-Band frequency with a chipping rate of 1.023 MHz, designated for use by civilian and commercial users. C/A-code consists of a sequence of 1023 pseudorandom binary biphasic modulations on the GPS carrier and has a repetition period of one millisecond.

Carrier - A radio wave having at least one characteristic (e.g., frequency, amplitude, phase) which may be varied from a known reference value by modulation.

Carrier Beat Phase - The phase of the signal, which remains when the incoming Doppler-shifted satellite carrier signal is beat (the difference at which the frequency signal is generated) with the nominally constant reference frequency generated in the receiver.

Channel - A channel of a GPS receiver consists of the radio frequency and the digital hardware and software required to track the signal from one GPS satellite at one of the two GPS carrier frequencies.

Chipping Rate- the rate at which an electronic circuit produces binary digits.

Clock Dithering - timing errors in the onboard satellite clocks. Slow clock dither introduces user position errors. Fast clock dither results in user velocity errors.

C/No – Carrier to Noise Ratio, indicates the strength of the GPS signal received by the antenna.

Differential Navigation - uses two receivers that continuously exchange navigation information with each other in real time. One receiver acts as a base station and navigates relative to the base station. This capability eliminates any common navigation errors and as a result acquires more accurate navigation data. The shuttle does not currently use differential GPS.

Dilution of Precision - A description of the purely geometrical contribution to the uncertainty in a dynamic position fix, given by the expression: $DOP = (\text{TRACE}(A^T A)^{-1})^{(1/2)}$, where A is the design matrix for the solution (dependent on satellite/receiver geometry). The DOP factor depends on the parameters of the position fix solution. Standard terms in the case of kinematic GPS are: GDOP (Geometric DOP), PDOP (Position DOP, three coordinates), HDOP (Horizontal DOP, two horizontal coordinates), VDOP (Vertical DOP, height only), TDOP (Time DOP, clock offset only), and HTDOP (Horizontal Position and Time DOP).

Doppler Shift - The apparent change in frequency of a received signal due to the rate of change of the range between the transmitter and receiver.

Ephemeris Constants - a group of parameters used to define the orbit of celestial bodies or man-made satellites. The Earth's ephemeris constants vary over time due to the equatorial bulge, mass concentrations, and other variations. Satellite ephemerides can only be acquired from a given satellite (ie ephemerides for satellite X can only be obtained by tracking satellite X. Ephemerides take approximately 30 seconds to download and are valid for 4-6 hours.

Ephemeris Errors - errors in the satellite position, which result in user position errors.

Figure of Merit (FOM) - The FOM is an integer ranging from 0 to 9 which represents a rough measure of the current navigational error of a receiver. (FOM will be displayed to the Crew, as well as MCC, in terms of Estimated Position Error, in feet). See FOM table (TABLE II) in the systems brief.

Frequency Band - A range of frequencies in a particular region of the electromagnetic spectrum.

GDOP - "Geometric Dilution of Precision" - is a measure of the purely geometrical contribution to the uncertainty in a dynamic position fix, for the three position coordinates plus clock offset in the solution. A good GDOP means a favorable geometric separation of satellites (ideally at least 4 satellites separated by at least 60 degrees). Also see DOP.

Global Positioning System (GPS) A satellite based navigation system.

GPS Receiver to be utilized by the SSP will be a military version (P-code version) of the Collins brand INS/GPS Receiver with 6 channels (final number of channels is TBD). A signal receiver that uses NAVSTAR RF signals to compute a position and velocity at a given time.

Handover Word - The word in the GPS message that contains the time synchronization information for the transfer from the C/A code to the P-code. (Also called the HOW word)

INS GPS (Inertial Navigation System GPS)- is a method of determining a craft's position, velocity, and acceleration (State Vector) by using inertial rate gyros/accelerometers in conjunction with GPS. The INS and GPS complement each other in this integrated system. The INS allows for faster GPS acquisition and re-acquisition, and improves dynamic GPS operation and jamming resistance. The GPS in return, provides periodic, real-time updates to the INS.

Ionospheric Refraction - A signal traveling through the ionosphere experiences a propagation time different from what would occur in a vacuum due to inhomogenates in the atmosphere.

Kinematic Positioning - refers to applications in which a trajectory is determined.

L-Band - The radio frequency band extending from 390 MHz to 1550 MHz.

NAVSTAR Global Positioning System Satellite Constellation operates in a 10,980 NM orbit in six planes, and an inclination of 55 degrees with respect to the equator. The complete constellation consists of 21 NAVSTAR satellites plus 3 active on-orbit spare in 12 hr orbits.

NAVSTAR satellites and their support equipment are financed by the Department of Defense (DoD), but their navigation signals are available free of charge to anyone, anywhere. The NAVSTAR satellite constellation control center facility is located at Falcon Air Force Base in Colorado Springs, Colo.

P-Code or "Precision Code" or "Protected Code"- is a high precision L-Band frequency that operates at a chipping rate of 10 million bits per second, and is reserved for use by the military. P-Code is protected from unauthorized users via encryption techniques. P-code is a very long (about 1014 bit) sequence of pseudorandom binary biphase pulse modulations on the GPS carrier and does not repeat itself for 267 days. Each one-week segment of the P-code is unique to one GPS satellite, and is reset once each week by the DoD. The SSP will utilize a P-Code receiver due to the increased precision/processing rate that the P-Code provides, and the control of the L-Band frequencies by the DoD.

PPS or "Precise Positioning System" - is the highest level of dynamic position accuracy that will be provided by GPS, and is available only via the dual frequency P-code. The Space Shuttle Program will employ PPS for GPS navigation.

PRN Code - Pseudorandom Noise code is any of a group of binary sequences that exhibit noise-like properties.

PRN Number - corresponds to the satellite number in the GPS constellation, and is used to identify satellites in the ephemeris matrix.

Pseudorange - The time shift required to align (correlate) a replica of the GPS code generated in the receiver with the GPS code, scaled into distance by the speed of light. This time shift is the difference between the time of the signal reception (measured in the receiver time frame) and the time of emission (measured in the satellite time frame). Pseudoranges change due to variations in the satellite-to-receiver propagation delay, and are biased by the time offset between satellite and receiver clocks. The resolution of pseudorange measurements depends on the accuracy with which the incoming and replicated codes can be aligned. A time accuracy of a few nanoseconds is equivalent to meter-level range position resolution.

Relative Positioning - The determination of relative positions between two or more receivers, which are simultaneously tracking the same radio positioning signals (e.g. from GPS).

Satellite Configuration - The state of the satellite constellation at a specific time, relative to a specific user or set of users.

Selective Availability - a mechanism by which the DoD can introduce small timing and position errors into the onboard GPS satellites to introduce small navigation errors into the SPS GPS user set. The two primary methods of introducing navigation errors via selective availability are by clock dithering and introducing ephemeris errors.

SPS or Standard Positioning System - the level of kinematic positioning accuracy that will be provided by GPS based on the single frequency C/A-code.

Switching Channel - A receiver channel which is sequenced through a number of satellite signals (each from a specific satellite and at a specific frequency) at a rate which is slower than, and asynchronous with, the message data rate.

UERE or User Equivalent Range Error - The contribution to the range measurement error from an individual error source, converted into range units, assuming that the error source is uncorrelated with all other error sources.

ZEROIZE - The Zeroize position on the Encryption switch inhibits P-Code operation of the GPS Receiver, defaulting to the C/A Mode, until reset by ground equipment. The Zeroize switch precludes unauthorized use of a P-Code GPS receiver (at an ELS).

1 σ Error - A one Sigma error is typically 50 feet or less and is a function of GDOP and User Equivalent Range Error (UERE). UERE is a line of sight vector connecting the user to each satellite.

B: IN FLIGHT ANOMALY LOG

| STS | IFA |
|------------|---|
| 91 | DPS-01B: GPS Command Lock-Up resulted in bad GPS State Vector propagation induced cyclic GPC errors. This in turn overloaded ICC communications between GNC GPC and SM GPC, which prevented accurate SM antenna management. |
| | GNC-01A: Loss of Tracking in Ascent. Poor ascent performance due to a filter reinit (per design) at MM 101 to MM 102 transition, coupled with < 4 SV track just after the reinit. |
| | GNC-02B:GPS 13 SV TILTS documented as Nav Data Good failures (5 TILTS) |
| 95 | GNC-03B: GPS 13 SV TILTS Conditions occurred (4 TILTS) |
| | GNC-04: TILT occurred during OPS 2 Pre-AMP Test, resulting in 100kft position and 150 fps velocity errors. Iono Model Singularity |
| 88 | GNC-001:Velocity Noise Numerous GPS downtrack and cross track velocity spikes (~ 3 fps) and QA3 limit violations |
| | GNC-002:TILT occurred during OPS 2 Pre-AMP Test. Due to more than 8 alternate SVs |
| | GNC-003C: GPS 13 SV TILTS Conditions occurred (4 TILTS) |
| 96 | GNC-001: Numerous bits unexpectedly set for one cycle |
| 103 | GNC-002: Tilt due to incorrect logical channel mapping. Only tracking satellites on UPPER ANT, progressive S/W failure |
| | GNC-002: Tilt due to both Pre-Amps being turned off |
| 99 | PRE-LAUNCH: S/N 13 - Failed Cold Key due to Low Battery |
| | PRE-LAUNCH: S/N 14 - Failed Self-Test due to Receiver Processor Failure |
| | GNC-001: PGSC reset and stopped recording data |
| 98 | GNC-01: Tilt due to > 15 satellites available in the alternate SV list (link 5 fixed) |
| 108 | GNC-02: S/Test failure during FCS C/O. Downlisted data indicated 4 bits where set in 3 separate fault log words, which were as follows (in Hex): fault log word 1 = 0804; fault log word 10 = 0002; and fault log word 11 = 009F. Fault log words 1 (0004) and 10 (0002) indicate a possible signal processor problem with GPS 2. Per the FCS C/O procedure, GPS 2 was deselected from the selection filter. After the completion of the self-test, GPS was commanded to NAV mode and continued with nominal performance. |

C: NOTES OF INTEREST

GPS PRT User Notes:

GITR's GPS153

DESCRIPTION:

If Power Up Self Test is interrupted by a short term power outage or TILT, the MAGR may not recognize the Shuttle MDM interface. The user will see a commfault on the SPEC55 display for this MAGR indicated by an over bright M.

USER RESPONSE:

If a commfault occurs for the reason cited above, recovery may be attempted as follows:

After at least 30 seconds, perform an I/O RESET.

If still no response, power off the MAGR, wait at least 30 seconds, power on the MAGR, wait at least 30 more seconds, and perform an I/O RESET.

If no response at this point then this MAGR cannot be used; power off.

SAIL IDR DIT996234

DESCRIPTION:

Any valid INIT data which is sent to the receiver while it is in INIT mode, is LOST if the receiver is powered down prior to going back to NAV mode, except time, which is initialized while still in INIT.

It has been observed in JAEL tests that after 30 minutes being in other-than-NAV mode, the velocity from the last NAV cycle is zeroed (and, position remains that of last NAV, but there is no propagation from that point). This is substantiated in the MAGR software function "nnavinit1.j73", where the code is remarked, "IF solution is over 1/2 hour old, ZERO velocity".

If the MAGR is moded out of NAV mode without the MAGR having had 4 satellite NAV track during the current NAV mode cycle, when moded back to NAV without valid aiding, the velocity will be zero and the position will remain that of the last valid NAV cycle. The MAGR propagation mode will be Ballistic Prop, and the MAGR state will propagate as if "falling" through to the center of the earth. In this scenario the MAGR will likely experience a "Type Conversion" error, and "tilt."

The effect on the system is that the MAGR may not be able to reacquire satellites for proper navigation.

USER RESPONSE:

During the normal power up 6 second INIT cycle, the receiver will accept valid* aiding as INIT data. If valid* aiding data is available to the MAGR, the loss of INIT data and/or zeroed velocity will not occur. If valid* aiding is not available, the user should not power cycle the MAGR or command the MAGR to other than NAV mode. Keeping the MAGR in NAV mode while aiding data is not available will help ensure the MAGR continues to track satellites for navigation.

* "Valid" aiding must pass time senescence check (<2 seconds old, <5 msec future).

SAIL IDR DIT996221

DESCRIPTION:

During an On-Orbit simulation run in SAIL, two MAGR receivers showed several ~20 fps velocity spikes at the time of a satellite switch. Both receivers were in PVA mode and tracking a (simulated) satellite in state 3 (low signal condition). The remaining MAGR receiver was in INS mode and did not show the spikes.

This is expected MAGR receiver operation under the noted conditions. The MAGR Integration Guide supplied by the vendor explains that, "When operating in the PVA (unaided) mode, the foreground

navigation solution is propagated by integrating the GPS best estimate of acceleration (assumed to be constant) at a 10 hertz rate using GPS best estimates of position and velocity as initial conditions." The receiver's best estimates of position and velocity are degraded if one or more satellites are being tracked in state 3. The spikes are a result of the propagation of the (assumed) constant acceleration. Further degradation could occur following a satellite switch. During the more dynamic Entry phase of flight, operating in the PVA mode could produce noisier velocity data than that observed while On-Orbit.

The MAGR receivers would normally be operating in INS mode for all flight phases. PVA mode is only expected during OPS transition or in an anomalous condition when the orbiter supplied aiding is erroneous or missing. Erroneous aiding is the result of either a grossly erred onboard navigation state or an erred shuttle system time which would affect all installed receivers. Missing aiding data is the result of a data transfer error between the shuttle and the MAGR and would affect only the involved receiver. During normal operation, the receivers may occasionally track a satellite in state 3 for brief periods depending upon satellite elevation and shuttle orientation with respect to the satellite. During the "blackout" region of Entry, the receivers are more likely to track satellites in state 3 due to plasma effects on the satellite RF signals.

In summary, short term PVA operation followed by a transition from PVA to INS mode is a normal event expected due to OPS transition, while extended PVA operations are unexpected events resulting from erroneous or missing aiding data. In all cases, adequate FSW checks are in place to protect the orbiter navigation system from significantly degraded GPS errors.

USER RESPONSE:

During an OPS transition, when only a subset of the installed receivers is in PVA – Deselect only the receiver(s) in PVA mode.

If all receivers are in PVA, no action required.

SAIL IDR DIT996137

DESCRIPTION:

During an On-Orbit simulation run in SAIL, GPS UVW error spikes of ~100 ft and ~34 ft/sec were seen following OPS transition at the transitions between PVA and INS receiver modes, and prior to the OPS transition filter restart. This behavior is also seen in flight data. The PVA/INS mode transitions typically occur for approximately 15 seconds following an OPS transition. The error spikes cease at the transition back to INS mode.

This is expected MAGR receiver operation following an OPS transition. Since GPS Rcvr State Proc executes every 3.84 sec and the duration of the spike is always < 3.8 sec, the orbiter navigation system is protected from taking an erroneous GPS state vector (spike > QA3 limits) and making it the selected GPS state. Furthermore, the GPS to Nav command is initialized to INHIBIT after an OPS transition and an additional crew command to take GPS to Nav out of INHIBIT is required before the selected GPS state vector could be incorporated into NAV.

USER RESPONSE:

No user response required.

GITR GPS237

DESCRIPTION:

Noisy or erroneous satellite measurements, combined with poor geometry and/or an underdetermined navigation solution (less than 4 SVs), could cause the MAGR-S3S's Navigation Kalman Filter to produce erroneous data that would prohibit acquisition or reacquisition of satellites. GITR GPS237 cites an example of this problem (STS-88, FOM Chimney #8), where erroneous satellite measurements occurred, and Space Station blockage caused poor satellite geometry.

There are currently no other known examples of this problem and, with corrections incorporated in the current MAGR firmware link, no known specific causes for erroneous satellite measurements.

The problem would be identifiable by the following sequence/conditions:

- FOM > 5
- MAGR state significantly diverging from the orbiter NAV state, followed by
- < 4 satellites tracked, followed by
- Kalman filter asleep (identified by zero state error)
- Subsequent inability of the MAGR to acquire/re-acquire satellites

USER RESPONSE:

If this problem is suspected, and recovery time is not critical, the crew should be instructed to perform an INIT. An INIT would cause the MAGR to drop any satellites being tracked, but would clean-up internal states and allow satellite tracking without further problem. Expect up to 2 minutes recovery time following an INIT-NAV transition. If recovery time is more critical, a Filter Restart should be attempted first. A Filter Restart should clear the condition while allowing the MAGR to continue to track any satellites it already has for a quicker recovery. Expect up to 1 minute recovery time. If the problem persists after performing a Filter Restart, the crew should be instructed to perform an INIT.

SAIL IDR DIT026179

Large navigation errors were observed in BFS after engage with GPS in AUTO and the MAGRS-3S GPS receiver configured to use the new GPS residual edit limits.

DESCRIPTION:

Large navigation errors were observed in BFS after engage with GPS in AUTO and the MAGRS-3S GPS receiver configured to use the new GPS residual edit limits.

This was observed in SAIL during a BFS engage case in MM304 with GPS for STS114. In this case GPS had been incorporated in BFS NAV earlier. At BFS engage, a transient was introduced in the GPS aiding state as a result of switching from the PASS GPS reference state to the BFS UPP state. The transient, coupled with the new MAGR Residual Edit limits, initiated an instability in the BFS GPS Nav system. Subsequent SAIL runs under the same conditions but with the old MAGR Default Edit limits did not produce the instability.

USER RESPONSE:

Do not incorporate GPS data into the BFS Nav if the MAGR Default Edit limits are not active.

MAGRS-3S DRs (Link-007)

PRN '32' is presented as PRN '0' (Preliminary User Note as of 11/06)

DESCRIPTION:

Due to a receiver five bit limitation, when PRN '32' is being tracked, it is displayed as PRN '0' in both PASS and BFS SPEC 55 displays. Both PASS and BFS downlist will show PRN '0' whenever a receiver channel is tracking or attempting to track PRN '32'.

This is a display and downlist feature only; all other operational functions (including satellite deselection) are not affected.

USER RESPONSE:

The user should be aware that the receiver is actually tracking satellite PRN '32' when the SPEC55 display shows the receiver is tracking satellite PRN '0'. No other operational functions are affected. Accordingly, should the need arise to deselect satellite PRN '32', then '32' should be entered (not '0').

In downlist, an indicated PRN of '0' will actually represent PRN '32' if the channel state is '3' or '5'. For any other channel state, PRN '0' may represent either an idled channel, or PRN '32'. No action is required.

DR 110645

MAGRS-3S Not Authorized Although Loaded with Valid Encryption Key

DESCRIPTION:

After being loaded with a valid Group Unique Key (GUK), a GPS receiver may fail to validate the daily encryption key, resulting in the inability to acquire and track PPS Y-code signals. After tracking one or more satellite signals for more than 6 seconds, the downlisted PPS status word for an affected receiver will be the undefined value of hex '088X' or '0C8X' (where X can be of any value). This condition has only been observed to occur after extended power outage (> 7 days). This condition has been observed to clear upon reception of the next daily key. This condition should also clear if the receiver is zeroized and the GUK is reloaded.

Off-line analysis and flight tests have demonstrated acceptable navigation performance with a receiver operating unauthorized with the SPS using C/A code only.

USER RESPONSE:

Monitor the PPS status word for each receiver to ensure that the daily key is and continues to be validated (hex '888X'). If a PPS status word value of '088X' or '0C8X' is observed, the following actions may be taken in order of preference:

- If sufficient time is available, wait until the next daily key is validated
- If pre-launch and sufficient time is available, zeroize the receiver and reload the GUK
- If a known software state is absolutely required, zeroize the receiver. Zeroizing the receiver while in flight will result in SPS operations and preclude returning to PPS for the remainder of the flight.

DR 110646

MAGRS-3S Hardware Channel Cannot Acquire Y-Code (YETI)

DESCRIPTION:

An affected GPS receiver hardware channel will not be able to acquire Y-code signals. The user may see failed Y-code acquisitions, failed C/A to Y-code handovers, and (most notably) failed ionospheric delay and channel calibration measurements. Ionospheric delay measurements are characterized by the peripheral channel tracking the same satellite as a navigation channel but in a different frequency. Channel calibration measurements are characterized by the peripheral channel tracking the same satellite as a navigation channel in the same code and frequency). Performance degradation may also occur.

The primary evidence of degraded performance will be elevated FOM and GDOP values. Degradation will be the result of C/A code tracking on a navigation channel, inability to perform ionospheric delay and

channel calibration measurements, and inability to switch requested satellites into the navigation constellation. The onset of this condition has not been tied to any specific stimulus.

USER RESPONSE:

Monitor for the occurrence of the described condition. If detected, the response will depend on a number of factors. If in a non-critical flight phase, recover the receiver to nominal operations by keyboard command of the receiver to INIT mode and then back to NAV mode. If in a GPS-critical flight phase, monitor the receiver for degraded performance as described above. If there is no detected performance degradation, do nothing. If receiver performance is degraded to the point of an out-of-service condition, recover the receiver to nominal operations by keyboard command of the receiver to INIT and then back to NAV. Be aware the receiver recovery time may exceed out-of-service constraints. Do not command INIT-NAV mode transitions during GPS signal blackout during entry.

DR 110640

Possible PASS GPS Initialization Command Lock-Up

DESCRIPTION:

Once an initialization command is issued in PASS, the variable INIT_STATE_SET is set true and is not reset until any of the following occurs:

- The MAGR responds
- A GPS bypass is detected
- The GPS I/O control option is "1"
- An OPS transition is performed
- The GPS reference state processor declares a command time-out.

While INIT_STATE_SET

USER RESPONSE:

Monitor for the occurrence of the described condition. If detected, the response will depend on a number of factors. If in a non-critical flight phase, recover the receiver to nominal operations by keyboard command of the receiver to INIT mode and then back to NAV mode. If in a GPS-critical flight phase, monitor the receiver for degraded performance as described above. If there is no detected performance degradation, do nothing. If receiver performance is degraded to the point of an out-of-service condition, recover the receiver to nominal operations by keyboard command of the receiver to INIT and then back to NAV. Be aware the receiver recovery time may exceed out-of-service constraints. Do not command INIT-NAV mode transitions during GPS signal blackout during entry.

DR 110669

SATELLITE TRACKING ON A CARRIER SIDEBAND CAN DEGRADE THE GPS SOLUTION

DESCRIPTION:

IN CERTAIN SIGNAL ENVIRONMENTS, THE MAGRS-3S CAN TRACK ON A SIDEBAND OF A COARSE ACQUISITION (C/A) CODE GPS SATELLITE SIGNAL. DURING MAGR DEVELOPMENT, TWO PRIMARY CAUSES OF THE SIDEBAND SIGNAL TRACKING WERE IDENTIFIED: LARGE SENSED JERK (RATE OF CHANGE OF ACCELERATION) OR SIGNAL MULTIPATH. SIDEBAND TRACKING OF A SATELLITE CAN AFFECT THE RECEIVER'S PERFORMANCE WHEN MANIFESTED UNDER TWO DIFFERENT CONDITIONS:

SIDEBAND TRACKING ON A PERIPHERAL CHANNEL CAN TRIGGER PARITY ERRORS DURING DATA COLLECTION. THIS CAUSES REPEATED DATA COLLECTION ATTEMPTS UNTIL THE AFFECTED CHANNEL LOSES TRACKING, THE DATA COLLECTION PROCESS COMPLETES, OR UNTIL THE PROCESS IS INTERRUPTED BY A HIGHER PRIORITY TASK (SUCH AS LOSS OF LOCK ON A NAVIGATION CHANNEL). PERSISTENT DATA COLLECTION ATTEMPTS BY THE PERIPHERAL CHANNEL, IN THE LONG-TERM, CAN CAUSE INCREASE IN THE FIGURE OF MERIT (FOM) AS THE GEOMETRY OF THE NAVIGATION (NAV) CONSTELLATION DEGRADES.

SIDEBAND TRACKING ON A NAV CHANNEL CAN LEAD TO RECEIVER NAVIGATION WITH 4 SATELLITES TRACKED BUT WITH AN UNDERDETERMINED (LESS THAN 4 SATELLITE

MEASUREMENTS) FILTER. THE RECEIVER WILL NOT MAKE ANY ATTEMPT TO REPLACE THE OFFENDING NAV SATELLITE UNLESS IT IS DUE TO NORMAL SATELLITE SWITCHING CONDITIONS, LOSS-OF-LOCK ON THE OFFENDING SATELLITE, OR THE SATELLITE BEING DROPPED OFF THE VISIBLE LIST. IF THE MAGRS-3S NAV SOLUTION IS BEING AIDED BY THE PRE-OI32 BFS SOFTWARE, WHILE THE MAGRS-3 HAS AN UNDERDETERMINED FILTER, EXTRANEIOUS VELOCITY ERRORS CAN BE INDUCED TO THE MAGRS-3S NAV SOLUTION DUE TO COUPLING OF INTERMITTENT USE OF GPS SATELLITE DATA AND UNANNOUNCED TRANSIENTS IN THE BFS AIDING STATE. INCORPORATION OF THIS DATA INTO THE PRE-OI32 BFS SOFTWARE CAN LEAD TO BOTH BFS AND GPS NAV STATE VECTORS WALK-OFF.

USER RESPONSE:

CONDITION 1, SIDEBAND TRACKING ON THE PERIPHERAL CHANNEL
CAN BE DETECTED BY:

GREATER THAN 2 MINUTES OF CONTINUOUS TRACKING OF THE SAME SATELLITE (PASS/BFS SPEC 55 AND D/L), AND GRADUAL INCREASE IN FIGURE OF MERIT (PASS SPEC 50, 3-STRING BFS SPEC 50, PASS/BFS SPEC 55, AND D/L).

CONDITION 2, SIDEBAND TRACKING ON A NAVIGATION CHANNEL
CAN BE DETECTED BY:

FREQUENT '<4 MEAS' (PASS/BFS D/L) INDICATIONS SIMULTANEOUSLY WITH FOUR SATELLITES INDICATING TRACK IN STATE 3 OR STATE 5. IN ADDITION, IF PASS D/L IS AVAILABLE, THE AFFECTED SATELLITE WILL BE TRACKING IN C/A-CODE.

IF GPS IS INCORPORATED WITH BFS ENGAGED (PRE-OI32 BFS SOFTWARE), BFS/GPS VELOCITY WALK-OFF CAN BE DETECTED BY THE GROUND. IN ADDITION, THE CREW (3-STRING SPEC 50) AND THE GROUND WILL SEE CONTINUOUS ELEVATED VELOCITY RESIDUAL AND RATIO.

PROBLEM WORKAROUND PROCEDURE:

A. IF GPS IS NOT NEEDED FOR ONBOARD OR GROUND NAV FILTER PROCESSING, NO ACTION IS REQUIRED. THIS PROBLEM WILL EVENTUALLY SELF-CORRECT.

B. DESELECT THE OFFENDING SATELLITE (PASS/BFS SPEC 55, ITEMS 43-50) TO CLEAR SIDEBAND TRACKING. RESELECTION OF THIS SATELLITE CAN BE DONE ONCE THE SATELLITE IS CONFIRMED TO BE DESELECTED. A RECURRENCE OF THIS CONDITION SHORTLY AFTER SATELLITE RESELECTION IS HIGHLY UNLIKELY, BUT THIS STEP MAY BE REPEATED IF NEEDED. IF IT IS NOT POSSIBLE TO ISOLATE THE OFFENDING SATELLITE, OR IF IN A 3-STRING GPS CONFIGURATION AND THE SAME SATELLITE IS ALSO USED AS A NAV SATELLITE FOR OTHER RECEIVER(S), AN ALTERNATE OPTION IS TO COMMAND THE AFFECTED RECEIVER TO 'INIT' AND BACK TO 'NAV'.

NOTE: FOLLOWING AN INIT-NAV TRANSITION, BE AWARE THE RECEIVER RECOVERY TIME MAY EXCEED OUT-OF-SERVICE CONSTRAINTS. DO NOT COMMAND INIT-NAV MODE TRANSITIONS WHILE IN GPS SIGNAL BLACKOUT DURING ENTRY.

C. BFS ENGAGED WITH PRE-OI32 BFS SOFTWARE: IF CONDITION 2 IS DETECTED DURING BFS GPS INCORPORATION, DESELECT THE AFFECTED RECEIVER(S) OR COMMAND BFS TO INHIBIT GPS INCORPORATION. PERFORM STEP B ABOVE TO CLEAR THE SIDEBAND TRACKING CONDITION. RESELECTION OF THE DESELECTED RECEIVER(S) OR GPS INCORPORATION MAY BE RESUMED ONCE THE CONDITION IS CLEARED.

D. GPS INCORPORATION WITH NO GROUND COMMUNICATION: NO ACTION IS REQUIRED BY THE CREW. FLIGHT DATA FILE PROCEDURES AND CREW TRAINING PROVIDE THE PROPER RESPONSE TO A 'NO UPDATE GPS' OR 'NAV EDIT GPS' GAX MESSAGE (PASS), OR A 'NAV EDIT' MESSAGE (BFS).

User Note 035211

I/O RESET WITH FAILED MTU OR GPS CAUSES BFS TO DOWNMODE A STRING

DESCRIPTION:

WHEN AN I/O RESET IS ENTERED IN THE PASS GNC SYSTEM WITH AN MTU ACCUMULATOR OR GPS RECEIVER FAILED, THE MTU/GPS ELEMENT WILL BE BYPASSED AFTER "N" ERRORS (N>2). THIS IS DUE TO ALL TRANSACTION COUNTERS BEING RESET FOR EACH OF THE SEVEN BUS/STRINGS RESTORED (ONE PER MAJOR CYCLE). SINCE MTU/GPS I/O IS PERFORMED ONCE PER MAJOR CYCLE, ITS COUNTER IS INCREMENTED AND CLEARED UNTIL ALL RESTORES ARE COMPLETE. THE ERROR COUNT THEN REACHES TWO. THE BFS BYPASSES THE MTU/GPS AFTER TWO ERRORS AND, BECAUSE PASS HAS NOT YET BYPASSES THE MTU/GPS (DUE TO COUNTERS BEING CLEARED), THE BFS DOWNMODES THE APPROPRIATE STRING.

USER RESPONSE:

IF AFTER ENTERING "I/O RESET EXEC" IN THE PASS SYSTEM WITH AN MTU ACCUMULATOR OR GPS RECEIVER FAILED AND A MESSAGE APPEARS ON THE BFS DISPLAY STATING THAT A STRING WAS DOWNMODED, PERFORM AN I/O RESET IN BFS AFTER THE PASS I/O RESET IS COMPLETE (WAIT FOR 10 SECONDS TO ELAPSE).

DR: 109184

GPS DILEMMA/FAILURE REANNUNCIATED AT OPS TRANSITION

Description:

THE GPS_DILEMMA_ANNUN_GAX AND GPS_QA_GAX FLAGS, WHICH INDICATE RESPECTIVELY THAT A GPS DILEMMA OR FAILURE SHOULD BE ANNUNCIATED, ARE INITIALIZED OFF FOLLOWING AN OPS TRANSITION. IF AN EXISTING DILEMMA OR QA FAILURE CONDITION PERSISTS ACROSS AN OPS TRANSITION, THE CORRESPONDING GAX FLAG WILL BE SET OFF AND THEN SET ON AGAIN AFTER THE APPROPRIATE I-LOADED COUNTER LIMIT IS EXCEEDED. THIS MAY RESULT IN A NEW MESSAGE ANNUNCIATION AT OPS TRANSITION FOR A PREVIOUSLY ANNUNCIATED DILEMMA OR FAILURE, WHICH IS CONTRARY TO EXPECTED GAX OPERATION.

User Response:

THERE IS NO WORKAROUND REQUIRED.

DR: 110337

GPS RECEIVER START-UP DELAY ON ENTERING NAV MODE

Description:

THERE IS A START-UP DELAY WHEN A GPS RECEIVER ENTERS NAV MODE BEFORE VALID STATES ARE AVAILABLE FROM THE RECEIVER FOR THE FSW TO PROCESS. A TYPICAL DELAY IS TWO MINUTES, BUT POTENTIALLY IT MAY LAST UP TO TEN MINUTES OR LONGER, REPRESENTING A SIGNIFICANT PERIOD FOR THE NAVIGATION SYSTEM TO BE WITHOUT GPS DATA. THIS SITUATION OCCURS WHEN: (1) THE GPS RECEIVER IS POWERED UP, OR (2) THE CREW COMMANDS THE RECEIVER TO NAV MODE.

THE CREW WILL SEE "DGF" DISPLAYED IN THE RECEIVER'S STAT FIELD ON THE

GPS STATUS DISPLAY. IN OPS MODES G2/G3/G8, MM602 AND MM603, WHERE THE GPS RECEIVER STATE PROCESSING PRINCIPAL FUNCTION IS ACTIVE, A DOWN ARROW WILL BE DRIVEN IN ITS DG FAIL FIELD. THE RECEIVER'S QA2 AND QA3 FIELDS WILL SUBSEQUENTLY BE BLANKED, AS WILL THE QA4 FIELDS INVOLVING THAT RECEIVER. THE RECEIVER'S SF CAND FIELD WILL DISPLAY A DOWN ARROW. IF THE START-UP DELAY EXCEEDS THE GPS ALERT DELAY (I-LOAD V97U3488C) AND NO OTHER GPS STATE IS AVAILABLE, A "NO UPDATE GPS" ALARM WILL BE ANNUNCIATED.

User Response:

POWER OFF/ON THE GPS RECEIVER, OR COMMAND VIA KEYBOARD ITEM ENTRY THE RECEIVER TO ANOTHER MODE AND BACK TO NAV MODE, AT TIMES WELL IN ADVANCE OF WHEN GPS DATA MIGHT BE NEEDED BY THE NAVIGATION SYSTEM.

DR: 110338

GPS TO NAV/G&C INTERFERES WITH LAMBERT GUIDANCE

Description:

DURING OR AFTER A FINAL ORBIT TGT LAMBERT-TARGETED BURN CALCULATION AND BEFORE THE COMPLETION OF THE CORRESPONDING LAMBERT-GUIDED BURN, IF EITHER GPS TO NAV OR GPS TO G&C WERE SET TO AUTO OR FORCE ON SPEC 55 (GPS STATUS DISP), AND/OR GPS TO NAV WAS SET TO AUTO OR FORCE ON SPEC 33 (REL NAV DISP), LAMBERT GUIDANCE MAY FAIL TO CONVERGE, OR LARGE CHANGES IN THE MANEUVER EXECUTE DISPLAY DELTA-V'S MAY BE SEEN, WHEN LOAD (ITEM 22) IS REQUESTED ON THE MANEUVER EXECUTE DISPLAY AND/OR DURING CYCLIC GUIDANCE. IF THIS OCCURS, A "TGT ITER" OR "TGT DELTA T" ALARM MAY BE ANNUNCIATED.

User Response:

ENSURE THAT BOTH GPS TO NAV AND GPS TO G&C REMAIN IN INHIBIT FROM FINAL TARGETING UNTIL AFTER EXECUTION OF A LAMBERT TARGETED BURN. NOTE THAT ON OI-27 AND SUBSEQUENT SYSTEMS, IF GPS TO NAV IS IN AUTO, GPS UPDATES TO NAV WILL NOT OCCUR IN MM 202 AND A DOWN ARROW WILL BE DISPLAYED IN THE GPS TO NAV STATUS FIELD.

DR: 110465

MLS FORCE BEFORE MLS ACQUISITION LOCKS OUT GPS

Description:

DURING MM602, IF THE MLS AUTO/INHIBIT/FORCE (AIF) CONTROL IS SET TO FORCE VIA ITEM ENTRY ON THE HORIZONTAL SITUATION DISPLAY (THREE GPS RECEIVER CONFIGURATION) PRIOR TO MLS ACQUISITION, GPS STATES AND DRAG AND BARO MEASUREMENTS ARE NO LONGER INCORPORATED INTO NAVIGATION. THE NAVIGATION STATE IS PROPAGATED UNTIL MLS ACQUISITION, OR MLS AIF IS CHANGED VIA ITEM ENTRY TO AUTO OR INHIBIT.

User Response:

DO NOT MAKE ITEM ENTRY ON THE HORIZONTAL SITUATION DISPLAY TO SET MLS AIF TO FORCE BEFORE MLS ACQUISITION. IF MLS AIF FORCE ITEM ENTRY IS MADE INADVERTENTLY PRIOR TO MLS ACQUISITION, RESTORE MLS AIF TO AUTO VIA ITEM ENTRY AS SOON AS POSSIBLE.

DR: 110486

UNEXPECTED GPC ERRORS LOGGED ON RECEIPT OF BOGUS GPS DATA

Description:

IF THE GPS RECEIVER SHOULD PROVIDE A GPS RECEIVER STATE WITH A VALID HEADER, NAV DATA VALIDITY FLAG AND TIME TAG WITH ZEROES OR NEAR-ZEROES FOR POSITION AND VELOCITY, GPC ERRORS ARE POSSIBLE WHEN GPS RECEIVER STATE PROCESSING ATTEMPTS TO PROPAGATE THIS STATE VECTOR TO CURRENT NAV FILTER TIME (QA2 AND QA3 FAILURES MAY ALSO BE OBSERVED). THE EXPECTED ERRORS ARE EXPONENT OVERFLOW (GROUP 3, CODE 11), SQUARE ROOT OF A NEGATIVE NUMBER (GROUP 4, CODE 5), AND ARGUMENT OF THE EXP FUNCTION TOO LARGE (GROUP 4, CODE 6). THESE ERRORS WILL HAVE A PDE OF THE CURRENT LFE: GELORB IN OPS 2, GERENT IN OPS 3, GEHRTL IN OPS 6 (OI27 AND SUBS ONLY), AND GEJORB IN OPS 8.

IF PROCEDURAL ERROR ALLOWS THE BAD GPS STATE TO BECOME THE SELECTED GPS STATE, THEN ADDITIONAL ERRORS OF THE ABOVE TYPES MAY BE OBSERVED. IF PROCEDURAL ERROR ALLOWS THE BAD GPS STATE TO BE INCORPORATED INTO THE NAVIGATION STATE, ADDITIONAL ERRORS OF THE ABOVE TYPES MAY BE SEEN AT THE LFE AND MFE PRIORITIES. SCALAR TO INTEGER CONVERSION OVERFLOW ERRORS (GROUP 4, CODE 15) MAY ALSO BE SEEN AT THE MFE LEVEL.

DIVIDE BY ZERO (GROUP 3, CODE 12) AND BOTH ARGUMENTS OF THE ARCTAN2 FUNCTION EQUAL ZERO (GROUP 4, CODE 62) ERRORS ARE POSSIBLE IN MAJOR MODES 304, 305, 602, AND 603 WHEN GPS UPP DISPLAY COMPUTATIONS ATTEMPT TO USE A ZERO STATE VECTOR TO COMPUTE RANGE, AZIMUTH, AND ALTITUDE PARAMETERS FOR OUTPUT TO THE HORIZONTAL SITUATION DISPLAY. PDE WOULD BE THE APPROPRIATE MFE: GEQENT IN OPS 3 AND GEGASC IN OPS 6.

User Response:

IF A TRANSIENT GPS RECEIVER STATE WITH VALID HEADER DATA AND ZERO OR NEAR-ZERO STATE VECTOR OCCURS AND PRODUCES GPC ERRORS, NO ACTION IS REQUIRED. THE GPC ERRORS WILL STOP WHEN A NEW VALID GPS RECEIVER STATE IS PROCESSED BY THE FSW. IF THE CONDITION PERSISTS AND GPC ERRORS CONTINUE TO OCCUR, POWER OFF THE GPS RECEIVER. THIS ACTION SHOULD STOP THE GPC ERRORS, WITH THE EXCEPTION OF GPC ERRORS FROM GPS UPP DISPLAY COMPUTATIONS, WHICH COULD CONTINUE AT A 0.52 HZ RATE.

IT IS ALSO RECOMMENDED THAT GPS TO NAV BE IN INHIBIT WHEN THE CREW PERFORMS A QA OVERRIDE OR GPS TO G&C FORCE, AND THE OVERRIDE OR FORCE BE REMOVED BEFORE SETTING GPS TO NAV TO AUTO OR FORCE. THE RESULTING GPS SELECTED STATE CAN THUS BE EVALUATED BEFORE IT IS INCORPORATED INTO THE NAV STATE.

DR: 110612

TRANSLATIONAL ACCELERATIONS DUE TO ROTATIONS NOT INCORPORATED INTO GPS SELECTED STATE

Description:

THE GPS STATE PROPAGATION SCHEME DOES NOT ACCUMULATE AND INCORPORATE SMALL TRANSLATIONAL ACCELERATIONS DUE TO ATTITUDE ROTATION MANEUVERS AS IS DONE IN THE ORBIT NAVIGATION SCHEME. THEREFORE, THE GPS SELECTED STATE WILL SLOWLY DIVERGE FROM THE BASELINE NAVIGATION STATE IF THE GPS STATE IS NOT UPDATED BY THE GPS RECEIVER FOR AN EXTENDED PERIOD OF TIME. THE CREW WILL SEE GPS MINUS NAV ERRORS ON THE GPS STATUS DISPLAY (SPEC 55) GRADUALLY INCREASE UNTIL A NAV RESET

OCCURS. ONCE THE GPS SELECTED STATE HAS BEEN PROPAGATED FOR AN I-LOADED PERIOD OF TIME WITHOUT GPS UPDATES FROM THE RECEIVER, A NO UPDATE GPS MESSAGE WILL BE ANNUNCIATED. GROUND CONTROL WILL ALSO SEE THE SLOW DIVERGENCE.

User Response:

GPS DATA SHOULD NOT BE INCORPORATED INTO THE NAV STATE IF A SELECTED STATE IS BEING PROPAGATED FOR AN EXTENDED PERIOD. THE AGE OF THE LAST SELECTED STATE IS DISPLAYED IN THE "LAST SEL FIL UPDATE" FIELD ON SPEC 55. BEFORE MODING GPS TO NAV TO AUTO OR FORCE, THE CREW SHOULD WAIT UNTIL A NEW SELECTED STATE IS AVAILABLE FROM A GPS RECEIVER. A QA OVERRIDE MAY BE REQUIRED IF QA FAILURES OCCUR DUE TO THE GRADUAL BUILD UP OF A LARGE DIVERGENCE.

DR: 110613

DETERMINING GPS SELECTED STATE STATUS THROUGH "LAST SEL FIL UPDATE"

Description:

THE "LAST SEL FIL UPDATE" FIELD ON THE GPS STATUS DISPLAY (SPEC 55) INDICATES THE HEALTH OF THE GPS SELECTED STATE. WHEN THE FIELD IS BLANK, A GPS SELECTED STATE IS NOT AVAILABLE FOR USE. WHEN THE VALUE IS '0.0', A SELECTED STATE IS AVAILABLE AND IS BEING UPDATED. WHEN THE VALUE IS SOMETHING GREATER THAN '0.0', A SELECTED STATE IS AVAILABLE BUT IS NOT CURRENTLY BEING UPDATED BY NEW GPS RECEIVER STATES. THE NUMBER INDICATES HOW LONG IT HAS BEEN SINCE THE SELECTED STATE WAS UPDATED.

IF THE SELECTED STATE IS NOT UPDATED IN GPS_ALERT_DT (ILOAD) SECONDS, A "NO UPDATE GPS" GAX MESSAGE WILL BE ANNUNCIATED. THIS ANNUNCIATION IS INDEPENDENT OF THE "LAST SEL FIL UPDATE" FIELD. THIS FIELD CAN BE EITHER A VALUE GREATER THAN ZERO OR BLANK AT THIS TIME.

IF AN OPS MODE RECALL IS PERFORMED, THE GPS SELECTED STATE WILL NOT BE AVAILABLE, SIMILAR TO A NOMINAL OPS TRANSITION. HOWEVER, THE GPS SELECTED STATE WILL NOT BE ZEROED DURING AN OPS MODE RECALL. THIS WILL BE SEEN AS A STATIC GPS SELECTED STATE IN THE DOWNLIST. THE "LAST SEL FIL UPDATE" FIELD WILL BE BLANKED UNTIL A NEW GPS SELECTED STATE IS AVAILABLE.

User Response:

BE AWARE OF THIS BEHAVIOR WHEN INTERPRETING "LAST SEL FIL UPDATE" RESULTS. IF THE FIELD DISPLAYS:

0.0 - SELECTED STATE IS CURRENTLY BEING UPDATED WITH GPS DATA

>0.0 - SELECTED STATE HAS BEEN UPDATED WITH GPS DATA IN THIS OPS (BUT IS NOT NOW BEING UPDATED)

BLANK - SELECTED STATE IS NOT AVAILABLE. THIS COULD BE THE RESULT OF ONE OF THE FOLLOWING:

- 1) A NEW OPS HAS BEEN ENTERED OR AN OPS MODE RECALL WAS EXECUTED AND A GPS SELECTED STATE HAS NOT YET BEEN MADE AVAILABLE.

-OR-

- 2) ALL RCVRS ARE IN SOME COMBINATION OF OFF, BYPASSED, OR

DESELECTED.

WHEN THE "LAST SEL FIL UPDATE" FIELD IS BLANK, THE GPS SELECTED STATE WILL NORMALLY BE ZERO BUT COULD BE STATIC IN THE OPS MODE RECALL SCENARIO.

DR: 110620

GPS DILEMMA STATUS/ANNUNCIATION PROBLEMS WITH A DATA GOOD FAILURE

Description:

IF A 3-WAY DILEMMA IS REDUCED TO A 2-WAY DILEMMA BY A GPS DATA GOOD FAILURE, THE GAX DILEMMA FLAG FOR THE RECEIVER WITH DATA GOOD OFF WILL REMAIN ON. IF THE 2-WAY DILEMMA IS RESOLVED AND THEN A NEW 2-WAY DILEMMA OCCURS AT A LATER TIME, THERE WILL NOT BE A NEW DILEMMA ANNUNCIATION DUE TO THE GAX DILEMMA FLAG REMAINING ON FROM THE ORIGINAL DILEMMA.

IF A 2-WAY DILEMMA OCCURS, AND THEN ONE OF THE RECEIVERS HAS A DATA GOOD FAILURE, THEN THE DILEMMA CONDITION WILL BE LOCKED IN. IF THE DATA GOOD FAILURE OCCURS BEFORE THE DILEMMA IS ANNUNCIATED, THEN, ON OI27 ONLY, THE RECEIVER THAT DOES NOT HAVE THE DATA GOOD FAILURE WILL SHOW A "?" STATUS WHEN THE DILEMMA IS ANNUNCIATED. NOTE THAT AS LONG AS THE DATA GOOD FAILURE EXISTS AND NEITHER RECEIVER IS DESELECTED, THE DILEMMA WILL MAKE ALL GPS DATA UNAVAILABLE FOR SELECTION.

User Response:

BE AWARE THAT DATA GOOD FAILURES ON GPS RECEIVERS INVOLVED IN DILEMMA CONDITIONS MAY LOCK IN SOME DILEMMA INDICATIONS. THIS MAY RESULT IN A FAILURE TO ANNUNCIATE A NEW DILEMMA CONDITION WHEN EXPECTED, OR THE ISSUING OF AN ANNUNCIATION WHEN ONLY ONE RECEIVER APPEARS TO BE AVAILABLE.

THE LATCHED DILEMMA CONDITIONS MAY BE CLEARED BY A MANUAL DESELECT OF GPS RECEIVERS THAT HAVE HAD A DATA GOOD FAILURE.

DR: 110621

POSSIBLE GPS COMMAND LOCK-UP ON GPS STATUS DISPLAY

Description:

IF A GPS RECEIVER IS STUCK IN A MODE (NAV, INIT, OR TEST) AND IS COMMANDED TO ANOTHER MODE, THEN THE COMMANDED MODE ASTERISK WILL MOVE TO THE NEW COMMANDED MODE, BUT THE RECEIVER MODE FIELD WILL REMAIN IN THE ORIGINAL MODE. SUBSEQUENT COMMANDED OF EITHER MODE WILL RESULT IN AN ILLEGAL ENTRY MESSAGE.

User Response:

DEPENDING ON THE CAUSE OF THE GPS FAILURE, IT MAY BE POSSIBLE TO REGAIN COMMAND OF THE RECEIVER BY COMMANDED THE THIRD MODE (THE "ODD" MODE THAT IS NOT THE CURRENT MODE NOR CURRENTLY COMMANDED MODE) OR BY POWER CYCLING THE RECEIVER.

DR: 110630

GPS COMMAND INDICATIONS WHEN GPS RECEIVER IS LOCKED-UP

Description:

NOTE: THE DESCRIPTION/USER RESPONSE IN THIS NOTE ARE ONLY CORRECT FOR SYSTEMS WITH CR'S 92288A OR 92289A (POST STS-91 CORRECTIONS) APPLIED (I.E., NOT STS-90, STS-91, AND STS-93).

WHEN A GPS RECEIVER FAILS TO RESPOND TO AN INTERNALLY GENERATED FSW COMMAND OR A CREW COMMAND (THE GPS RECEIVER IS LOCKED-UP), THE FSW WILL WAIT 16 SECONDS (KLOAD V97U6071C) BEFORE TERMINATING ALL CURRENT AND PENDING COMMANDS.

DEPENDING ON THE TYPE OF GPS COMMAND (GPS RECEIVER INITIALIZATION (INIT), GPS FILTER RESTART, GPS ANNOUNCED RESET, OR A GPS TO IMU RECONFIGURATION (FOR OI27 AND OI28)), USERS WILL HAVE DIFFERENT INDICATIONS OF THE CURRENT GPS STATUS.

IF AN INIT IS COMMANDED DURING A LOCK-UP, THE GPS STATUS DISPLAY WILL SHOW THE ASTERISK MOVE FROM THE CURRENT MODE TO INIT. THE MODE FIELD, HOWEVER, WILL RETAIN THE PREVIOUS VALUE. THIS MIXED DISPLAY WILL CONTINUE UNTIL THE LOCK-UP IS CLEARED. ONCE THE COMMAND LOCK-UP IS CLEARED, THE INIT WILL CONTINUE TO BE COMMANDED BY THE FSW.

IF A GPS FILTER RESTART IS COMMANDED DURING THE LOCK-UP, THE GPS STATUS DISPLAY WILL NOT SHOW AN '*' NEXT TO RESTART, INDICATING NO GPS FILTER RESTART IS IN PROGRESS.

IF A GPS ANNOUNCED RESET IS COMMANDED DURING THE LOCK-UP, THE COMMAND WILL NOT BE SENT TO THE GPS RECEIVER. THE CREW HAS NO INSIGHT INTO ANNOUNCED RESETS. THE GROUND CAN DETERMINE THAT AN ANNOUNCED RESET WAS MISSED BY LOOKING AT THE DOWNLIST AND COMPARING THE TIMES OF ANNOUNCED RESETS TO THE EXPECTED TIMES. ANNOUNCED RESETS ARE ON A TIME SCHEDULE THAT IS I-LOAD CONTROLLED.

(FOR OI-27 AND OI-28) - IF A GPS TO IMU RECONFIGURATION IS COMMANDED, THE ACTUAL GPS TO IMU RECONFIGURATION IN THE FSW WILL OCCUR INDEPENDENT OF WHETHER AN INDICATION CAN BE SENT TO THE GPS RECEIVER. THEREFORE, THE RECEIVER MAY NOT BE AWARE OF THIS IMU RECONFIGURATION. THE GPS STATUS DISPLAY WILL SHOW THE ACTUAL FSW ASSIGNMENTS.

FOR OI-29 AND SUBS, NO GPS TO IMU RECONFIGURATION IS POSSIBLE.

User Response:

THE RESPONSE DEPENDS ON THE COMMAND BEING ISSUED DURING THE LOCK-UP.

FOR AN INIT, THERE IS NO CREW INTERACTION REQUIRED. THE FSW WILL COMMAND THE INIT. SHOULD A LOCKOUT CONDITION PERSIST, SEE USER NOTE FOR DR DR110621

FOR A GPS FILTER RESTART, THE CREW WILL NEED TO RE-COMMAND THE FILTER RESTART AFTER THE APPROPRIATE NUMBER OF CYCLES HAS EXPIRED.

FOR A GPS ANNOUNCED RESET, NO ACTION IS NECESSARY.

(FOR OI-27 AND OI-28) - FOR A GPS TO IMU RECONFIGURATION, THE ACTUAL GPS TO IMU RECONFIGURATION WILL OCCUR INDEPENDENT OF THE INDICATION SENT TO THE GPS RECEIVER. NO USER INTERVENTION IS REQUIRED.

DR: 110640

POSSIBLE PASS GPS INITIALIZATION COMMAND LOCK-UP

Description:

ONCE AN INITIALIZATION COMMAND IS ISSUED IN PASS, THE VARIABLE INIT_STATE_SET IS SET TRUE AND IS NOT RESET UNTIL (1) THE MAGR RESPONDS, (2) A GPS BYPASS IS DETECTED, (3) THE GPS I/O CONTROL OPTION IS "1", (4) AN OPS TRANSITION IS PERFORMED, OR (5) THE GPS REFERENCE STATE PROCESSOR DECLARES A COMMAND TIME-OUT. WHILE INIT_STATE_SET IS TRUE, THE GPS SOP WILL COMMAND THE MAGR TO INIT MODE, AND NO OTHER MODE COMMANDS WILL BE HONORED.

IF THE MAGR DOES NOT RESPOND AFTER AN INIT COMMAND, IN OPS 9 OR MM 101 PRIOR TO NAV INIT, MANUAL INTERVENTION IS REQUIRED TO RECOVER THE COMMAND CAPABILITY SINCE THE GPS REFERENCE STATE PROCESSOR DOES NOT RUN DURING THIS TIME PERIOD.

User Response:

IF THE MAGR HAS NOT RESPONDED TO AN INIT COMMAND IN OPS-9 OR MM 101 PRIOR TO NAV INIT WITHIN 10 SECONDS, THE FOLLOWING ACTIONS MAY BE TAKEN TO BREAK THE MODE LOCK-UP. EITHER OPTION WILL WORK IN OPS 9, BUT OPTION A WILL NOT WORK IN OPS 1.

- A) CYCLE I/O CONTROL ON GPS STATUS DISPLAY
 1. CYCLE THE I/O CONTROL FROM 2 TO 0 TO 1 AND BACK TO 2.
- B) POWER CYCLE THE RECEIVER
 1. POWER OFF THE AFFECTED GPS RECEIVER.
 2. COMMAND A RECEIVER MODE OTHER THAN INIT OR THE GPS_MODE_CURRENT (AS INDICATED BY THE MODE FIELD ON THE GPS STATUS DISPLAY)
 3. POWER ON THE GPS RECEIVER.
 4. PERFORM AN I/O RESET.

ONCE THE LOCK-UP IS BROKEN, RECOMMAND THE INIT (IF REQUIRED).

SMS: B27863

DR: 110633

GPS STATE PROPAGATION ASSUMPTIONS NOT ALWAYS CORRECT

Description:

ASSUMPTIONS MADE IN THE BFS GPS NAVIGATION FUNCTIONS RELATING THE BUFFERED IMU TIMES AND GPS TIMES WILL LIKELY CAUSE THE GPS RESIDUALS TO EXCEED THEIR TOLERANCES IN THE LATTER PORTION OF MM 101 AND ALL OF MM 102, REGARDLESS OF WHETHER ONE OR THREE GPS RECEIVERS ARE INSTALLED. THE FIRST ASSUMPTION, WHICH IS ONLY VALID FOR BFS IN LISTEN MODE, IS THAT THE GPS TIME WILL BE VERY CLOSE TO THE BUFFERED IMU TIMES. WITH BFS IN LISTEN, THE GPS TIME WILL BE THE PASS AIDING TIME, WHICH MAY BE SKEWED FROM THE BUFFERED BFS IMU TIME. THE SECOND ASSUMPTION, APPLICABLE TO BOTH LISTEN AND ENGAGE MODES, IS THAT BUFFERED IMU TIMES WILL BE 0.16 SECONDS APART. IN BFS MM 101 & 102, THE IMU DATA IS DETERMINED AT 1.04 HZ AND THE BUFFERED IMU DATA IS SAVED AT 6.25 HZ, RESULTING IN IDENTICAL BUFFERED IMU TIMES FOR SIX CONSECUTIVE CYCLES. WHEN THE RESIDUALS EXCEED TOLERANCES, THE CREW WILL SEE A DOWN ARROW DISPLAYED FOR THE GPS RATIO STATUS INDICATOR ON

THE HORIZONTAL SITUATION DISPLAY. FOR OI-29 AND SUBSEQUENT FLIGHTS, THE CREW WILL ALSO SEE LARGE GPS RESIDUALS (GPS MINUS NAV) ON THE GPS STATUS DISPLAY.

THE GROUND WILL SEE LARGE RESIDUALS IN THE DOWNLISTED DATA.

User Response:

WHILE IN MM 101 AND 102, EXPECT A DOWN ARROW FOR THE GPS RATIO STATUS INDICATOR ON THE HORIZONTAL SITUATION DISPLAY AND LARGE GPS RESIDUAL VALUES ON THE GPS STATUS DISPLAY (FOR OI-29 & SUBS). GPS DATA SHOULD NOT BE INCORPORATED DURING THIS TIME.

GROUND SHOULD EXPECT LARGE GPS RESIDUALS IN MM 101 AND 102.

SMS: B20689
DR: 110650

STALE GPS SATELLITE NUMBER DISPLAYED MOMENTARILY

Description:

IF A GPS SATELLITE IS DESELECTED ON BFS SPEC 55 (GPS STATUS DISPLAY) (OI-30 ONLY) VIA ITEM 43+XX EXEC THROUGH ITEM 50+XX EXEC ("XX" REPRESENTS THE SATELLITE NUMBER), OR IF A GPS RECEIVER LOSES TRACK OF A GPS SATELLITE (OI-29 AND OI-30), THE SATELLITE NUMBER DISAPPEARS FROM THE 'SATELLITE TRKD' FIELD AS REQUIRED. HOWEVER, WHEN A NEW SATELLITE IS ACQUIRED TO REPLACE THE DESELECTED OR LOST SATELLITE, THE OLD SATELLITE NUMBER REAPPEARS FOR 960 MSEC BEFORE BEING REPLACED WITH THE NEWLY ACQUIRED SATELLITE NUMBER.

User Response:

NONE. THIS IS A TRANSIENT DISPLAY PROBLEM ONLY. THE CORRECT SATELLITE NUMBER IS DISPLAYED 960 MSEC LATER.

D: REFERENCES

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SECTION 21 TRAJECTORY CONTROL SENSOR

21.1 GENERAL

The Trajectory Control Sensor is a payload bay mounted unit that provides relative range, azimuth, elevation, and rate information on targets being approached by the Orbiter. The system's primary targets will be the MIR, the International Space Station and payloads which are equipped with optical retroreflectors (retros). The TCS unit generates the output data values by employing the use of two lasers. A continuous-wave (CW) laser diode modulated with three tones will be used to measure ranges from 1.5 meters (≈ 5 ft) to 400m (≈ 1300 ft) and a less accurate Pulsed laser will be used for ranges greater than 400 m up to at least 1500m (≈ 5000 ft). The laser beams are scanned using a three axis galvanometric beam scanning system (X, Y and Z axes) which provides for a 20 degree radius cone field of regard. The Z axis provides beam focusing which is used in part to provide automatic gain control to compensate for the dynamic range limitations between the near and far-field operations.

In the acquisition phase, the laser is scanned within the field of regard until a retroreflector is illuminated on the target vehicle. A quality retroreflector has the unique property that the incident and reflected laser beams are coaxial over about a 30 optical degree cone of angle. The laser energy returned by the retroreflector is detected and the range information for the near field CW operation is derived from the measured phase shift between the transmitted and received tones. This phase shift is due to the round trip travel time caused by the finite speed of light. Range rate is determined by back-differencing¹ the CW range data. The range data from the pulsed operation is derived by time of flight. A high speed counter is used to determine the elapsed time from when the leading edge of a pulse is transmitted and when it is received after reflection from a target vehicle. The time delay is due to the round trip travel time caused by the finite speed of light. Range rate in this mode is determined by back-differencing pulsed range data. The azimuth and elevation angles are determined by the position of the optical scanners at the time a retroreflector was encountered. The azimuth and elevation rates are determined by back-differencing of azimuth and elevation data.

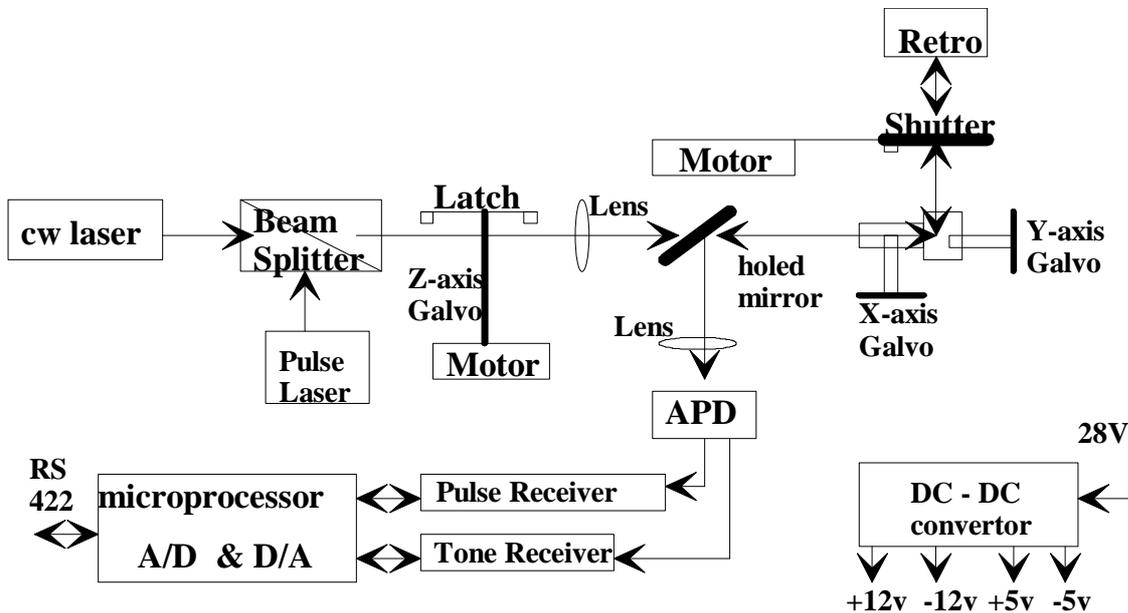
21.2 HARDWARE

The hardware description is divided into optics and electronics for both CW and pulse operation. Each TCS unit is approximately 450mm (18 inches) long by 225mm (9 inches) wide by 300mm (12 inches) tall and weighs 25Kg (55 pounds).

21.2.1 Optics

The optics are optimized for 850 +/- 10 nm operation. The horizontally polarized modulated CW laser beam first goes directly through a polarized beamsplitter. The horizontally polarized pulse laser beam is vertically polarized by a half-waveplate so that it turns 90 degrees in the beamsplitter and exits the beamsplitter aperture coaxial to the CW laser beam. Both beams then follow the same optical path. First they go through a quarter waveplate which circularly polarizes the beam in order to reduce back-reflections into the transmitting lasers. The beam then travels through the Z-axis galvanometer which is an electronically controlled 2x telescope used to

¹ Back-differencing: subtracting the current value from the previous value and dividing the difference by the elapsed time interval.



drawing courtesy of Alan Fox

Figure 21.1 TCS hardware interface diagram

collimate or diverge the laser beams to allow laser power level control at the target. Then the beams pass through the back of a holed mirror mounted at 45 degrees to the beams. From there the beams travel to the X then Y mirrors which electronically steer the beam to the target through a shutter aperture. The retroreflector on the target directs the laser beam directly back to the Y then X mirrors and the holed mirror. Some of the expanded returned signal is lost through the hole in the mirror. However, the majority of the beam is mirrored first to an 850 +/- 20 nm optical filter to reduce stray light (sun, floodlights, etc.) and then focused by a lens to an Avalanche Photodiode (APD) detector. In addition, a calibration mount is positioned near the shutter of the TCS to provide a temperature stable position and range reference for the electronics.

21.2.2 Electronics

The TCS is a scanning laser radar using both CW and pulsed rangefinders. The CW laser beam is modulated with 3 different related frequencies from the tone synthesizer in order to determine range to the target, by measuring the phase shift in the tone receiver between the transmitted and received signals. The tone synthesizer also generates the receiver detector mixing signals. The highest frequency modulation is divided into 512 detectable increments and provides a range resolution of 2.3 mm. The other 2 tones are divided into 256 increments and provide for a 152 meter unambiguous range measurement.

The pulse ranger's laser emits a series of 30 nanosecond-wide pulses. Each emitted pulse starts a digital electronic counter which is turned off by a returned laser pulse. The counter reading corresponds to the time delay between the pulses and therefore indicates the range to the

retroreflector. The pulse signal from the APD detector is preconditioned by a constant fraction discriminator circuit before reaching the high speed counter.

Scanning and divergence control of the laser beam is accomplished using a three-axis galvanometric scanning system. The galvos are analog devices which deflect in angle proportional to an input voltage. Mirrors are attached to the shafts of the X and Y galvos, providing a two dimensional scanning capability under computer control (azimuth and elevation). Both raster and random vectoring are supported by these devices. Position feedback voltages are used to derive the target bearing angles to a precision of about 0.05 degrees over the short term. Periodic angle calibrations against reference targets must be performed about every four minutes to compensate for long term temperature drift.

21.3 SOFTWARE

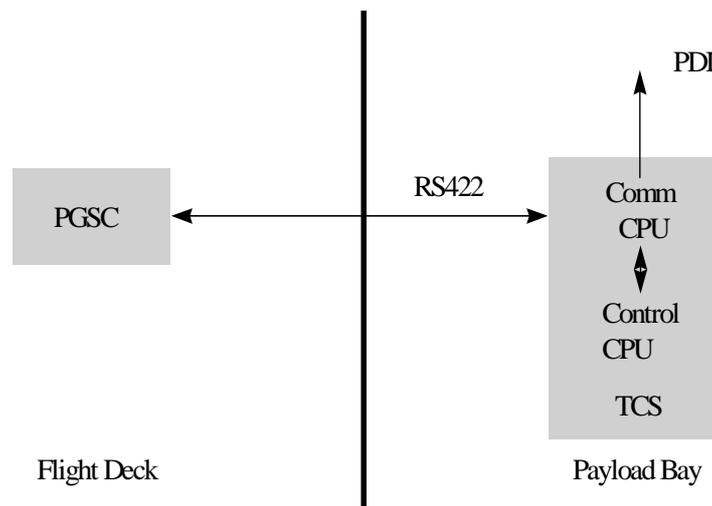


Figure 21.2 TCS software interface diagram

The TCS units are controlled by three separate programs. The Command and Display (CAD) program is a Windows application, executing in a Payload and General Support Computer (PGSC). This program is used by the User to command the TCS unit and display the current engineering unit values from the sensor. The remaining two programs reside in the sensor itself. The Control program is responsible for the testing and calibrating of the sensor subsystems, the implementation of run-time commands from the User, the updating of data parameters and the reporting of key sensor activities. The exchanging of the data and command information is performed by employing the third TCS program. The Communications program performs two major functions. The first function is to act as the buffer between the User (via the CAD) and the Control program by providing for the orderly transfer of command information from the CAD and for the orderly formatting and return of data and status information from the Control program to the User (CAD). The second major function associated with the Communications program is to format and transmit the TCS Payload Data Interleaver (PDI) data frames for insertion into the Orbiter telemetry stream.

21.4 FLIGHT CONFIGURATION

The operations of the TCS system will be dependent on the type of activity that is currently being performed. Generally the operation of the TCS firmware and hardware will be conducted with a minimum of user interaction. The following paragraphs will expand on the types of operations performed by the sensor and the level of user activity required for supporting the processes.

21.4.1 Both Lasers Operational

| <u>Range</u> | <u>Controlling Laser/Rational</u> |
|----------------------------|---|
| < 1.5m(5ft) | No tracking/CW receiver becomes saturated by return signal |
| 1.5m(5ft)-100m(328ft) | CW laser |
| 100m(328ft)-400m(1312ft) | Both lasers |
| 400m(1312ft)-1524m(5000ft) | Pulse laser |
| > 1524m(5000ft) | Pulsed laser/maximum distance tested on the ground was 1776m(5825ft). |

21.4.2 CW Laser Only (Pulsed Laser Failed)

| <u>Range</u> | <u>Controlling Laser/Rational</u> |
|------------------------|--|
| < 1.5m(5ft) | No tracking/CW receiver becomes saturated by return signal |
| 1.5m(5ft)-400m(1312ft) | CW laser |
| > 400m(1312ft) | CW laser/maximum distance tracked in space with the retroreflectors in the field of view was 1232 ft |

21.4.3 Pulsed Laser Only (CW Laser Failed)

| <u>Range</u> | <u>Controlling Laser/Rational</u> |
|-------------------------|--|
| < 20m(66ft) | No tracking/pulsed receiver is electronically blanked within this range interval |
| 20m(66ft)-1524m(5000ft) | Pulsed laser |
| > 1524m(5000ft) | Pulsed laser/maximum distance tested on the ground was 1776m(5825ft) |

21.4.4 Pulse Laser To CW Laser Handoff (Docking)

The normal operational mode for the TCS unit is to acquire the target to a minimum distance of 1524m (5000ft) with the pulsed laser. The system will continue to track the retroreflectors exclusively with the pulsed laser until the range is less than 400m (1312ft). At this distance, the TCS software will begin the process of handing off operational control to the CW laser. This

action is preferred due to the better accuracy of the CW laser. The handoff occurs in the background with no intervention by - or indication to - the User. Completion of the handoff must follow a series of tracking performance trials to ensure the CW system can continue to track the retroreflectors without a loss of tracking. This feature is assured by the number of trials that must be successfully completed. The number of trials is a function of the current range. The greater the range the greater the number of successful CW updates that are required. The tracking trials are grouped into sets of 2 update tests per trial. The following general requirement must be met for a successful handoff:

| <u>Range</u> | <u>Requirement</u> |
|--------------------------|---------------------------------|
| 400m(1312ft)-304m(997ft) | 3 sets of 2 updates (ie, 2,2,2) |
| 304m(997ft)-152m(499ft) | 2 sets of 2 updates (ie, 2,2) |
| 152m(499ft)-20m(66ft) | 1 set of 2 updates |
| < 20m(66ft) | 0 sets of 2 (immediate handoff) |

A counter counts the number of successful trials completed. If a failure occurs, the counter is decremented to a minimum of zero, if a success occurs, it is incremented. When the required number of successes has accrued, the CW ranger begins tracking on the next update, completing the handoff process.

If a loss of tracking with the CW laser occurs and the range is less than 100m (328ft), the TCS system will attempt to reacquire the retroreflector exclusively with the CW laser. If a loss of tracking with the CW laser occurs and the range is greater than 100m (328ft) and less than 400m (1312ft), the TCS system will attempt to reacquire the retroreflector with the CW laser. If these attempts are unsuccessful, the unit will promote to the pulsed laser. Following a successful acquisition with the pulsed laser, the unit will again attempt to handoff to the CW laser using the handoff criteria defined earlier in this section.

21.4.5 CW Laser to Pulsed Laser Handoff (Undocking)

The CW to Pulsed laser handoff will be slightly modified from the more structured Pulsed to CW handoff. The nominal method will be to continue tracking the retroreflectors with the CW laser until a loss of tracking occurs. The unit will attempt to reacquire the target with the CW laser. If an immediate reacquisition is not successfully completed, the TCS will begin a full scan in an attempt to resume tracking. If the last range was less than 100m(328ft), the CW laser will be used for the full scan. If the last range was greater than 100m(328ft) the TCS system may begin the full scan with the pulsed laser. After the acquisition is successful with the pulsed laser and if the range is less than 400m(1312ft), the TCS system will again attempt to handoff to the CW laser, according to the criteria defined in the previous section. If the range is greater than 400m(1312ft), the unit will continue to use the pulsed laser exclusively.

21.5 CAD APPLICATION

21.5.1 Platform

The Command and Display (CAD) program is a Windows 3.x application requiring a Windows (3.x or higher) operating system for execution. The application adheres to the standard Windows Graphical User Interface (GUI) style. Some data presentation styles that are CAD-specific include:

21.5.1.1 Reverse background

Reversing the background for the displaying of dynamic fields implies that the data value currently being displayed is in a “bad” status. This condition may be caused by the lack of a valid data sample from the sensor within a reasonable period of time (currently ten seconds) or the failure of some hardware subsystem during a reset or self-test action. The presentation of the data in this manner is solely to assist the User in the identification of bad data values. **(Note: the reverse background feature does not work for this CAD release on the Win95 platform)**.

21.5.1.2 Asterisks

Data displayed between asterisks implies that the control function that is generally applied to the value has been overridden (see the **Override** menu function) by the operator. This override activity was performed by the operator during the current session.

21.5.1.3 Failed Status

The inability of a hardware subsystem to complete a reset, self-test or a required calibration action results in the display of a failed status value. This field is displayed in a reversed background mode. The absence of the hardware subsystem will also cause a failed status to be displayed.

21.5.2 Program Execution:

The CAD application is activated via a double click on the *cad.exe* file in the file manager or by a double click on the TCS icon in the Program Manager window. Upon activation, the application initializes the monitor with the TCS OPS window and displays a timed popup window acknowledging the software developers that have worked on the TCS task. This window will be removed automatically after five seconds or may be manually removed by the User at any time by clicking the **OK** button. If the AutoSeed parameter of the initialization file is enabled, then the Rendezvous and Proximity Operations Program (RPOP) must already be running, else a popup window occurs informing the User that the Automatic Acquisition feature cannot be performed (see 21.5.x).

The CAD main window is divided into five display areas needed to interface with the User. The five display areas include the Title Bar, the Menu, the Tool Bar, the View, and the Status Bar.

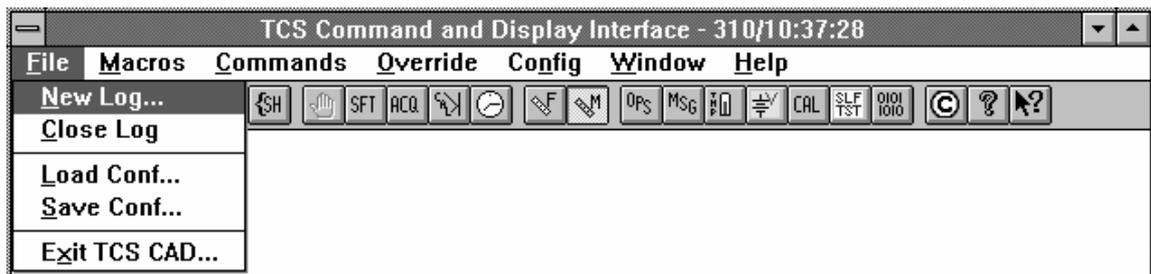
21.5.2.1 Title Bar

The Title Bar contains the CAD application name "**TCS Command and Display Interface**" and the current internal computer time value in *ddd/hh:mm:ss* format. The time value will change reflecting the time value currently operating on the computer and will be displayed in the same *ddd/hh:mm:ss* format.

21.5.2.2 The Menu

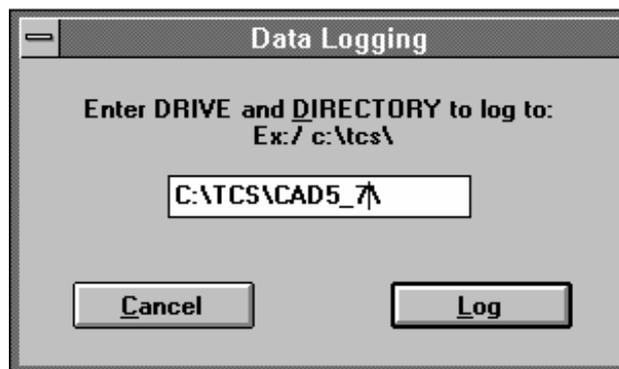
The Menu is the primary user interface with the application. The available top level selections include **F**ile, **M**acros, **C**ommands, **O**verride, **C**onfig, **W**indow and **H**elp. The exact definition of the different User selection options will be expanded in the following paragraphs. The presentation of this information in this document will be by the main menu, followed by all the submenu selections, proceeding from left to right.

21.5.2.2.1 File



File commands are used to control TCS data logging and manage CAD configurations. The available commands are:

New Log...



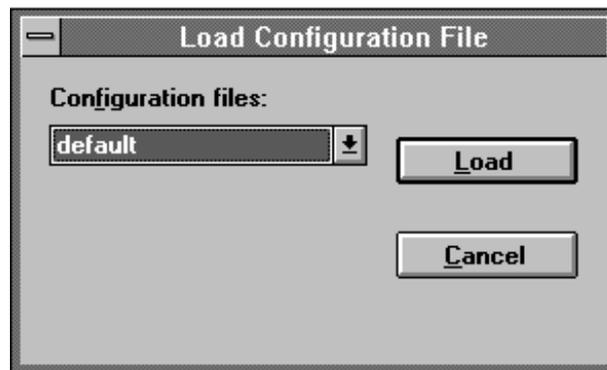
This command presents the Data Logging dialog box that allows the User the capability of opening a new log file with user selected recording parameters. Selection of the output file name, the pathname, and a **C**ancel and **L**og button is available in the box. Log files are defaulted to the same directory, where the executable file is resident. The filing sequence begins with the

TCS000.LOG file name. Additional files will increment the 000 portion of the filename by one. Each data file will contain approximately 50,000 bytes of ASCII formatted data according to different data strings defined in the TCS Software Interface Control Document. If data files already exist in the current directory, the first file number from (TCS000 to TCS999) not found by CAD will become the default log file name.

Close Log

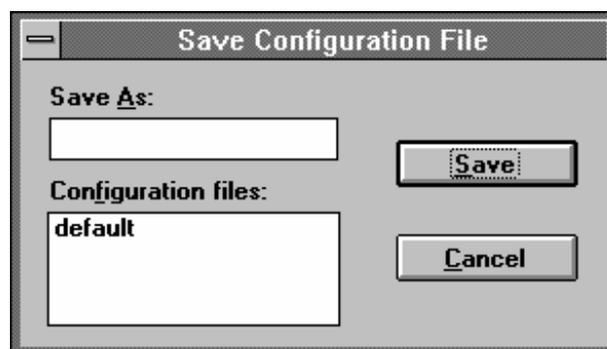
This command closes of the current TCS log file and stops data logging.

Load Conf...



This command presents the Load Configuration File dialog box that allows the User to define what configuration to use in during CAD initialization. All configurations are stored in the file *cad.ini*. The **default** configuration is used upon bootup or if no User input is defined. A selection to **L**oad or **C**ancel the operation is permitted via a button entry.

Save Conf...



This command presents the Save Configuration File dialog box that allows the User the capability to save the current CAD configuration into *cad.ini*. A selection to **S**ave the current configuration parameters will result in the file *cad.ini* being updated with the new parameters. A selection to **C**ancel the operation is permitted via a button entry and will not alter the content of the current *cad.ini* file.

Exit TCS CAD...

This command presents the User the Exit TCS CAD dialog box and an inquiry concerning the exiting of the CAD application or the cancellation of the exit request. This feature is presented to the User via buttons. The selection of the **OK** button closes any open log files and then exits the application, closing all open windows (including Help windows). The selection of the **Cancel** button permits the application to continue execution.

21.5.2.2.2 Macros



Macro commands are designed to reduce the amount of user interfacing activity by automatically executing several TCS commands to the sensor. The execution of the command sequences is controlled by time delay and data strings reflecting the current status of the sensor. Three macro commands are supported by CAD. The **INITIALIZATION** and **ACQUISITION** macros must be issued while the TCS system is in the Standby mode. The **SHUTDOWN** macro may be issued at any time. The execution of the macros may require additional user input via dialog boxes.

INITIALIZATION

The **INITIALIZATION** macro performs the activities necessary to download to the sensor information defining the initial levels of processing parameters. The CAD requires the sensor be in Standby mode. The automatic issuing of the TCS sensor commands necessary to perform the series of initialization commands generally requires responses from the TCS unit upon completion of the previous TCS command. The following actions will be performed via the initialization macro:

- 1) Set the TCS clock to the current PGSC time
- 2) Send the TCS temperature and voltage limits as defined in the *cad.ini* file
- 3) Send the default TCS initialization data as defined in the *cad.ini* file
- 4) Verifies that the Z-latch is unlocked

ACQUISITION

The **ACQUISITION** macro performs the activities necessary to begin the process of acquiring a target retroreflector. The sensor must be in Standby mode for the macro activation. Automatic activation of this macro may be requested by the User. This action will be expanded later in this document under the **Config>Automatic** menu option. Upon activation of this macro the CAD performs the following activities:

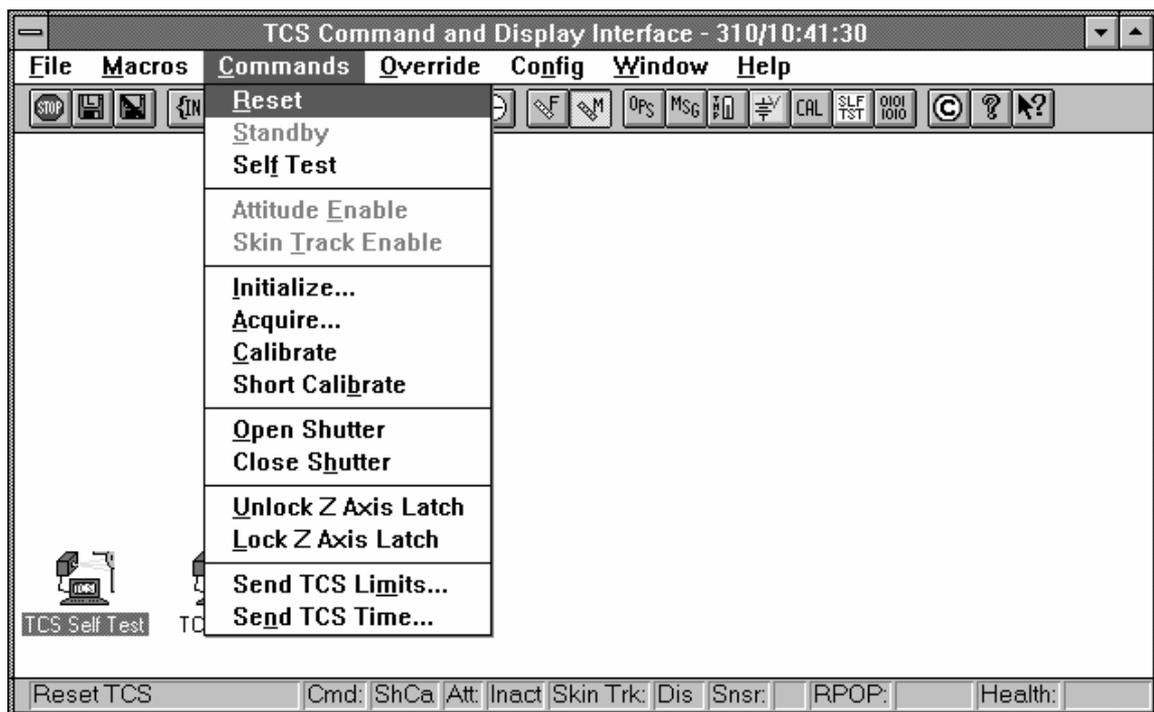
- 1) Opens the TCS shutter (takes approximately 22 seconds from closed position)
- 2) Displays a popup window requesting an estimated target range and bearing angles
- 3) Issues an Acquire command to the sensor to search for a target retroreflector(s)

SHUTDOWN

The **SHUTDOWN** macro performs the actions necessary to terminate a TCS session. The macro may be activated at any time or in any mode during TCS operations. Upon activation the CADS performs the following activities:

- 1) Commands the TCS to Standby mode (if tracking, a popup appears)
- 2) Closes the TCS shutter (which takes approximately 22 seconds from open position)

21.5.2.2.3 Commands



Reset

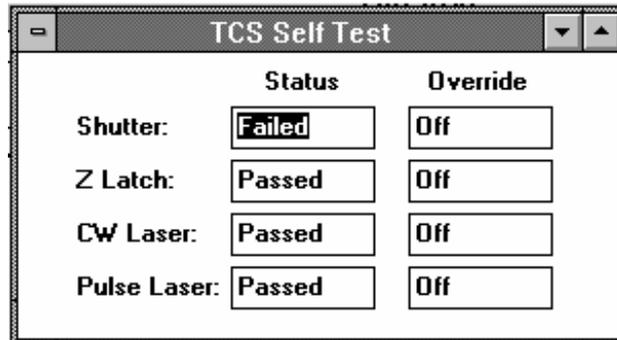
The **Reset** command will cause the complete resetting of the TCS, affecting current values of any displayed status flags. Appropriate messages are displayed to the User.

Standby

The **Standby** command will terminate the current activities that the TCS unit is performing. In the event that the sensor is currently tracking a retroreflector, a popup window will be displayed

to the User confirming the request to break tracking. Upon completion of the command, the mode in the TCS OPS window will reflect the **Stdby** status and most of the commands available through the main menu will become available.

Self Test



The **Self Test** command requests the sensor to perform a Built-in Test (BIT) of the major subsystems in the TCS unit. The performance of the tests result in the display of a passed/good or failed status for the CW Laser, the Pulsed Laser, the Shutter and the Z-Latch. The results of the tests are reflected in the TCS OPS window and in the Self Test window that pops up for the User to acknowledge. A "**TCS Self Test completed**" message in the OPS window informs the User that the command has completed execution. This command may be issued only while the unit is in the Standby mode.

Attitude Enable

The **Attitude Enable** command will allow the TCS to begin three retro acquisition and tracking, whenever the target is less than a required range (target dependent). This tracking allows the TCS to estimate relative attitude (roll, yaw, and pitch angles) and attitude rates. Per program agreement, this command option is currently disabled, as well as the attitude enable flag in the Initialization window.

Skin Track Enable

The **Skin Track Enable** command will direct the TCS unit to attempt to acquire and track a target that is not equipped with retros (using the pulsed laser only). Per program agreement, this command option is currently disabled.

Initialize...

The **Initialize** command is used to download initialization/configuration parameters to the TCS as defined in the *cad.ini* file. The **Initialize** command must be successfully completed before the TCS unit will allow itself to enter the Active mode (scanning and tracking). Upon selection in the Standby mode, the CAD will display the Initialization Parameters popup window, containing the current parameter values. The User may change the existing values prior to the implementation of the command. The transmitting of the parameters to the sensor firmware will be performed via a **Send** button option. Once selected, the parameter string will be sent to the TCS. An “**TCS Initialization completed**” message will be displayed in the TCS OPS Message Window informing the User that the TCS unit has been successfully initialized. A second button is available to **Cancel** the sending of the command. Once selected, any changes to the parameter values entered by the User will be lost, and must be re-entered.

Acquire...

The **Aquire** command is used to begin the process of searching for a target, by commanding the TCS to the Active mode. Upon activation in any mode, the CAD will display the popup window containing the current seed (best guess) values for the range distance, azimuth angle and elevation angle. The User may change the existing values prior to executing the command. The transmitting of the parameters to the sensor firmware is performed via the **Send** button option. Once the command is recognized, the TCS will enter/restart the Active mode, and the CAD will change the displayed mode to **Acq** (for Acquire). A second button is available to the User to **Cancel** the transmission of the command. If selected by the User, any changes to the seed range distance, azimuth angle or elevation angle will be lost.

Calibrate

The **Calibrate** command may be executed in any TCS mode. Once activated, this option forces the TCS unit to perform a long-cycle (full) calibration of the hardware. Upon completion of the command (approximately 5 seconds), the status of the calibration results and a message “**Calibration completed**” are displayed in the OPS Window. The current values of the updated calibration parameters are displayed in the Calibration and Temperatures and Voltages windows. The execution of this command occurs automatically in the background every time the TCS unit resets, or is commanded to acquire, and at about 4 minute intervals, while the TCS unit has power applied.

Short Calibrate

The **Short Calibrate** command may be executed in any TCS mode. Once activated, this option forces the TCS unit to measure and update the system temperatures and power supply voltages. Upon completion of the command, a “**Calibration completed**” message will be displayed in the OPS Window. The data values associated with the short calibration may be monitored in the TCS Temperatures and TCS Voltages windows. The execution of this command occurs automatically in the background every time the TCS unit resets, or is commanded to acquire, and at 30 second intervals, while the TCS unit has power applied.

Open Shutter

The TCS **Open Shutter** command is available in the Standby mode. Upon execution, the TCS unit will open the protective optical cover. Upon completion of the command (approximately 22 seconds) the status of the shutter is changed to **Opened** in the TCS OPS window. The shutter will be opened automatically as part of the **ACQUISITION** macro described above.

Close Shutter

The TCS **Close Shutter** command is available in Standby mode. Upon execution, the TCS unit will close the protective optical cover. Upon completion of the command (approximately 22 seconds) the status of the shutter is changed to **Closed** in the TCS OPS window. The shutter will be closed automatically as part of the **SHUTDOWN** macro described above.

Unlock Z Axis Latch

The **Unlock Z Axis Latch** command is available in the Standby mode. Upon execution, the command causes the TCS unit to unlock the latch mechanism on the Z-axis telephoto assembly. Upon completion of the command the status of the Z-latch is changed to **Unlock** in the TCS OPS Window. The Z-axis latch will be unlocked automatically when the sensor is powered up as part of the TCS self-test.

Lock Z Axis Latch

The **Lock Z Axis Latch** command is available in the Standby mode. Upon execution, the command causes the TCS unit to lock the latch mechanism on the Z-axis telephoto assembly. Upon completion of the command, the status of the Z-latch is changed to **Locked** in the TCS OPS Window.

Send TCS Limits...

The image shows a 'Limits Data' popup window with two main sections: 'Temperature (deg C)' and 'Voltage'. The temperature section has three columns: 'Low Alert', 'Alert', and 'High Autosafe'. The voltage section has two columns: 'Low' and 'High'. There are 'Send' and 'Cancel' buttons on the right side of the window.

| Limits Data | | | |
|---------------------|-----------|-------|---------------|
| Temperature (deg C) | | | |
| | Low Alert | Alert | High Autosafe |
| CW Laser: | -40. | 45. | 50. |
| APD: | -40. | 65. | 70. |
| CPU: | -40. | 80. | 85. |
| DC Power: | -40. | 80. | 85. |
| Galvos: | -40. | 65. | 70. |

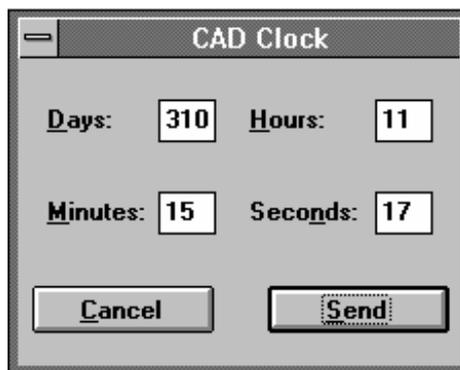
| Voltage | | |
|-----------|-------|-------|
| | Low | High |
| 5 Volt: | 4.75 | 5.25 |
| 5.5 Volt: | 5. | 6. |
| 12 Volt: | 11. | 12.75 |
| 15 Volt: | 14.25 | 15.5 |

Buttons: Send, Cancel

The **Send TCS Limits** command permit the User to change and send to the TCS unit the limit checking values affecting the unit's temperatures and voltages. Upon activation in Standby mode, the Limits Data popup window will appear on the screen. The User may select to change

the values for the low alert, high alert and high autosafe² temperatures. Upon completion of the parameter changes, the transmitting of the new parameters occurs by selecting the **S**end button option. Once sent, a message will be displayed in the TCS OPS window informing the User that “**New limits have been set**” in the TCS unit. A second button is available to **C**ancel the sending of the command. Once selected, any changes entered by the User to the limit values will be lost. The setting of the TCS limits occurs automatically as part of the **I**NITIALIZATION macro described above

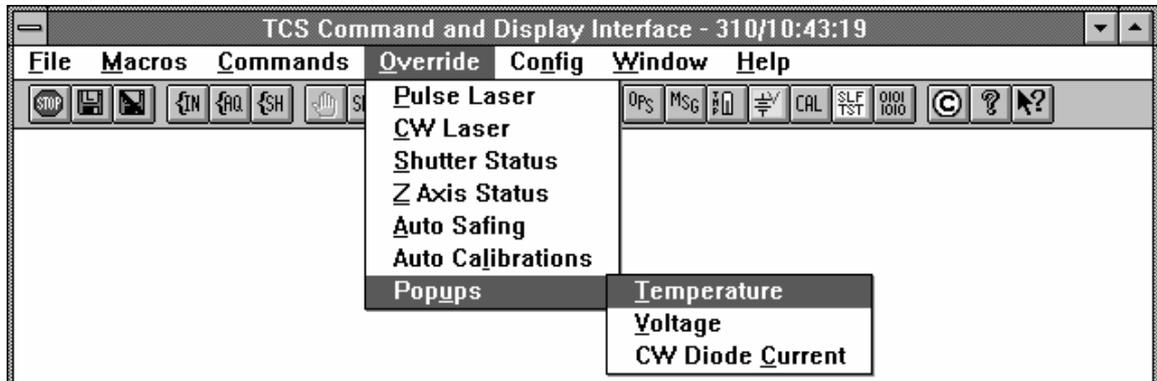
Send TCS Time...



The **S**end TCS Time command permits the User to change the TCS unit time to any value desired. Once the command is activated in the Standby mode, the CAD Clock popup window will appear on the screen with the current local (PGSC) system time values displayed. The User may elect to change the current day, hour, minute and second values before transmitting the data to the TCS by selecting the **S**end button option. Once selected, the time values will be sent to the TCS, resetting its time base. A message will be displayed in the TCS OPS Window that the “**TCS clock has been set**”. The time base of the CAD will also be automatically reset by this command. A second button is available to **C**ancel the resetting of the time. Once selected by the User, any changes to the day, hour, minute and second values in the display will be lost. Generally the time value is set upon activation of the sensor in the **I**NITIALIZATION macro to the time value in the PGSC. The User then performs a manual operation to reset the time to the Current Mission Elapsed Time (MET).

² Autosafe: when this temperature level is exceeded for over a minute, TCS will shut down the overheating subsystem and enter Autosafe mode (a subset of the Standby mode).

21.5.2.2.4 QVERRIDE



The **Override** command permits the User to change/alternate the current operational status of TCS subsystems, which are originally determined by TCS during the self-test. Subsystems with an override enabled will be identified on the TCS OPS Window display with a double asterisk “**” before and after the subsystem/option status. An enabled override status in the menu will be identified with a check mark to the left of the menu option.

The override option in the menu is enabled by selecting and clicking on the **Override>xxxxxxx** subsystem. Upon activation, the status on the screen will be flagged with the double asterisk before and after the status and the menu item with a check mark to the left of the item. The message “**Override enabled**” will be displayed in the TCS OPS window. Reselecting an overridden subsystem with the override menu option will cancel the original override. Upon reselection, the CAD will display the status of the subsystem without the double asterisk and the check mark to the left of the menu will be removed. The message “**Override disabled**” will appear in the TCS OPS window.

The principle use for this feature is to force the TCS system to continue to use a subsystem that has exceeded the normal safeguarding guidelines. This situation may exist when the current mission activity would be impaired if the sensor failed. Continued forced use of a failed subsystem could permanently damage the sensor.

Pulse Laser

The overriding of the pulsed laser forces a **Failed** sensor to become ****Avail**** or an **Avail** or **Active** sensor to become ****Failed****. If the current state is **Active** and TCS is tracking, a popup will occur requesting confirmation that the User wants to continue with the **Override** command.

CW Laser

The overriding of the CW laser forces a **Failed** sensor to become ****Avail**** or an **Avail** or **Active** sensor to become ****Failed****. If the current state is **Active** and TCS is tracking, a popup will occur requesting confirmation that the User wants to continue with the **Override** command.

Shutter Status

The overriding of the shutter status forces the open/closed microswitch status or the erroneous transit/dilemma states to be displayed as "***override***". A popup will occur requesting confirmation that the User wants to continue with the **Override** command.

If the TCS is going to be used for target acquisition/tracking imminently, a visual inspection should be made (if possible) to insure the laser has a chance to clearly exit the sensor. Obviously, a partially occluded shutter aperture will adversely affect TCS performance.

Z Axis Status

The overriding of the Z axis status causes the lock/unlock microswitch status or the erroneous transit/dilemma state to be displayed as "***override***". A popup will occur requesting confirmation that the User wants to continue with the **Override** command. There is currently no method to manually status the Z-latch after the subsystem has been overridden.

Auto Safing

The overriding of the autosafing features prevents the TCS unit from automatically shutting down TCS subsystems due to autosafe temperature limit exceedences (hot temps). Prolonged used of the sensor under these temperature extremes could damage the sensor.

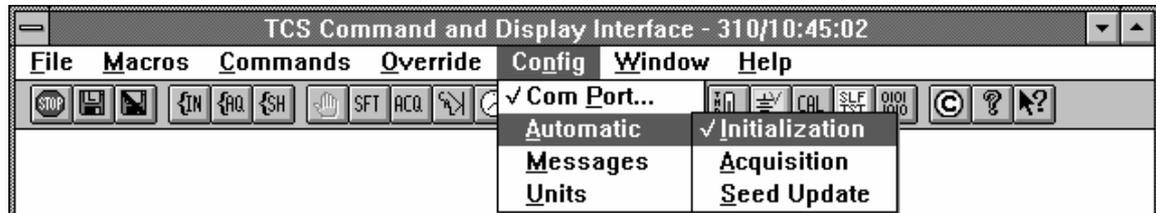
Auto Calibrations

The overriding of the auto-calibrations inhibits the performance of the long calibrations performed automatically about every 4 minutes. The execution of the short calibration cycle (temperature and voltage updates) continues to be performed every 30 seconds. Generally this option would be implemented when the loss of sensor tracking (particularly during terminal docking situations) would be detrimental to the mission, since the long calibration cycle requires that the laser be steered off the target for about 5 seconds. The prevention of the long calibrations may reduce the accuracy of the sensor due primarily to electronic drift associated with changing temperatures.

Popups

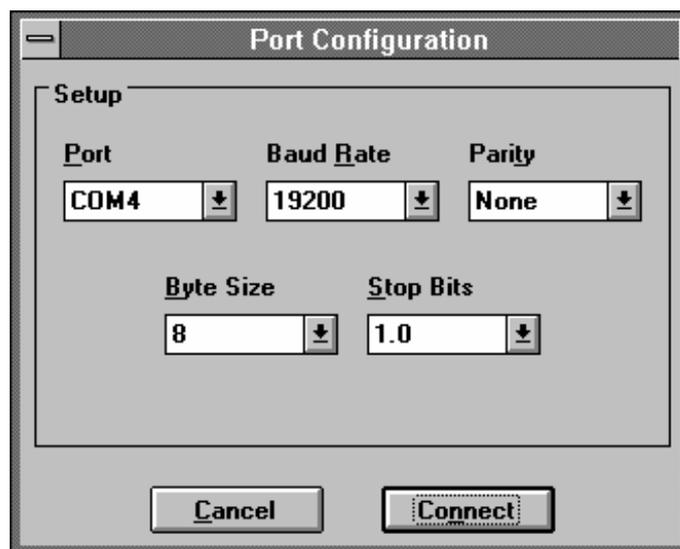
The overriding of the popup windows suppresses the displays associated with out of limit conditions for temperatures, voltages and the CW Diode Current. The selection of this menu results in the displaying of a submenu with 3 selectable items. The selection of any of the override conditions (**Temperature, Voltage or CW Diode Current**) is identified by the presence of a checkmark to the left of the selected item(s). The overriding of this option does **NOT** inhibit the automatic shutdown process due to exceeded autosafe temperature limits (See **Override>AutoSafing**).

21.5.2.2.5 Config



The **Config** menu permits the User to change the normal operational environment affecting the CAD application. Permitted changes include the selection of the RS422 communications port, the automatic execution of commanding activities, the complexity of informational messages and the type of engineering units for the range/range rate display. The defaulting of all the options in the **Config** menu are contained in the *cad.ini* file, delivered as part of the software load.

Com Port...



The selection of the **Com Port** submenu item during any TCS mode, results in the presentation of the Port Configuration popup window defining the current port assignment, baud rate, parity and number of bits. Upon presentation, the User may modify the values and via a **Connect** button option, connect to the port using the changed information. A **Cancel** button is also available to exit the option. Changes made to the communications port parameters prior to using the **Cancel** button option will be lost. Generally the Orbiter TCS system uses COM2, at a 19200 baud rate with 8 data bits, no parity and 1 stop bit (8N1). The Shuttle Mission Simulator (SMS) uses COM2, at a 9600 baud rate with 8 data bits, no parity and 1 stop bit (8N1). While generally stable, the possibility exists for changes to these values depending on the mission and upgrades to the SMS facility.

Automatic

The Automatic option permits the User to execute common TCS actions independent of individual user commands. Upon selection, the popup menu displays a submenu defining three automated functions. The Initialization, Acquisition and Seed Update features are supported. The presence of a check mark to the left of the submenu option implies that the particular feature is currently enabled. The initial settings for the 3 options are defined in the *cad.ini* file. The User may manually request the enabling or disabling of the automatic features by selecting the submenu item and clicking on the desired option. The selection of a submenu item with the checkmark forces a deselection of the item and the removal of the check mark.

The enabling of the Initialization option, forces the TCS to execute the INITIALIZATION macro automatically after the receiving the “TCS is performing a RESET” and the “TCS is entering STANDBY mode” messages from the sensor. The enabling of the Seed Update option requires that the RPOP application be active and providing initial seed values to the CAD application. The absence of the RPOP application will result in the automatic disabling of this feature. The enabling of the Acquisition option, will result in the sensor beginning the acquisition process automatically when the seed value provided by RPOP is less than the user defined start range for this feature. The Initialization and Seed Update options may be enabled in the *cad.ini* file using an editor (flight files are already configured). However, **the Acquisition feature must be manually activated by the User after the CAD is active.** The selection of this option calls up the Automatic Acquisition dialog box, allowing the User to configure/edit the parameters. The Maximum Range parameter determines when the sensor will automatically switch from Standby to Acquisition mode based upon the RPOP seed range values, and **it is recommended that this is the only parameter that the User edits.** The User then selects the OK button to enable the ACQUISITION macro and the modified parameters. The selection of the Cancel button, will disable the auto ACQUISITION macro and the user inputs to the parameters will be lost.

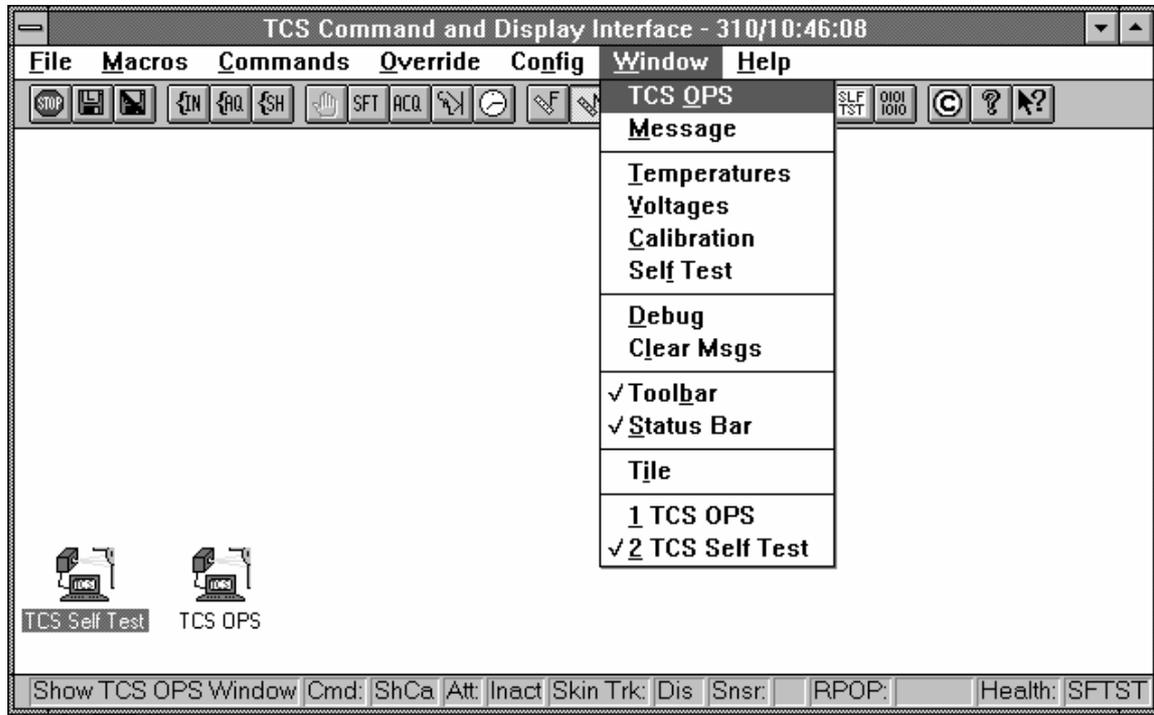
Messages

The Messages option allows the setting of the message display capability to Filtered or Unfiltered mode. The default setting in the *cad.ini* file is filtered. This mode reduces the complexity of the messages from the TCS unit and the CAD application. Generally, unfiltered mode is used during TCS performance/acceptance testing by the TCS engineering staff.

Units

The Units option allows the User to select English (feet and feet/second) or Metric (meters and meters/second) as the engineering units presentation on the TCS display windows. The selection of this menu results in the presentation of the English/Metric selection. A check mark to the left of the menu item identifies the current presentation mode. The selection of the submenu item is mutually exclusive of the remaining item. The *cad.ini* file sets the default setting for the engineering units to English (feet and feet/second).

21.5.2.2.6 Window



This menu is used to call up any of the different display windows defined for the CAD task. The selection of the window causes the display of the selected item. Selection of the toolbar and status bar causes the presence or removal of the individual display area. The presence of the display item is defined by a checkmark next to the name.

Currently defined display windows/options include:

| | |
|----------------------------|--------------------------------------|
| <u>TCS OPS</u> | operations window (default window) |
| <u>M</u>essage | message window |
| <u>T</u>emperatures | temperature window |
| <u>V</u>oltages | voltage window |
| <u>C</u>alibration | calibration window |
| <u>S</u>elf Test | self test window |
| <u>D</u>ebug | debug/raw packet window |
| <u>C</u>lear Msgs | clears all display windows |
| <u>T</u>oolbar | toggles display of toolbar/bitmap |
| <u>S</u>tatus Bar | toggles display of the status bar |
| <u>T</u>ile | tiles all open non-minimized windows |

21.5.2.2.7 Help



The help option permits the User to get additional information concerning the CAD application by calling up the online help utility.

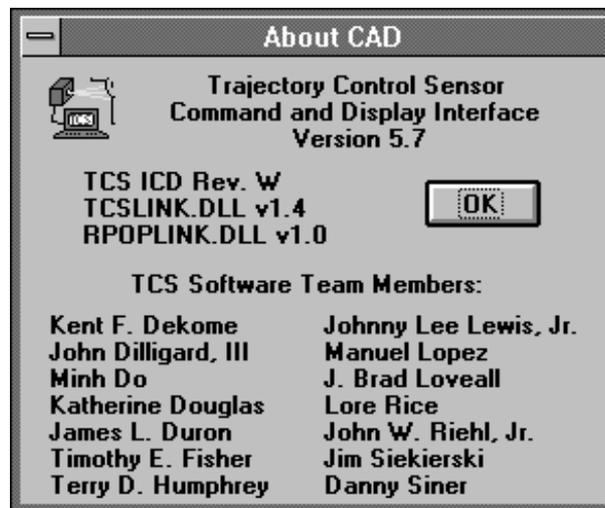
Index

This option allows the selection and presentation of help information of the major menu items according to windows procedures.

Using Help

This feature presents information on how to use the help facility in accordance with windows procedures.

About TCS CAD...



The menu presents the User some current information about the CAD program, including the release versions of the application, the current version of the ICD, the current versions of the dynamic link libraries and the names of individuals that have participated in the software development and management effort.

21.5.2.3 The Toolbar



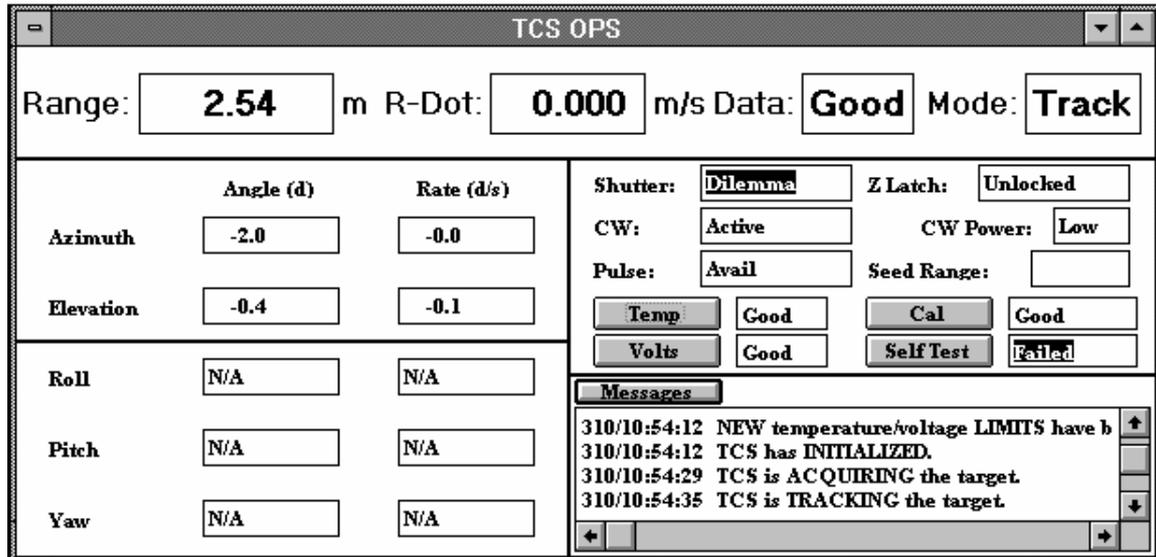
The Toolbar is the secondary Command and Display application interface with the User. This window is directly under the main menu in the main window. This window contains a subset of the different user-selectable options available via the main menu. For more information on the command see the main menu definition of the equivalent command. The icons will be described from left to right, just as they appear in the Toolbar window. The gaps in the placement of the icons indicates a change in their general function. This gap will be demonstrated in the definition of the icons by the presence of a blank line between the icon definitions.

| ICON | MAIN MENU EQUIVALENT | DESCRIPTION |
|-----------------------|-----------------------------------|---|
| Stop | File>Exit TCS CAD... | Exit the CAD application |
| New Log | File>New Log... | Define and open TCSxxx.LOG file |
| Close Log | File>Close Log | Closes the current TCSxxx.LOG file |
| Initialization | Macro>Initialization | Execute the INITIALIZATION macro |
| Acquisition | Macro>Acquisition | Execute the ACQUISITION macro |
| Shutdown | Macro>Shutdown | Execute the SHUTDOWN macro |
| Standby | Command>Standby | Issue a Standby command |
| Self Test | Command>SelfTest | Issue a Self Test command |
| Acquire | Command>Acquire | Issue an Acquire command |
| Calibration | Command>Calibration | Issue a Calibrate command |
| Set Time | Command>Set Time | Issue a Set TCS Time command |
| English | Config>Units>English | Display data in English units (feet) |
| Metric | Config>Units>Metric | Display data in metric units (meters) |
| OPS | Window>TCS OPS | Display TCS operations data window |
| MSG | Window>Message | Display Messages window |
| TEMP | Window>Temperatures | Display TCS Temperatures window |
| VOLT | Window>Voltages | Display TCS Voltages window |
| CAL | Window>Calibration | Display TCS Calibration window |
| SLFTST | Window>Self Test | Display TCS Self Test window |
| 01011010 | Window>Debug | Display Debug and Raw Debug windows |
| About CAD | Help>About TCS CAD... | Display information about the CAD version |
| CAD Help | Help>Index | Calls up the CAD Help utility |
| Context Help | Help>Index | Calls up context-sensitive Help |

21.5.2.4 The View

The CAD view will contain different child windows used to present the TCS data to the User. Child windows in this area may be moved and resized within the main CAD window.

21.5.2.4.1 TCS OPS

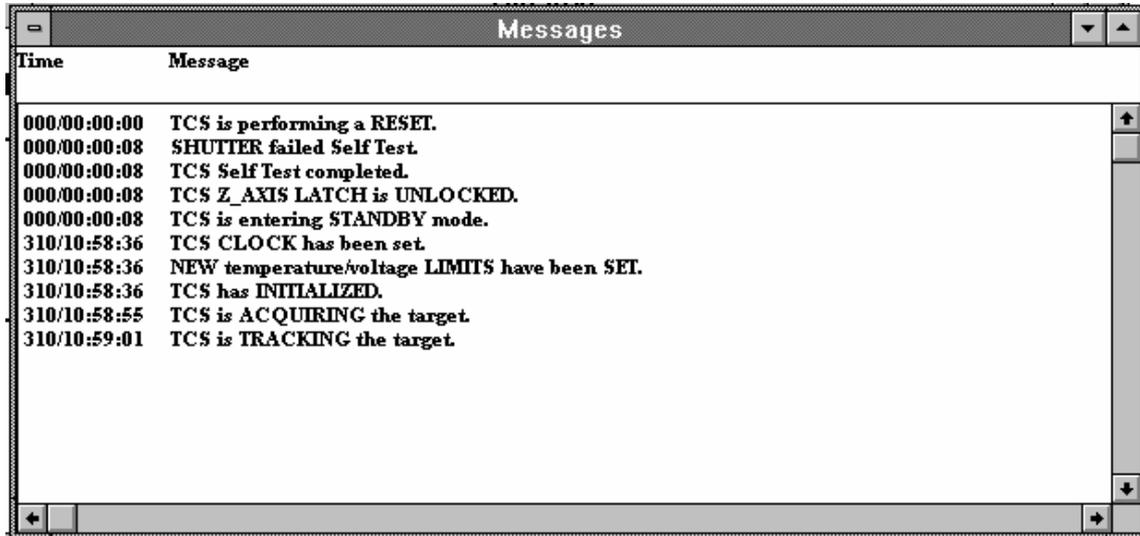


The TCS OPS (operations) window is the main data display window used during TCS operations. The TCS OPS window contains several major data display areas including:

- Range/range rate, data status and current TCS mode
- Bearing/bearing rate data (azimuth and elevation)
- Attitude/attitude rate data (roll, pitch and yaw)
- Subsystem status
- Small message area
- Call-up buttons for temperature, voltage, calibration, self test and messages windows

Any data item construed to be an error condition will be displayed in reverse video.

21.5.2.4.2 Messages



The **Messages** window is the expansion of the small window available on the TCS OPS page used for displaying the messages (M strings) sent from the sensor. The display consists of a large chronologically ordered page showing the latest messages. The window has the capability to be scrolled in either direction. This permits the viewing of all messages received during the session, in whatever mode is currently selected (Filtered or Unfiltered).

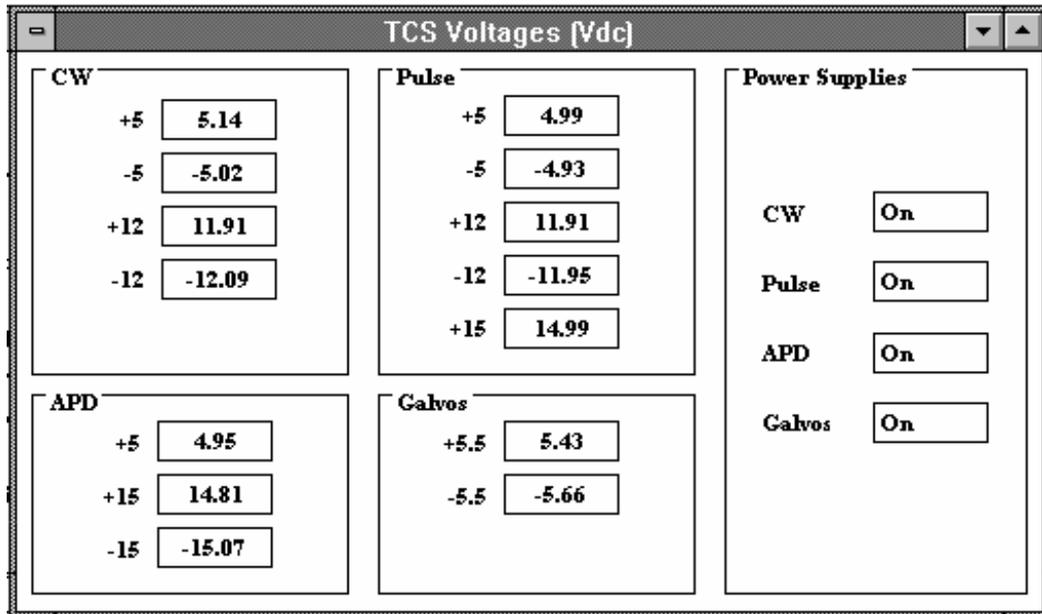
21.5.2.4.3 Temperatures

The screenshot shows a window titled "TCS Temperatures (deg C)" containing a table of temperature readings for various sensors.

| Sensor | Temperature (deg C) |
|----------|---------------------|
| CW Laser | 24.39 |
| APD | 28.37 |
| CPU | 0.00 |
| DC Power | 0.00 |
| X Galvo | 28.50 |
| Y Galvo | 27.89 |
| Z Galvo | 27.76 |

The **TCS Temperatures** window displays the system temperatures in degrees Centigrade for the seven temperature sensors located within the TCS. The values are updated every time a short cycle calibration is performed (every 30 seconds). If a temperature exceeds a limit defined by the last **Send TCS Limits** command, it is displayed in reverse video.

21.5.2.4.4 Voltages



The **TCS Voltages** window displays the data values associated with the TCS sensor. The 14 voltages are divided into four groups. The four groups include the CW and Pulsed subsystems, the APD, and the galvos, showing the individual measured polarities for the supplies that make up the group. The Power Supplies area shows the operational status (**On/Off**) of each group. The values are updated every time a short calibration is performed (30 seconds). If a voltage exceeds a limit defined by the last **Send TCS Limits** command, it is displayed in reverse video.

21.5.2.4.5 Calibration

| TCS Calibration | | | | | |
|-----------------|-------------|--------|-------------------|-------------------------|--------------------------|
| Brg Cal Result: | No Error | | Time of last Cal: | Long 310/11:07:11.52 | Short 310/11:07:41.61 |
| CW Cal Result: | No Error | | | | |
| | Slope | Offset | Noise Floor | | |
| X-axis | 0.760 | 63.890 | | AC | DC |
| Y-axis | 0.761 | 43.363 | CW Ranger | 126 | 8143 |
| ADC: | 0.999 | 18.0 | Pulsed Ranger | 106 | -14130 |
| | Brg Cal Tgt | | CW Rng Cal Tgt | | CW Averaged Tone Offsets |
| X-axis Centroid | 0 | 1 | | | Fine: |
| | -29747 | 29842 | -30236 | | 138.08 |
| Y-axis Centroid | 29519 | -29633 | -29919 | | Intermediate: |
| Z-axis Centroid | -32767 | -32767 | 19654 | | 74.00 |
| Reflectance | 2018 | 2002 | 16642 | | Coarse: |
| | | | | | 238.00 |

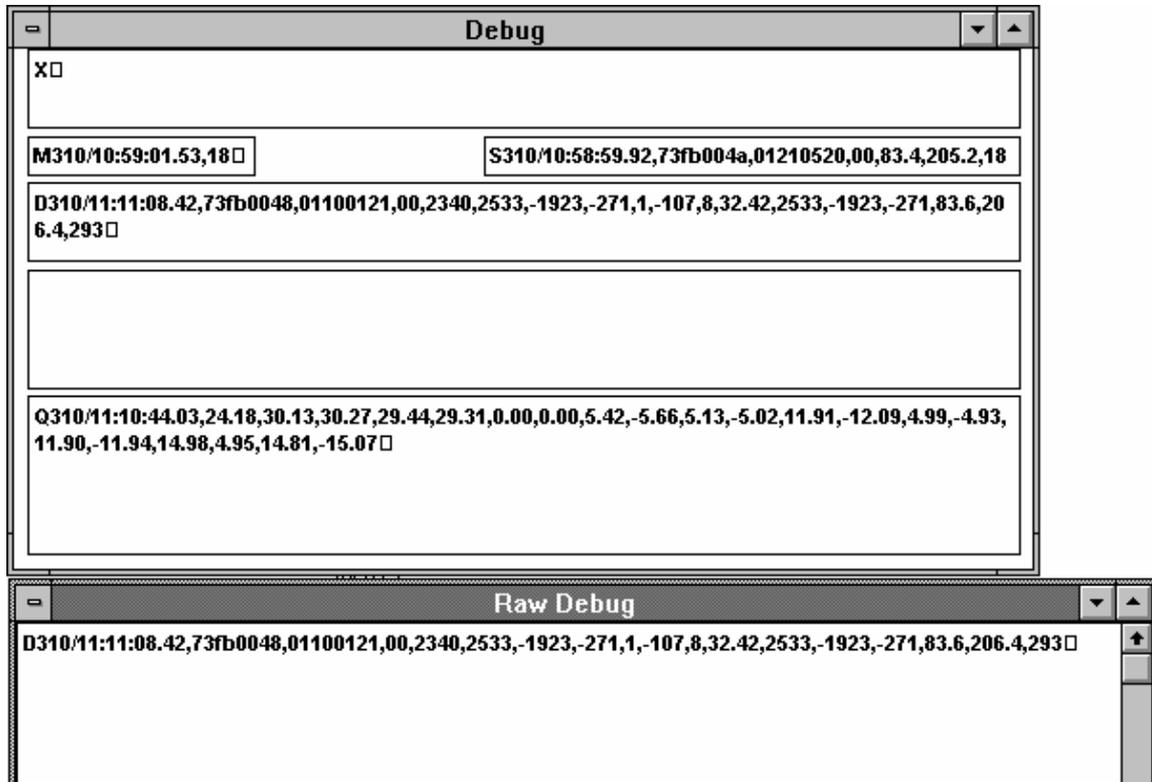
The **TCS Calibration** window contains the parameters associated with the last long calibration performed by TCS. These values are generally updated every four minutes. The values will also be updated every time a **Reset, Self Test, Acquire** or **Calibrate** command is requested. The time tag of the last long calibration data string is also displayed in the window. This window also displays the time of the last short calibration string. If a calibration error occurs, the Brg Cal Result and/or the CW Cal Result data items will be in reverse video.

21.5.2.4.6 Self Test

| TCS Self Test | | |
|---------------|--------|----------|
| | Status | Override |
| Shutter: | Failed | Off |
| Z Latch: | Passed | Off |
| CW Laser: | Passed | Off |
| Pulse Laser: | Passed | Off |

The **TCS Self Test** window contains the values associated with the four main TCS subsystems. The current status of these subsystems (**pass/fail**) and the **Override** command status of the individual subsystems is displayed. If a subsystem fails it's self-test, it is displayed in reverse video.

21.5.2.4.7 Debug



The debug windows are used to display the duplex communication strings on the RS 422 port that occurs between the CAD running in the PGSC and the Communications Task running in the sensor. The **Debug** window shows the last command issued from the CAD application to the sensor and the last string of each data string type received by the CAD application. The **Raw Debug** window shows the last data string received by the CAD application.

21.5.2.5 Status Bar



The Status Bar is located at the bottom of the TCS CAD display window. This window is used by the CAD to display the current status of various TCS items. This window is defaulted to be present when the CAD begins execution. The User may remove the window by selecting and clicking on the **Window>Status Bar** menu item. The following information will be displayed in the Status Bar area.

For TCS Help, press F1

This informational message informs the User that the CAD context-sensitive help facility is available by depressing the F1 function key.

Cmd:

The command field identifies the name of the last command that was been sent from the CAD to the TCS unit. The command resides in the display area only briefly, while the TCS unit processes the request. Generally the field displays the value of None. The following list expands on the displayed values and the associated command of each:

| <u>DISPLAY</u> | <u>MEANING</u> |
|----------------|-----------------------------------|
| None or blank | No active command |
| UNALL | Command not allowed/invalid |
| Cal | Long calibration |
| Stdby | Standby |
| SfTst | Self Test |
| Unlk | Unlock Z axis latch |
| Lock | Lock Z axis latch |
| Open | Open shutter |
| Close | Close shutter |
| Acq | Acquire target |
| AtEna | Attitude enable (disabled in CAD) |
| Time | Time set |
| Reset | Reset sensor |
| Diag | Diagnostic |
| SkTrk | Skin track (disabled in CAD) |
| Init | Initialize |
| Limit | Set limits |
| Oride | Override |
| ShCal | Short calibration |
| What | Whatever |

Att:

This field displays whether TCS is in Attitude measurement mode, which occurs only by a command from the User. Per agreement the attitude capability will not be used during TCS operations, and the command option is disabled. The status for this will always be set to **Inact** for Inactive.

Skin Trk:

This field displays whether TCS is in Skin Tracking mode, which occurs only by a command from the User. Per agreement, the skin tracking feature will not be used during TCS operations, and the command option is disabled. The status for this item will hence always be set to **Dis** for Disabled.

Snsr:

The Sensor field is used to display the TCS number that matches the unique sensor identification value in the *cad.ini* initialization file. There are four possible display values associated with this field. The confirmation of the sensor currently being used requires the processing of one long calibration string from the TCS unit.

| <u>DISPLAY</u> | <u>MEANING</u> |
|----------------|--|
| 0 or blank | The sensor identification number for the TCS unit has not been received via a long calibration string or the sensor number does not match any of the sensor identification numbers in the <i>cad.ini</i> file. |
| 1 | The sensor identification number for the TCS unit matched the value in the <i>cad.ini</i> file for the TCS1 (crew backup) sensor. |
| 2 | The sensor identification number for the TCS unit matched the value in the <i>cad.ini</i> file for the TCS2 (crew prime) sensor. |

RPOP:

The RPOP field displays the status of the Rendezvous Proximity Operations Program currently executing in the same computer. Acceptable choices include:

| <u>DISPLAY</u> | <u>MEANING</u> |
|----------------|---------------------|
| None or blank | Not initialized yet |
| Good | RPOP is running |
| Bad | RPOP is not running |

Health:

The health field displays the status of the TCS sensor. Possible choices include:

| <u>DISPLAY</u> | <u>MEANING</u> |
|----------------|------------------------------|
| NONE or blank | Not initialized yet |
| OK | TCS is functioning correctly |
| TEMP | Temperature limit exceedence |
| VOLT | Voltage limit exceedence |
| CAL | Long Calibration failure |
| SFTST | Self Test failure |

21.6 OPERATIONAL DATA

TCS was first flown on STS-64 and was tested using SPARTAN as the target. Some software problems were encountered, however, it was determined that a mislabeled cable was the cause. Since the TCS unit flown on STS-64 did not contain a pulse laser and had slight misalignments caused by vibration testing, acquisition of the target was not expected outside 500 feet. At a range of 690 feet, a momentary "hit" was received, but the TCS broke track immediately. Several other intermittent tracks occurred until approximately 390 feet at which time the sensor locked on and tracked successfully. The reason for the intermittent tracks is unknown, however, SPARTAN itself or the RMS could have been obscuring the retroreflector. While performing the "Shutdown" macro, a message was received from TCS that the Z latch had failed to lock. This message occurred because the latch was not locked prior to powerdown of the unit. Instead of repowering the unit and performing the appropriate steps, the crew issued an override command to the Z latch and then commanded it to lock, which drives the Z latch motor in the lock direction for a set time regardless of microswitch indication. The crew repeated deactivation steps which further drove the Z latch motor in the lock direction, possibly damaging the mechanism.

2 TCS units were flown on STS-71, the first docking with the Russian Space Station Mir. TCS 1 was activated first and acquired Mir at 474 feet about an hour after activation. TCS 2 was then activated and immediately acquired Mir at 430 feet. Since TCS 1 APD temperature indicated a trend that the auto shutdown limit would be reached prior to docking, and because a long period of close-in stationkeeping was expected, TCS 1 was powered down to allow adequate cooling such that the unit could be repowered for docking. The temperature decreased 19 degrees over a 38 minute period, after which TCS 1 was repowered. Both units performed successfully during docking and undocking phases. TCS 1 and 2 acquired prior to undocking and remained in track until 350 feet and 375 feet, respectively. The units dropped lock because the Mir was in free drift and drifted out of attitude. The units were powered once again during the mission for thermal concerns (units were getting too cold).

Both TCS units performed nominally on STS-74. TCS 1 locked on at 682 feet and maintained a solid lock during the approach. TCS 2 locked on immediately following power up and maintained lock as well. Both units provided a solid lock during the undocking phase. Tracking was intermittent during the flyaround phase but TCS 1 remained locked on until about 1200 feet (a new record).

STS-79 was the first flight TCS used the pulse laser tracking mode. Mir was acquired at 6010 feet using the pulse laser mode and tracked until approximately 400 feet at which time the system switched to CW mode. During undocking, the TCS would not initially lock on until the crew manually entered an initial range based on an estimate of actual range to reflector. PGSC software error was believed to be the cause. During both approach and departure the TCS showed alliterating range values with about an 18 foot delta caused by switching between tracking the intended hemi reflector and an ESA reflector which had been optically coated to prevent TCS tracking.

APPENDIX of INITIALISMS/ABBREVIATIONS

| | |
|-------|--|
| A/D | Analog/Digital Converter |
| APD | Avalanche Photodiode |
| ASCII | American Standard Code for Information Interchange |
| BIT | Built-In Test |
| CAD | Command and Display application |
| CPU | Central Processing Unit |
| CW | Continuous Wave |
| D/A | Digital/Analog Converter |
| DC | Direct Current |
| ft | feet |
| GUI | Graphical User Interface |
| MET | Mission Elapsed Time |
| m | meters |
| mm | millimeters |
| nm | nanometers |
| PDI | Payload Data Interleaver |
| PGSC | Payload and General Support Computer |
| RPOP | Rendezvous and Proximity Operations Program |
| SBC | Single Board Computer |
| SMS | Shuttle Mission Simulator |
| TCS | Trajectory Control Sensor |

Table Of Contents

| | |
|---|----------|
| 22.0 SSME REPOSITION | 1 |
| 22.1 INTRODUCTION..... | 1 |
| 22.2 BACKGROUND..... | 1 |
| 22.3 HARDWARE | 2 |
| 22.3.1 <i>APU and Hydraulic Systems</i> | 2 |
| 22.3.2 <i>Thrust Vector Controllers and Aero Surfaces</i> | 2 |
| 22.3.3 <i>Command and Feedback</i> | 2 |
| 22.4 SOFTWARE..... | 3 |
| 22.4.1 <i>Post MECO</i> | 3 |
| 22.4.2 <i>Drag Chute</i> | 4 |
| 22.4.3 <i>Post Landing</i> | 5 |
| 22.5 SEQUENCE AND OPERATIONS | 9 |
| 22.5.1 <i>Post-MECO</i> | 9 |
| 22.5.2 <i>Drag Chute</i> | 11 |
| 22.5.3 <i>Post Landing</i> | 11 |
| 22.6 BFS | 13 |
| 22.7 REFERENCES | 13 |

List of Tables

| | |
|--|----|
| Table 22 - 1 AI Error Messages..... | 7 |
| Table 22 - 2 Final Actuator Positions..... | 10 |
| Table 22 - 3 Final Actuator Positions (cont.)..... | 10 |
| Table 22 - 4 Final AI Actuator Positions | 12 |

List of Figures

| | |
|--|----|
| Figure 22 - 1 SSME Reposition Signal Flow..... | 3 |
| Figure 22 - 2 SPEC 51..... | 5 |
| Figure 22 - 3 SPEC 105..... | 6 |
| Figure 22 - 4 SPEC 113..... | 6 |
| Figure 22 - 5 SSME Post-Landing Reposition Sequence..... | 12 |
| Figure 22 - 6 BFS Override SPEC 51..... | 13 |

SECTION 22

SSME REPOSITION

22.1 INTRODUCTION

The purpose of Space Shuttle Main Engine (SSME) repositioning is to place the engine bells and aero surfaces in an optimal position to support vehicle ground and flight operations. This system brief will describe hardware, software, and operations used to achieve reposition during post-landing and flight. The information presented addresses the Primary Avionics Software System (PASS). A separate description of reposition in the Backup Flight System (BFS) is included in a section 22.6.

22.2 BACKGROUND

Generically speaking, repositioning is a capability available in GNC OPS 1/6, 3, and 9 that provides for commanding of Orbital Maneuvering System (OMS) Thrust Vector Controllers (TVC), Body Flap (BF), Rudder/Speedbrake (RUD/SB), SSME Ascent Thrust Vector Controllers (ATVC), Solid Rocket Booster (SRB) TVC, and Nose Wheel (NW). The flight sequences in OPS 1/6 and 3 reposition the SSME Engine bells only. The ground sequence can be executed anytime in OPS 9, and is of interest to flight operations due to its utilization post-landing, when it is used to command the SSME engine bells to a desired turnaround position. In OPS 9, there are a total of seven actuator initialization (AI) modes available, with five final positions achievable. These are 1) Ferry, 2) Rain Drain, 3) Gravity, 4) Null, and 5) Turnaround. Only rain drain is utilized post-landing for a variety of reasons. The most significant being it places the engine bells in a stable position that allows access to systems requiring servicing. The turnaround position, while also allowing the same service access, ends with the engine bells in an elevated position, and may result in drifting of the ATVC's after the Auxiliary Power Units (APU) are shut down if lock rings are not installed. This drift creates obstructions with areas that require access for servicing of the OMS and RCS. The rain drain position also provides protection against rain migrating into the engine prior to the Orbiter being towed to a protected area. However, the primary benefit of any post-landing reposition (rain drain or turnaround) is to provide access for the SSME turbopump bearing purge, which is required within 48 hours of landing to avoid a significant turnaround impact. To perform the purge, a minimum repositioning of the BF and RUD/SB is acceptable if the SSME engine bells are in the drag chute deploy position, otherwise a full reposition sequence is required.

Ferry, gravity, and null positions are also available for various ground tasks requiring a unique position of the previously mentioned actuators. The ferry (FERRY) position places the actuators in a position optimal for ferry operations with the Boeing 747 Shuttle Carrier Aircraft (SCA). Null (NULL) places the actuators at their null (zero) positions. Gravity (GRAV) is not used post landing, but was once used to position the engine bells in a 4 degree down position in preparation for roll out. However, this position prevented access to aft compartments and use was discontinued. The GRAV position is used, however, after pad abort if the engines have been lit.

During flight operations, SSME repositioning nominally occurs post Main Engine Cut Off (MECO) to the stow position in OPS 1, and at Mach 8 for the drag chute in OPS 3. Other periods where engine reposition is expected include after failure of an engine during powered flight, to the drag chute deploy position post MECO during Return to Launch Site (RTLS) aborts, and during hydraulic repressurization prior to entry.

22.3 HARDWARE

Repositioning utilizes hardware from three systems. The Auxiliary Power Units (APU's) generate hydraulic pressure for the three Shuttle hydraulic systems and provide the mechanical energy necessary to move the engine bells. Ground support equipment (GSE) is used to generate hydraulic power for the SRB TVC during pre-launch checkout operations.

The SSME Ascent Thrust Vector Controller (ATVC) repositions the bells to the desired pitch and yaw positions, the OMS TVC repositions the OMS engine bells, the Aero Surface Actuators (ASA) control elevon, body flap, and rudder/speedbrake positions, and SRB TVC rock and tilt actuators position the SRB engine nozzles (during pre-launch operations only).

Finally, the Shuttle Data Processing System (DPS) controls the process through GPC commands, Flight Critical (FC) Multiplexer/Demultiplexer (MDM) outputs, and Operational Instrumentation (OI) MDM and Discrete Signal Conditioner (DSC) inputs. During ground testing, data and commanding is also provided through the Launch Data Bus (LDB), however during post-landing operations the only available interface is through GPC command (SPEC Item entry or DEU equivalent), and telemetry downlink services.

22.3.1 APU and Hydraulic Systems

The Shuttle Auxiliary Power Units (APU) and Hydraulic Systems (HYD) provide the mechanical force to move all orbiter actuators with the exception of the OMS TVC, which is electrical. The Main Propulsion System (MPS) TVC Isolation Valve (MTIV) must be open in order for the SSME actuators to receive hydraulic pressure. Referencing SSSH drawing 12.3, the MTIV requires hydraulic pressure to open and close the valve, and a minimum pressure of 100 psi in order for the MTIV to move. Each SSME ATVC is driven by two of the three hydraulic systems (Center SSME by 1 and 3, Left SSME by 2 and 3, Right SSME by 1 and 2), and results in a requirement that a minimum of two APU/HYD systems be available for a reposition of all three SSME ATVC's. The MTIV can be reconfigured using either a switch throw on panel R4A4 or by FA MDM command. Control bus failures affect switches only, however main bus (MN A(B,C) ALC 1(2,3)) failures impact both MDM and switch commands, and result in a closed telemetry indication regardless of the actual position.

22.3.2 Thrust Vector Controllers and Aero Surfaces

Actuators used during repositioning sequences include the SSME ATVC, OMS TVC, ASA (including RUD/SB and BF), NW, and SRB TVC. During post-MECO, drag chute, and post-landing repositioning, NW and SRB TVC are not affected. In order for the sequence to complete, the ATVC's and ASA's must be powered, along with the associated MDM. The FCS channels must be in AUTO or O/R to provide hydraulic pressure to the associated actuator. Detailed drawings of these actuators are available in SSSH drawings 9.14, 9.15, and 9.16.

22.3.3 Command and Feedback

Commanding originates from the General Purpose Computers (GPC) and is transmitted through the FC FA MDM's to the intended Bus Terminal Unit (BTU) (e.g. SSME ATVC, ASA, etc.). Feedback for the OMS TVC, BF, and RUD/SB is through the FC FA MDM's. SRB TVC feedback (if being repositioned) is via MDM LL 2 and LR 2. All position feedback for the SSME ATVC is through the OI MDM OA 1, with signal conditioning provided by DSC OA 1. The GPC's then receive position data through 1 of 2 Pulse Code Modulation Master Units (PCMMU). Note: This is the only time the ATVC position feedback is used in closed loop control of the engine bells. In flight, a mechanical position linkage (reference SSSH drawing 9.14) nulls command errors from the GPC, and provides position feedback. Data and command paths are illustrated in Figure 22-1.

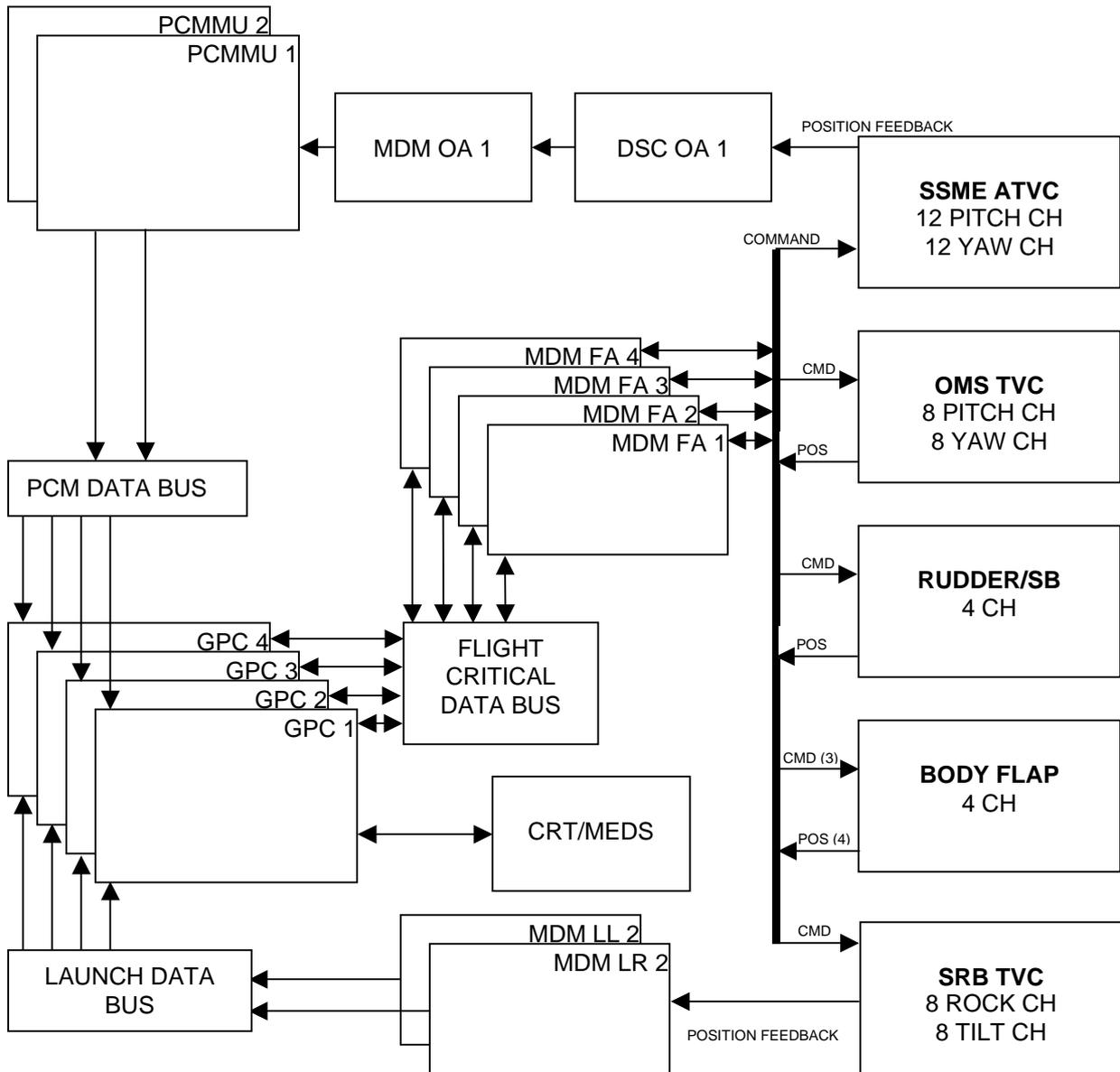


Figure 22 - 1 SSME Reposition Signal Flow

22.4 SOFTWARE

22.4.1 Post MECO

In OPS 1 and 6, the reposition sequence for post-MECO repositioning is controlled by the MPS TVC CMD Subsystem Operating Program (SOP) based on the status of the stow flag from either the MPS DUMP or ET SEP SEQ programs, depending on whether nominal MECO or an abort is declared. The stow flag is set to "1" initiating the stow sequence upon either 1) completion of the MPS dump, 2) MM 602, or 3) SSME pre-valves indicate closed and RTLs or TAL has been declared. There is no check of MTIV position prior to execution of the sequence since the valves are already assumed open for ascent. Once initiated, the SOP cyclically increments the TVC pitch and yaw commands until the desired position is

commanded (an open loop process). The commands for pitch and yaw are, respectively, defined by the K-Loads MPSPDUMP1(2,3) and MPSYDUMP1(2,3). Final stow positions are defined by MPSPSTOW1(2,3) and MPSYSTOW1(2,3). The values of these K-loads are currently set equal to the stow values listed in Tables 22-2 and 22-3. During RTLS, the positions are commanded to the I-loads CHUTESTOW1(2,3) (MSID V97U4271(2,3)C), which are also equal to the values listed in the tables.

The resulting rate of the gimbals is determined by the K-loads MPS_P and MPS_Y, which are both set to 0.08 deg/increment. The processing rate of the SOP post-MECO reduces from 25 to 12.5 Hz and results in a gimbal rate of $12.5 \times 0.08 = 1$ deg/sec. The SOP terminates once the SSME stow positions are reached in OPS 1 or 6.

Upon transitioning to OPS 3 (MM 301), the SOP resumes commanding the SSME gimbals to the entry stow positions.

22.4.2 Drag Chute

Drag Chute repositioning is performed in PASS MM 304 and MM 602.

In the case of OPS 3, the MPS TVC CMD SOP initiates the reposition sequence if the SSME_GMBL_POS flag from the LDG GEAR VLV CNTL SEQ is set true (= "1"), and the HYD SYS SOP declares more than 1 hydraulic system is operational. During MM 304, LDG GEAR VLV CNTL SEQ commands open the MTIV's at Vrel between 8000 and 3500 fps (Mach 8 to 3.5) as long as the sequence has not been inhibited from the SPEC 51 Item 19 (Figure 22-2). At Mach 8, if 2 hydraulic systems are considered operational by Priority Rate Limiting (PRL) logic in the HYD SYS SOP, all three MTIV are simultaneously commanded to open via MDM's FA 1, 2, and 3(4). The MTIV 3 OPEN ENABLE command is channelized through FA 4, while the OPEN command is issued through FA 3. MTIV 1 and 2 OPEN ENABLE commands are redundant through FA's 4 & 2, and 1 & 3, respectively. Therefore, it only takes one failure of either FA 3 or 4 to fail MTIV 3 (MTIV 1 and 2 require two failures to leave stuck). Status feedbacks for the MTIV's are through FA 1, 2, and 3, respectively. The valves will not be commanded to open below Mach 3.5, and will be commanded back to close upon 1) completion of the SSME reposition (SSME STOW COMPLETE flag = "1" from MPS TVC CMD SOP), 2) less than two hydraulic systems become available, or 3) less than two MTIV's are confirmed open when commanded. Hydraulic system status (HYD SYS 1(2,3) FAIL STATUS), MTIV status (MPS TVC ISO VLV 1(2,3)), and commfaults (FA 1(2,3) COMMFAULT) are considered to determine if two hydraulic systems are available to complete the reposition sequence. The conditions required for a system to be considered available is 1) HYD SYS not failed by PRL, 2) associated MTIV open, and 3) associated MDM (FA 1, 2, or 3) for the MTIV status is not commfaulted. If at least two systems are not available at the beginning or any time during execution, the sequence is halted. If halted, an SSME REPO FAIL SM alert is annunciated on the PASS fault page.

For RTLS (MM 602), the drag chute position is commanded upon entry into MM 602.

| XXXX/ 051 | | OVERRIDE | | XX X | | DDD/HH:MM:SS | |
|------------------|----|-------------------|------|----------------|-------|-----------------|---------|
| | | | | | | DDD/HH:MM:SS | |
| ABORT MODE | | ELEVON | | ENTRY FCS | | ATMOSPHERE | |
| TAL | 1X | | | FILTER | | | |
| ATO | 2X | AUTO | 17X | NOM | 20X | NOM | 22X |
| ABORT | 3X | FIXED | 18X | ALT | 21X | N POLE | 23X |
| MAXTHROT | 4X | SSME REPOS 19 XXX | | | | S POLE | 24X |
| PRPLT DUMP | | IMU | STAT | ATT | DES | PRL | |
| XXX ICNCT 5 XXXX | | 1S | XXXX | X | 25X | SYS | AUT DES |
| OMS DUMP | | 2S | XXXX | XXX | 26X | 1S | 28X 31X |
| ARM | | 3S | XXXX | | 27X | 2S | 29X 32X |
| START | | | | | | 3S | 30X 33X |
| STOP | | | | | | | |
| 9 QUAN/SIDE | | | | | | | |
| OMS DUMPTTG | | | | | | | |
| AFT RCS 13 | | | | | | | |
| 14 TIME | | | | | | | |
| FWD RCS 15 | | | | | | | |
| 16 TIME | | | | | | | |
| | | ADTA | H | | α | | M |
| | | L | 1S | XXXXXX | ±XX.X | X.XX | DES |
| | | | 3S | XXXXXX | ±XX.X | X.XX | 34X |
| | | R | 2S | XXXXXX | ±XX.X | X.XX | 35X |
| | | | 4S | XXXXXX | ±XX.X | X.XX | 36X |
| | | ET SEP | | ROLL MODE | | XXXXXX | |
| | | AUTO | | 38X | | AUTO SEL 42X | |
| | | SEP | | 39X | | WRAP MODE 4XXXX | |
| | | ET UMB DR | | VENT DOOR CNTL | | | |
| | | CLOSE | | 40X | | OPEN 43XXX | |
| | | RCS RM MANF | | CLOSE | | 44XXX | |
| | | CL OVRD | | 41X | | | |

(XX)

Figure 22 - 2 SPEC 51

22.4.3 Post Landing

SSME post-landing repositioning can only be accomplished in OPS 9, and executed using the Test Control Supervisor Sequences (TCS-S) software. The Actuator Initialization Process is controlled by the software module VXI_NPUT_CONTROL and performs six separate functions involving preconditioning and final actuator positioning. The software module controls the actuators through the sub-modules VRP_RAMP_CALL (ramp processing), VXE_ERROR_ROUTINE (error processing), and VXF_FEEDBACK_PROCESS (position feedback processing). Control and monitoring of TCS-S is available from OPS 9 SPEC 105 (Figure 22-3), with flight crew inputs made from SPEC 113 (Figure 22-4). SRB and nose wheel positioning is available if enabled through SPEC 113. Software resides on the Mass Memory Unit (MMU), and is fetched during the post-landing OPS 9 transition. Monitoring of the sequences requires PCM format 129 and GN format 44. It can be run either with a GPC redundant set or a simplex GPC.

22.4.3.1 Actuator Initialization (AI) Modes

The seven AI modes are designed to execute automatically after manual initiation from SPEC 105 or 113. AI mode 1 and 0 are common to all sequences and serve the purpose of placing the actuators in a known configuration prior to final positioning. Termination of the reposition sequence occurs upon either: 1) a TCS Cancel (SPEC 105 Item 5) is issued, 2) a STOP (SPEC 113 Item 32) is issued, 3) an OPS transition, or 4) a terminate condition is met by actuator positions being out of tolerance. Error messages associated with each sequence are provided in Table 22 - 1.

Table 22 - 1 AI Error Messages

| Display | Failure | Corrective Action |
|-----------------------------------|---|---|
| AI MODE 0 FAILED | Occurs after AI MODE IN PROGRESS message. Indicates actuators are not in proper Mode 0 position after moving from stow tolerance range. | If display occurs prior to attempting AI mode 6, no corrective action is required. An automatic rerun of mode 0 occurs. If display occurs during automatic AI mode 0/6 rerun, program will indicate that it has stopped and will display a crew option. Select AI – 1/0/6 for a retest or select CANCEL option. |
| AI MODE 0 RERUN IN PROGRESS | None. In process message | None |
| AI MODE 0 FAILED RETEST | Indicates the 1P or 2P actuators are still not in proper Mode 0 position. | Program will indicate that it has stopped and will display a crew option select. Select AI – 1/0/6 for a retest or select cancel option. |
| PROGRAM IN STOP MODE | None. In progress message for AI-6 failure. Program actually stops after crew option display. | Program actually stops after display of crew option select. Select appropriate crew option for particular failure. |
| ITEM 5 EXEC TO CANCEL TCS | None – crew option select | Option will terminate TCS |
| ITEM 7+0010 EXEC TO DO AI – 1/0/6 | None – crew option select | Option will re-enter program at step 10 and re-execute the normal AI-1, AI-0, AI-6 sequence. |
| ITEM 7+0034 EXEC TO DO AI – 0/6 | None – crew option select | Option will re-enter program at step 34 and re-execute an ECP call for AI-0 and AI-6. |
| AI MODE 6 FAILED | 1P/2P/3P actuators did not achieve AI-6 positions. | If occurs after first AI mode 6 attempt, no action. Automatic re-run of AI Mode 0 and AI Mode 6 occurs. If it occurs after automatic AI Mode 0/6 re- |

| | | |
|------------------------|---|--|
| | | run, program will indicate that it has stopped and will display a crew option select AI-1/0/6 for a re-test or select cancel option. |
| ECP ERROR 102 | ECP did not execute due to preconditions not satisfied. | If during Mode 0, Item 5 EXEC (Cancel) and recall. If during AI Mode 6 Item 6 EXEC 9 (Resume), program will retry AI Mode 0 and AI Mode 6. |
| ECP ERROR 103 | ECP detected actuator positions out of tolerance | Same as above. |
| ECP ERROR 104 | ECP detected actuator due to I/O Errors | Same as above. |
| ERROR CODE 11 | Sequence not found or nonexistent. | Item 5 Exec (Cancel) and Recall. |
| ERROR CODE 18 | Sequence request/retrieval miscompare. | Item 5 Exec (Cancel) and Recall. |
| ERROR CODE, ALL OTHERS | TCS-S Instruction Errors | Item 5 Exec (Cancel) |

22.4.3.2 AI 1

AI mode 1 is performed following completion of the OPS 9 transition. MDM commands are initialized to a value equal to the actuator position feedbacks for aero surfaces (RM selected value), OMS (selected TVC value), and SSME TVC's (MDM OA 1 value). Additionally, BF commands are inhibited. AI mode 1 does not affect SRB TVC or the nose wheel, if enabled on SPEC 113. If any of the positions exceed a table of maximum values, the output command is then set to the maximum value.

Prior to updating any actuator commands in AI mode 0, the C SSME yaw position is evaluated for collision with the OMS pod. If the clearance as determined from the SSME ATVC pitch and yaw exceeds a coded limit ($(pitch + ABS(0.75 * yaw)) > 593$ counts), the sequence is aborted in software. This equation is designed to ensure that yaw does not exceed 8.5 degrees if the SSME bell is pitched down greater than -5 degrees, and contact the OMS pod structure. Related to this check, a change was made to software that inhibits OMS TVC commands upon the transition to OPS 9, and prevents inadvertent commanding of the OMS. Rationale for this change was 1) to ensure no inadvertent movement of the OMS during ground processing, and 2) more Space Shuttle Program risk is assumed with not completing a post landing reposition due to an OMS instrumentation problem than in colliding bells. As long as the center SSME is within 2 degrees of zero yaw, there is no risk of collision. Additionally, as long as the OMS yaw TVC is not past zero degrees yaw inboard, the center SSME bell will collide with OMS pod structure before the OMS engine bell. An extensive list of SSME and OMS gimbal combinations which result in collision (zero clearance) is provided in the Shuttle Operational Data Book (SODB) Volume 1 section 3.4.3.1.

A check is also performed on the RUD/SB positions. If the delta between RUD and SB command is less than -705 counts, or the sum of RUD and SB command is greater than 556 counts, the reposition sequence is again aborted. The limits are designed to protect the RUD/SB conical seals from damage if the RUD commands cause the panels to drive into the seals. Due to combining effects of the RUD and SB, and differences in calibration, it is difficult to equate the software PCM count limits to a meaningful value in degrees/percent (the RUD and SB engineering units), so a transformation is not attempted in this discussion. However, a detailed discussion of the RUD/SB, and issue with the conical seal is available in SCP 5.7 Single APU Landing - Manual Speedbrake Setting.

22.4.3.3 AI 0

AI mode 0 is a precondition for AI modes 2 through 6. The TVC's and aero surfaces are configured in a specific sequence in positions to avoid collision. Like AI 1, AI mode 0 does not affect the SRB TVC or the nose wheel positions, if enabled on SPEC 113. Failure to meet any of these positions will result in the automatic re-try of the sequence, and display of a failure message on SPEC 105 (AI MODE 0 RERUN IN PROGRESS). A second failure results in automatic termination, and another SPEC 105 error message (AI MODE 0 FAILED RETEST).

A successful completion of AI mode 0 results in success messages on SPEC 105, and initiation of the AI mode associated with the selected final position. Failure during the subsequent AI modes (2-7), results in a re-execution of AI Mode 0, and re-try of the selected AI mode.

22.4.3.4 AI 2 through 6

AI modes 3 and 6 move the bells and aero surfaces into the rain drain and turnaround positions, respectively. AI modes 2, 4, and 5 are used during ground processing to command the engine a aero surface actuators to the ferry (FERY), gravity (GRAV), or null (NULL) positions, respectively.

22.5 SEQUENCE AND OPERATIONS

22.5.1 Post-MECO

The nominal post MECO repositioning to the dump position occurs automatically after MECO CONFIRM and ET SEP CMD. After the MPS dump is complete, the engines are commanded to the entry stow position. Today, the dump and stow positions are the same, however early in the Space Shuttle program, the dump and entry stow I-loads were different. For dump, all engines were commanded to $P = +10.5^\circ$ and $Y = 0^\circ$, while for stow they were commanded to the current entry stow positions. Deleting a separate dump position has the advantage of positioning the engines earlier, and avoiding having them stuck in a sub-optimum position for entry. Any differences in vent forces resulting from the two different positions during the MPS dump are minor. In both sequences, commanding is by open loop (no position feedback is used as in post-landing reposition). Termination of the MPS LH2 dump is indicated by the BF AUTO/MAN pbi being extinguished (equal to AUTO on ground displays). This indicates the position of the engine bells is up and away, preventing collision between the bells and the BF during entry. However, if a satisfactory stow position is not achieved, there are no constraints to taking down the APU's on time, since ample time is available on-orbit to devise a strategy for working around a failure to reach the stow positions.

Procedures affecting stow operations are located in the APU/HYD SHUTDN procedure in the POST OMS 1 tab of the Ascent Checklist. In this procedure, each MTIV is commanded to close prior to APU shutdown. The timing of this action is important since the MTIV will only move if hydraulic pressure is equal to or greater than 600 psia. Hydraulically isolating the TVC's prevents excessive drift while on-orbit due to cooling of the hydraulic fluid in the lines. Actuator stow must be complete prior to the OPS 2 transition since the MPS TVC SOP is not active during OPS 2. In OPS 8 and 3 (except during and after drag chute repositioning), the TVC's are again commanded to the entry stow position. During TAL aborts, the engines are commanded to stow position during MM 104. For RTLS, the drag chute position is entered immediately upon entering MM 602 (see paragraph 22.5.2).

In addition to stowing after MECO, a failed engine is commanded to the engine out position during powered flight. Since the remaining engines (or engine) will now be used to compensate for the loss of the other (or other two) engine(s), this places the non-running engine in a position away from the operating engine(s) in an attempt to prevent bell collision.

Tables 22 - 2 and 3 provide details on engine positions during different flight phases.

Table 22 - 2 Final Actuator Positions

| Trans/ event | OPS 9 (g) | | OPS 101 | | | | OPS 102 (c) | | OPS 103 | | OPS 104 | | OPS 105-304 (d) | | |
|-----------------|--------------|------------|------------|-----|--------------|--------|----------------|-----------------|-----------------|-------|----------------------|-------|----------------------|-------|-------|
| | Actr | In | Out | In | Start (a) | Launch | Out | In | Out | In | Out (ENT STOW) | In | Out (ENT STOW) | In | Out |
| P1 | AI | AI | AI | 0.0 | 0.0 | 0.0 | 0.0 | A C S M | A C S M | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | +10.0 |
| P2 | M O D E | M O D E | M O D E | 0.0 | 0.0 | 0.0 | 0.0 | C D | C D | -10.0 | -10.0 | -10.0 | -10.0 | -10.0 | 0.0 |
| P3 | | | | 0.0 | 0.0 | 0.0 | 0.0 | D V A A | D V A A | -10.0 | -10.0 | -10.0 | -10.0 | -10.0 | 0.0 |
| Y1 | | | | 0.0 | 0.0 | 0.0 | 0.0 | P L U E S | P L U E S | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 |
| Y2 | 1 | 1 | 1 | 0.0 | -3.5 | -3.5 | -3.5 | | | -3.5 | -3.5 | -3.5 | -3.5 | -3.5 | -3.5 |
| Y3 | | | | 0.0 | 3.5 | 3.5 | 3.5 | | | +3.5 | +3.5 | +3.5 | +3.5 | +3.5 | +3.5 |

Table 22 - 3 Final Actuator Positions (cont.)

| Trans/event | OPS 601-602 (e) | | OPS 602 (e) | | OPS 601 (f) | | ENG OUT STOW | DRAG CHUTE STOW | |
|-------------|--------------------|-------|----------------|-------|----------------|-----------------|--------------------|-----------------------|-------|
| | Actuator | In | Out | In | Out | In | | | Out |
| P1 | A C S M | 0.0 | 0.0 | 0.0 | 0.0 | A C S M | 0.0 | -3.0 | +10.0 |
| P2 | C D | -10.0 | -10.0 | -10.0 | -10.0 | C D | -10.0 | 3.0 | 0.0 |
| P3 | D V A A | -10.0 | -10.0 | -10.0 | -10.0 | D V A A | -10.0 | 3.0 | 0.0 |
| Y1 | P L U E S | 0.0 | 0.0 | 0.0 | 0.0 | P L U E S | 0.0 | 0.0 | 0.0 |
| Y2 | | -3.5 | -3.5 | -3.5 | -3.5 | | -3.5 | 3.0 | -3.5 |
| Y3 | | +3.5 | +3.5 | +3.5 | +3.5 | | +3.5 | -3.0 | +3.5 |

^aMPS slew check, which is initiated via ground inputs, commands SSME's to START position on completion of slew test. If slew check not performed, SSME's are commanded to START positions on event 6 (GO for auto sequence start).

^bSSME's are commanded to launch position at event 18A (MPS thrust > 90 percent).

^cLaunch positions maintained until tower clearance.

^dActuator stow must be completed in OPS 1 prior to entering OPS 201. MPS TVC command SOP does not run in OPS 2. OPS 8 (VU) S/W commands SSME to entry stow position, as does entry into OPS 3. Drag chute position is commanded at M 8-3.5 and maintained through MM 305.

^eAt termination of LH2 dump (event 43A), the SSME's are commanded to the stow position by MPS dump sequencer. For contingency abort (OPS 104 to OPS 602 transition), MM initialization commands SSME's to stow on entry to 602.

^fFor fast SEP in OPS 102, ET SEP sequencer sets SSME stow on entry to 602.

^gAI Mode 1 (CMD = MVS position fbk) performed at initial entry into G9 then again at L-3 hr.

Note: For recycle back to OPS 9, the following G9 initializations apply:

- 1) if SSME slew not performed: AI Mode 1 CMD from G9-G1 transition
- 2) if slew performed but no ignition: Start position
- 3) if ignition/pad abort: Gravity position

When a recycle count is requested, the SSME's are moved from wherever they are (i.e., either the launch or start positions) to the gravity positions.

22.5.2 Drag Chute

With the addition of a drag chute on STS-49, a requirement has existed to reposition the engine bells prior to drag chute deploy to prevent unnecessary wear to the center engine bell from the drag chute riser cords. This is performed between Mach 8 and 3.5 in MM 305 if enabled on SPEC 51 in Figure 22-3. Item 19 provides a toggle capability to alternately DIS or ENA the function. The function is initialized to ENA at the OPS 3 transition.

The drag chute reposition sequence occurs automatically at Mach 8. Redundancy Management (RM) will allow the sequence to execute if 1) two hydraulic systems are available according to Priority Rate Limiting (PRL), and 2) the associated MTIV's are open at the start of the sequence, and 3) the MTIV status FA MDM are not commfaulted. A detailed discussion of the impacts of system failures on drag chute repositioning is provided in SCP 8.5 OPS 3 SSME REPOSITION FOR DRAG CHUTE DEPLOY.

Drag chute repositioning during RTLS occurs automatically after the MM 602 transition. For TAL, the sequence is the same as nominal entry.

22.5.3 Post Landing

Procedures for post-landing SSME repositioning are maintained in the POST LANDING tab of the ENTRY C/L. As with drag chute repositioning, flight rules require fault tolerance prior to executing the sequence. Failures that are cause to no-go reposition include loss of MDM/DSC OA 1, bad aerosurface position feedbacks, body flap anomalies, or BFS engage. Position feedback for post landing reposition is through MDM OA 1, so any failures affecting that LRU will be cause to no-go the reposition sequence. Erroneous aerosurface feedbacks may also result in a large step command (greater than 2 degrees), and result in damage to hydraulic lines from a water hammer effect. Body flap anomalies causing interference with the final left and right SSME stow positions are also reason to no-go the sequence. However, if reconfiguration of the system can work around these failures (FCS channel management, MDM commfault, etc.), then the sequence may be recoverable. The final condition to no-go the reposition sequence is BFS engage, since the BFS has no reposition capability.

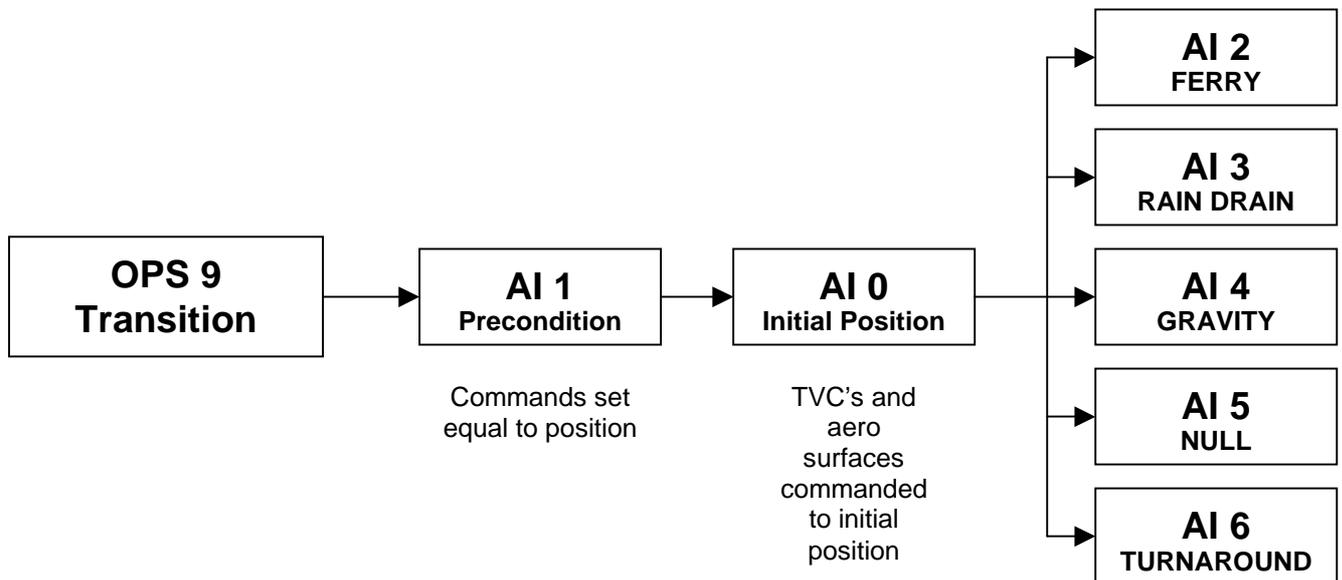


Figure 22 - 5 SSME Post-Landing Reposition Sequence

Table 22 - 4 Final AI Actuator Positions

| | | Commanded Position (Tolerances) | | | | |
|----------|---------------|---------------------------------|-------------|---------------------|--------------------|---------------------|
| | | AI | 1* | 0 Initial | 2 Ferry | 3 Rain |
| SSME TVC | CP (Deg) | | -10.0/+9.3 | -3.3 (-5.1/-1.4) | +10.0 (+7.8/+11.0) | +9.3 (+11.0/+7.1) |
| | CY (Deg) | | -8.1/+8.1 | +0.2 (-1.1/+1.5) | +0.2 (-1.1/+1.5) | +0.2 (-1.1/+1.5) |
| | LP (Deg) | | -9.3/+10.0 | +0.9 (-0.8/+2.5) | -3.4 (-5.3/-1.6) | +10.0 (+7.8/+11.8) |
| | LY (Deg) | | -8.1/+8.1 | +5.1 (+3.6/+8.9) | -0.6 (-2.0/+0.7) | +0.2 (-1.1/+1.5) |
| | RP (Deg) | | -10.0/+10.0 | +0.2 (-1.5/+1.8) | -4.0 (-5.9/-2.0) | +9.3 (+7.1/+11.0) |
| | RY (Deg) | | -8.1/+8.1 | -4.9 (-3.3/+8.9) | +0.9 (-0.4/+2.3) | +0.2 (-1.1/+1.5) |
| ASA | Elevons (Deg) | | -3.5/+20.0 | +0.4 (-1.4/+2.3) | N/A | N/A |
| | RUD (Deg) | | -27.0/+27.0 | 0 (-2.3/+2.3) | 0 (-2.3/+2.3) | 0 (-2.3/+2.3) |
| | SB (%) | | -10.0/+1.6 | +21.3 (+15.0/+27.5) | +5.1 (-1.2/+11.9) | +5.1 (-1.2/+11.9) |
| | BF (%) | | INH | +41.4 (+34.2/+49.0) | +94.6 (-1.9/+87.4) | +34.2 (+26.9/+41.4) |
| OMS TVC | L/R P (Deg) | | -8.4/+8.4 | +5.8 (-7.3/+7.2) | +6.1 (+5.3/+6.9) | +5.8 (+5.1/+6.1) |
| | L Y (Deg) | | -8.4/+8.4 | +6.8 (-0.8/+8.1) | -6.3 (-7.2/-5.4) | +6.9 (+6.1/+7.2) |
| | R Y (Deg) | | -8.4/+8.4 | +6.8 (-0.8/+8.1) | -6.3 (-7.2/-5.4) | +6.9 (+6.1/+7.2) |

* Maximum Initial Limits

| | | AI | 4 Gravity | 5 Null | 6 T/O |
|----------|---------------|----|---------------------|---------------------|----------------------|
| SSME TVC | CP (Deg) | | +9.2 (+11.0/+7.0) | +0.3 (-0.7/+1.3) | -6.2 (-4.0/-8.3) |
| | CY (Deg) | | +0.2 (-0.8/+1.2) | +0.2 (-0.8/+1.2) | +0.2 (-1.1/+1.5) |
| | LP (Deg) | | +6.5 (+4.5/+8.5) | +0.3 (-0.7/+1.3) | -5.8 (-7.8/-3.4) |
| | LY (Deg) | | -2.0 (-3.4/-0.5) | +0.2 (-0.8/+1.2) | +0.2 (-1.1/+1.5) |
| | RP (Deg) | | +5.8 (+3.9/+7.7) | +0.3 (-0.7/+1.3) | -6.2 (-4.0/-8.3) |
| | RY (Deg) | | +2.3 (+0.8/+3.8) | +0.2 (-0.8/+1.2) | +0.2 (-1.1/+1.5) |
| ASA | Elevons (Deg) | | N/A | N/A | N/A |
| | RUD (Deg) | | 0 (-2.3/+2.3) | 0 (-2.3/+2.3) | 0 (-2.3/+2.3) |
| | SB (%) | | +5.1 (-1.2/+11.9) | +5.1 (-1.2/+11.9) | +5.1 (-1.2/+11.9) |
| | BF (%) | | +34.2 (+26.9/+41.4) | +34.2 (+29.6/+38.4) | +94.6 (+87.4/+101.9) |
| OMS TVC | L/R P (Deg) | | +5.8 (+5.1/+6.2) | +5.8 (+5.1/+6.2) | +5.8 (+5.1/+6.2) |
| | L Y (Deg) | | +6.0 (+5.3/+6.6) | +6.0 (+5.3/+6.6) | +6.0 (+5.3/+6.6) |
| | R Y (Deg) | | +6.0 (+5.3/+6.6) | +6.0 (+5.3/+6.6) | +6.0 (+5.3/+6.6) |

22.6 BFS

In general, the BFS is capable of the same reposition capabilities as PASS. The major difference occurs during post-landing when BFS has no reposition capability. For all other periods, the BFS program MPS_TVC_CMD_HIP will reposition engines in the same timeframe as PASS (MM 104, 601/602, 301, and 304), and for the same reasons (engine out, engine stow, or drag chute repositioning). Stow in 104 will occur after ET SEP (MPS_DUMP_POS_FLAG = "1" from ET_SEP_SEQ, or AS43A_TERM_MPS_DUMP_N_STOW_SSMES = "1" from MPS_DUMP or ET_SEP_SEQ). Stow during RTLS is to the drag chute position and occurs after ET SEP (AB36D_STOW_SSMES = "1"), but interestingly does not require MM 602 be active prior to commanding the drag chute position (different from PASS). Fault tolerance for HYD SYS, MTIV, and FA commfaults is the same as for PASS. As in PASS, a crew input ENA or INH Item 19 is available on BFS SPEC 51 illustrated in Figure 22-7.

| XXXX / 051 | OVERRIDE | XX X | DDD / HH : MM : SS |
|-------------------|-------------------|----------------|--------------------|
| | | BFS | DDD / HH : MM : SS |
| ABORT MODE | | ENTRY FCS | |
| TAL 1X | ELEVON | FILTER | ATMOSPHERE |
| ATO 2X | AUTO 17X | NOM 20X | NOM 22X |
| ABORT 3X | FIXED 18X | ALT 21X | N POLE 23X |
| MAX THROT 4X | SSME REPOS 19 XXX | | S POLE 24X |
| | IMU DES ATT | AA | RGA SURF |
| PRPLT DUMP | 1S 25X X | LRU DES | DES DES |
| | 2S 26X | 1 | 31X 35X 39X |
| XXX ICNCT 5 XXXXX | 3S 27X | 2 | 32X 36X 40X |
| OMS DUMP | | 3 | 33X 37X 41X |
| ARM | | 4 | 34X 38X 42X |
| START | | | |
| STOP | | | |
| 9 QUAN/SIDE XXX | R | | |
| OMS DUMPTTG XXX | | | |
| | ET SEP | ROLL MODE | XXXXXXXX |
| | AUTO 28X | AUTO SEL | 42X |
| AFT RCS 13 XXX | SEP 29X | WRAP MODE | 4XXXX |
| 14 TIME XXX | ET UMB DR | VENT DOOR CNTL | |
| | CLOSE 30X | OPEN | 43XXX |
| FWD RCS 15 XXX | | CLOSE | 44XXX |
| 16 TIME XXX | | | |

(X X)

Figure 22 - 6 BFS Override SPEC 51

22.7 References

This discussion was developed using a variety of sources. Documentation consisted primarily of Space Shuttle software requirements and specification documents valid for OI 28, copies of the software code, general personal notes, and conversations with knowledgeable sources. Subsequent software modifications have the potential to render the data contained in this brief out-of-date. The documents listed here can be found in NASA, USA, or contractor libraries, and in some cases on the internet. For those that can be referenced electronically, a URL is provided.

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SSSH Drawing 9.14 ATVC ACTRS

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SSSH Drawing 9.16 AERO ACTRS

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APPENDIX A
ACRONYMS AND ABBREVIATIONS

| | |
|---------|--------------------------------------|
| A/D | analog to digital |
| A/L | approach and landing |
| AA | accelerometer assembly |
| ABC | automatic brightness control |
| ABO | artificial bright object |
| ac | alternating current |
| ACA | annunciator control assembly |
| ADI | attitude director indicator |
| ADP | air data probe |
| ADS | air data system |
| ADT | aerosurface drive test |
| ADTA | air data transducer assembly |
| AGC | automatic gain control |
| alpha | angle of attack |
| ALT | alternate altitude |
| AMI | alpha/Mach indicator |
| APU | auxiliary power unit |
| AS | artificial star |
| ASA | aerosurface amplifier |
| ATT | attitude |
| ATVC | ascent thrust vector control |
| AUTO | automatic |
| AV | avionics |
| AVE | average |
| AVVI | attitude/vertical velocity indicator |
| AZ | azimuth |
| | |
| BCE | bus control element |
| BF | body flap |
| BFS | backup flight system |
| BITE | built-in test equipment |
| BOA | bright object alert |
| BOS | bright object sensor |
| | |
| C&W | caution and warning |
| CAL | calibration |
| CAPRI | capacity reset integrator |
| CAS | calibrated air speed |
| cb | circuit breaker |
| CDI | course deviation indicator |
| CDR | commander |
| CF | communication fault |
| c.g. | center of gravity |
| CIL | critical items list |
| COMPOOL | common data pool |
| CRT | cathode ray tube |

| | |
|-------|--|
| CSS | control stick steering |
| DAP | digital autopilot |
| DAS | digital air speed |
| DB | deadband |
| dc | direct current |
| DDD | |
| DDS | detailed design specification |
| DDU | display driver unit |
| DES | deselect |
| DG | data good |
| DME | distance measurement equipment |
| DPS | data processing system |
| DR | discrepancy report |
| EAS | equivalent airspeed |
| ECLSS | environmental control life support system |
| EL | elevation |
| EPDCS | electrical power distribution and control system |
| ET | external tank |
| FA | flight aft |
| FC | flight control |
| FCOS | flight computer operating system |
| FCS | flight control system |
| FD | fault detection |
| FDF | flight data file |
| FDI | fault detection identification |
| FDIR | fault detection, isolation, and reconfiguration |
| FDO | flight dynamics officer |
| FF | flight forward |
| FI | fault identification |
| FMCA | forward motor control assembly |
| FMEA | failure modes and effects analysis |
| FOH | first order hold |
| FOV | field of view |
| FPCA | forward power control assembly |
| FSM | fault summary message |
| FSSR | Functional Subsystem Software Requirements |
| G&C | guidance and control |
| G&N | guidance and navigation |
| GNC | guidance, navigation, and control |
| GHz | gigahertz |
| GMT | Greenwich mean time |
| GN&C | guidance, navigation, and control |
| GPC | general purpose computer |
| GPS | global positioning system |
| GRTLS | glide return to launch site |
| GSE | ground support equipment |
| GSI | glide slope indicator |
| GYPTO | gyro pulse torquing |

| | |
|------|--|
| HAC | heading alignment circle |
| HIP | hardware interface program |
| HLV | high level video |
| HPU | hydraulic power unit |
| HSD | horizontal situation display |
| HSI | horizontal situation indicator |
| HUD | head up display |
| HUDE | head up display electronics |
| | |
| I/O | input/output |
| I/OP | input/output processor |
| IAS | indicated airspeed |
| IAU | interface adapter unit |
| ID | identification |
| IDT | image dissector tube |
| IF | intermediate frequency |
| IFOV | instantaneous field of view |
| IGS | inner glide slope |
| ILS | instrument landing system |
| IMU | inertial measurement unit |
| INH | inhibit |
| IOM | input/output module |
| | |
| JSL | jet select logic |
| | |
| KAGC | |
| KSC | John F. Kennedy Space Center |
| | |
| LCC | launch commit criteria |
| LED | light emitting diode |
| LIB | left inboard |
| LLV | low level video |
| LOB | left outboard |
| LOS | line of sight |
| | loss of signal |
| LRU | line replacement unit |
| LS | light shade |
| LSB | least significant bit |
| LSO | landing site officer |
| LVDT | linear variable differential transformer |
| | linear voltage differential transformer |
| LVLH | local vertical/local horizontal |
| | |
| M | magnitude |
| MDM | multiplexer/demultiplexer |
| ME | main engine |
| MECO | main engine cutoff |
| MEP | minimum entry point |
| MET | mission elapsed time |
| MHz | megahertz |
| MLS | microwave landing system |
| MM | major mode |

| | |
|----------------|---|
| MNVR | maneuver |
| MPS | main propulsion system |
| MSB | most significant bit |
| MSBLS | microwave scanning beam landing system |
| MSBLS-GS | microwave scanning beam landing system ground station |
| MSBLS-NS | microwave scanning beam landing system navigation set |
| MTBF | mean time between failures |
| MVS | midvalue select |
| | |
| NAGC | noise automatic gain control |
| NAS | no available sensor |
| NAV | navigation |
| NAVDAD | navigationally derived air data |
| NB | navigation base |
| NEP | nominal entry point |
| n. mi. | nautical mile(s) |
| NOM | nominal |
| N _z | normal acceleration |
| | |
| OGS | outer glide slope |
| OI | operational instrumentation |
| OMS | orbital maneuvering subsystem |
| OPS | operational sequence operations |
| ORBRG | Orbiter rate gyro |
| OSC | optic shutter closed |
| | |
| P | pitch pressure |
| PASS | primary avionics software system |
| pbi | pushbutton indicator |
| PCM | pulse code modulation |
| PDU | pilot display unit power drive unit |
| PFB | position feedback |
| PFS | preflight sensor self-test primary flight software |
| PL | payload |
| PLT | pilot |
| PNL | panel |
| POD | power-on discrete |
| POR | power-on reset |
| PRF | pulse repetition frequency |
| PRI | primary |
| PRL | priority rate limiting |
| PROC | processor procedure |
| PROM | programmable read only memory |
| PSI | parameter status indicator pounds per square inch |
| | |
| q | dynamic pressure |

| | |
|-------|----------------------------------|
| QMVS | quad midvalue select |
| R | roll |
| R/T | receiver/transmitter |
| RA | radar altimeter |
| RCS | reaction control system |
| RCV | receive |
| REF | reference |
| RF | radio frequency |
| RFOV | reduced field of view |
| RGA | rate gyro assembly |
| RGM | redundant gyro motor |
| RHC | rotational hand controller |
| RIB | right inboard |
| RM | redundancy management |
| RNG | range |
| ROB | right outboard |
| ROM | read only memory |
| RPC | remote power controller |
| RPTA | rudder pedal transducer assembly |
| RTLS | return to launch site |
| | |
| SBTC | speedbrake/thrust controller |
| SCU | servocontrol unit |
| SEC | secondary |
| SEP | separation |
| SF | selection filter |
| | selection filtering |
| SFC | SF command |
| SM | systems management |
| SMRD | spin motor run detector |
| SODB | Shuttle Operational Data Book |
| SOP | standard operating procedure |
| | subsystem operating program |
| SPEC | specialist function |
| SPI | surface position indicator |
| SRB | solid rocket booster |
| SSME | Space Shuttle main engine |
| SRC | sensitivity range control |
| SSSHB | Space Shuttle Systems Handbook |
| ST | star tracker |
| STAT | status |
| | |
| T | temperature |
| to | |
| TIO | target insertion orbit |
| T/R | transmit/receive |
| TACAN | tactical air navigation |
| TAEM | terminal area energy management |
| TAS | true airspeed |
| TCA | temperature control amplifier |
| TFOV | total field of view |
| THC | translational hand controller |

| | |
|--------|------------------------------|
| TLV | track level video |
| TS | target suppressed |
| TVC | thrust vector control |
| | |
| UPP | user parameter processing |
| UPS | uninterruptible power system |
| | |
| WONG | weight on nose gear |
| WOW | weight on main gear (wheels) |
| WOWLON | weight on wheels latched on |
| | |
| Y | yaw |