

Shuttle Flight Operations Manual

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Volume 7
Applications Software

Flight Operations Directorate
Crew Training and Procedures Division

Final
August 1, 1980

NASA

National Aeronautics and
Space Administration


Lyndon B. Johnson Space Center
Houston, Texas

FLIGHT OPERATIONS DIRECTORATE

FINAL
SHUTTLE FLIGHT OPERATIONS MANUAL
VOLUME 7: APPLICATIONS SOFTWARE

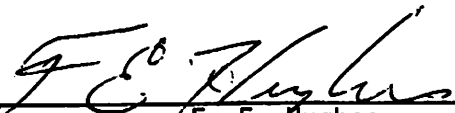
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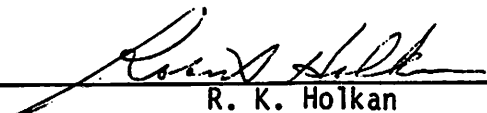


M. Al Brekke
Ascent/Entry Section


APPROVED BY:



F. E. Hughes
Head, Ascent/Entry Section

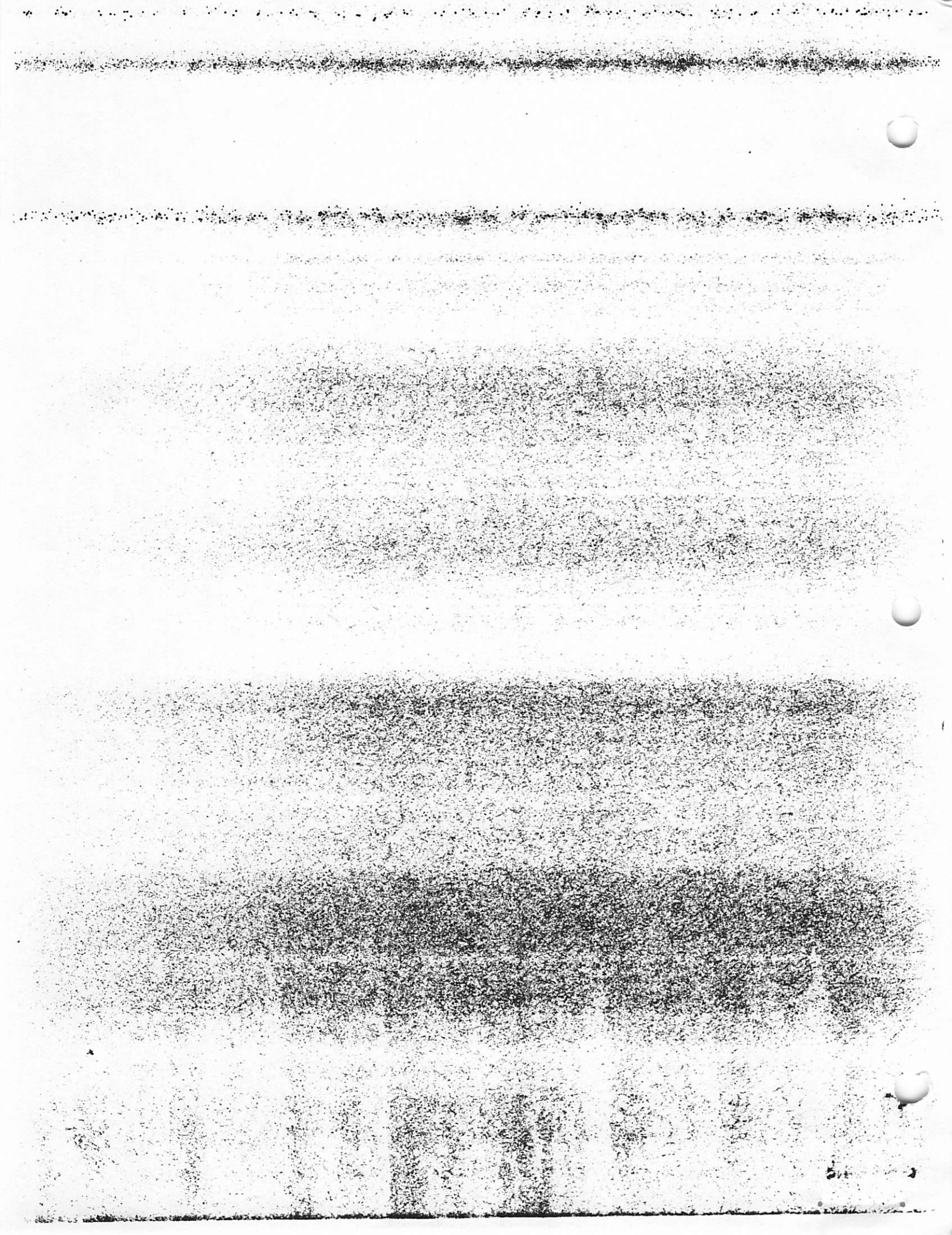


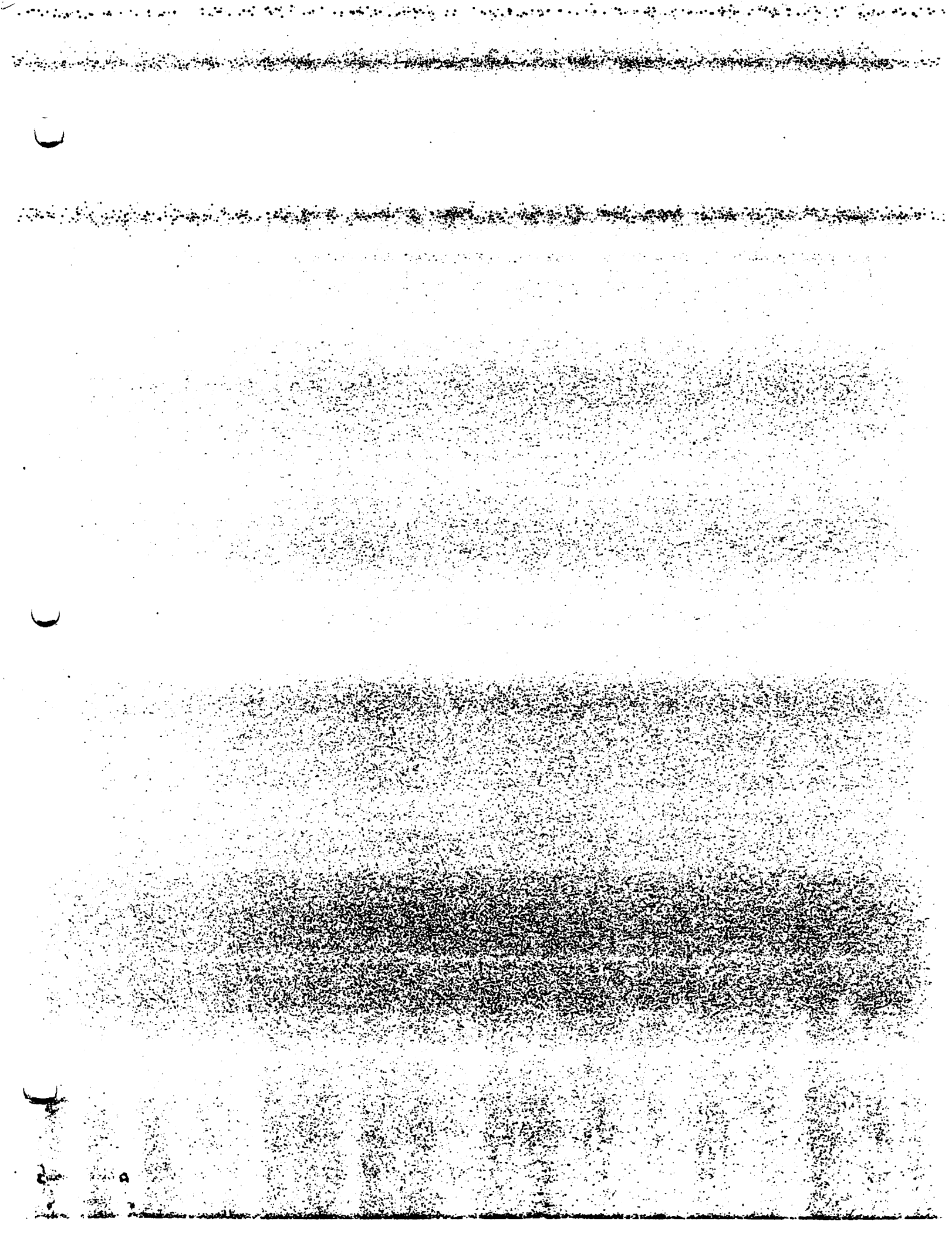
R. K. Holkan
Chief, Flight Training Branch



J. W. Bilodeau
Chief, Crew Training and Procedures Division

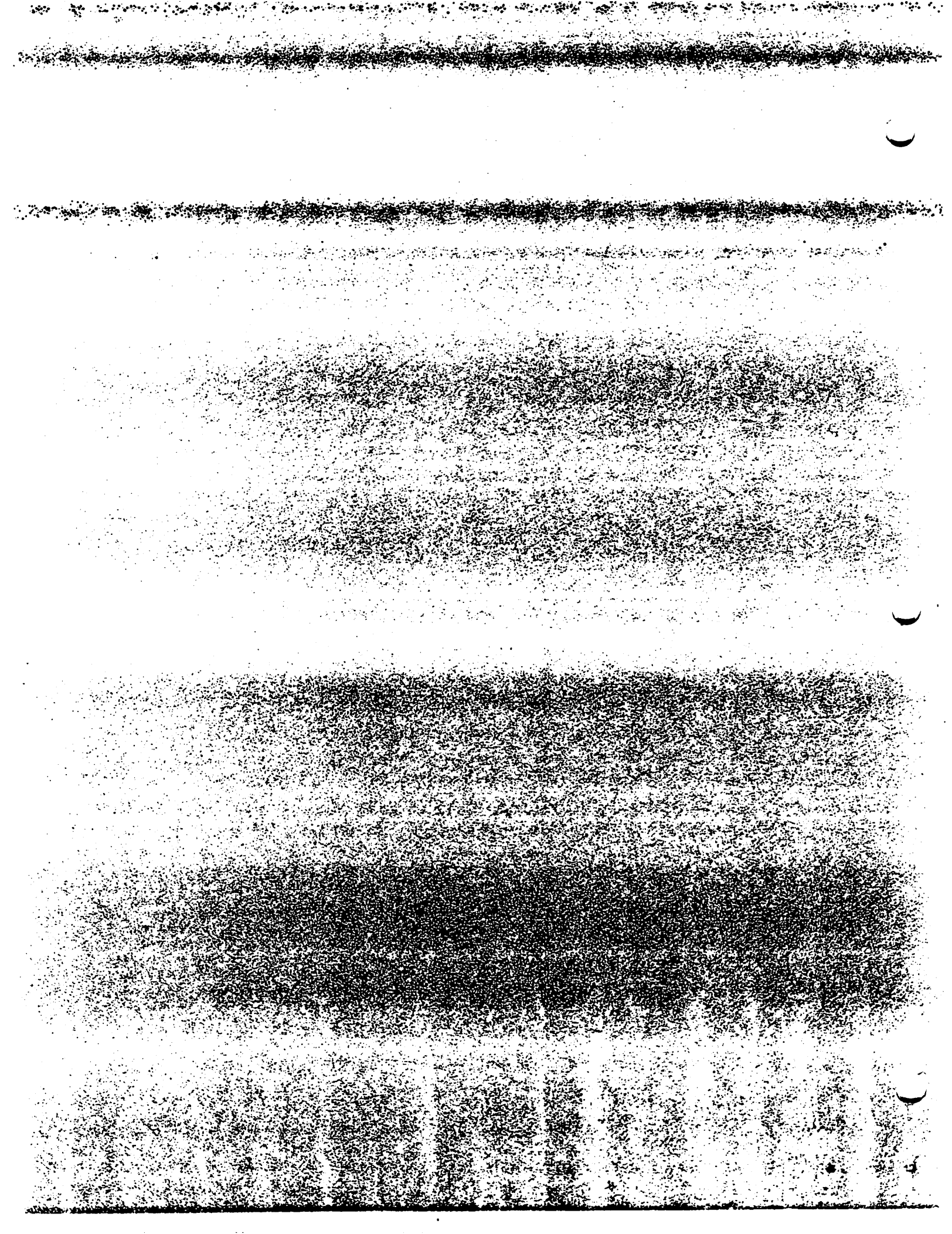
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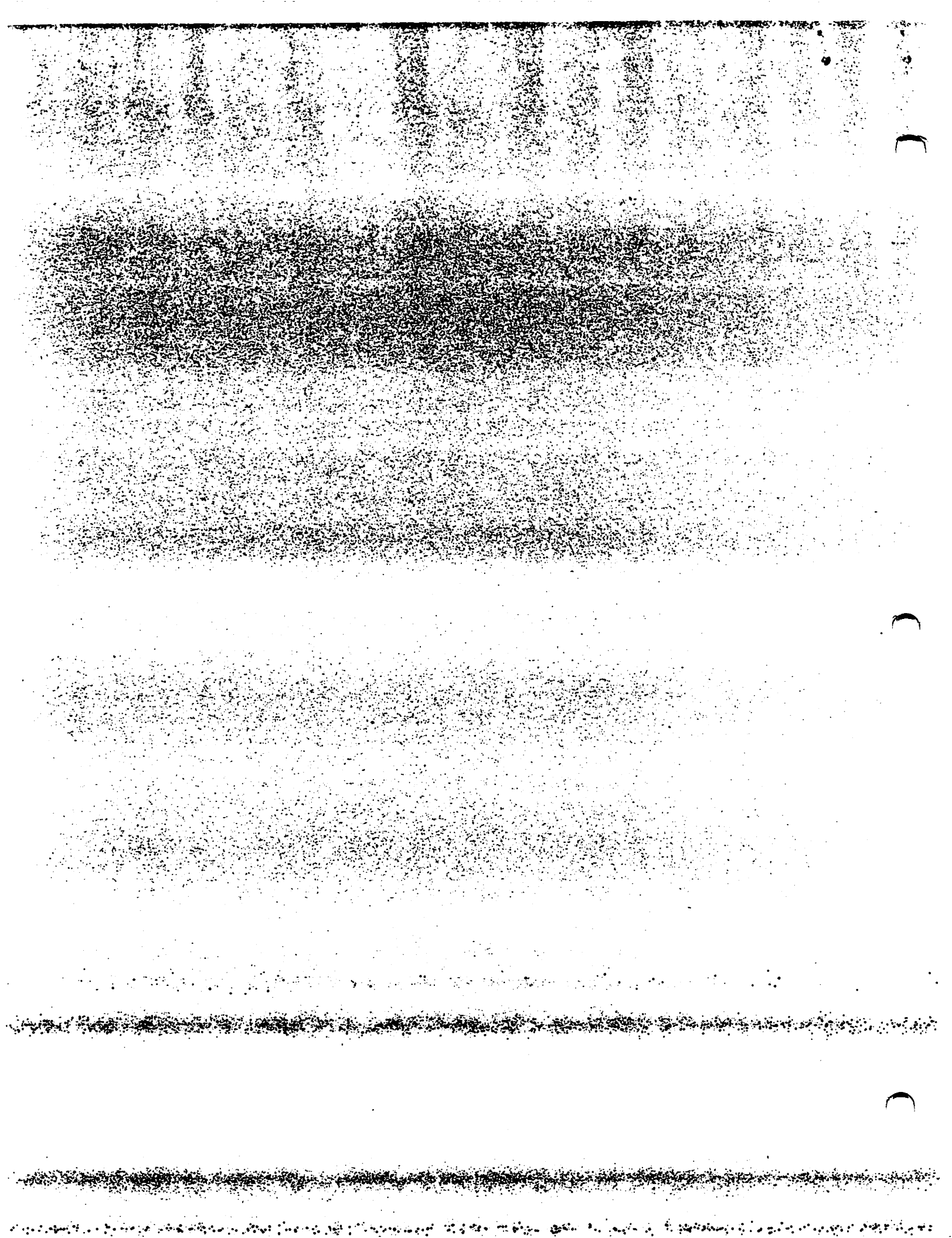




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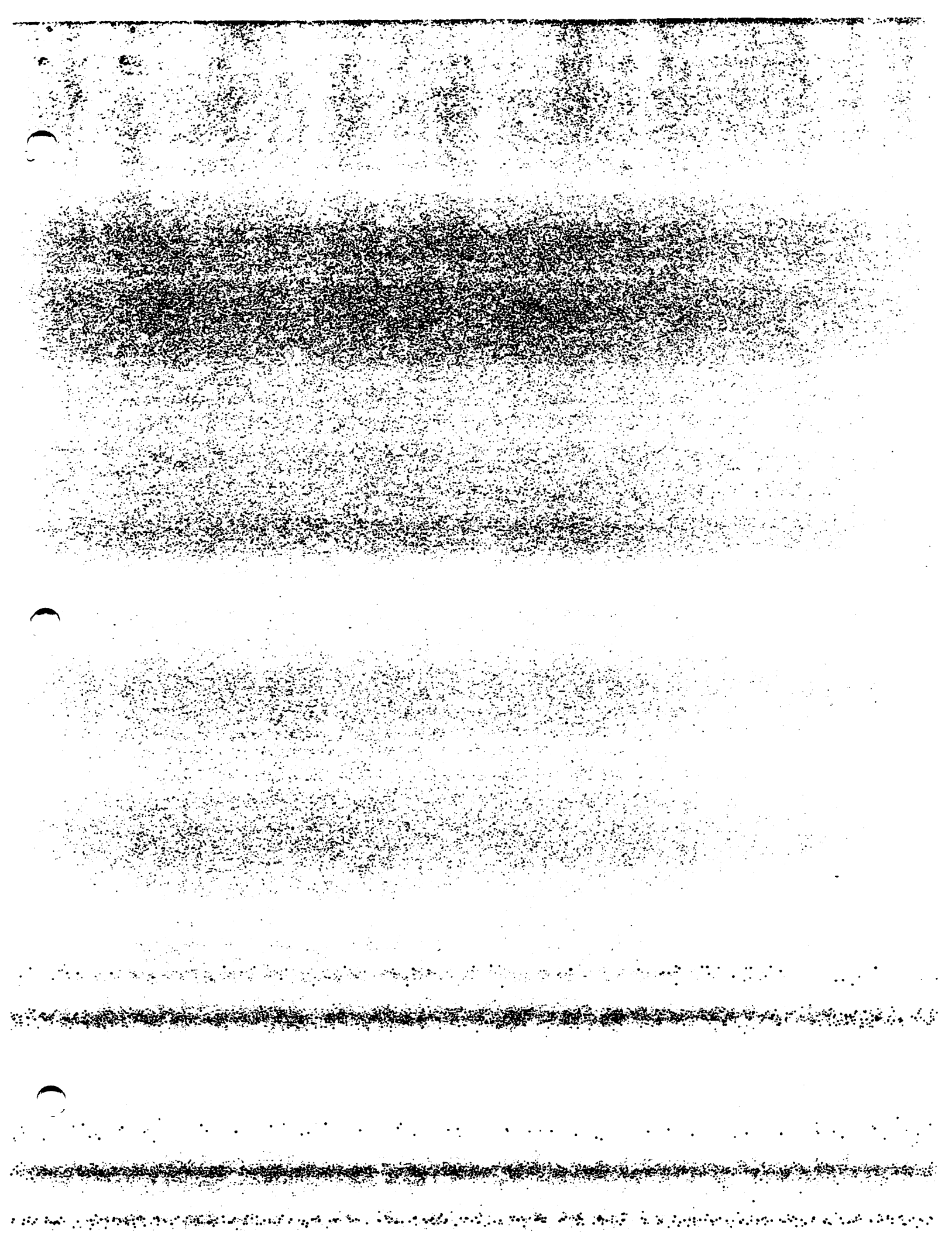


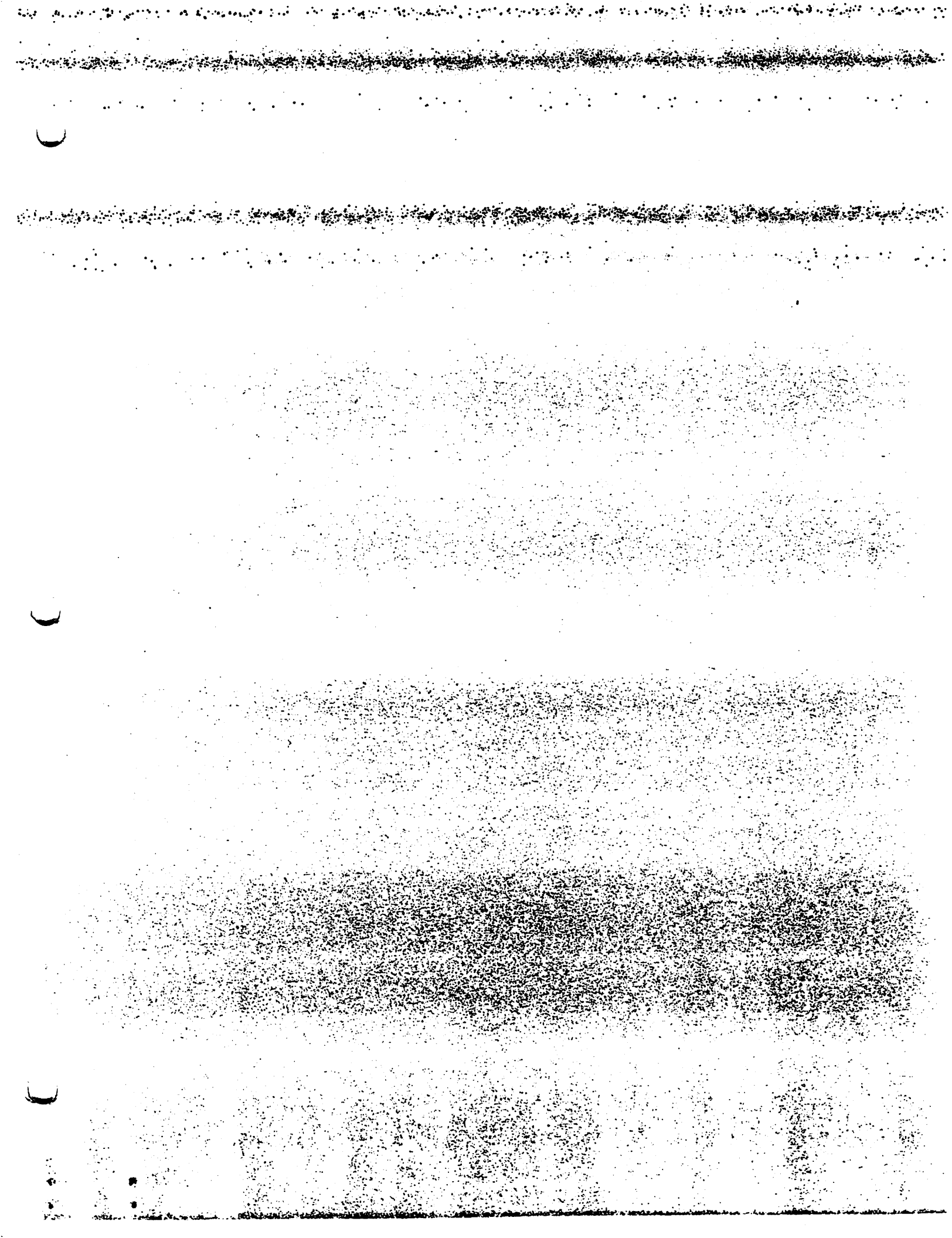
DOCUMENT ORGANIZATION

The Shuttle Flight Operations Manual (SFOM) will be published in separate individually bound volumes. Publication of the updated editions of the volumes will be staggered throughout 1980.

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16	Payload Deployment and Retrieval Systems	CG5/B. Kain	-2611
17	Inertial Upper Stage System	CG6/G. Laski	-4794
18	Spinning Solid Upper State System	L. Vande Zande	-4794
19	Spacelab Support Systems	A. Holt	-4794





SECTION 2
GUIDANCE, NAVIGATION, AND CONTROL APPLICATIONS SOFTWARE

The Shuttle GN&C applications software consists of several principle functions. They are; guidance, navigation, flight control, redundancy management, subsystem operating programs and moding, sequencing, and control. An overview of these functions relating inputs and outputs is shown in figure 2.0-1.

Software processing of these principle functions is highly dependent upon flight phase. Therefore, each section contains a general functional description followed by descriptions for each Operational Sequence (OPS) where appropriate.

For those functions for which the BFS processing differs from the Primary Avionics Software System (PASS), an additional paragraph is included following the PASS description.

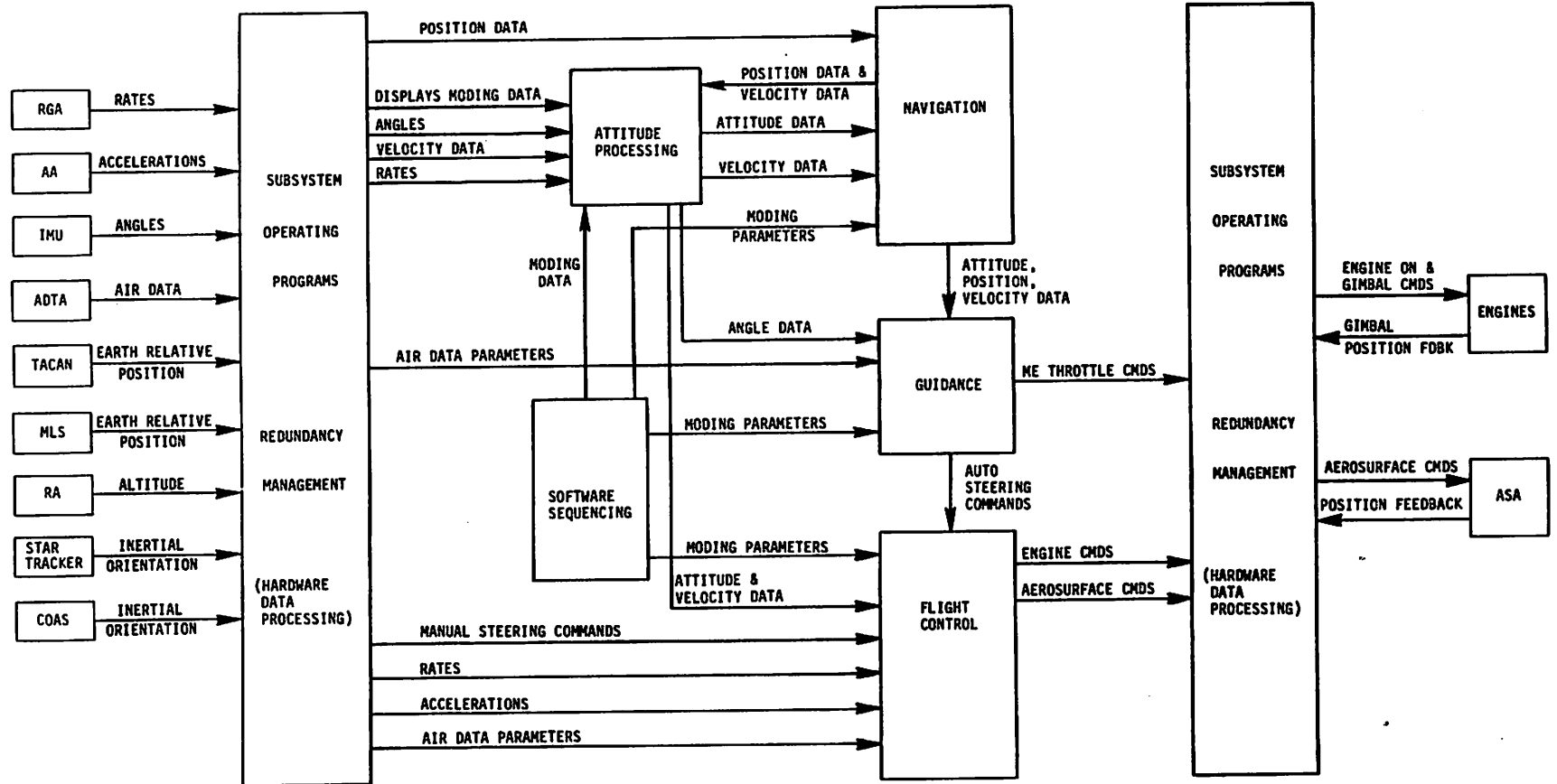


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2.3.6	D. J. Murphy/W. C. Brown	X4637
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2.5.7	M. A. Brekke	X4637
2.5.8	T. J. Gluszek	X3048
2.5.9	D. C. Zwingman	X4637
2.6	A. J. Petro	X4637
2.7	W. A. Mathews	X4637
3	J. E. Sidwell	X2651

SECTION 1 INTRODUCTION

1.1 PURPOSE

The purpose of the Shuttle Flight Operations Manual, Vol. 7 - Applications Software, is to provide a compilation of the functional description of the Space Transportation System (STS) application's software, including Guidance, Navigation, and Control (GN&C), Redundancy Management (RM), software sequencing, Subsystem Operating Programs (SOP), hardware data processing, and Systems Management (SM) software.

1.2 SCOPE

This SFOM provides GN&C and SM applications software reference information specifically for the flight crew and flight operations personnel. It is comprised of the following sections:

- Section 1, Introduction - This section describes the purpose, scope, authority, control, publication, and revisions of this volume.
- Section 2, GN&C Applications Software - This section provides a functional description of the guidance, navigation, and flight control software, software sequencing, RM, and SOP, along with the related hardware data processing for each flight phase.
- Section 3, Systems Management Software - This section provides a functional description of the software which is responsible for acquiring subsystem data, performing Fault Detection and Annunciation (FDA), and processing the data for display.

1.3 AUTHORITY

The Crew Training and Procedures Division (CTPD) within the Flight Operations Directorate (FOD) has the responsibility to develop crew procedures and to provide documentation necessary to support training operations as defined in the Flight Operations Directorate Program Support and Management Plan (JSC 10750).

This document, SFOM Volume 7, is CTPD's partial fulfillment of this responsibility for STS/OPS in the applications software area.

1.4 CONTROL

The STS/OPS Shuttle Flight operations manual (SFOM) is controlled by the Training Control Board as defined in the Space Shuttle Training Management Plan (JSC 0993L).

1.5 FORMAT

This document covers all applications software pertaining to the Space Shuttle. GN&C software is discussed in section 2 and SM software is covered in section 3. Sections 2 and 3 are further divided into multiple subsections to facilitate reference.

The Backup Flight System (BFS) software is not covered in a section of its own. Instead, BFS deltas are covered immediately following the PASS discussion of each section.

1.6 PUBLICATIONS AND REVISIONS

This document will be published in Preliminary and Final editions. Each edition will be revised as required via Page Change Notices (PCN's) or, if extensive changes are required to a milestone edition, the document will be issued again as a Revision. The Flight Training Branch of the CTPD has the responsibility for preparing, coordinating, and maintaining this document.

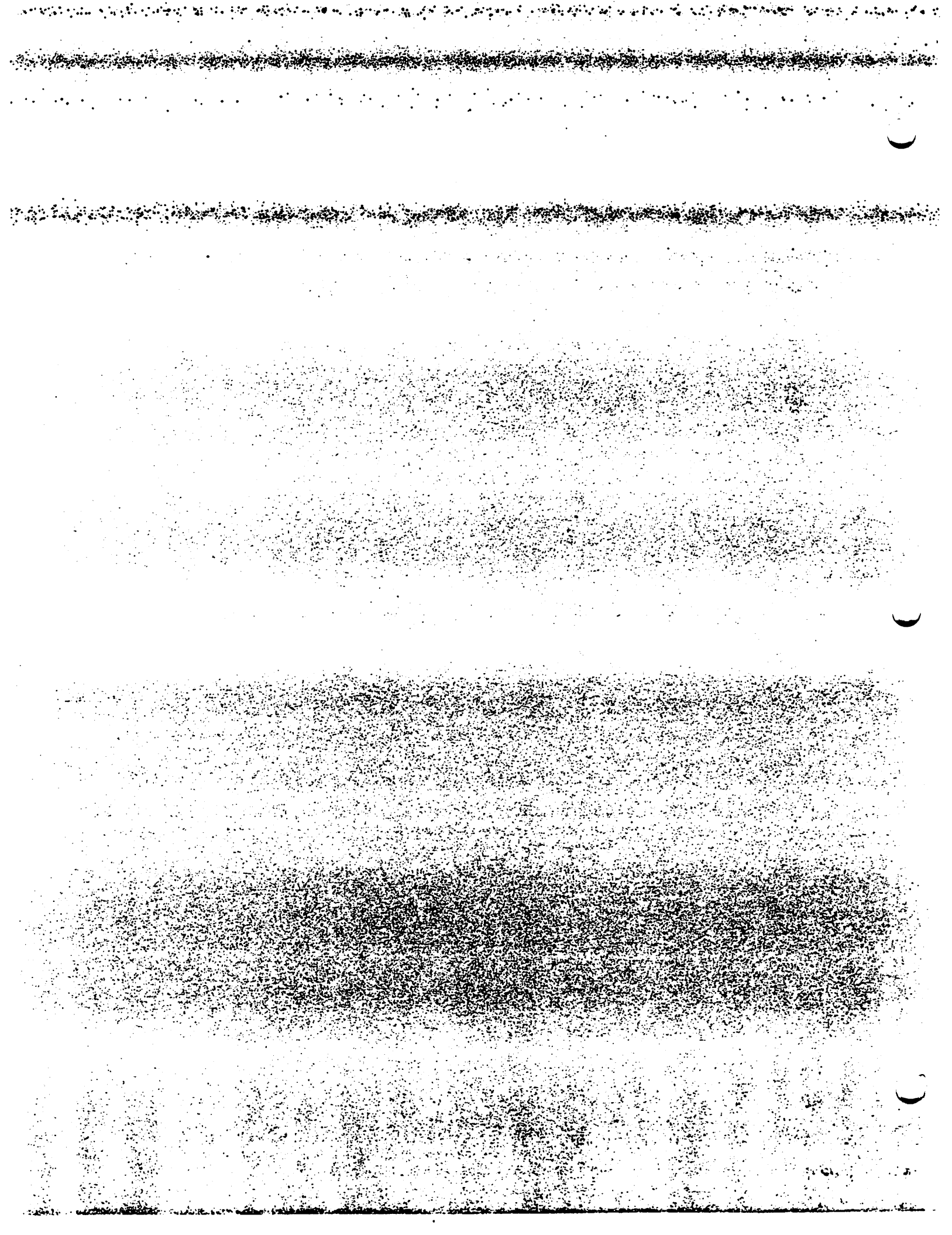
Any organization having requests for changes in distribution, comments, questions, suggestions, or proposal changes concerning this document should contact M. A. Brekke (phone 713-483-4637 or FTS 525-4637).

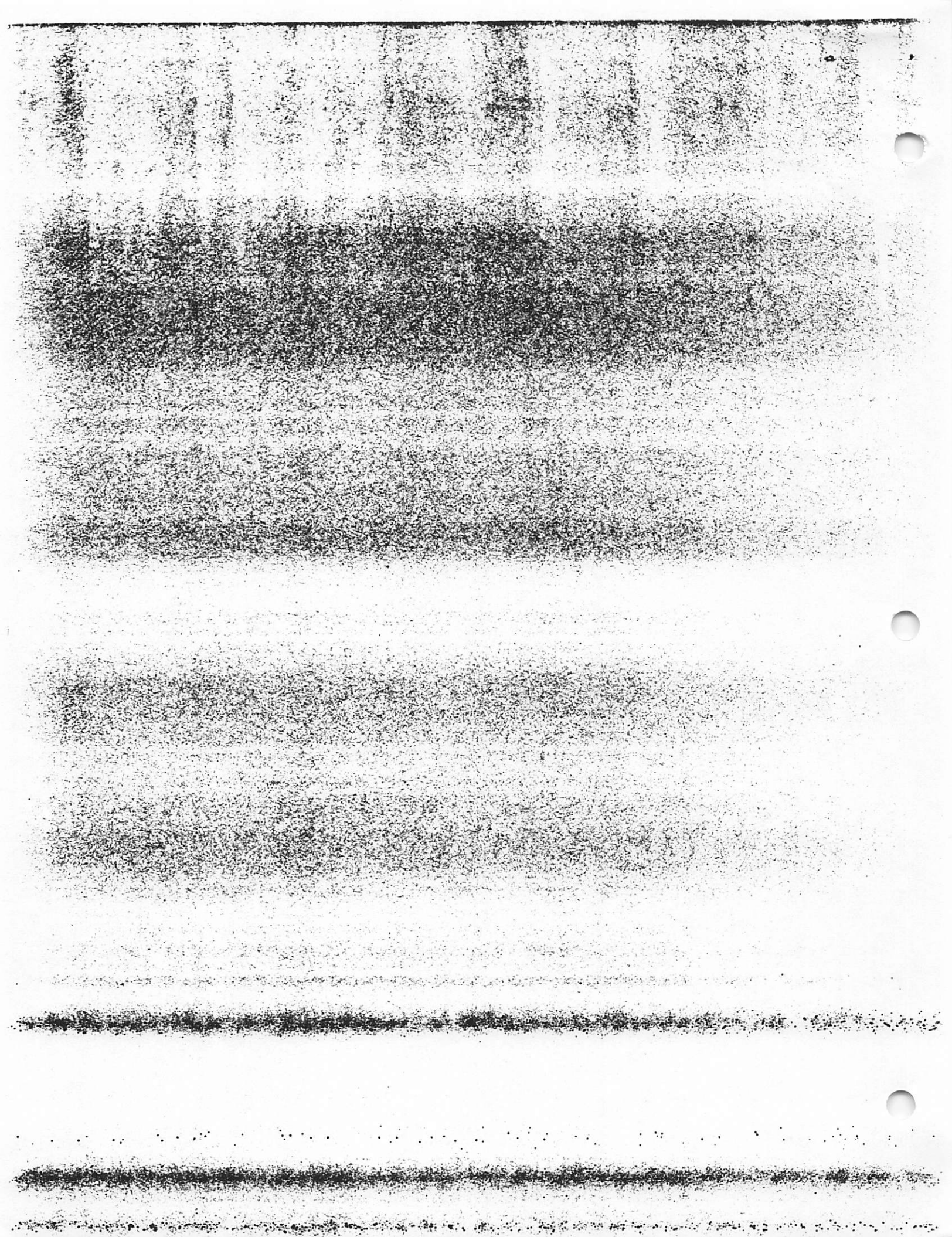
1.7 ACKNOWLEDGMENTS

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Authors of the various subsections are listed below. All technical questions should be directed to them.

<u>Section</u>	<u>Author</u>	<u>Extension</u>
2.1.1	A. S. Marino	X4637
2.1.2	A. M. Patterson	X3048
2.1.3	A. M. Patterson	X3048
2.1.4	A. M. Patterson	X3048
2.1.5	A. M. Patterson	X3048
2.1.6	D. W. Johnson	X4637
2.1.7	T. H. Browder	X3048
2.1.8	W. A. Chanis	X3048
2.1.9	W. E. Powers	X3048
2.1.10	W. E. Powers	X3048
2.1.11	T. J. Gluszek	X3048
2.1.12	P. J. Beauregard	X5348
2.1.13	P. J. Beauregard	X5348
2.1.14	A. Cavello	X3048
2.1.15	A. Cavello	X3048
2.1.16	A. Cavello	X3048
2.1.17	A. Cavello	X3048
2.1.18	J. E. James	X4637





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2.1 HARDWARE DATA PROCESSING

2.1.1 Inertial Measurement Unit

Subsystem operating program: The Inertial Measurement Unit (IMU) SOP provides accumulated velocity and compensated attitude data. Also, the SOP processes both hardware and software Built-in Test Equipment (BITE). The SOP sends the three sets of IMU attitude, velocity, and BITE data to RM and other GN&C users (fig. 2.1-1 and fig. 2.1-1a).

Redundancy management: The IMU RM software is discussed in terms of attitude RM and velocity RM. RM is then further discussed in terms of the Selection Filter (SF) and Fault Detection Identification and Reconfiguration (FDIR).

All IMU RM functions require knowledge of which IMU's are candidates for RM processing. An IMU is defined as a candidate for RM processing if the following are satisfied.

- Crew selection - The IMU must be crew selected or if crew selection is not available, the selection status shall be configured as it was at the end of the most recent flight phase (OPS) in which selection capability was available. Selection status shall be carried across OPS transition in protected memory.
- No previous failure - The IMU must not have been previously declared failed by either the attitude or velocity fault identification functions.
- No communications fault - The IMU commfault flag must not be valid.
- No IMU slewing in progress - The IMU is not being slewed (realignment in progress).

Velocity SF: The purpose of the velocity SF is to select single-velocity values to provide to navigation.

- Three good IMU's - The SF middle-value selects the three total velocity vectors on a component-by-component basis.
- Two good IMU's - The SF averages the two total velocity vectors on a component-by-component basis.
- One IMU - The SF will pass the velocity data.
- Zero IMU - The selected velocity will be set equal to its previous value.

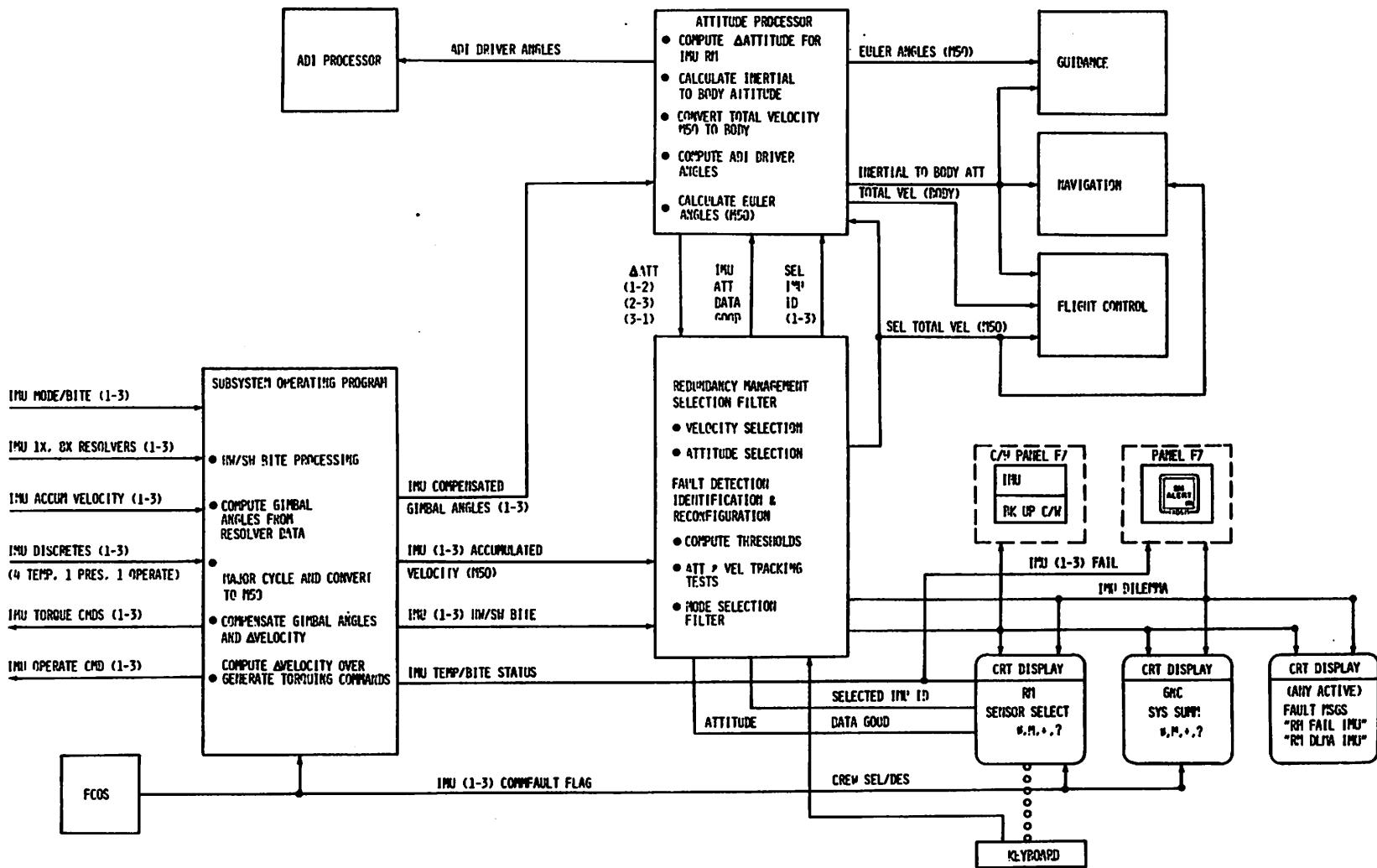


Figure 2.1-1.- Ascent (GNC OPS 1 & 6) IMU data processing.

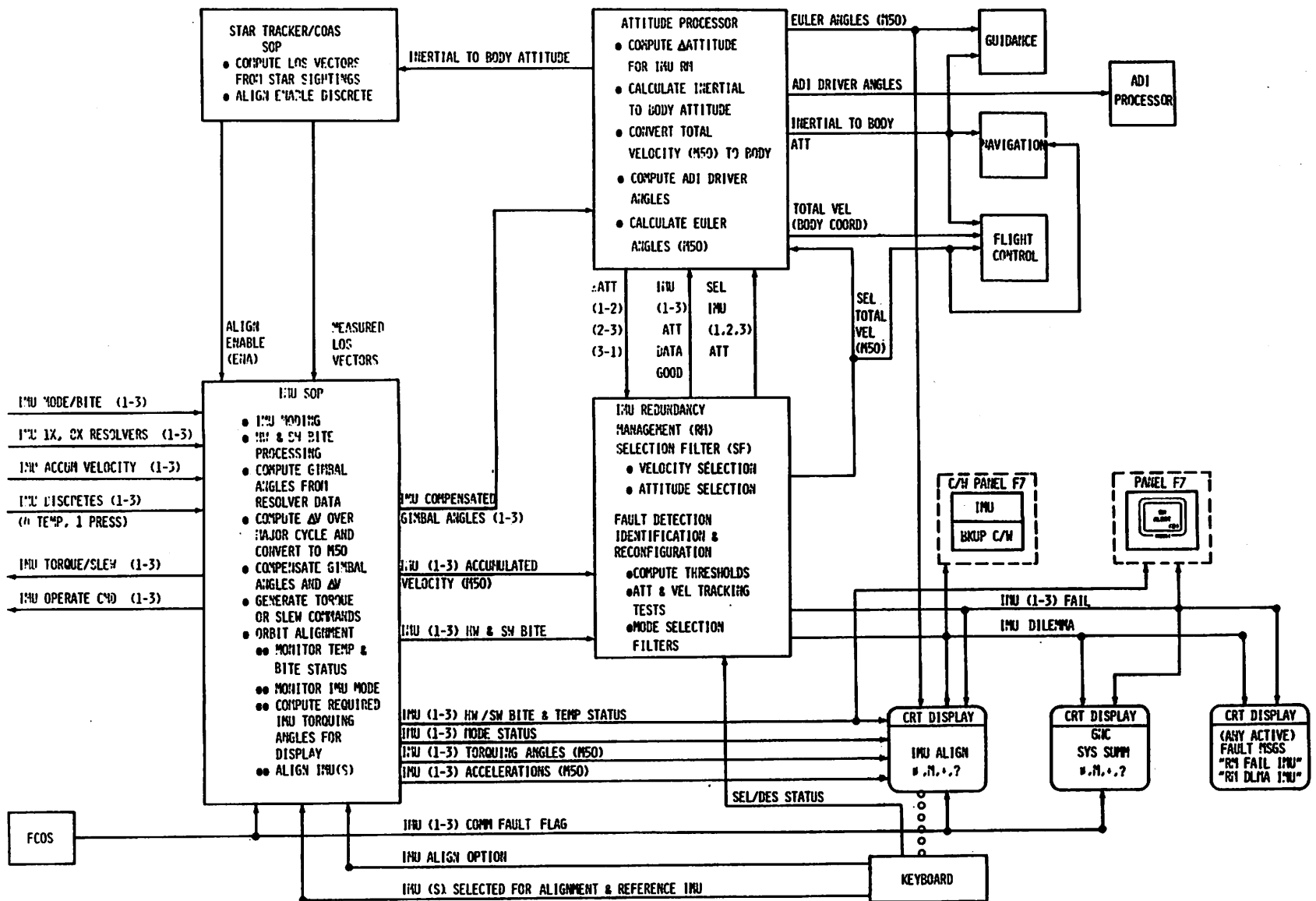


Figure 2.1-1a.- Orbit (GNC OPS 2 and MM 301) IMU data processing.

Velocity FDIR: The purpose of velocity FDIR is to determine which IMU (1, 2, or 3) should be declared failed for velocity purposes.

- Three good IMU's - FDIR compares each of the IMU relative velocity error differences (e_{ij}) to a threshold tolerance (k). Where e_{12} is a representation of the difference vector between total velocity vectors for IMU 1 and IMU 2, etc. FDIR will declare velocity failures on the following bases:

IMU 1 failed if $e_{12} > k$ and $e_{13} > k$
IMU 2 failed if $e_{12} > k$ and $e_{23} > k$
IMU 3 failed if $e_{23} > k$ and $e_{13} > k$

In each case, the failure is declared after three consecutive mismatches.

- Two good IMU's - FDIR compares the IMU relative velocity vector difference (e_{ij}) to the threshold tolerance (k). If $e_{ij} > k$, a fault is detected, but identifying the faulty IMU presents a dilemma situation. If BITE indication is present for an IMU, that IMU is declared failed. If not, the dilemma still exists and identification of the fault proceeds into a scheme (skew logic) where each individual accelerometer (four accelerometers for two IMU's) is checked for bias error. If one accelerometer is found to be bad, that associated IMU is declared failed.
- One IMU - None.
- Zero IMU - None.

Attitude SF: The function of the attitude SF is to select an IMU for attitude purposes.

- Three good IMU's - The SF compares each of the IMU relative attitude vector differences (e_{ij}) to each other, where e_{12} is a representation of the difference vector between total attitude vectors for IMU 1 and IMU 2. The SF selects the middle value IMU, using the following scheme:

If $e_{12} > e_{13}$ and $e_{12} > e_{23}$,
then IMU 3 is selected.

If $e_{13} > e_{23}$ and $e_{13} > e_{12}$,
then IMU 2 is selected.

If $e_{23} > e_{13}$ and $e_{23} > e_{12}$,
then IMU 1 is selected.

- Two good IMU's - The SF chooses the last IMU selected if this IMU is a candidate; otherwise, the SF chooses the lowest numbered IMU. The attitude data good threshold is calculated and compared against the attitude error vector (e_{ij}). If the threshold is exceeded, the attitude data good flag is set and modes the attitude processor not to use IMU data until the attitude data good flag is set "good" again.

- One good IMU - The SF will pass that IMU's attitude data.
- Zero IMU - The selected attitude data will be set equal to its previous value and attitude data good flag will be set "bad".

Attitude FDIR: The purpose of attitude FDIR is to determine which IMU (1, 2, or 3) should be declared failed for attitude purposes.

- Three good IMU's - FDIR compares each of the IMU relative attitude vector differences to a threshold tolerance (k). Where e_{12} is a representation of the difference vector between total attitude vectors for IMU 1 and IMU 2, FDIR will declare attitude failures on the following basis:

IMU 1 failed if $e_{12} > k$ and $e_{13} > k$
 IMU 2 failed if $e_{12} > k$ and $e_{23} > k$
 IMU 3 failed if $e_{23} > k$ and $e_{13} > k$

In each case, the failure is declared after three consecutive mismatches.

- Two good IMU's - FDIR compares the IMU relative attitude vector difference (e_{ij}) to the threshold tolerance (k). If $e_{ij} > k$, a fault is detected but identifying the faulty IMU presents a dilemma situation. If BITE indication is present for an IMU, that IMU is declared failed. If not, the dilemma still exists and identification of the fault proceeds into a scheme (skew logic) where each individual gyro (four gyros for two IMU's) is checked for a drift misalignment. If one gyro is found to be bad, that associated IMU is declared failed.
- One good IMU - None.
- Zero IMU - None.

OPS 9 inertial measurement unit: The IMU SOP is primarily concerned with ground calibration and alignment of the three IMU's during preflight (GNC OPS 9). The SOP will also monitor IMU hardware BITE, communications fault, and some software BITE (fig. 2.1-2).

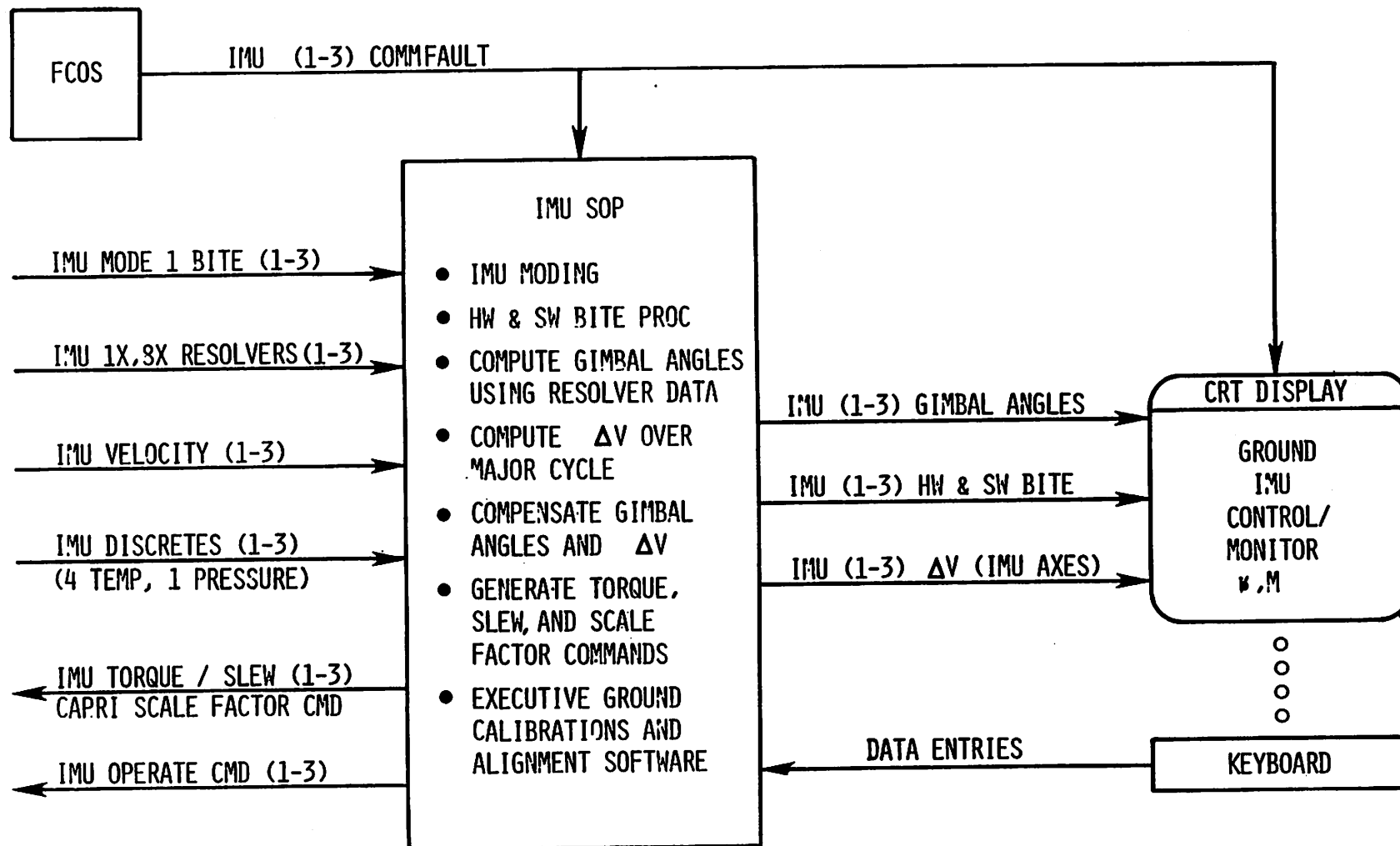


Figure 2.1-2.- Preflight(GNC OPS 9) IMU data processing.

During ground operations, the SOP processes attitude and velocity data. This data is used to generate calibration and alignment terms since there are no GNC users during GNC OPS 9.

Each IMU GNC 9 SOP submode can be described as follows:

- Terminate/Idle - This submode monitors IMU BITE and status discretes, sets torque and slew commands to zero, and sets Capacitance Reset Integrator (CAPRI) scale factors to low gain. TERM/IDLE is called by software whenever another submode is terminated or defaulted.
- Attitude Determination - This submode uses attitude and velocity data to determine the transformation from nav base to North/West/Up (NWU) for each IMU. This submode is required whenever the Orbiter has been moved on the ground while the IMU was not inertial. The correct value of the nav base to NWU transformation for the current navigation base attitude must be available prior to performing IMU calibration and alignment. It requires about four minutes and is done simultaneously in all operating IMU's. At completion of attitude determination, the TERM/IDLE default submode is automatically called.
- Hangar Calibration A - This submode torques each operating IMU to 13 cluster orientations and, using the measured acceleration and acceleration drift at each position, updates the accelerometer and gyro compensation parameters. All IMU's are commanded simultaneously and the submode takes \sim 5.5 hours. At completion, the TERM/IDLE submode is automatically called.
- Hangar Calibration B - Each IMU is torqued to four positions, with rates applied in each position. Comparison of resolver data with a reference IMU provides an update of gimbal angle offset errors, outer roll non-orthogonality errors, and gyro/accelerometer misalignment error terms. The calibration sequence is IMU 1-2-3; the reference IMU's are 3-3-1 respectively (a minimum of two IMU's are required). The time required is \sim 150 minutes. At completion, the TERM/IDLE submode is automatically called.
- Hangar Calibration C - This submode computes the IMU to IMU transformations for the selected launch orientation. Relative attitude error angles are computed for use in gyro compassing to make the gyro compass goodness test more sensitive. For three IMU's in operate mode, the pairwise calibration will be IMU's 1:2; 1:3; 2:3 respectively and takes \sim 3 minutes. At completion, the TERM/IDLE submode is automatically called.
- Preflight Calibration A - Each IMU is torqued to the seven cardinal positions to collect vertical acceleration and acceleration drift data. The terms updated are accelerometer biases and scale factors and gyro restraints. The gyro mass unbalance error terms are computed, but are not used to update the gyro mass unbalance compensation parameters. This submode will calibrate all IMU's (in the operate mode) that are selected and takes \sim 1.5 hours. At completion, the TERM/IDLE submode is automatically selected.

- Gyrocompass Alignment - This submode provides an accurate, defined cluster alignment for each IMU prior to platform release by updating/refining for each IMU a transformation relating NWU to an aligned cluster. Gyrocompass alignment will compute azimuth misalignment for each cluster, the relative azimuth errors between clusters, and a new cluster to reference transformation for each IMU. The relative azimuth misalignment is used in a reasonableness test performed with IMU pairs (1:2; 1:3; 2:3 nominally). At the end of this submode, the program will automatically advance to the velocity tilt submode.
- Velocity Tilt - This submode provides an estimation of tilts and drift between gyrocompass completion and platform release. This data is used to generate an updated cluster to M50 transformation (REFSMAT) for use during in-flight operation.

IMU backup flight system: The BFS will not control the IMU's during warm-up or runup. The BFS assumes that upon BFS engagement, the IMU's are in the operate mode; hence, the BFS is not required to initialize the IMU's. However, in the operate mode, the BFS will control the IMU's for inertial operation using primary system transferred IMU errors. The BFS IMU Hardware Interface Program (HIP) does no IMU moding; however, the BFS must issue operate commands upon engagement to all IMU's (fig. 2.1-3).

The BFS will assume IMU low gain CAPRI operation, and will therefore only command low gain operation.

All PASS IMU errors are compensated in the BFS software except for the gyro misalignment errors which were determined to be negligible.

The IMU HIP will monitor the hardware BITE and commfault for possible IMU downmoding. None of the IMU software BITE is processed by BFS.

For IMU RM in the BFS, there is no Fault Detection and Identification (FDI). The only protection provided is the IMU SF processing. Generally, the BFS will downmode for BITE and commfault's and generate the "RM FAIL IMU" class 3 message, tone, and down arrow ("↓") on the GNC SYS SUMM display.

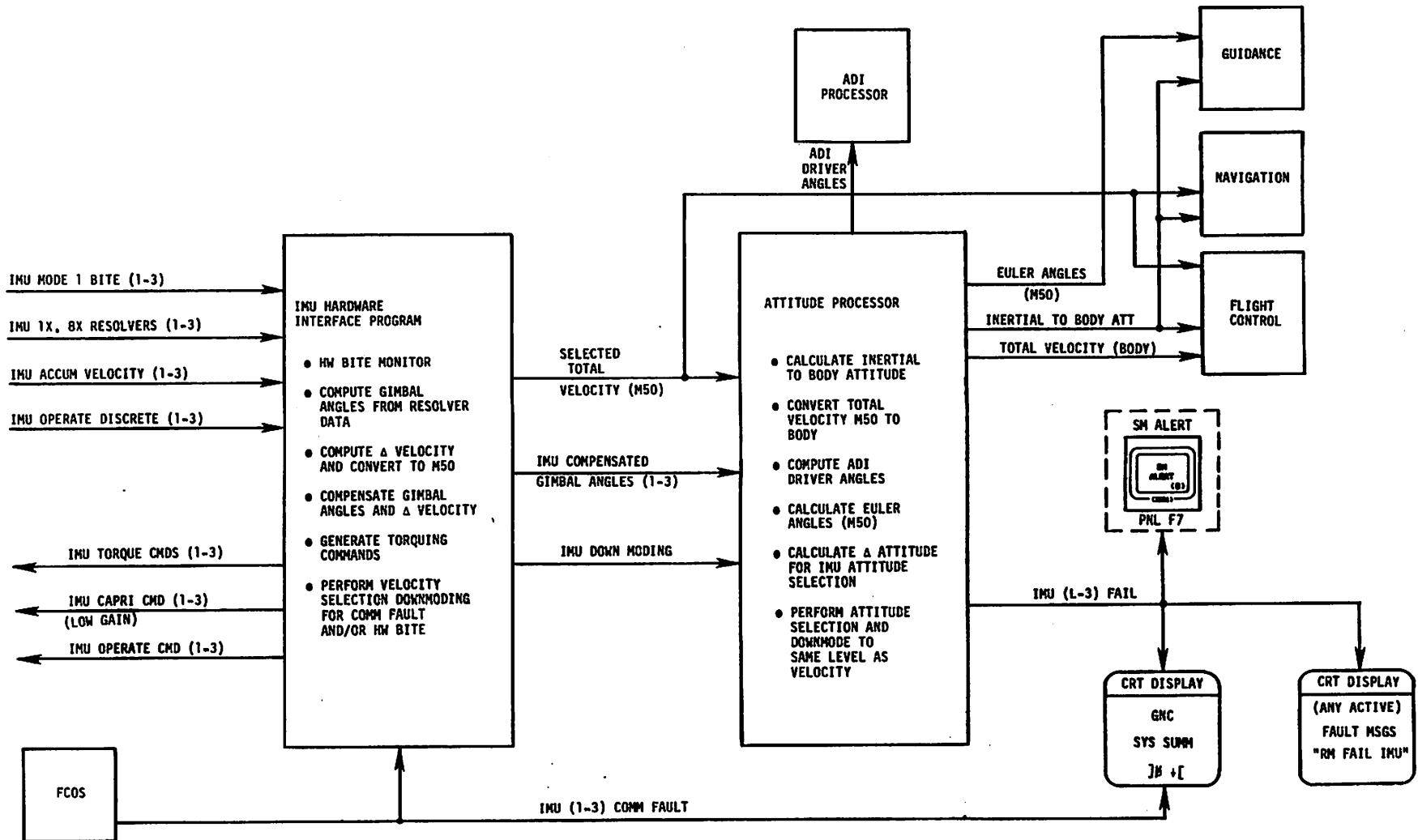


Figure 2.1-3.- BFS IMU data processing.

Specifically, the velocity data is mid-value selected at the three level, prime selected to the lowest numbered IMU at the two level, and prime selected at the one level. The attitude data is selected similar to the primary system at the three level. Any downmoding from the velocity SF will downmode the attitude SF accordingly. At the two level, the attitude SF will use the same IMU that the velocity SF is using. The attitude SF will pass the attitude data at the one level. Upmoding will occur upon detection of the absence of commfault's during a given cycle. There is no select/deselect capability in the BFS for IMU's. If all three IMU's have commfault's/BITE detected simultaneously, then IMU attitude processing will not be performed in BFS.

2.1.2 Orbiter Rate Gyro Assemblies

Subsystem operating program: The four Orbiter Rate Gyro Assemblies (RGA's) provide body rates for flight control stability augmentation and Attitude Direction Indicator (ADI) rate needle signals. The RGA SOP functions are unit conversion from voltages to deg/s and rate compensation. RGA self-test is carried out in OPS 9 (prelaunch) and OPS 8 (on-orbit). The RGA SOP sends RGA 1, 2, and 3 roll, pitch, and yaw rates to RM for selection filtering and fault identification. Roll, pitch, and yaw rates from RGA 4 are held in standby until RGA 1, 2, or 3 is declared failed (fig. 2.1-4).

Redundancy management: The status of each RGA can be found on the GNC SYS SUMM 1 display, which receives failure and commfault indications from FDIR.

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.

- Candidates are roll, pitch, and yaw rates from RGA 1, 2, and 3.
- FDIR will detect the first failure only. Once the failed RGA is identified, the failure will be annunciated through a fault message, "RM FAIL RGA", an SM alert light and tone, and status indication on GNC SYS SUMM 1 of a "+" on "M" to indicate which RGA has failed. Rates from RGA 4 will then be incorporated into the selection filter and FDIR will terminate.
- When all four RGA's are operating, RGA 4 rates are held in standby until RGA 1, 2, or 3 is commfaulted or deselected. For each parameter type (roll, pitch, and yaw rate), the following comparisons are made:

$$\begin{aligned} P_1 - P_2 &> LIM \\ P_1 - P_3 &> LIM \\ P_2 - P_3 &> LIM \end{aligned}$$

where LIM = SELECTED RATE • SCALE FACTOR + BIAS

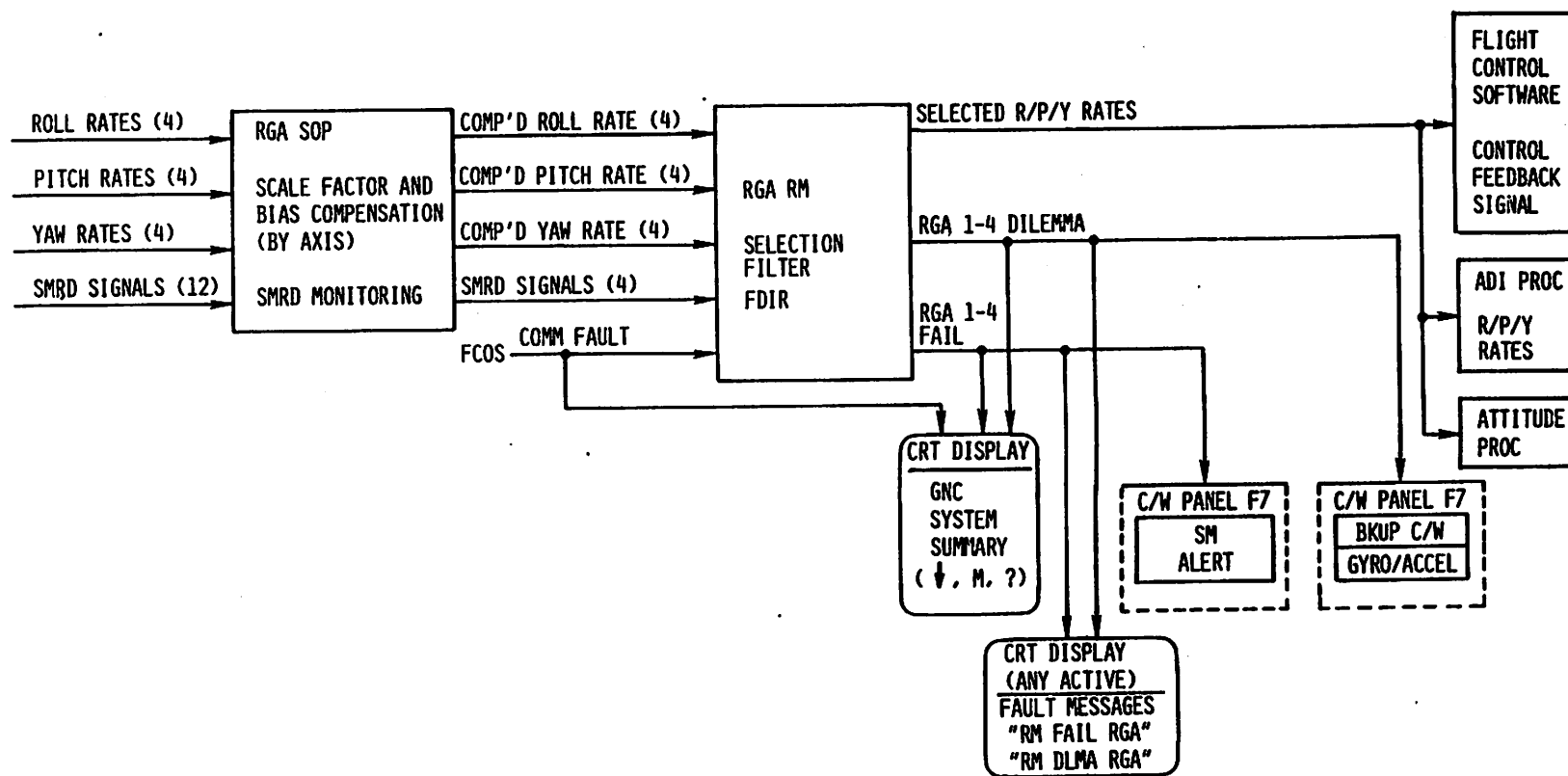


Figure 2.1-4.- Orbiter rate gyro data flow.

An RGA is declared failed if any parameter of that RGA is the common parameter in two miscomparisons for three consecutive times. FDIR substitutes the RGA 4 rates for the failed RGA and then FDIR terminates. No subsequent failures will be detected, only commfaults will downmode the selection filtering scheme.

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

Selection filter: The purpose of the RGA SF is to select single values of roll, pitch, and yaw rates to provide to users.

- Candidates are the rates from RGA 1, 2, and 3. RGA 4 rates are held in standby until RGA 1, 2, or 3 is commfaulted or declared failed by FDIR.
- Selection filter is downmoded for commfault. If the commfault disappears, the selection filter will upmode.

MVS - When three good rate sets are available, the SF will select the middle values of the three roll rates, three pitch rates, and three yaw rates. Subsequent to the first failure, the SF will always Middle Value Select (MVS) except for commfaults. No second or third failures will be detected.

AVE - When two good rate sets are available, the SF will average the two roll rates, two pitch rates, and two yaw rates. This mode is achieved only through commfaults.

PASS - When only one good rate set is available, the SF will pass the good roll, pitch, and yaw rates to the users. This mode is achieved only through commfaults.

OPS 8 RGA self-test procedures: Self-test for the Orbiter RGA's is carried out in OPS 8 using the SENSOR SELF-TEST SPEC 40. After the display has been called, vehicle rates are to be nulled using the RCS JETS VERN, and minimum accelerations are to be established by placing the vehicle in free drift (DAP MAN/PULSE mode). After the previous rate nulling has been accomplished, verify that the rates are within the limits listed below.

ROLL	< 0.02 deg/s
PITCH	< 0.02 deg/s
YAW	< 0.02 deg/s

Next, the hardware bias data needs to be verified by examining the sensed rates shown on the display. It should be within the limit listed below.

ROLL (R) ± .25 deg/s
PITCH (P) ± .165 deg/s
YAW (Y) ± .165 deg/s

If bias data are outside these limits, the option exists to deselect the appropriate hardware unit. Note that the hardware will be tested, regardless of whether a unit is selected or deselected.

After the above verifications have been accomplished, the high or low test is to be initiated. When the test has been completed, the data are checked to verify that they are within the proper limits (listed below). If the data are out of limits for either high or low tests, deselection of the appropriate RGA is recommended. If an RGA had been previously deselected and the test looks good, the option exists to reselect that RGA. When one test (high or low test) is finished, that test must be terminated by a keyboard item number entry before the other test is initiated. Status symbols will be displayed beside the appropriate data indicating any of the following.

↑ Data above limit
↓ Data below limit
M Data missing

A SMRD status indication will be displayed under the STAT column to indicate if a spin motor is not spinning synchronously.

OPS 9 prelaunch rate gyro assembly self-test: The display that will be used for prelaunch self-test for the RGA's is shown in figure 2.1-5. Items 7 through 10 enable an RGA for self-testing. Under test control, item 15 initiates self-testing with high values on the enabled sensors; item 16 initiates self-testing with low values on the enabled sensors; item 17 initiates self-testing with both high and low values on all sensors listed; item 18 terminates testing and inhibits all sensors; and item 19 inhibits all sensors with regard to self-testing. A SMRD indication will appear under the STAT column if the spin motor on any RGA is spinning unsynchronously. The limits used for on-orbit self-test are also used for the prelaunch self-tests.

9011/102/

RCS/RGA/ADTA TEST

3 021/11:26:37
000/00:01:19

RCS DRIVERS

RJDF	1	START	TEST	4	STAT
RJDA - A	2	CONT		5	JET
- B	3	TERM		6	

SENSORS

	ENA	STAT		R		P		Y
R 1	7							
G 2	8*	SMRD		+23.5	↑	+15.8		+08.7
A 3	9							
4	10							

TEST CONTROL:

HIGH	START	15
LOW	START	16
AUTO	SEQ	17
TERM		18*

INH ALL 19

	ENA	STAT		P α C		PS		P α U		P α L		TT
A 1	11			29.553		29.553		29.553		29.553		30.00
D 3	12			29.553		29.553		29.553		29.553		30.00
T 2	13			29.553		29.553		29.553		29.553		30.00
A 4	14			29.534		29.553		29.553		29.553		30.00

2-16

Figure 2.1-5.- RGA self-test SPEC.

BFS Orbiter RGA: The main functional module for the Orbiter rate gyro assembly backup flight system is the Orbiter RGA Hardware Interface Program (ORB RGA HIP). The ORB RGA HIP consists of two submodules that perform selection filtering and rate compensation. In the ORB RGA HIP, selection filtering is performed before rate compensation. In the primary system, rate compensation is performed before selection filtering.

Selection filtering in the ORB RGA HIP is performed using mid-value selection of the measured rate signals from RGA 1, 2, and 3 only. There is no capability for use of rates from RGA 4 in the selection filtering. One roll, one pitch, and one yaw rate are selected and sent for rate compensation. Mid-value select is the only mode of selection filtering used for ORB RGA HIP. There is no FDIR contained in the ORB RGA HIP. If failures or bad rates occur, mid-value selection is performed using the bad rates; for commfaults, the last value that was seen in compool will be used as the selection filter input for the commfaulted 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. The scale factor and bias values are I-loaded. The compensated rates are then output to the appropriate users.

2.1.3 Solid Rocket Booster Rate Gyro Assembly

SRB RGA SOP: The six Solid Rocket Booster (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. The SRB RGA SOP sends the compensated rates to RM for selection filtering and fault identification (fig. 2.1-6).

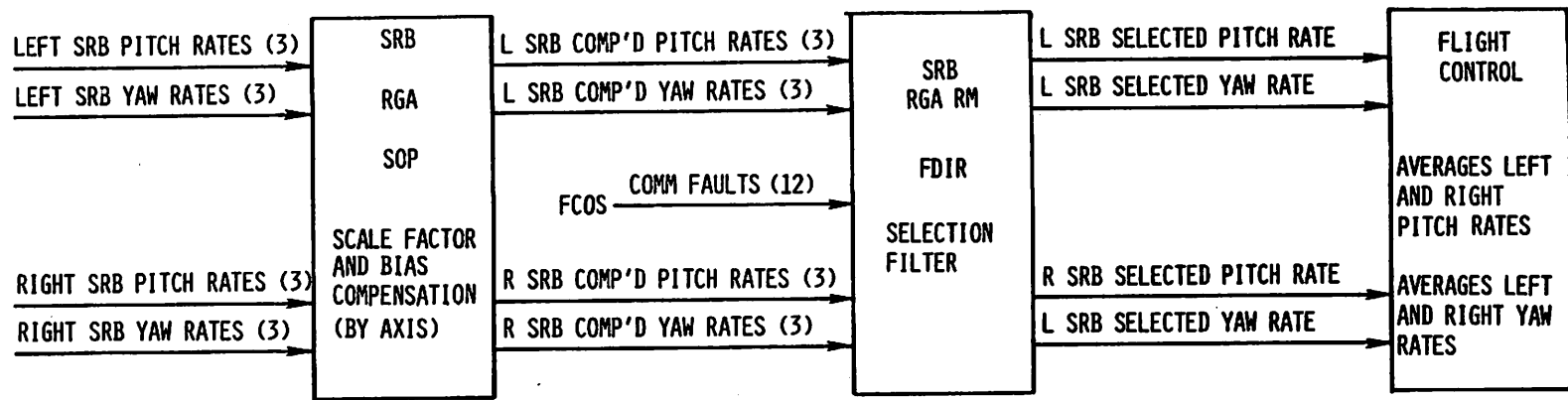


Figure 2.1-6.- SRB RGA data flow.

SRB RGA RM: SRB RGA RM consists of FDIR and an SF; however, there is no provision for fault annunciation to the crew. The SF will select a rate from the left SRB RGA's and a rate from the right SRB RGA's and flight control will average them.

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

$$\begin{aligned}P_1 - P_2 &> LIM \\P_1 - P_3 &> LIM \\P_2 - P_3 &> LIM\end{aligned}$$

where LIM = $\frac{\text{SELECTED}}{\text{L or R SRB}} \cdot \text{SCALE FACTOR} + \text{LIMIT CONSTANT}$
RATE

An SRB RGA is declared failed if any parameter of that RGA is the common parameter in two miscomparisons for three consecutive times. The selection filter will be notified which SRB RGA has been failed. After the first failure, FDIR terminates processing for the SRB (L or R) where the failure was detected. Subsequent failures for that particular SRB will not be detected. When one RGA is declared failed, a "fourth" set of rates will be incorporated. This "fourth" rate set will be the selected value for pitch and yaw, from the other SRB; e.g., if the left SRB has a failure, the selected pitch and yaw rates from the right SRB will replace the failed rates for use in left SRB selection filtering.

Selection filter: The SRB RGA SF shall be considered as a four input SF with an SF command indicating the usability status of its inputs. The fourth signal shall be ignored by the SF until one of the three SRB RGA signals is not usable. The selected SRB rate from the other side, about the same axis, shall be the fourth parameter and will be used only when one or more RGA parameters are not available. The fourth input, when active, shall not be declared unusable.

The SF will select rates through use of mid-value selection such that when three good rate sets are available, the SF will select the middle values of the three pitch and three yaw rates for the left SRB and right SRB. The SF will not downmode except for a commfault; in which case, the SF would average two good rate sets or pass one good rate set through to the users.

BFS SRB RGA: The main functional module for the SRB RGA BFS is the SRB RGA HIP. The SRB RGA HIP consists of two submodules that perform selection filtering and rate compensation. In the SRB RGA HIP, selection filtering is performed before rate compensation. In the primary system, rate compensation is performed before selection filtering.

The SRB RGA HIP selection filter performs mid-value selection only on the measured pitch and yaw rates from each SRB. Mid-value selection is the only mode of selection filtering available to the SRB RGA HIP as there is no FDIR available to the SRB RGA HIP. If failures or bad rates occur, mid-value selection would be performed using the bad rates; for a comm-fault, the last value that was seen in compool will be used as the selection filter input for the commfaulted RGA.

Selected rates are then processed through rate compensation which multiplies the selected mid-value 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.

2.1.4 Accelerometer Assembly

Subsystem operating program: The four Accelerometer Assemblies (AA's) provide normal and lateral body accelerations for load relief in ascent first stage, flight control stability augmentation in entry, and for ADI error needle signals during entry Major Modes (MM's) 304 and 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. 2.1-7).

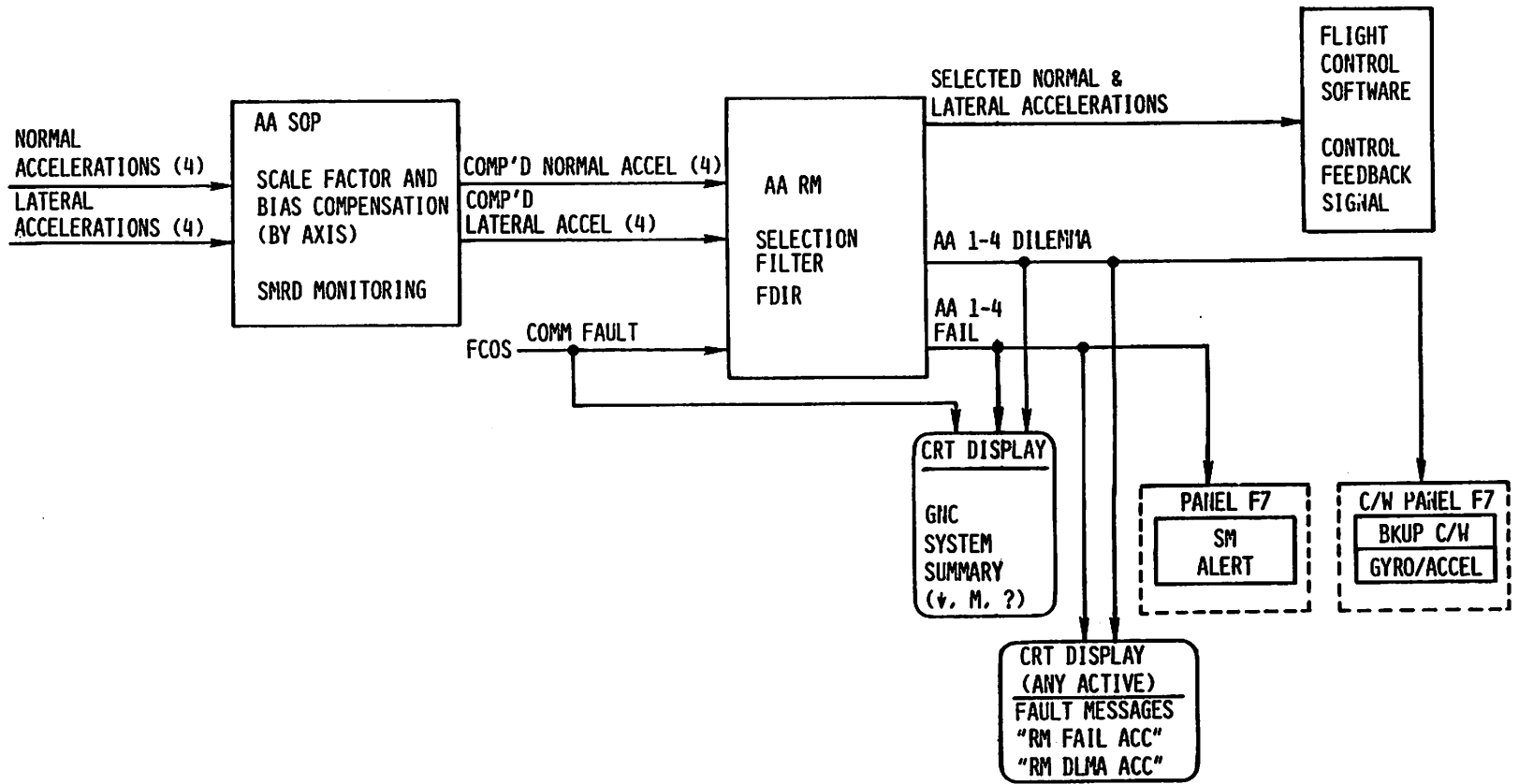


Figure 2.1-7.- Accelerometer assembly data flow.

RM: Status indications for the AA's are displayed on the GNC SYS SUMM 1, (for ascent and entry) which receives commfault and failure status from FDIR.

FDIR: The purpose of AA FDIR is to determine which AA should be declared failed.

- Candidates are normal and lateral accelerations from AA's 1, 2, or 3.
- FDIR will detect the first failure only. Once the failed AA is identified, the failure will be annunciated through a fault message, "RM FAIL ACC", an SM alert light and tone, and status indications on GNC SYS SUMM 1 of a "+" or "M" to indicate which AA has failed. Accelerations from AA 4 will then be incorporated into the selection filter and FDIR will terminate.
- FDIR downmoding is on an LRU basis. An AA that has been declared failed may not become a candidate for FDIR again unless it is reselected during OPS 8, self-test.
- Three good parameters sets - When all four AA's are operating, AA 4 accelerations are held in standby until AA 1, 2, or 3 is commfaulted or declared failed. For each parameter type (normal and lateral accelerations), the following comparisons are made:

$$\begin{aligned}A_1 - A_2 &> \text{CONSTANT} \\A_1 - A_3 &> \text{CONSTANT} \\A_2 - A_3 &> \text{CONSTANT}\end{aligned}$$

where the constant term is an I-loaded value.

An AA is declared failed if any parameter of that AA is the common parameter in two miscomparisons for three consecutive times. FDIR substitutes the AA 4 accelerations for the failed AA and FDIR terminates. No subsequent failures will be detected, only commfaults will downmode the selection filtering scheme.

- An AA is downmoded (withheld from FDIR, but not declared failed) for commfault. Should the commfault disappear, the AA will become an RM candidate again. Should a commfault occur before any failures, the commfault would be treated as a "first failure" by FDIR, such that FDIR would terminate upon recognition of that commfault. Should that commfault be resolved, FDIR would resume processing.

Selection filter: The purpose of the AA selection filter is to select single values of normal and lateral acceleration to provide to users.

- Candidates are the accelerations from AA 1, 2, and 3. AA 4 accelerations are held in standby until AA 1, 2, or 3 is commfaulted or declared failed by FDIR.
- The selection filter is downmoded for commfault. If the commfault disappears, the selection filter will upmode.

- MVS - When three good acceleration sets are available, the SF will select the middle values of the three good normal accelerations and three good lateral accelerations. Subsequent to the first failure, the SF will always MVS except for commfaults. No second or third failures will be detected.
- AVE - When two good acceleration sets are available, the SF will average the two normal and two lateral accelerations. This mode is achieved only through commfaults.
- PASS - When only one good acceleration set is available, the SF will pass the good normal and lateral accelerations to the users. This mode is achieved only through commfaults.

OPS 8 accelerometer assembly self-test: When the AA SELF-TEST command is initiated, an external signal is applied to the pendulum causing it to deflect a known amount and thus allowing light from the pickoff lamp to hit the photodiodes. The on-orbit sensor self-test (SENSOR TEST) CRT display will indicate the test results and test data. After the SENSOR TEST display has been called, procedures require that the vehicle rates must be nulled through use of the RCS JETS VERN and minimum accelerations are to be established by placing the vehicle in free drift (DAP MAN/PULSE mode). Vehicle rates must be verified according to the limits listed below:

ROLL < 0.02 deg/s
 PITCH < 0.02 deg/s
 YAW < 0.02 deg/s

The next step is to verify that hardware bias data are within the limits listed below.

LATERAL Y $\pm 0.9 \text{ ft/s}^2$
 NORMAL Z $\pm 0.34 \text{ ft/s}^2$

If bias data is not within the listed limits, the appropriate AA is to be deselected. Initiators of the sensor test are next. When the sensor test is complete, the self-test limits are to be checked.

	LOW LIMIT	HIGH LIMIT
LATERAL Y	11.43 ft/s^2	20.73 ft/s^2
NORMAL Z	45.90 ft/s^2	82.80 ft/s^2

Status indicators will show whether test results are above (†) or below (‡) limits or if the data are missing (M). Deselection of the appropriate AA is recommended if the test results are above, below, or missing. If the self-test data looks good for an AA that had previously been deselected, the option exists to reselect the AA. When satisfied with the self-test results, terminate the sensor test (fig. 2.1-7a).

OPS 9 prelaunch sensor test: The display used for self-testing the AA's (and some other sensors) during OPS 9, prelaunch, are shown in figure 2.1-8. Self-test information for the accelerometers is at the bottom of the display.

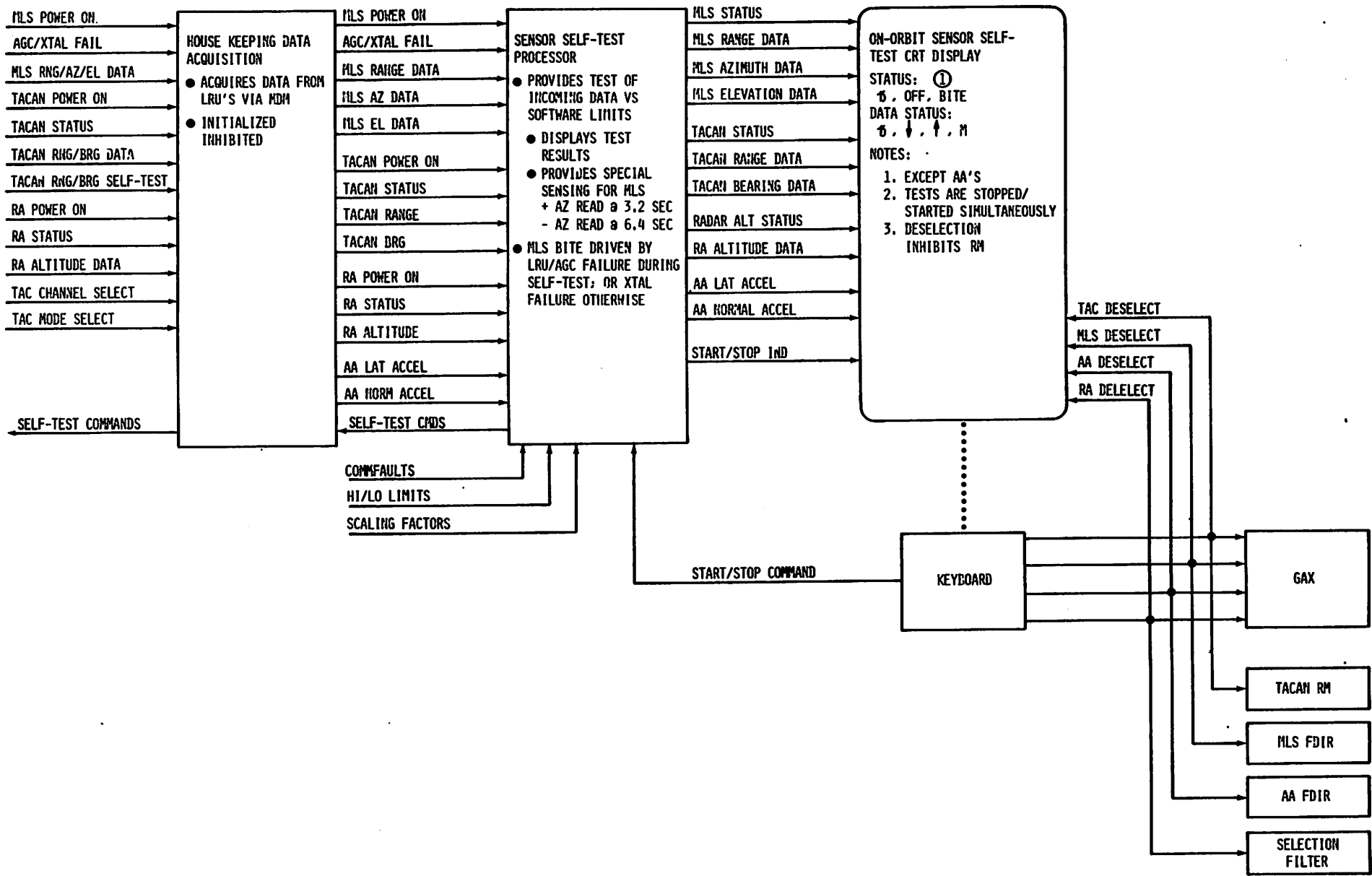


Figure 2.1-7a.- On-orbit sensor self-test (GNC OPS 8).

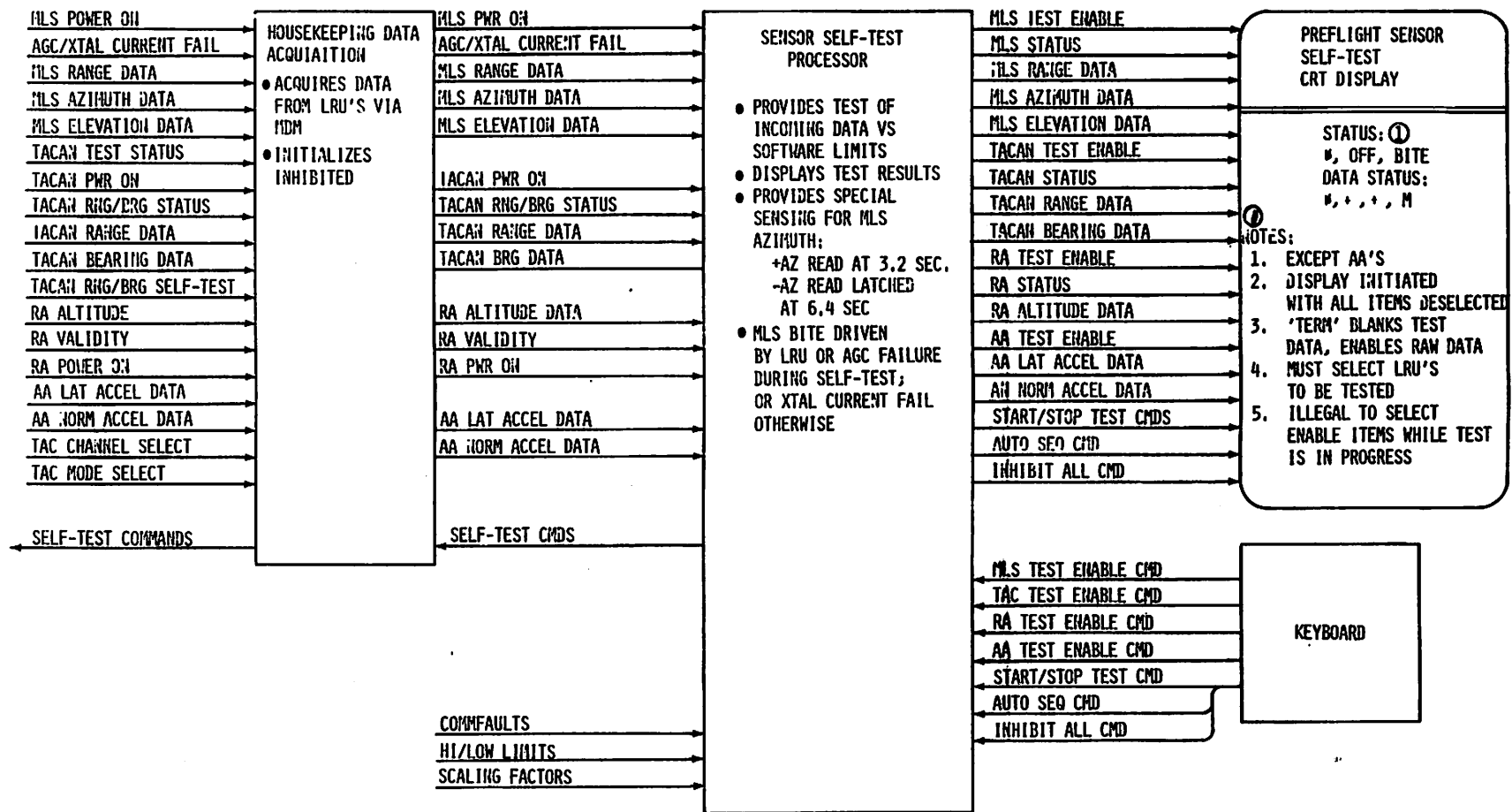


Figure 2.1-8.- Preflight sensor self-test.

Prelaunch self-testing requires that the particular unit to be tested must be selected or enabled by item number entry. For example, if one wanted to test AA 2, AA 3, and AA 4, the AA's would have to be enabled by key-in:

ITEM 10 EXEC
ITEM 11 EXEC
ITEM 12 EXEC

If a unit is enabled, an * will appear next to the item number.

To initiate the sensor self-test on the enabled sensors, the START command (under test control) must be executed by item number entry. The auto sequence (AUTO SEQ) command initiates self-testing on all sensors, whether enabled or not. The terminate (TERM) command terminates testing and inhibits all of the sensors. The inhibit all (INH ALL) command inhibits all of the listed sensors with regard to self-testing. Limits to be checked for prelaunch self-test are the same as those for on-orbit self-test.

BFS accelerometer assembly: The main functional module for the AA BFS is the ORB AA HIP. The ORB AA HIP consists of two submodules that perform selection filtering and acceleration compensation. In the ORB AA HIP, selection filtering is performed before acceleration compensation; in the primary system, acceleration compensation is performed before selection filtering.

Selection filtering in the ORB AA HIP is performed using mid-value selection of the measured accelerations from AA 1, 2, and 3. There is no capability for use of accelerations from AA 4 in the selection filtering. One lateral and one normal acceleration are selected and sent for acceleration compensation. Mid-value select is the only mode of selection filtering used for the ORB AA HIP. There is no FDIR contained in the ORB AA HIP. If failures or bad accelerations occur, mid-value selection is performed using the bad accelerations; for commfaults, the last value that was seen in commpool will be used as the selection filter input for the commfaulted 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.

2.1.5 Controllers/Switches

Three-axis RHC SOP: During MM 102 through 106, 301 through 303, and 601, the three-axis Rotational Hand Controller (RHC) SOP performs the following functions for the left (CDR) and right (PLT) RHC's (fig. 2.1-9).

- Changes the RHC voltage output to degrees.
- Outputs zero commands for all three axes of an RHC if any one of the pitch, roll, or yaw data good flags is "bad".

- Applies a deadband to pitch, roll, and yaw command signals for each RHC.
- Generates an auto to manual downmode signal by axis for flight control when an RHC deflection exceeds a predetermined detent threshold in an axis. (While the ascent Digital Autopilot (DAP) is active, a Control Stick Steering Pushbutton Indicator (CSS PBI) must be depressed for manual inputs to be implemented in flight control. Downmoding occurs for all axes.)
- Generates a software softstop indicator by axis for flight control to change from a rate command mode to an acceleration command mode when an RHC deflection exceeds the software softstop in an axis (only affects Reaction Control System (RCS) jets - no effect on Orbital Maneuvering System (OMS).
- Sums the deadbanded commands by axis from both RHC's to generate single manual roll, pitch, and yaw commands.
- For the aft station RHC, transforms the aft RHC outputs based on the sense switch position prior to any station selection (table 2.1-1).

TABLE 2.1-1.- AFT RHC OUTPUTS

Aft RHC inputs (RHC axis)	Transformed aft RHC outputs (to body axis)	
	-X sense	-Z sense
+Roll	-Roll	-Yaw
-Roll	+Roll	+Yaw
+Pitch	-Pitch	-Pitch
-Pitch	+Pitch	+Pitch
+Yaw	+Yaw	-Roll
-Yaw	-Yaw	+Roll

2 axis RHC SOP: During MM 304, 305, 602, and 603, the two axis RHC SOP performs the above functions for the left (CDR) and right (PLT) RHC's for pitch and roll commands only.

SBTC SOP: During MM 102, 103, and 601, the Speed Brake Thrust Controller (SBTC) SOP performs the following functions for main engine throttling (fig. 2.1-10).

- Converts the pilot's compensated manual throttle command to a manual throttle setting in percent for guidance.

- Provides AUTO and MANUAL flags to illuminate both SBTC AUTO PBI's and the pilot's MAN PBI.

During MM 304, 305, 602, and 603, the Speed Brake/Thrust Controller (SBTC) SOP performs the following functions for manual speed brake control.

- Converts the selected manual speed brake command from volts to degrees for flight control.
- Provides takeover latch flags for flight control to identify which crewman is presently in control.
- Performs station selection for manual speed brake 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 data good 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.

RPTA SOP: During MM 304, 305, 602, and 603 the RPTA SOP performs the following functions (fig. 2.1-10):

- Converts the selected transducer commands from volts to degrees.
- Performs station selection by selecting either the right or left Rudder Pedal Transducer Assembly (RPTA) signal provided the associated data good flag is good, and the command has the largest magnitude of the two. If both data good flags are bad, the selected signal shall be set at zero.
- Applies a breakout deadband to the station selected command.

Translational Hand Controller (THC) processing: THC processing begins post Main Engine Cutoff (MECO) and continues through MM 303 (fig. 2.1-11).

- Monitors the sense switch position and transforms the aft station THC commands from the -X or -Z sense to the forward station sense (table 2.1-2).

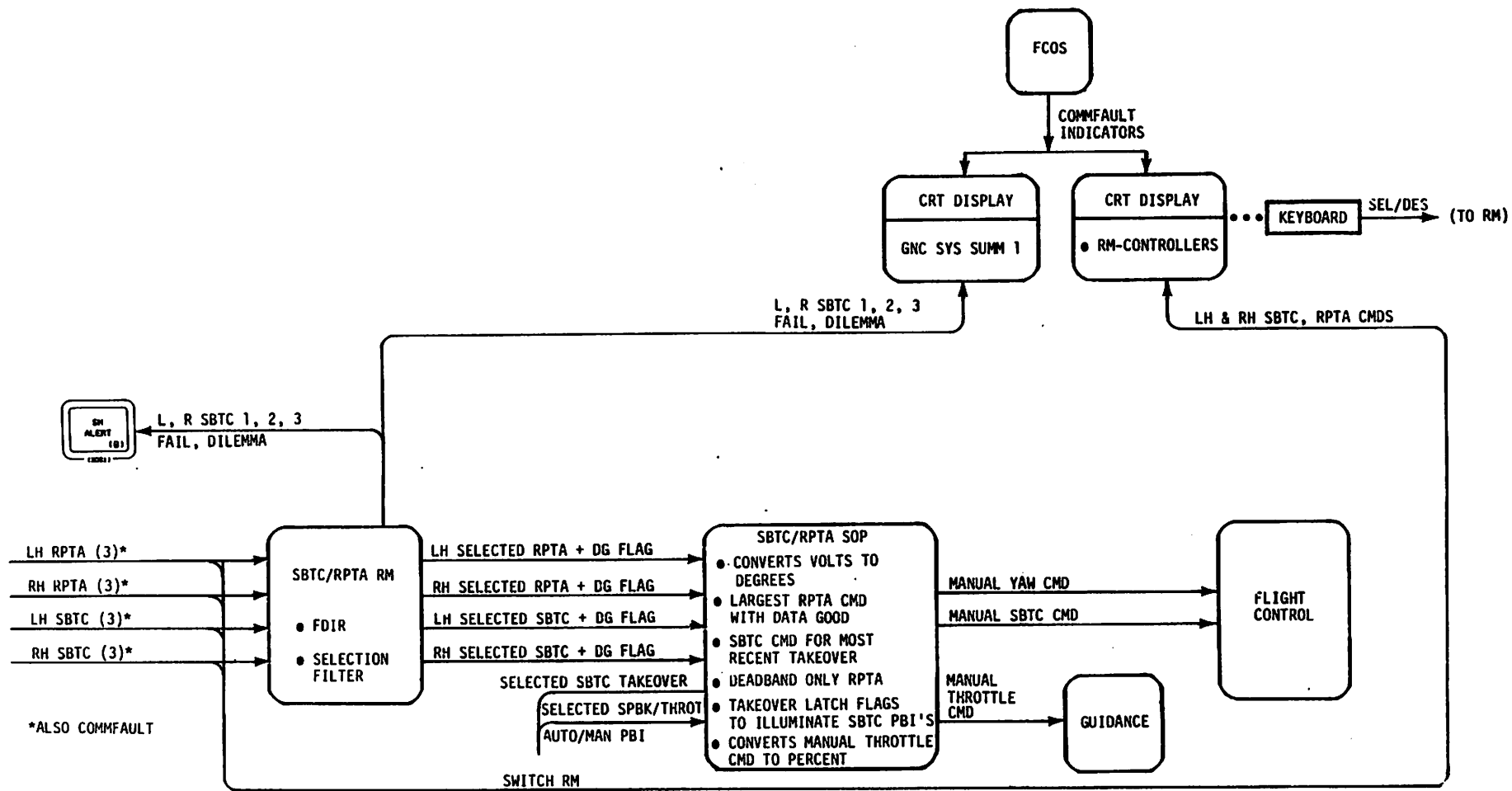
TABLE 2.1-2.- AFT THC OUTPUTS

Aft RHC inputs (THC axis)	Transformed aft THC outputs (to body axis)	
	-X sense	-Z sense
+X (in)	-X	-Z
-X (out)	+X	+Z
+Y (rt)	-Y	-Y
-Y (lt)	+Y	+Y
+Z (dn)	+Z	-X
-Z (up)	-Z	+X

- Generates a resultant positive signal if a forward THC positive signal or a transformed aft THC positive signal is present.
- Generates a resultant negative signal if a forward THC negative signal or a transformed aft THC negative signal is present.
- Outputs a zero command when both crew stations (CDR and aft) generate conflicting translation commands in an axis.

Checksum SOP: During all flight phases, a checksum word based on certain General Purpose Computer (GPC) output signals is generated by the checksum SOP. The checksum word is used by system software to determine the integrity of the flight software.

The checksum word is generated by summing predetermined parameters whenever the principal function of their origin is active. At the beginning of each process cycle, the checksum word shall be nulled. The checksum word has the least significant bit tolerance, and a disagreement for three consecutive times in even the least significant bit of a GPC sum word causes crew annunciation to occur.



Note: RPTA processing does not apply in UPS 1.

Figure 2.1-10.- SBTC/RPTA SOP data flow.

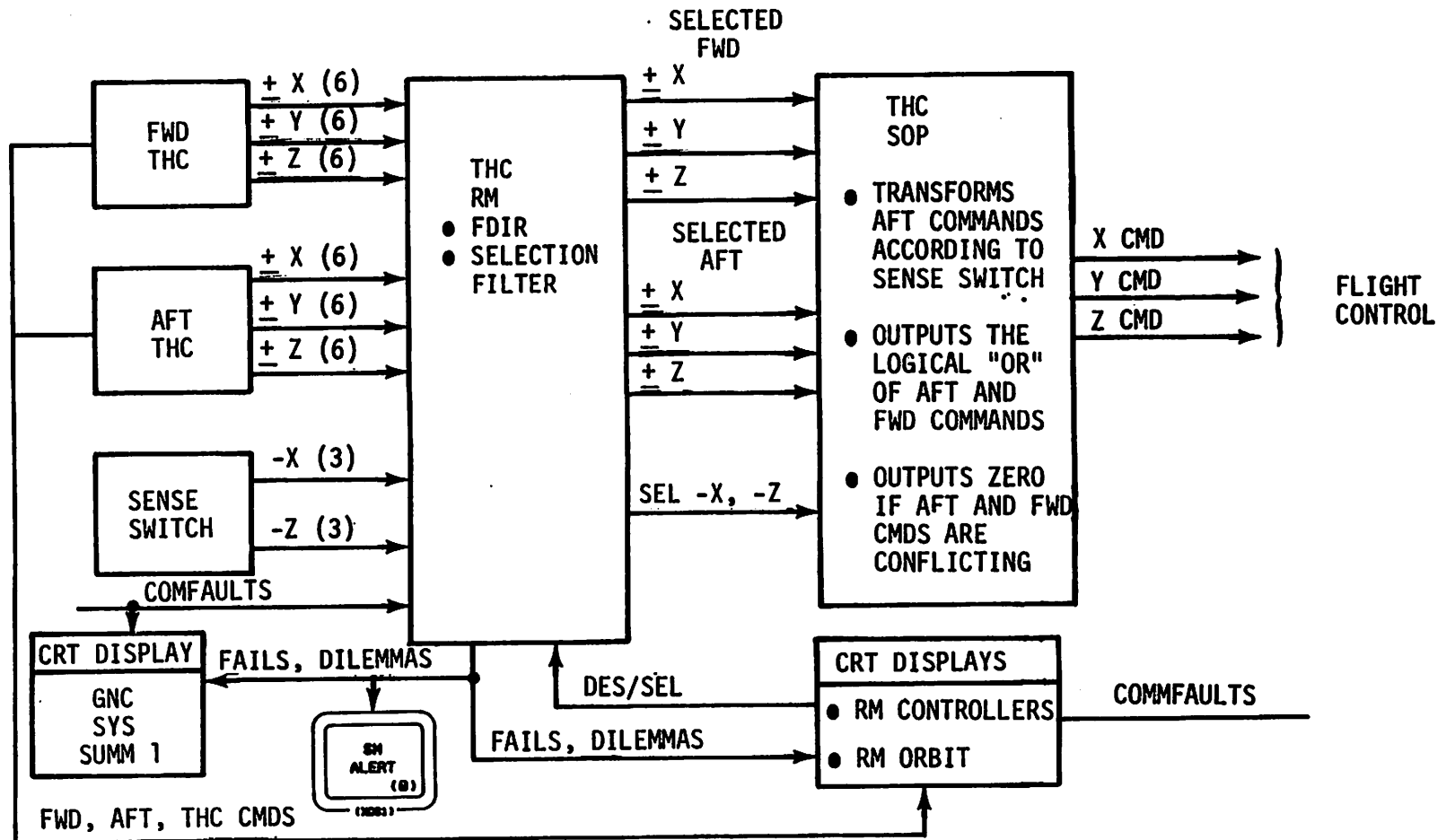


Figure 2.1-11.- THC SOP data flow.

Flight control switch SOP: During OPS 2 and 8, the flight control switch SOP performs the following functions (fig. 2.1-12):

- Performs station selection (the logical "OR") of the forward and aft orbital DAP switches and the manual mode translation and rotation switches.
- Transforms the aft manual mode rotational switches according to the position of the aft sense switch.

TABLE 2.1-3.- AFT ROTATION SENSES

Aft mode inputs*	Transformed aft mode outputs	
	-X sense	-Z sense
Roll	Roll	Yaw
Pitch	Pitch	Pitch
Yaw	Yaw	Roll

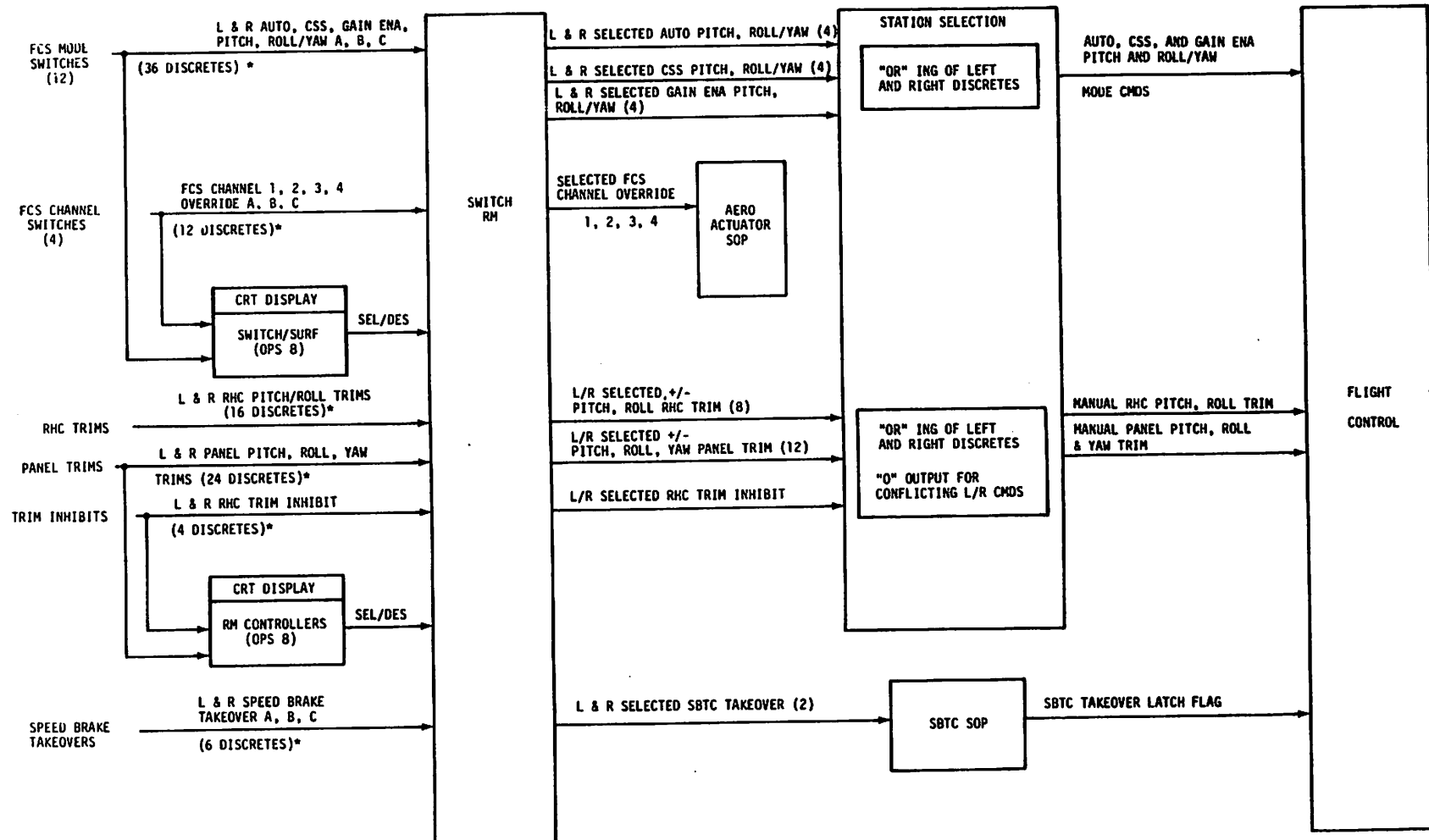
*For each of the discrete, accel, and pulse modes.

- According to the position of the sense switch, transforms the manual mode rotational PBI commands from the on-orbit DAP which are used to illuminate the aft rotational PBI's.

TABLE 2.1-4.- AFT DAP PANEL SENSES

Aft PBI inputs*	Transformed aft PBI outputs*	
	-X sense	-Z sense
Roll	Roll	Yaw
Pitch	Pitch	Pitch
Yaw	Yaw	Roll

* For each of the discrete, accel, and pulse modes.



* AND COMFAULTS

Figure 2.1-12.- FCS switch processing.

Controller RM: Controller RM receives triple redundant signals from the left, right, and aft controllers, and provides the controller SOP with the selected signal. RM receives the following controller signals:

- Left, right, and aft RHC pitch commands A, B, C
- Left, right, and aft RHC roll commands A, B, C
- Left, right, and aft RHC yaw commands A, B, C
- Left and right SBTC commands A, B, C
- Left and right RPTA commands A, B, C

Three Good Signals (Nominal Case)

The middle valued signal is selected in the selection filter for output to the SOP. For the three level RM, the data good indicator is always set good. The FDIR subtracts pairs of signals and compares the results to an I-load limit:

$$|A-B| < I\text{-LOAD}, |B-C| < I\text{-LOAD}, |C-A| < I\text{-LOAD}$$

A signal is declared failed by the FDIR if it differs from the other two signals by more than the I-load limit for three consecutive times.

First Failure (Two Good Signals)

When a signal is declared failed by the FDIR, it is deselected from the selection filter and the FDIR. Fault annunciation occurs for failed RHC and SBTC signals. (For an RHC, a failure in one axis causes the comparable signals in all axes of that RHC to be deselected from the selection filter and FDIR.) RM is downmoded to the two level and the two signals are averaged and output to the SOP. The data good test is implemented by subtracting the two signals and comparing the results with an I-load limit. The data good flag is set good if the data good limit is not exceeded. The FDIR continues to subtract the two signals and compare the results to an I-load limit. If the limit is exceeded for three consecutive times, a dilemma condition is annunciated for RHC and SBTC signals. (This is the second failure.) A failure caused by a commfault will also downmode RM, but if the commfault clears, RM will upmode back to the previous level.

Second Failure (One Good Signal)

A second failure caused by a bad signal will not cause RM to downmode, but will result in the selection filter averaging the remaining good signal with the bad signal for output to the SOP. However, if the data good limit is exceeded, the data good flag will be set bad and all signals from that controller will be set to zero by the SOP. Fault annunciation occurs for RHC and SBTC dilemma conditions.

A second failure caused by a commfault will result in the selection filter and the FDIR downmoding to the one level. The remaining signal (good or bad) is passed through RM with no processing. The data good flag is set good for one level RM. If the commfault clears, RM will upmode to the two level.

Third Failure (No Good Signals)

If RM is at the two level:

- A third failure by a commfault downmodes the selection filter to the one level and the remaining signal (good or failed) is passed through RM with no processing. The data good flag is set good. If the comm-fault clears, RM will upmode to the two level.
- A third failure caused by the remaining good signal failing will not change RM and the selection filter will average the two failed signals. The data good flag will be set good until the data good limit is exceeded.

If RM is at the one level:

- A third failure by a commfault downmodes the selection filter and FDIR to the zero level. The data good flag is set bad. The SOP sends zeros to the flight control system for the controller output. If the comm-fault clears, RM upmodes to the one level.
- A third failure caused by the remaining signal failing will not affect RM. The signal is passed through RM with no processing. The data good flag is set good.

THC RM: RM is performed on the six positions of the left and aft THC's (Fig. 2.1-13). Each position (+X, -X, +Y, -Y, +Z, -Z) has three contacts and is analogous to a three contact switch. (RM writeup follows.) THC failures are annunciated for the crew.

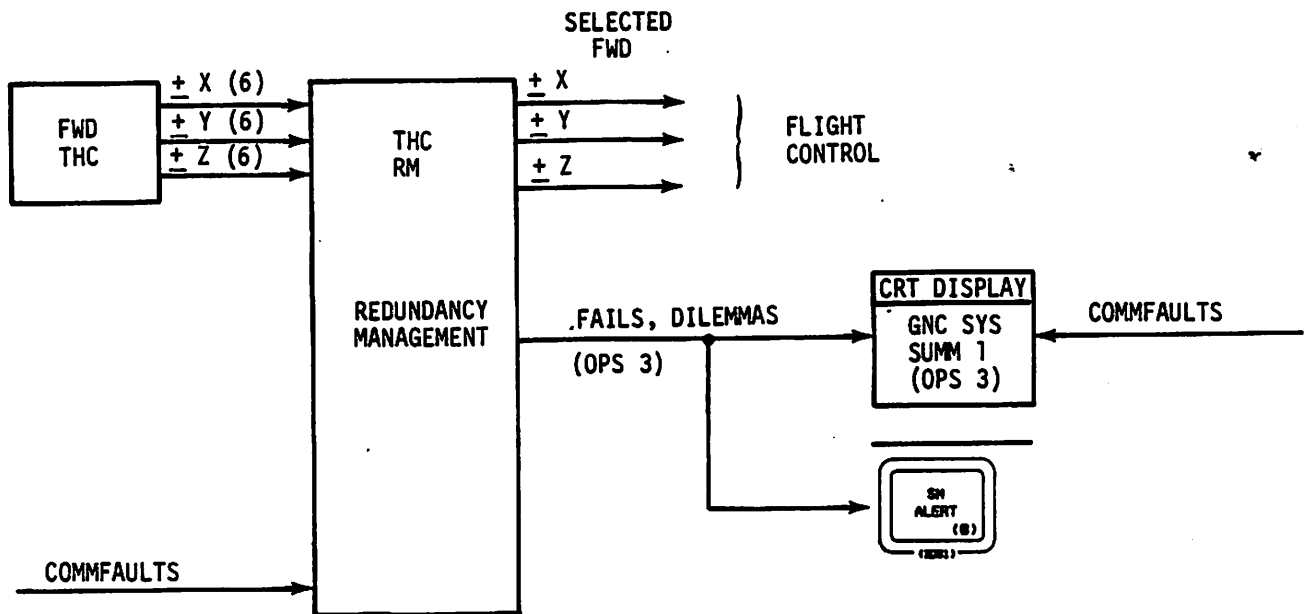


Figure 2.1-13.- THC data processing.

GNC switch RM: Switch RM monitors switch contacts and determines the outputs for certain switch positions. An example is shown in figure 2.1-14.

- Three Contacts Available - The selection filter output is the majority vote of the three contacts. That is, two of the three contacts must be closed (made) for the switch to be in the monitored position. The FDIR compares all three contacts and declares a contact failed if it differs from the other two contacts for three consecutive FDIR cycles. A failed contact is deselected from the selection filter and the FDIR, and thus downmodes RM to the two contact level.
- Two Contacts Available - Both contacts must be closed (made) for the selection filter output to indicate that the switch is in the monitored position. The FDIR compares the two contacts and declares a dilemma if they miscompare (one closed and one open) for three consecutive FDIR cycles.
- One Contact Available - The selection filter output is the same as the single contact input for the monitored position. There is no FDIR processing at the one contact level.

Commfaults downmode the selection filter and FDIR for as long as the commfault is present. If the commfault goes away, the selection filter and FDIR upmode to the next higher level.

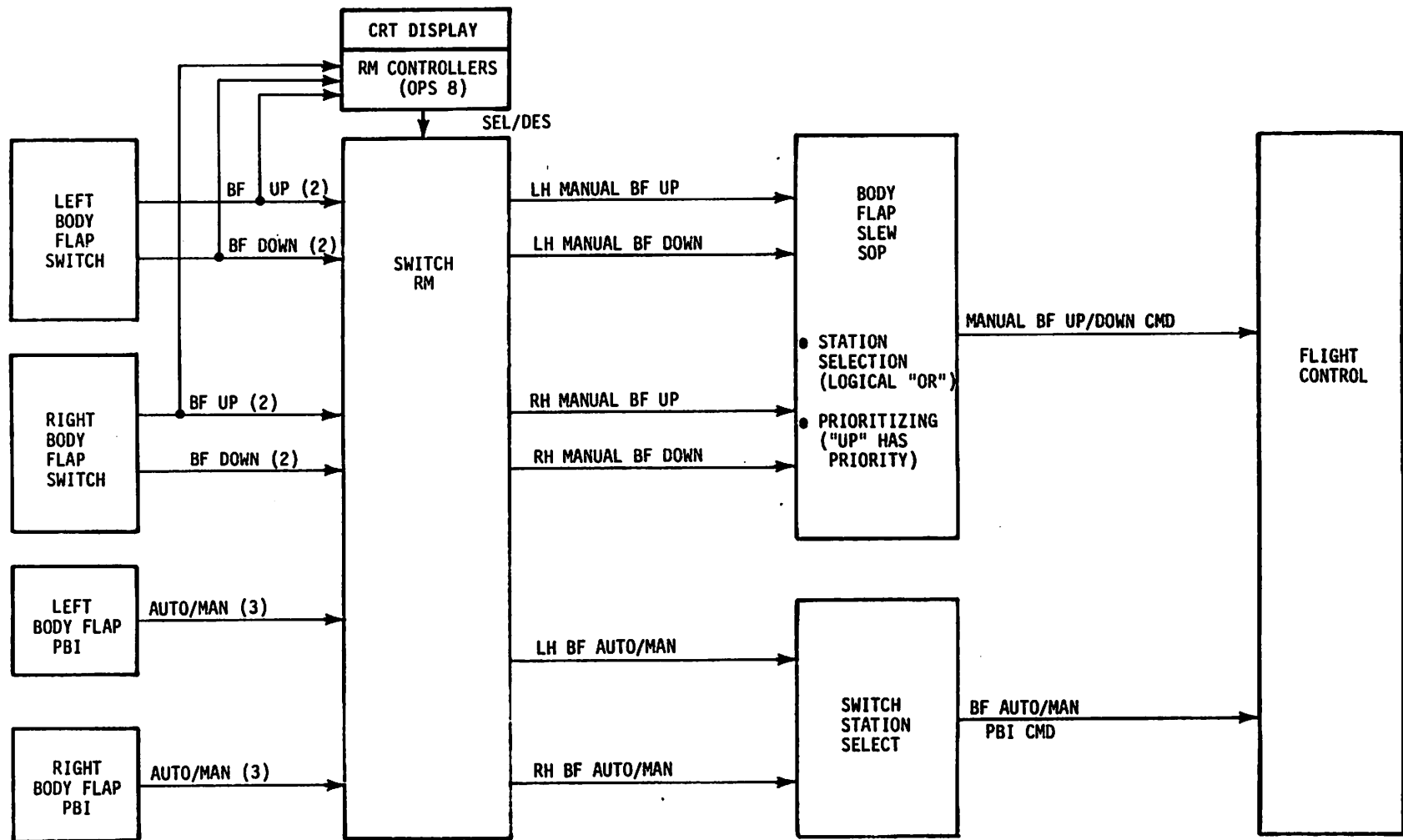


Figure 2.1-14.- Body flap switch processing.

2.1.6 Aerosurface Actuators

The aerosurface hardware data processing modules and system interface are shown in figures 2.1-15, 2.1-16, 2.1-17, 2.1-18, 2.1-19. The aeroactuator command SOP, the primary delta pressure feedback SOP, primary delta pressure feedback RM, body flap command RM, four position feedback SOP's, and position feedback RM are the modules that are used to perform the interface. They are described below.

Aero actuator command SOP: Processing of flight control commands for the Aerosurface Servo Amplifiers (ASA's) is performed by the aero actuator SOP which accomplishes the following functions.

- Changes the aerosurface position commands from degrees to volts for the left inboard elevon, left outboard elevon, right inboard elevon, right outboard elevon, speed brake, and rudder actuators.
- Changes the flight control generated nosewheel steering command from degrees to volts and applies compensation factors.
- Converts body flap commands to enable, up, and down solenoid commands.
- Processes bypass commands accomplished through GPC read/writes and override commands from the Flight Control System (FCS) channel switches (located on panel C3) for the elevon, rudder, and speed brake.

Elevon primary delta pressure feedback SOP: During MM 102, the elevon primary delta pressures are used by flight control for load relief indications. The elevon delta pressure feedback SOP converts the RM-selected primary delta pressures for the left and right inboard and outboard elevons from volts to psi for use by the ascent DAP.

Elevon primary delta pressure feedback RM: During MM 102, the elevon primary delta pressures are used by flight control for load relief indications. The elevon delta pressure feedback RM receives four redundant delta pressure feedback signals from each of the elevons and provides the SOP with a selected delta pressure feedback for each elevon.

There are four levels of RM which process signals as follows:

- Nominal three level RM (mid-value select) is performed for the four good input case with the fourth signal held in standby, and for the first failure case where the failed signal is replaced with the standby fourth signal. This is done for all elevons on a Line Replaceable Unit (LRU) basis.
- Two level RM is performed after the second failure occurs. The two values are averaged.
- One level RM is performed when a commfault occurs on an Multiplexer/Demultiplexer (MDM) associated with one of the two existing signals. The remaining signal is merely passed through the selection filter.

- Zero level RM occurs when a commfault exists on the MDM associated with the final signal. At this level, the last valid signal is continually output by the selection filter.

Body flap command RM: Body flap command RM enables and disables the body flap pilot valves based on crew selection of individual pilot valves. (Nominally, the crew selects all three pilot valves.) The change in position of the body flap and the direction of motion or lack of motion is computed based on the body flap position feedbacks.

If the body flap is not being commanded to move by the flight control system, but movement of the body flap is detected (body flap position change is greater than 1.5 degrees for an accumulated error count of six), RM sets a hold on channel 2. That is, channel 2 is commanded in the opposite direction of the detected body flap motion. The hold on channel 2 is removed if the body flap motion is not stopped, and a hold is then set on channel 1. If the motion continues, all holds are removed for six cycles and then alternately applied to channels 2 and 1 as RM continues to attempt to isolate the failure. Annunciation will occur to alert the crew of the problem.

If the body flap is being commanded to move by flight control system, but the direction of body flap motion is not following the flight control command for a rate count of three, RM checks for a hold on pilot valves 1 or 2. If no HOLD's are set, crew annunciation occurs. If a HOLD does exist, the direction of the HOLD is compared to the direction of body flap motion. If it is the same, the HOLD is removed. If the HOLD is opposite to the direction the body flap is moving, crew annunciation occurs.

Position feedback SOP's: The four position feedback SOP's are the elevon POS FDBK SOP, the rudder POS FDBK SOP, the speed brake POS FDBK SOP, and the body flap POS FDBK SOP. These SOP's receive RM selected position feedbacks and perform the following functions:

- Elevon
 - Changes the selected elevon feedback data from volts to degrees for flight control and the crew interfaces.
 - Computes the elevator position for flight control by averaging the selected elevon position feedbacks for the four elevon panels.
- Rudder
 - Changes the selected rudder feedback data from volts to degrees for flight control and the crew interfaces.
- Speed Brake
 - Changes the selected speed brake feedback data from volts to degrees for flight control and the crew interfaces.
 - Computes the percentage of maximum speed brake position for crew displays.
- Body Flap
 - Changes the selected body flap feedback data from volts to degrees for flight control and the crew interfaces.

Aerosurface position feedback RM: Position feedback RM receives four redundant actuator position feedback signals from the elevons (four panels), rudder, speed brake, and body flap; and provides the position feedback SOP with a selected actuator position feedback.

Four Good Actuator Position Feedback Signals (Nominal Case)

Channel 4 is held in standby, and the middle-valued signal is selected in the selection filter from the remaining three signals for each actuator. The FDIR subtracts pairs of feedback signals and compares the results to an I-load limit.

$$|PFB1-PFB2| < I-LOAD, |PFB2-PFB3| < I-LOAD, |PFB3-PFB1| < I-LOAD$$

First Failure (Three Good Signals)

When a signal is declared failed by the FDIR because it differs from the other two by more than the I-load limit for three consecutive times, the failed signal is deselected from the selection filter and the FDIR and is replaced by the standby fourth signal. (For the elevons, deselection of failed signals is done on an LRU basis.) The selection filter and FDIR continue as in the nominal case.

Second Failure (Two Good Signals)

The second failure causes the selection filter and FDIR to downmode to the two level. The two remaining good signals are averaged in the selection filter for each actuator and subtracted with the results compared to an I-load limit by the FDIR. If a result exceeds the limit three consecutive times, a dilemma is declared. (This is the third failure.)

Third Failure (One Good Signal)

A third failure caused by a bad signal will result in the selection filter averaging the remaining good signal with the bad signal (since it is impossible for RM to isolate the failure) for output to the SOP. FDIR declares a dilemma. A third failure caused by a commfault will cause the selection filter and FDIR to downmode to the one level, and the remaining signal is passed through RM with no processing. If the commfault clears, RM will upmode to the two level and operate as described above for two level RM. (Note: A commfault is treated as a failure and will always cause RM to downmode to the next lower level. However, unlike a bad signal failure, if the commfault clears, RM upmodes back to the next higher level.)

Fourth Failure (No Good Signals)

● If RM is at the two level:

- A four failure by a commfault downmodes the selection filter and FDIR to the one level and the remaining signal (good or failed) is passed through RM with no processing.
- A fourth failure caused by the remaining good signal failing will not change RM; the selection filter will average the two failed signals.

- If RM is at the one level:

- A four failure by a commfault downmodes the selection filter and FDIR to the zero level. The selection filter outputs the last valid signal value to the SOP.
- A fourth failure caused by the remaining signal failing will not affect RM. The signal is passed through RM with no processing.

Hydraulic thermal conditioning: The Orbiter hydraulic systems may be thermally conditioned for entry. The purpose of the hydraulic thermal conditioning function is to generate simultaneous slew position commands to the four elevons, rudder, speed brake, and body flap upon receiving a crew indicator, and speed brake commands use a triangular function and the body flap uses integer commands.

The slew position commands for all the aerosurfaces but the body flaps are incremented from the initial position to the upper limit by a unique constant delta and then decremented by delta to the lower limit. For the body flap, all three pilot valves are enabled and the slew command is cycled between upper and lower limits by integer commands +1. The aerosurface commands are generated cyclically between upper and lower limits until the surface drive terminate flag is set. At this time, the commands remain unchanged from the previous commanded positions.

2-43

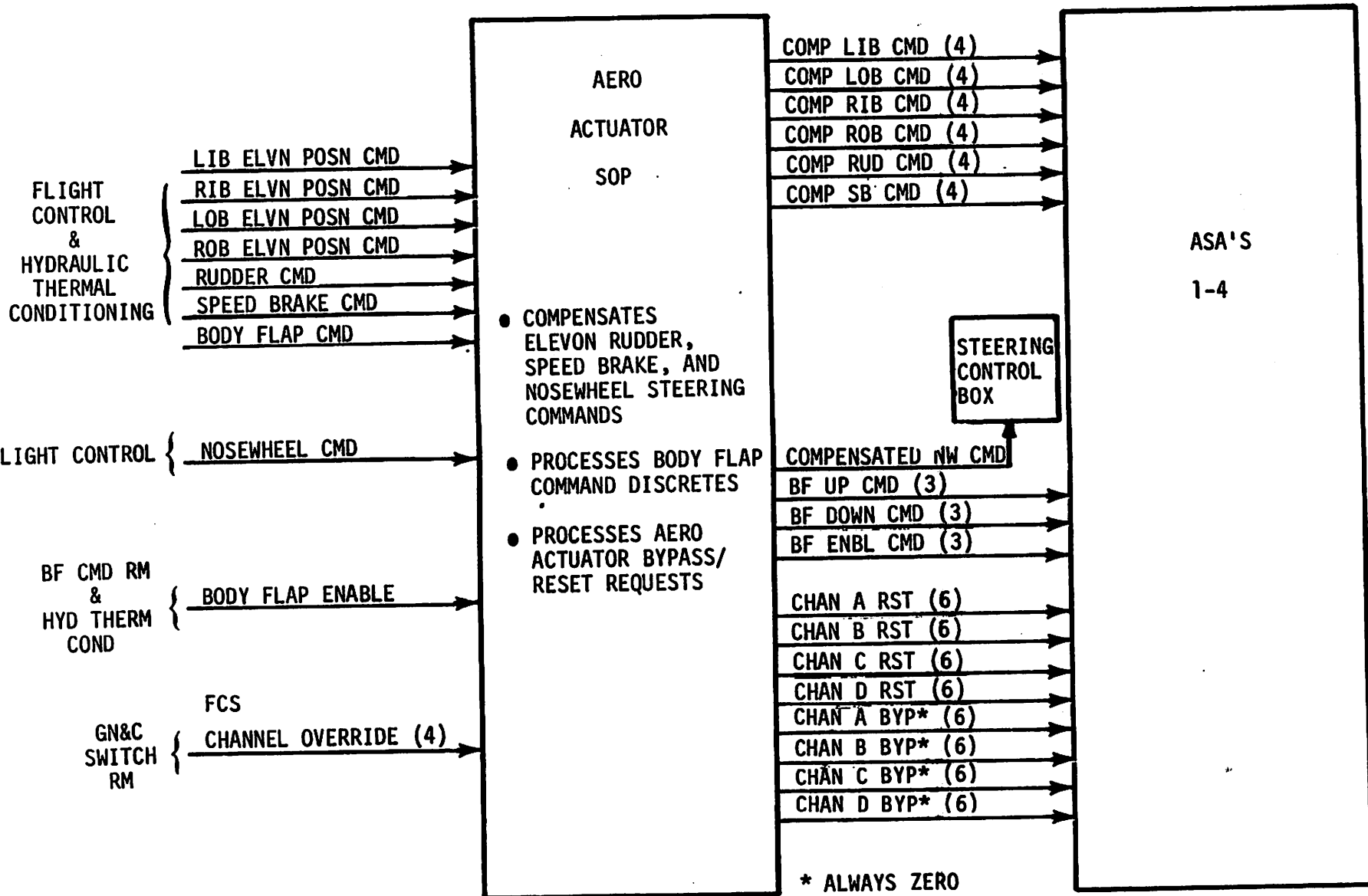


Figure 2.1-15.- Aero actuator command SOP data flow.

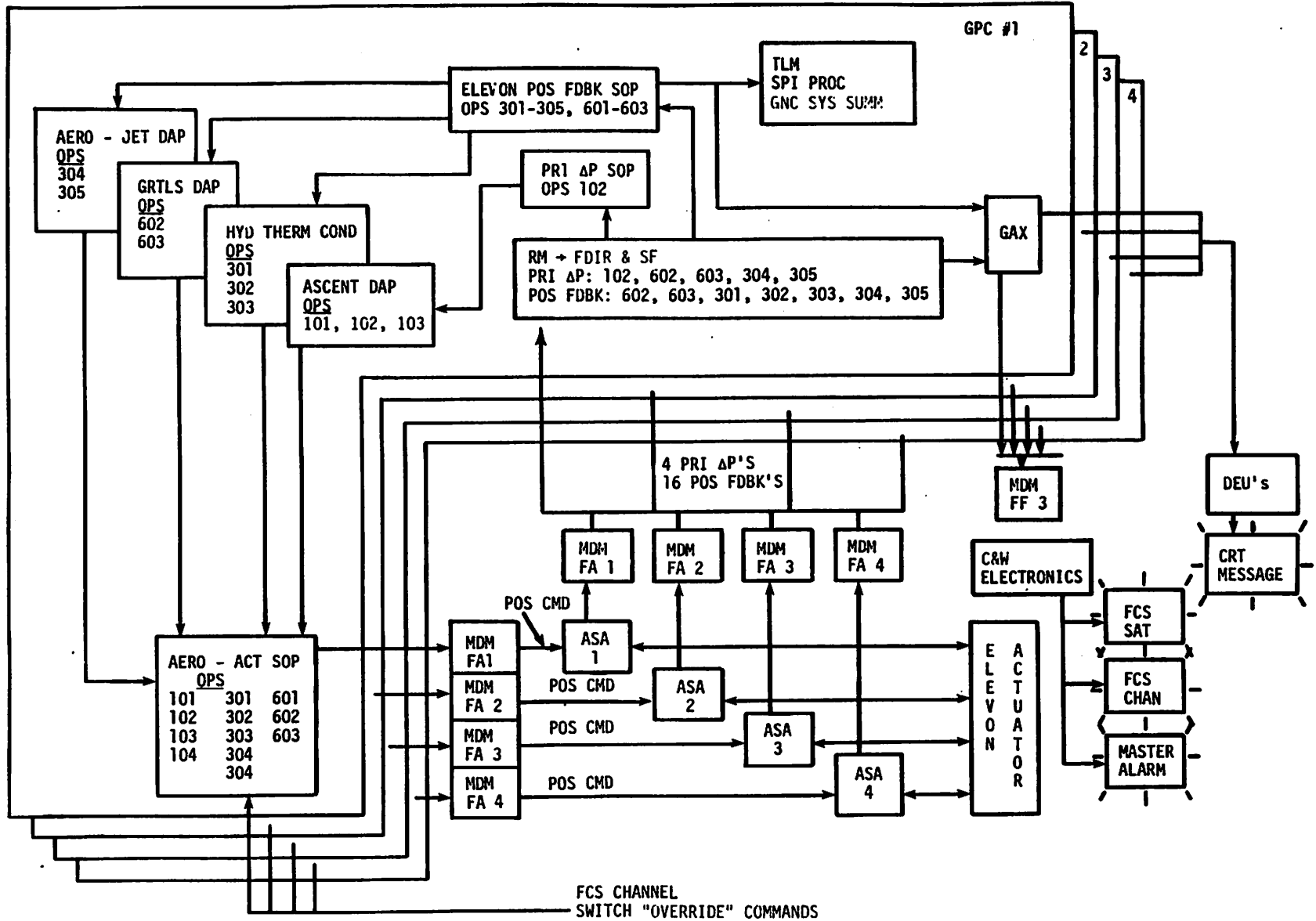


Figure 2.1-16.- Hardware/software interface (elevons).

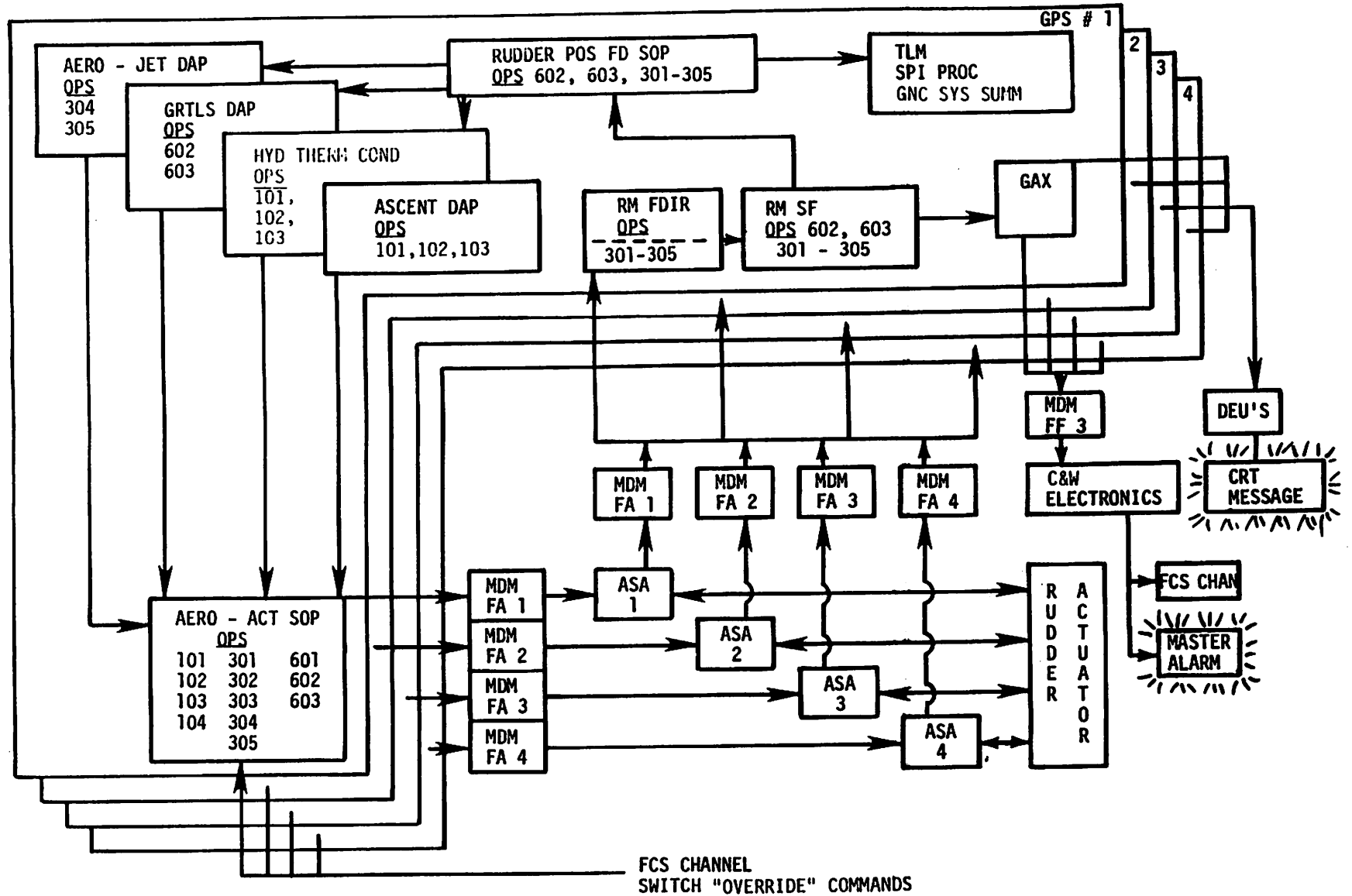


Figure 2.1-17.- Hardware/software interface (rudder).

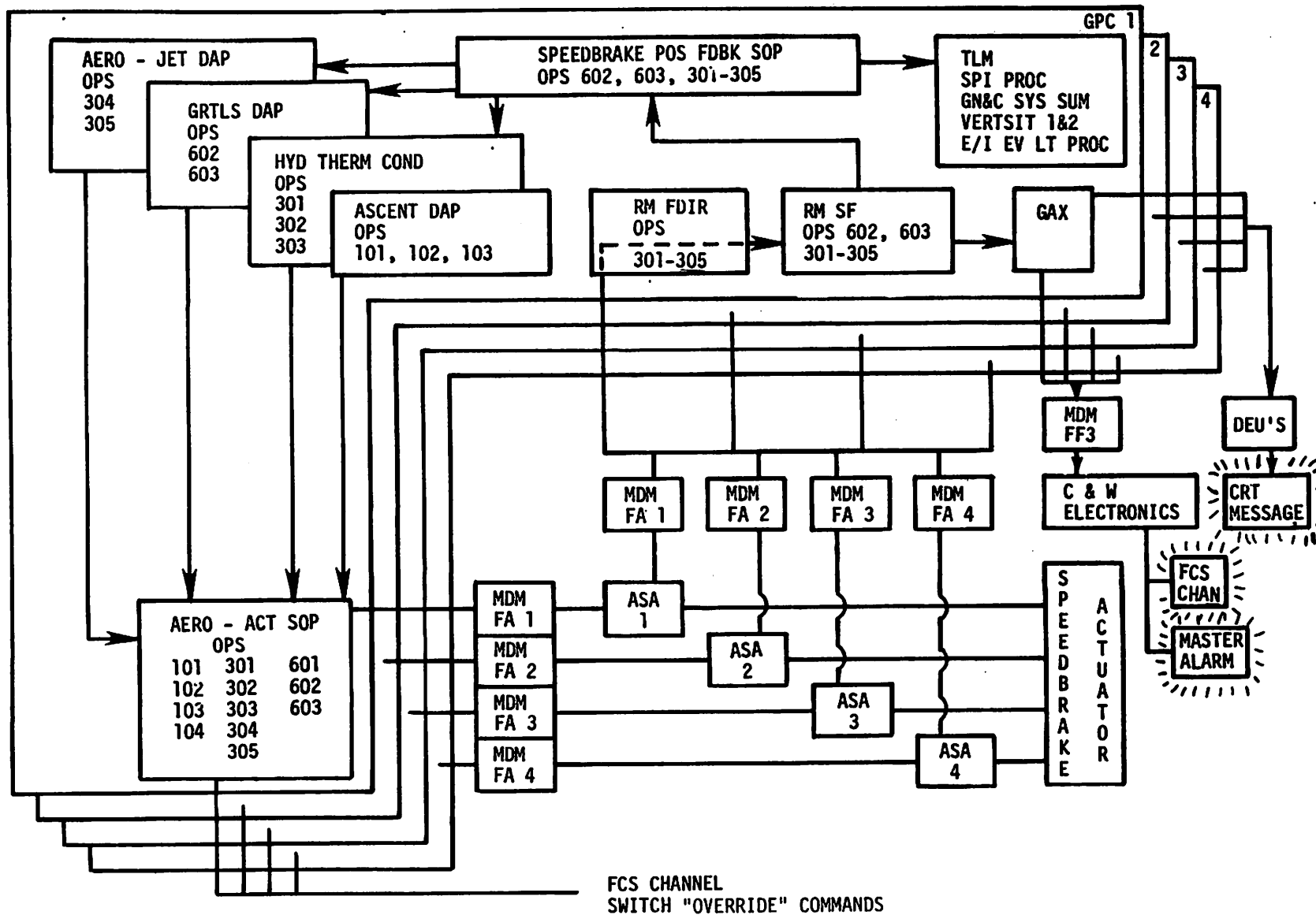


Figure 2.1-18.- Hardware/software interface (speed brake).

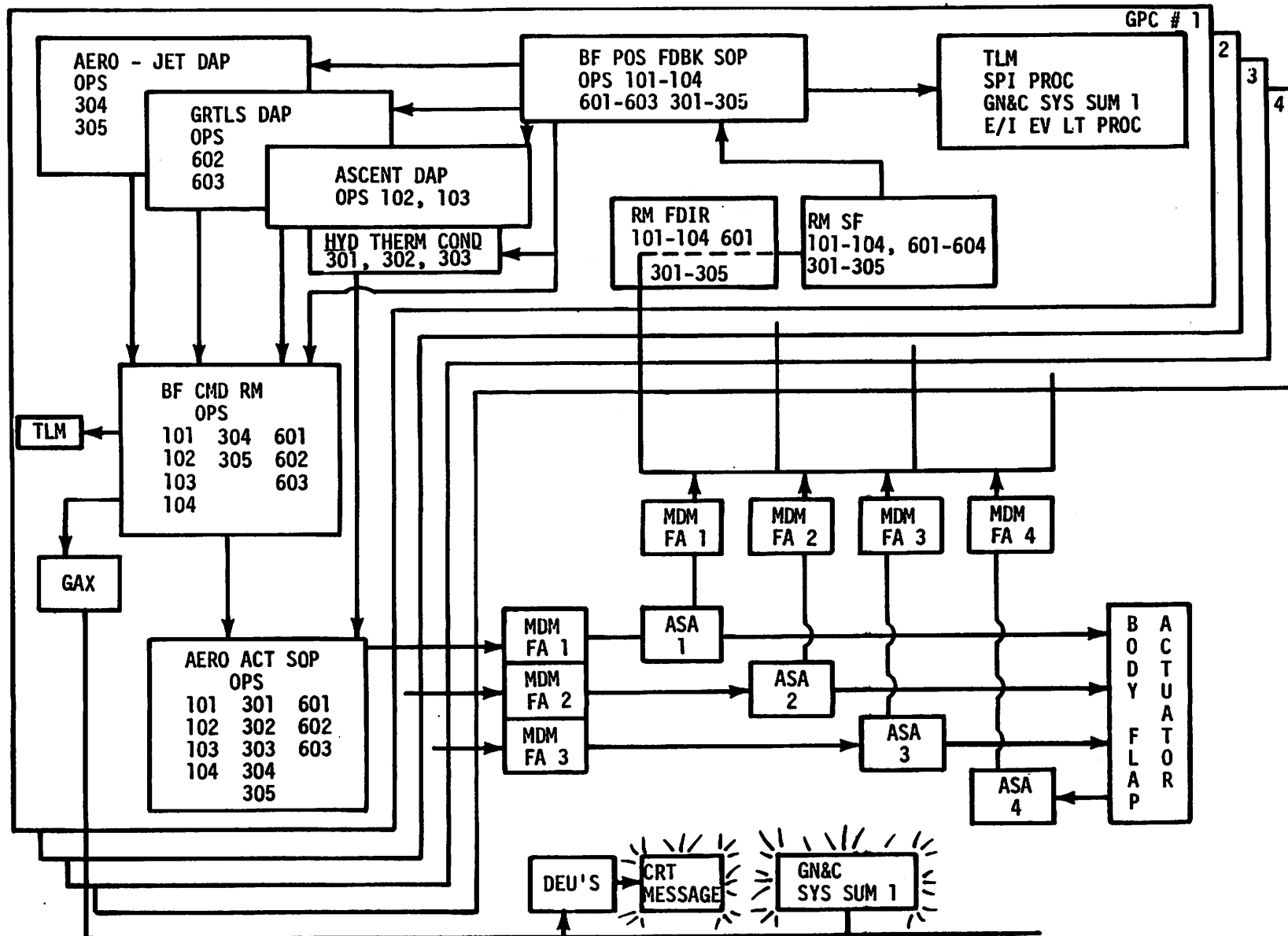


Figure 2.1-19.- Hardware/software interface (body flap).

2.1.7 Hydraulic System

The hydraulic system SOP monitors triply-redundant hydraulic system pressures to determine fail status of each of three hydraulic systems and determines the total number of good systems available for Priority Rate Limiting (PRL) in flight control. The nosewheel and main landing gear/door discretes are also monitored to determine the gear/door status. The hydraulic system SOP is executed at a 25 Hz rate. Following are the functions, in more detail, of the hydraulic system SOP (fig. 2.1-20):

- When the AUTO/MAN failure determination flag is selected AUTO by the crew via the item entry on the OVERRIDE display, the SOP monitors each hydraulic system for a low pressure condition by comparison to a pressure low limit value, provided the data good flag is "good". If the pressure is less than the low limit value for a given number of cycles, a hydraulic system fail status indicator is output to flight control. If the pressure subsequently exceeds the low limit value, the hydraulic system status will be set to "not failed".
- When the AUTO/MAN flag is in the manual mode, the hydraulic system fail status will be output when the crew deselects the hydraulic system using the RM SENSOR display.
- The total number of good hydraulic systems is calculated by subtracting the number of failed hydraulic systems from the total number of hydraulic systems and is output to flight control.
- Data good flags are monitored from the selection filter and when the flag indicates "bad" data, the pressure data from the previous cycle is sent for processing.
- If one or more hydraulic systems has been declared failed, the uplock discretes and associated commfaults are monitored to determine if a "gear not up" condition exists. A "gear not up" discrete is sent to flight control for PRL if three noncommfaulted discretes or more of the six uplock discretes from proximity sensors indicate a "not up-locked" state. If the "gear not up" discrete is set for a given number of cycles, a downlock indication is set and output to flight control. Below a given altitude when a "gear not up" condition exists, a gear down discrete is sent to flight control.

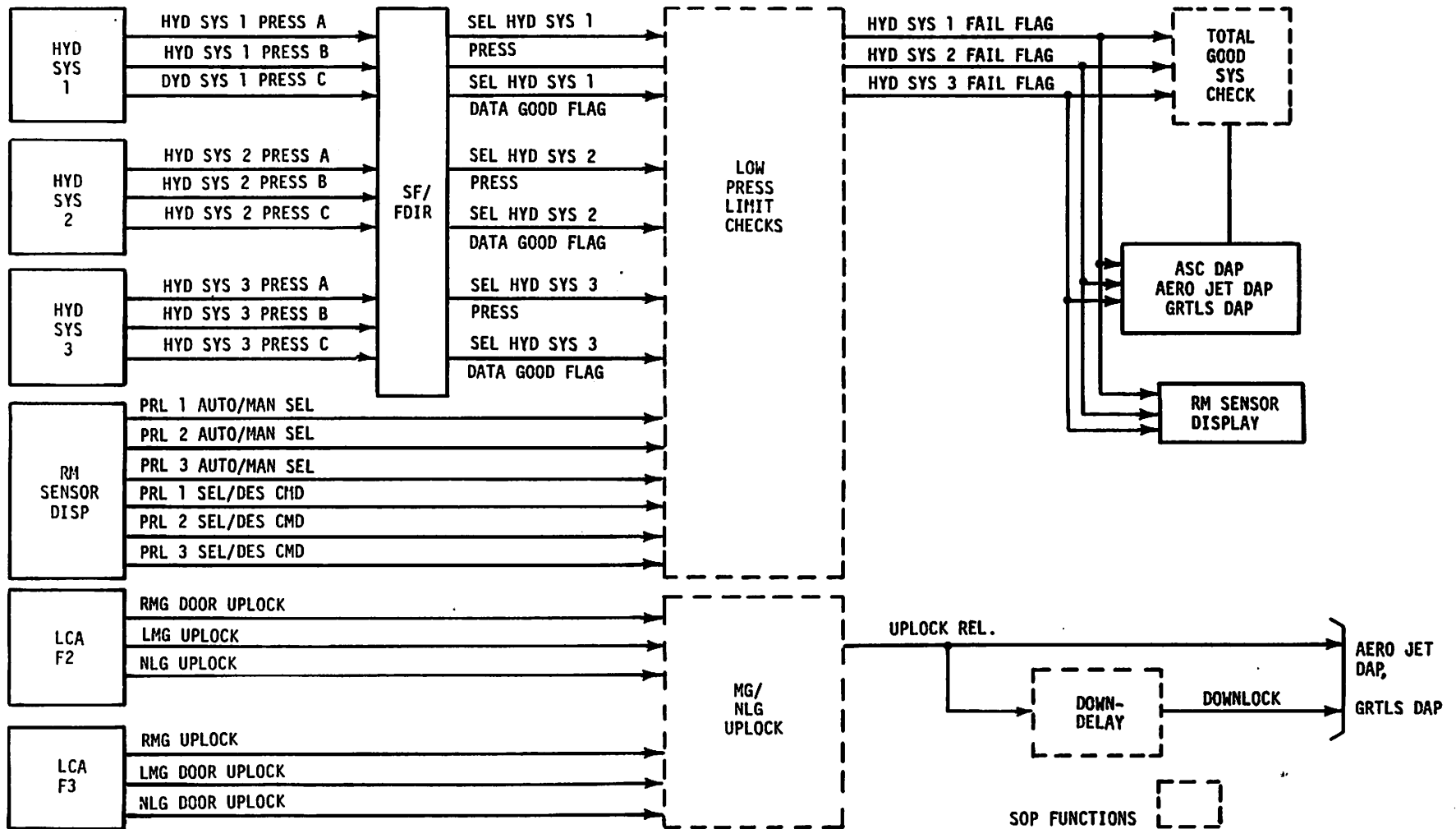


Figure 2.1-20.- Hydraulic system data flow.

When BFS is engaged, the hydraulic system HIP (fig. 2.1-21) performs the same functions as the hydraulic system SOP with the following exceptions:

- There is no failure determination flag and therefore no crew capability to manually deselect a hydraulic system, causing a system fail status.
- No monitoring of altitude or "gear not up" condition is done to set the gear down discrete.
- All uplock discrettes are monitored, whether commfaulted or not, in determining "gear not up" condition.

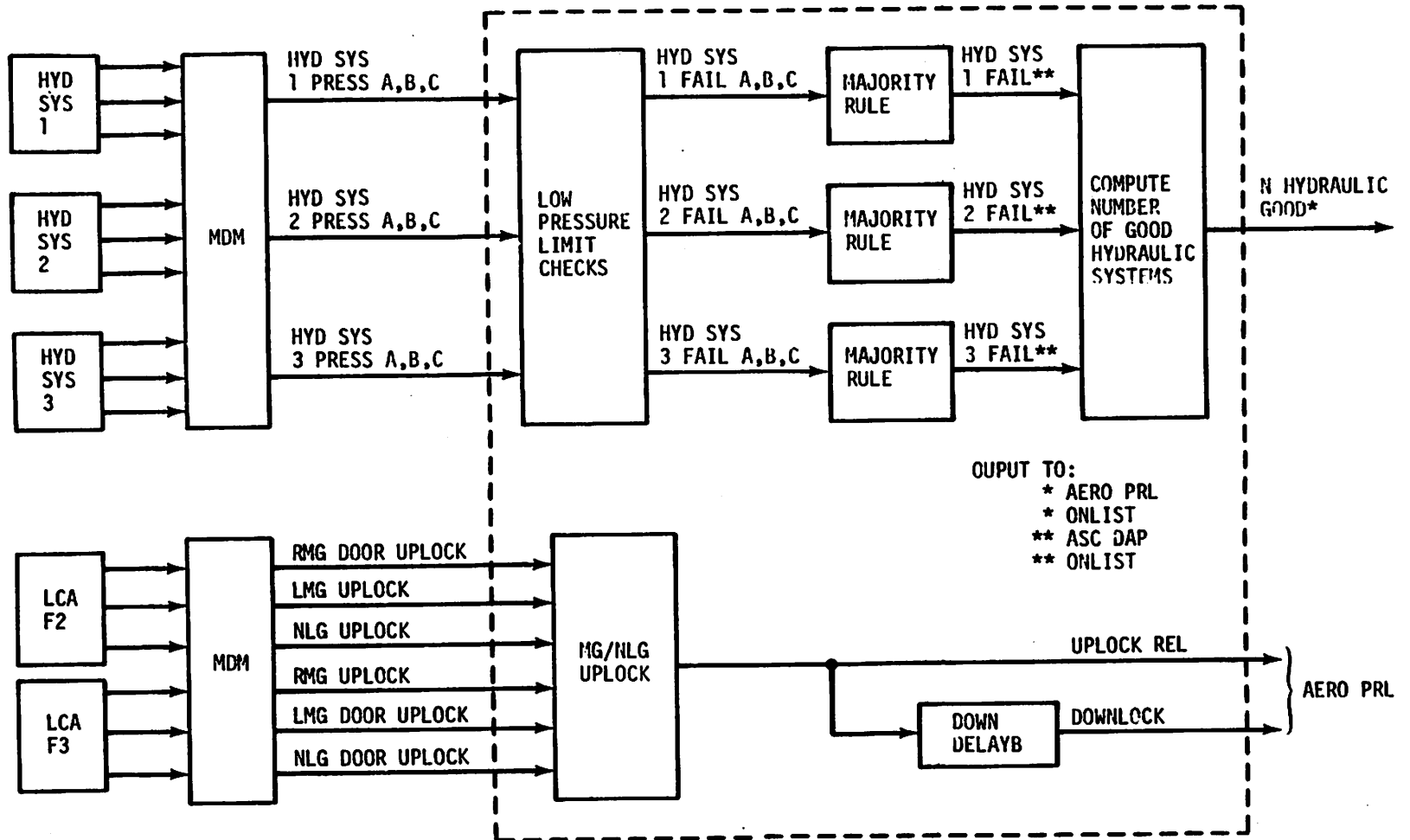


Figure 2.1-21.- Hydraulic system HIP.

2.1.8 Main Propulsion System

The Main Propulsion System (MPS) Thrust Vector Control (TVC) Command (CMD) SOP has two major functions:

- Processes the MPS pitch and yaw actuator position commands.
- Processes the MPS actuator bypass/override requests.

The MPS TVC CMD SOP monitors dump, stow, and control law main engine commands. Compensation and biasing is performed on position commands from the ascent DAP when the dump flag from ET SEP SEQ is not set and the stow flag from MPS DUMP SEQ or ET SEP SEQ is not set.

When the dump flag is set, the MPS TVC CMD SOP cyclically increments (0.20) the last pitch and yaw position commands until each actuator is in the designated dump position. The dump position commands take precedence over the ascent DAP position commands.

When the stow flag from MPS DUMP or ET SEP SEQ is set, the MPS TVC SOP cyclically increments (0.20) the last pitch and yaw position commands until each actuator is in the stowed position.

The stow position commands will take precedence over the dump position commands. During MM 301, the Main Engine (ME) pitch and yaw actuators shall be cyclically commanded to the stow positions.

The outputs of MPS TVC CMD processing are four redundant commands to each of the three main engines' pitch and yaw actuators.

The MPS TVC CMD SOP also performs actuator bypass/override processing. This function shall generate bypass and override secondary channel command discretely to the three main engine pitch and yaw actuators.

The override discrete will be generated by monitoring the following override requests:

- When a channel switch is in OVERRIDE, override discretely are sent to all actuators on that channel.
- Forced override command from Redundant Set (R/S) launch sequence is monitored and results in all the pitch and yaw actuators for all three ME's receiving the override discrete.
- ME shutdown flags are monitored for each engine. When an engine is in shutdown, all pitch and yaw actuators for the engine receive the override discrete for (I-load) cycles (equivalent to 4 sec). The discrete is then removed.

SSME SOP: The Space Shuttle Main Engine (SSME) SOP monitors ME data for engine status and for detecting data path and command failures. ME status and data/command path failure indications are sent to applications software for sequencing and display (fig. 2.1-22).

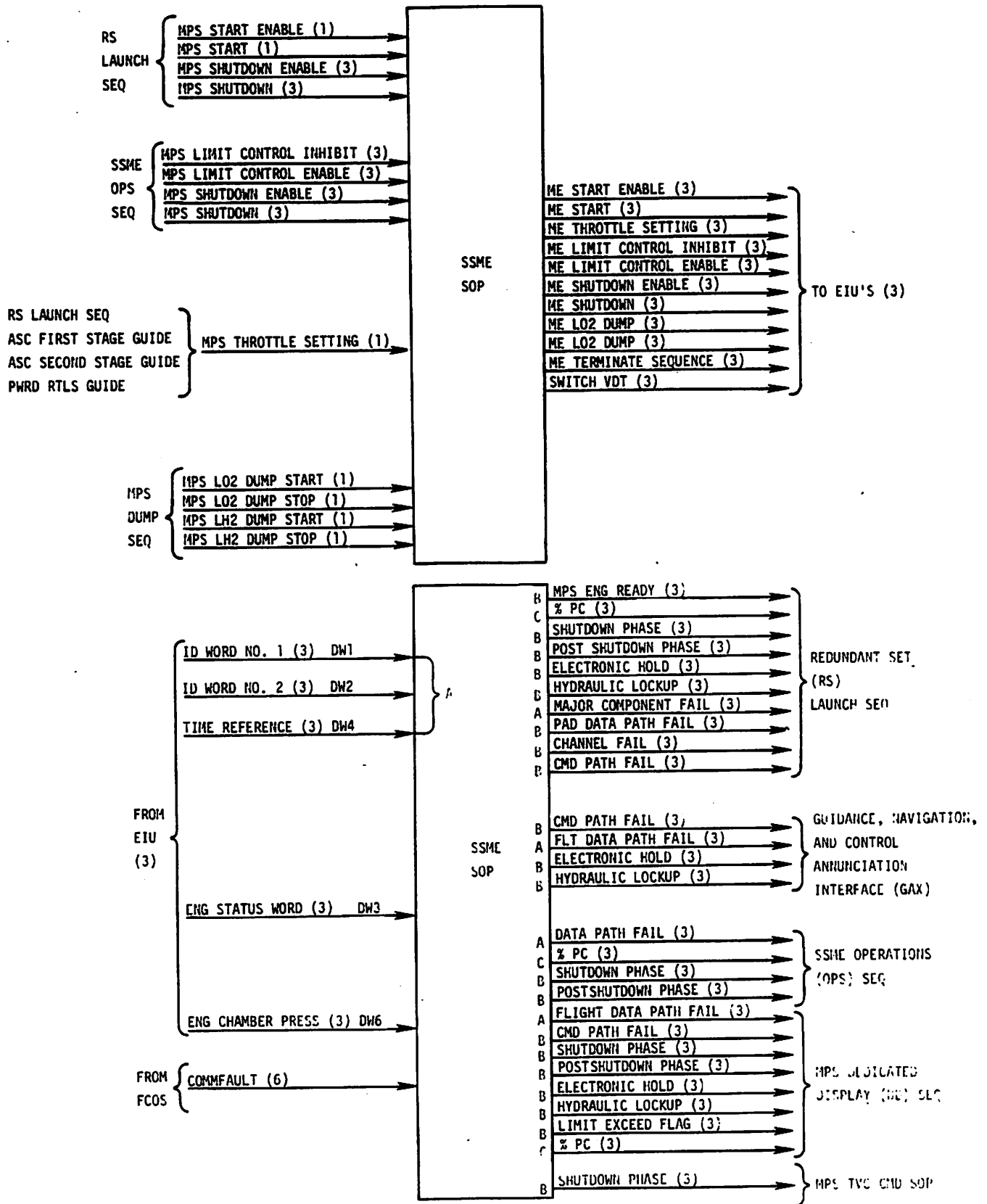


Figure 2.1-22.- SSME SOP.

The SSME SOP disc monitors for command flags. When a command flag is set, the SSME SOP will send from a mass memory a command corresponding to the flag that was set. The commands output by the SSME SOP are sent to the ME controllers via the Engine Interface Units (EIU's). Refer to tables 2.1-5 and 2.1-6 for definitions.

TABLE 2.1-5.- SSME SOP COMMAND DEFINITIONS

<u>GPC Command</u>	<u>Engine Controller Operating Phase</u>	<u>Function/Engine Controller Acceptability</u>
START ENABLE	Checkout Start preparation Postshutdown	Valid START ENABLE command required by controller for acceptance of START command. If any command is transmitted other than START, effectivity of START ENABLE is terminated. For START command then to be accepted by controller, START ENABLE must be retransmitted. (START ENABLE issued at T_0 -5 seconds.)
START	Next Command after START ENABLE	(1) Initiates start phase; (2) Terminates purge sequence 4 (fuel purge); (3) Closes bleed valves. (START issued at T_0 -3.8 seconds.)
THROTTLING SETTING	A11	Sets main engine chamber pressure level. Given in percent P_c (100 percent thrust level defined as 2960 psia). Range is 60 percent to 109 percent in 1 percent increments.
LIMIT CONTROL INHIBIT	A11	(1) Inhibits controller-initiated engine shutdown; (2) deactivates emergency shutdown solenoid valve. (Controlled by LIMIT SHUTDN switch, Pnl C3.)
LIMIT CONTROL ENABLE	A11	(1) Enables controller-initiated engine shutdown; (2) allows activation of the emergency shutdown solenoid valve. (Controlled by LIMIT SHUTDN switch.)
SHUTDOWN ENABLE	A11	Valid SHUTDOWN ENABLE command required for acceptance of SHUTDOWN command. If any command is transmitted other than SHUTDOWN, the effectivity of SHUTDOWN ENABLE is terminated. For SHUTDOWN command then to be accepted by controller, SHUTDOWN ENABLE must be retransmitted.
SHUTDOWN	Next command after SHUTDOWN ENABLE	Initiates engine shutdown phase if in start or mainstage phases. (Sent next cycle after SHUTDOWN ENABLE is sent.)
LO2 DUMP	Checkout Start preparation Postshutdown	Starts oxidizer dump and maintains Main Oxidizer Valve (MOV) open. (Requires OMS 1 ignition signal or MPS PRPLNT DUMP LO2 sw-START at MECO + 10 sec (earliest time) before MPS DUMP sequence issues command request to SSME SOP.)
LH2 DUMP	Checkout Start preparation Postshutdown	(1) Terminates oxidizer dump; (2) performs fuel system purge; (3) initiates fuel dump; and (4) then maintains Main Fuel Valve (MFV) open.
TERMINATE SEQUENCE	Checkout Start preparation Postshutdown	(1) Terminates any sequence during start preparation or postshutdown and closes all propellant valves; (2) transfer phase to the postshutdown phase.
SELECT CHANNEL B	Mainstage	Selects main engine controller channel B VDT for output to that controller's EIU.

TABLE 2.1-6.- SSME SOP DATA DEFINITIONS

<u>Data Indication</u>	<u>Data Word</u>	<u>Definition</u>	<u>Destination</u>	<u>Function/Use</u>
MPS Engine Ready	3	Following parameters within limits: <ul style="list-style-type: none"> ● Low Press Fuel Turbop Disch Temp 37° R - 40° R ● Low Press Fuel Turbop Disch Press 42 - 50 psia ● Hi Press Oxid Turbop Disch Temp 163° R - 170.5° R ● Low Press Oxid Turbop Disch Press 101 - 113 psia ● Emergency Shutdown Press 20 - 50 psia 	RS LAUNCH SEQ	Enables RS launch seq to issue START commands.
Percent Pc	6	Percent thrust (range 0 - 109 percent)	(a) RS LAUNCH SEQ	Used for checking that all three SSME's > 90 percent enabling SRB ignition. If all three > 90 percent after 4 sec, SSME's shutdown and SRB ignition inhibited.
			(b) SSME OPS SEQ	TBD
			(c) MPS DD SEQ	Drives Pc meter. When percent Pc > 30 percent inhibits manual and GPC preclude close logic; i.e., preclude cannot be commanded closed by crew or GPC. Inhibit removed when percent Pc < 30 percent.
Shutdown Phase	3	Affected main engine is in process of shutting down.	(a) RCS LAUNCH SEQ	SRB ignition inhibited. Remaining two engines shutdown after a time delay.
			(b) SSME OPS SEQ	(1) Applies time delays for preclude closure depending upon cause of engine shutdown and if RTLS selected. (2) Used for terminating MAIN ENGINE SHUTDOWN pb commands. (3) If LIMIT SHUTDOWN switch - ENABLE, then sends inhibits to other two engines to prevent controller initiated shutdown.
			(c) MPS DD SEQ	Drives red main engine status light.
Postshutdown Phase	3		(a) RS LAUNCH SEQ	Same as 3
			(b) SSME OPS SEQ	Same as 3
			(c) MPS DD SEQ	Same as 3

TABLE 2.1-6.- Concluded

<u>Data Indication</u>	<u>Data Word</u>	<u>Definition</u>	<u>Destination</u>	<u>Function/Use</u>
Electronic Hold	3 (part)	Loss of flow meter data used mixture ratio control results in main engine controller freezing engine at last commanded position.	(a) RS LAUNCH SEQ	Results in SSME shutdown and SRB ignition inhibit.
			(b) GAX	Sends electronic hold indication to systems software for Class 3 alarm (SM Alert).
			(c) MPS DD SEQ	Used to turn on amber main engine status light.
Hydraulic Lockup	3	One or more of the following engine valves is outside of position limits 6 percent channel A; 10 percent channel B - engine controller: <ul style="list-style-type: none"> ● main fuel valve ● main oxid valve ● fuel preburner oxid valve ● oxid preburner oxid valve ● chamber coolant valve 	(a) RS LAUNCH SEQ	Results in SSME's shutdown and SRB ignition inhibit.
			(b) GAX	Sends electronic hold indication to systems software for Class 3 alarm (SM ALERT).
			(c) MPS DD SEQ	Used to turn off amber main engine status light.
Major Component Fail	3	One of the following has failed: <ul style="list-style-type: none"> ● One controller channel ● Any shutdown parameter out-of-limits ● Any main engine propellant valve actuator failure 	(a) RS LAUNCH SEQ	Results in SSME shutdown and SRB ignition inhibit.
Limit Exceeded Flag	3	One of the following engine limits exceeded: <ul style="list-style-type: none"> ● Pc (Ignition to 3.2 sec) = 450 - 3312 psia (>3.2 sec) = 1183 - 3312 psia ● Hi Press Fuel Turbop Shaft Speed - 39,100 rpm ● Hi Press Oxid Turbop Shaft Speed - 32,300 rpm ● Hi Press Fuel Turbine Disch Temp = 760 - 2010° R ● Hi Press Turbine Disch = 760 - 2140° R ● Hi Press Oxid Turbop Interm Seal Purge Press = 40 - 270 psia 	MPS DD SEQ	Used to turn on red engine status light.
Pad Data Path Fail	1,2,4	Either PRI or SEC data lost (prelaunch only - MM 101).	(a) RS LAUNCH SEQ	Results in SSME shutdown and SRB ignition inhibit.
Data Path Fail	1,2,4	Both PRI and SEC data lost (MM 102, 103, and 601).	(a) GAX	Sends data path fail indication to systems software for Class 3 alarm (SM ALERT).
			(b) SSME OPS SEQ	If LIMIT SHUTDN Switch - AUTO, sends inhibit command to other two engines, preventing controller-initiated shutdown.
			(c) MPS DD SEQ	Used to turn on amber main engine status light.
Channel Fail	3	Single command channel lost (MM 101 only).	RS LAUNCH SEQ	Results in SSME shutdown and SRB ignition inhibit.
Command Path Fail	3	Two command channels lost (MM 102, 103, and 601).	(a) GAX	Sends command path fail indication to systems software for Class 3 alarm (SM ALERT).
			(b) MPS DD SEQ	Used to turn on the amber main engine status light.

2.1.9 Solid Rocket Boosters

The SRB Pc RM receives three compensated/calibrated Pc signals from each SRB. The RM uses a selection filter to output to the SRB SEP SEQ one selected Pc value from each SRB. For no failures, the MVS mode is used on each set of Pc signals. For one failure, the average of the two good Pc signals in each set is used. For two failures, the single good Pc signal is passed through with no selection taking place (PASS mode).

Solid rocket booster thrust vector control command subsystem operating program: The purpose of the SRB TVC SOP shall be to perform the following functions:

- Data compensation and unit conversion of the rock and tilt deflection commands.
- Processing of the SRB actuator bypass/override request.

The SRB TVC actuator bypass/override processing shall generate bypass and override secondary channel command discretetes to the two SRB engines, rock and tilt actuators.

The override discrete shall be generated by monitoring the following override requests:

- When the switch is in override, override discretetes are sent to all actuators on the given channel.
- The SRB override flag from Launch Processing System (LPS) shall be monitored during Auxiliary Power Unit (APU) start-up. When the flag is set, all the roll and tilt actuator override discretetes shall be set. When the SRB slew indicator from LPS is set, the SRB override discretetes are unset.
- When the SRB SLEW COMPLETE flag is set, all the SRB rock and tilt actuator override discretetes are set. When the SRB FDIR ON flag is set, the override discretetes are unset.
- All SRB rock and tilt actuator override discretetes are set when the SRB SEP MODE flag is received from SRB SEP SEQ.

2.1.10 Reaction Control System

The RCS RM monitors the RCS jets' chamber pressures, temperatures, reaction jet driver output discretetes, jet fire commands, and manifold valves status. It also provides a limited amount of automatic jet deselection and alerts the crew when a fault is detected.

The Data Processing System (DPS) software provides status information on MDM Input/Output (I/O) errors to the RCS RM software, referred to as MDM commfaults (communications faults).

Commfaults are included in the RCS RM requirements to help prevent the redundant GPC's from moding to dissimilar software, to optimize the number of jets available for use, and to prevent the RCS RM from generating additional alerts to the Flight Control Operational Software (FCOS)-generated alerts associated with commfaults. The RCS RM uses the MDM and Line Replaceable Unit (LRU) commfaults (where LRU is defined to be either one RCS jet or one RCS manifold), and will reconfigure for commfaults, regardless of whether the commfault is permanent, permanent and subsequently removed, or transient. Note that moding for RCS manifold LRU commfaults occurs only in OPS 1.

A jet with an LRU commfault will not have any of its status flags or counters modified as long as the fault exists, except by subsequent crew action. LRU commfaults or transducer failures will cause the quantity monitor to use substitute measurements or constants, and the CRT's will show an "M" to indicate missing data. If a substitute is not available, or a constant is used, the calculations are suspended, a "+" appears on the CRT, and a class 3 alarm is output.

The input signals associated with each RCS jet are a chamber pressure discrete, fuel and oxidizer injector temperatures, and reaction jet driver output discrete. The input signals associated with each manifold are the open and close discrettes for the fuel and oxidizer manifold isolation valves.

Jet failed on monitor: The jet failed on monitor uses the Reaction Jet Driver (RJD) output discrettes and the jet fire command discrettes provided by the RCS CMD SOP to detect jets failed on.

The jet failed on monitor uses the jet fire command A discrettes, the RJD output discrettes, the jet RM inhibit discrettes, and the jet LRU commfault discrettes as inputs; and outputs the jet failed on indicator discrettes and the jet failed on counter discrettes. There are 44 of each of these discrettes.

The jet failed on logic AND's the reaction jet driver output discrete with the complement of the jet fire command A discrete, and declares the jet failed on if this calculation is true for three consecutive cycles. Consecutive passes are not affected by commfaults or by cycles in which there are fire commands for the affected jets. The three consecutive cycle logic will be reset, however, if the noncommanded jet has its reaction jet driver output discrete reset to indicate the jet is not firing. A jet failed on will not be automatically deselected by RM, nor will the DAP reconfigure the jet selections.

A jet failed on determination will set the jet failed on indicator discrete and the jet failed on counter discrete. These discrettes will be reset when the associated jet's RM inhibit discrete is reset. The jet failed on monitor outputs the jet failed on indicators to displays and controls and to the jet fault limit module.

The jet failed on monitor design is valid for a minimum jet fire command pulse of 80 msec on and 80 msec off. The crew will be alerted by a class 2 alarm, the backup C&W light and RCS jet light on the Caution/Warning

(C&W) matrix on panel F7, a fault message on the CRT fault message line, and jet on indications on the RCS SPEC display and the GNC SYS SUMM 1 and 2 displays. The jet failed on monitor is active in OPS 1, 2, 3, 6, and 8 in the PASS; and 1, 3, and 6 for the BFS, but only if BFS is engaged.

Jet failed off monitor: The jet failed off monitor uses the jet fire command discretes provided by the RCS CMD SOP, and the jet chamber pressure feedback discretes provided by the RJD's to detect jets failed off.

The jet failed off monitor uses the jet fire command A discretes, the jet chamber pressure discretes, the jet RM inhibit discretes, and the jet LRU commfault discretes as inputs; and outputs the jet failed off indicator discretes and the jet failed off counter discretes. There are 44 of each of these discretes.

The jet failed off logic AND's and the jet fire command A discrete with the complement of the jet chamber pressure discrete, and declares the jet failed off of this calculation is true for three consecutive cycles. Consecutive passes are not affected by commfaults or by cycles in which there are no fire commands for the affected jets. However, consecutive passes leading to a failed off indication must begin anew if, prior to reaching the third consecutive cycle, the fire command and its associated pressure discrete indicates that the jet has fired. The RCS RM will automatically deselect a jet which has failed off, and the DAP will reconfigure jet selection accordingly unless the jet deselection limit for that RCS pod has been reached.

A failed off jet determination will set the associated jet failed off indicator and the jet failed off counter discretes. These discretes will be reset when the associated jet's RM inhibit discrete is reset. The jet failed off monitor outputs these jet failed off indicator discretes to the jet fault limit module and to displays and controls. The jet failed off monitor will be inhibited for that jet which has failed off until the crew resets the RM inhibit discrete.

The jet failed off monitor design is valid for a minimum jet fire command pulse mode of 80 msec on and 80 msec off. The crew is alerted to a failure by a class 2 alarm, the backup C&W light and RCS jet light on the C&W matrix on panel F7, a fault message on the CRT fault message line, and a jet off indication on the RCS SPEC display and the GNC SYS SUMM 1 and 2 displays. The jet failed off monitor is active in OPS 2, 3, 6, and 8 in the PASS; and 1, 3, and 6 for the BFS, but only if BFS is engaged.

Jet leak monitor: The jet leak monitor uses the jet fuel and oxidizer injector temperature transducer outputs of each jet to detect a leaking jet.

The jet leak monitor uses the jet fuel and the jet oxidizer injector temperatures, the jet RM inhibit discretes, and the jet LRU commfault discretes as inputs, and outputs the jet failed leak indicator discretes and the jet failed leak counter discretes. There are 44 of each of these discretes.

The jet leak logic compares the jet fuel injector temperatures and the jet oxidizer injector temperatures with the specified temperature limit of 300° F for three consecutive cycles. Consecutive passes leading to jet failed leak indication will begin anew if the fuel and oxidizer temperatures are both greater than 300° F before the jet leak counter reaches 3. The RCS RM will automatically deselect a jet which is declared leaking and the DAP will reconfigure jet selection accordingly unless the jet deselection limit has been reached for that RCS pod.

A jet failed leak determination will set the associated jet failed leak indicator and jet failed leak counter discretes. These discretes will be reset when the associated jet's RM inhibit discrete is reset. The jet leak monitor outputs the jet failed leak indicator discretes to the jet fault limit module and to crew displays.

The crew is alerted to a failure by a class 2 alarm, the backup C&W light and the RCS jet light on the C&W matrix on panel F7, a fault message on the CRT fault message line, and a jet leak indication on the RCS SPEC display and the GNC SYS SUMM 1 and 2 displays.

The jet leak monitor is active in OPS 2, 3, and 8 for the PASS; and 1, 3, and 6 for the BFS, but only if BFS is engaged.

Jet fault limit module: The jet fault limit module limits the number of jets which can be automatically deselected in response to failures detected by RCS RM. The limits are modifiable by crew input on the RCS SPEC display (RCS F, L, R jet fail limit integers - one integer per pod). This module also reconfigures a jet's availability status (jet deselect output discretes (44)) in response to crew inputs on the RCS SPEC display (jet RM inhibit discretes (44) and jet deselect input discretes (44)).

An automatic deselection of a jet occurs if all of the following are satisfied:

- Jet detected failed off or leak (jet on failures do not result in automatic deselection)
- Jet select/deselect status is "SELECT"
- Jet's manifold status is "OPEN"
- RM is not inhibited for this jet
- Jet failure has not been overridden
- The number of automatic deselections of primary jets on this pod is less than the associated jet fail limit (no limit on vernier jets)

All jet failures detected will be announced to the crew, even if they do not cause automatic jet deselection. If multiple failures occur on a jet, only the last failure will be annunciated. Failure indications are as indicated in the jet failed off and leak monitors.

The jet fail limit counter is incremented by the number of jets which has been automatically deselected for that pod by the RCS RM, and it is decremented by one for each automatically deselected jet that is reselected. The vernier jets do not increment or decrement the jet fail limit counter. The jet fail limits are individually changeable in major modes 2 and 3 on the RCS SPEC display. An increase in the limit allows previously failed jets to be deselected, providing the above requirements are met. A decrease in the limit will not cause a change in the status of any jet. Note that setting the jet fail limit equal to or less than the number of jets which have been automatically deselected will effectively inhibit the RCS RM for that pod.

A jet's status can be changed from deselect to select only by item entry on the RCS SPEC page. Failure resets or reductions in the jet fail limit will not cause the status to be reset to select. The select item entries cause the override to be invoked if there is a declared failure for that jet, and will make those failures inoperative in the jet fault limit module. An overridden failure will remain overridden until the applicable failure is reset.

Automatic deselection of a jet can be prevented by the use of the inhibit item entries on the RCS SPEC page. Changing the inhibit to not inhibited will reset a jet's failures, but will not cause the jet fail limit to be incremented or decremented. Reset by use of the RM inhibit of a failure which has been overridden will reset the override. Jet failures are unordered; that is, if there are more candidates for automatic deselection than is permitted by the jet fail limit, there is no preference as to which of the candidates will be deselected.

Manifold status monitor: The manifold status monitor uses the open and close discrettes of the oxidizer and fuel manifold isolation valves (provided by the motor control assemblies) to determine the open/close status for each jet manifold.

The manifold status monitor uses the fuel and oxidizer manifold valve open discrettes (15 of each discrete), the fuel and oxidizer close discrettes (15 of each), the manifold status discrete (15 discrettes), the manifold LRU commfault discrettes (15 discrettes), the MDM commfault discrettes (8 discrettes), and the manifold status override discrete (one discrete) as inputs; and outputs the manifold open/close status discrettes (15 discrettes), the RCS manifold RM dilemma discrettes (15 discrettes), and the RM power fail discrete (one discrete).

The manifold status monitor monitors the open and close discrettes for each manifold for any changes of state. A change of state in any one or more of these discrettes will cause a redetermination of that manifold's open/close status, independent of status changes made by the crew. This redetermination also contains logic which will determine if a power failure has occurred, and will determine whether a dilemma exists on a manifold (tables 2.1-7 and 2.1-8). A power failure condition exists when all of the open and close discrettes on a manifold are false for three consecutive cycles, and will cause the RM power fail flag to be set. The manifold sets identified in table 2.1-8 are the only manifolds which require power failure determination. This flag will remain set until the GAX honors

it and will then be reset. There is only one RM power fail flag, and all manifolds are capable of setting it, but each can set the flag only once. Whenever a dilemma exists for three consecutive passes, the RCS manifold RM dilemma flag for that manifold will be set. MDM or LRU comm faults will not modify the dilemma pass counter or the dilemma flag. The flag will be reset, however, if any of the four manifold open/close discrettes change state.

TABLE 2.1-7.- MANIFOLD STATUS

LOGICAL STATE OF						MANIFOLD STATUS	RCS MANIFOLD RM DILEMMA
INPUT		DISCRETES		POWER FAILURE			
OPEN Fuel	Ox	CLOSE Fuel	Ox				
0	0	0	0	Yes	(Previous) ^a	No	
0	0	0	0	No	Close	No	
0	0	0	1	N/A	Close	No	
0	0	1	0	N/A	Close	No	
0	0	1	1	N/A	Close	No	
0	1	0	0	N/A	Close	Yes	
0	1	0	1	N/A	Close	Yes	
0	1	1	0	N/A	Close	Yes	
0	1	1	1	N/A	Close	No	
1	0	0	0	N/A	Close	Yes	
1	0	0	1	N/A	Close	Yes	
1	0	1	0	N/A	Close	Yes	
1	0	1	1	N/A	Close	No	
1	1	0	0	N/A	Open	No	
1	1	0	1	N/A	Open	No	
1	1	1	0	N/A	Open	No	
1	1	1	1	N/A	Close	Yes	

^aThe manifold status from the previous pass is to be maintained.

TABLE 2.1-8.- MANIFOLD SETS FOR POWER FAILURE DETERMINATION

- Forward No. 3 and Forward No. 4

- Aft Left No. 1 and Aft Right No. 1

- Aft Left No. 2 and Aft Right No. 2

- Aft Left No. 3 and Aft Right No. 3

- Aft Left No. 4 and Aft Right No. 4

The transition of an MDM commfault discrete from false to true will cause the status of all affected manifolds to be set to close in all major modes. In Major Mode 1, the same is true of an LRU commfault. In Major Modes 2 and 3, the transition of an LRU commfault will cause no change in manifold statuses.

The crew is able to override the status of all manifolds on an individual basis by item entries on the RCS SPEC display via the manifold status override discrettes. The setting of this discrete for a manifold will change that manifold's status to its complementary state, and will then reset the override discrete. The use of the manifold status override feature will not inhibit or modify any of the other functions of the manifold status monitor. The module will continue to honor subsequent changes in the affected manifold's input signals (open/close discrettes, commfaults, override discrete) as specified in this section.

The manifold close status override is used in OPS 1 and 3 to open all manifolds whose status is closed and whose open/close discrettes are in dilemma. This discrete can be set by item entry on the override page, and will be reset to false after the reconfiguration is complete. The use of the manifold close status override feature will not inhibit or modify any of the other functions of the manifold status monitor.

Available jet status table: The available jet status table module provides a list of jets available for use by the Flight Control Systems (FCS).

The available jet status table uses the manifold open/close discrettes (15 discrettes) from the manifold status monitor, and the jet deselect output discrettes (44 discrettes) from the jet fault limit module as inputs; and outputs the jet available discrettes (44 discrettes) and the jet status change discrete (one discrete).

The available jet status logic AND's the jet deselect output discrete with the manifold open/close status discrete and statuses a jet as available to the FCS if the discrettes indicate select and open, respectively. The available jet status table will be computed each time that the jet status change discrete is true.

In the BFS, jet failures are detected only when BFS is engaged. Jets failed leaking and jets failed off are detected in the BFS the same as in the PASS, but the jet chamber pressure feedback discrete is used for fail on detection in the BFS, rather than the RJD output discrete which is used in the PASS.

2.1.11 Orbital Maneuvering System

OMS engine FDI: The objective of the OMS engine Fault Detection and Identification (FDI) is to detect and identify off-nominal thrust performance of the OMS engines. The OMS engine FDI notifies the crew of the following conditions:

- Off - failures during OMS burns
- On - failures prior to commanded ignition
- On - failures after commanded shutdown

The OMS engine FDI is initialized at:

- SRB ignition and terminated at either the end of purge after OMS 1 burn or at OPS transition from MM 602 to 603 if in RTLS abort
- OMS ignition minus 15 seconds during any OMS burn and terminated after end of purge

OMS ENGINE FDI PROCESS SUMMARY

1. DETERMINE WHICH ENGINE(S) SHOULD FIRE, IF ANY

These indications are obtained from guidance (ENGINE ON command) and from MNVR Display Interface Processor (DIP) (engine selection for the maneuver)

2. DELTA VELOCITY CHECK

If post-MECO, continue; if pre-MECO, go to step 3.

- a) Check 2-ENG ΔV threshold, or 1-ENG ΔV threshold according to the number of engines that should be firing.
- b) If ΔV check fails, set VEL FAIL flag = true.

3. CHAMBER PRESSURE CHECK

- a) If no commfault in the chamber pressures reading, continue; if commfault is present, go to step 3c.
- b) Check left and right OMS engines' chamber pressures (P_c) against threshold. Set P_c fail flag if $P_c > 76$ percent and engine should be off, or $P_c < 76$ percent and engine should be on. (The left (or right) P_c fail flag results in a class 2 alarm with left (or right) OMS P_c message, left (or right) OMS C&W light, master alarm, and tone.)
- c) If P_c fail flag is true and VEL FAIL flag is true, set OMS ALERT flag "true" for failed engine. (OMS ALERT flag is used in the MNVR DIP to display a "+" by the failed engine. This flag is also used in the next step. Note: the OMS ALERT flag cannot be set if P_c is commfaulted.)

4. REQUEST ENGINE SHUTDOWN

If the OMS ALERT flag is true for an engine and the engine switch (panel C3) for that engine is off, set left (or right) OMS fail indication to be used by the OMS firing sequence and guidance.

- S TRK self-test monitoring
- S TRK display processing processes the following:
 - S TRK magnitudes from hardware
 - Input vertical displacements from hardware
 - Input horizontal displacements from hardware
 - Status line messages
 - Built-in Test Equipment (BITE) indications
 - Angle differences and angle errors for sighting table
 - Sighting table clear keyboard command
 - Sighting table delta times
- COAS control processing processes:
 - COAS MARK command
 - COAS ACCEPT keyboard command
 - Calls COAS processing module if either a current MARK or ACCEPT command has been received

S TRK self-test: Consists of determining if there is a bright-object-alert. If there is a bright-object-alert, an OCCULT status message is displayed and the test is terminated. If there is no bright-object-alert, then self-test provides a series of test conditions to test the hardware, after which pass/fail signals are sent to the supervisor.

COAS processing: This will be discussed in the following section.

S TRK processing: First by using selected IMU data, determines if the S TRK is in an OCCULT position by calling the occult-test module. If there is no occultation, then either a crew-input star or software selected star of opportunity will be tested to determine if it is within the S TRK field of view. If no star is found, OUT-FOV is displayed. If the candidate star is within the S TRK field of view, SEARCH will be displayed.

- Two types of searches may occur - offset scan or full field scan
- If, during a search period, a star is locked onto, data filter processing will commence.
- If the data filter processing determines that false tracking has occurred. "FALSE TK" is displayed, a "BREAK TRACK" command is issued, and searching is resumed. If no false tracking occurs, searching does not resume.
- If no good star is locked onto in either search interval, NO STAR will be displayed.
- If there is occultation of the S TRK LOS, OCCULT is displayed. Last pass logic is executed when all other S TRK processing is complete:
 - If the -Z S TRK processing is in progress, then the star to COAS processing module is called.

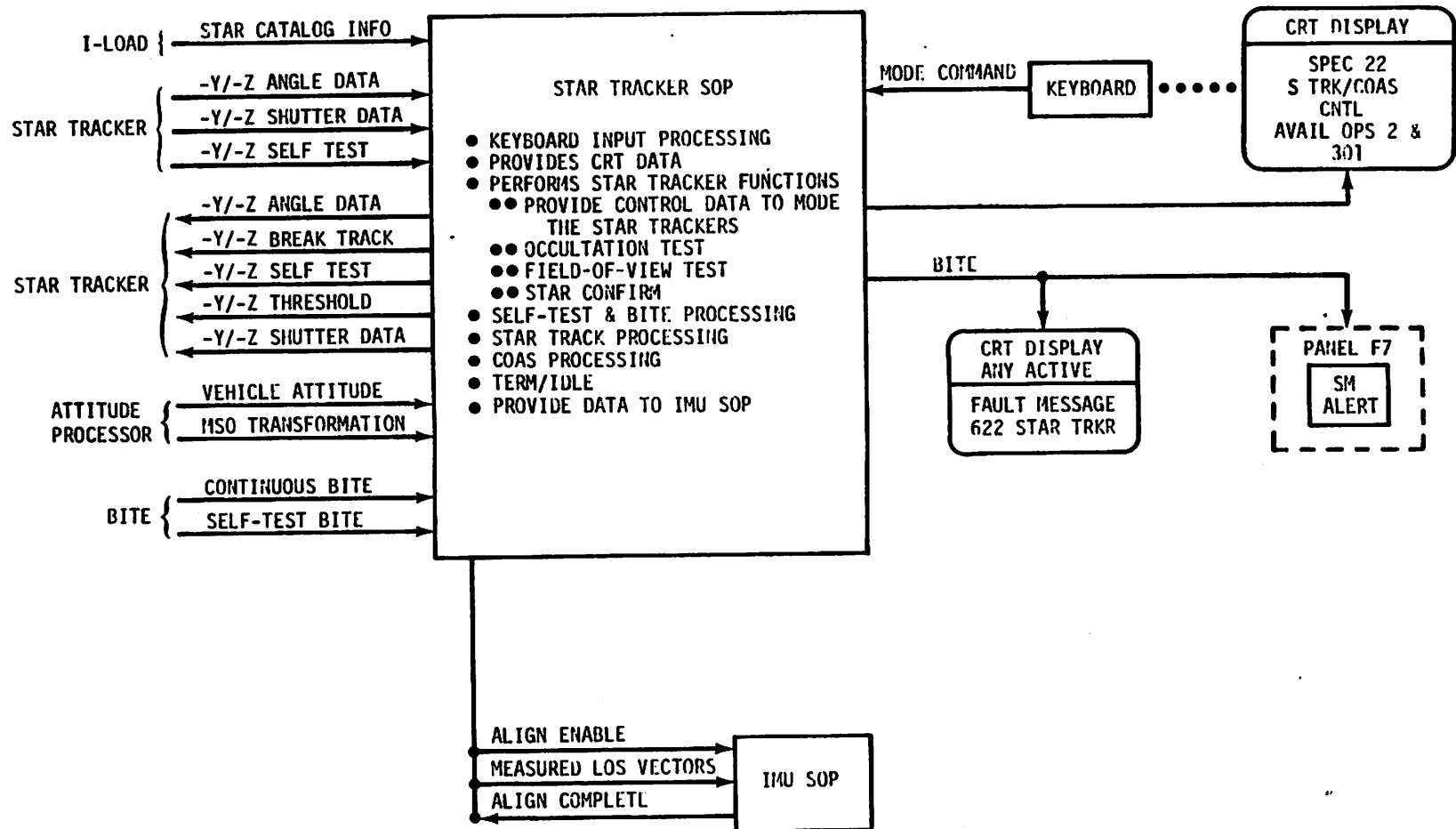


Figure 2.1-23.- Star tracker software data flow.

- If the crew provided a candidate star, the software automatically modes to TERM/IDLE.
- Cleans up and reinitializes

S TRK TERM/IDLE: Initializes to zero the output commands to the S TRK. However, it initializes the horizontal and vertical offset commands equal to the smallest positive angular values available. TERM/IDLE also initializes software flags, including mode indicator for display. As long as a S TRK is commanded to TERM/IDLE. TERM/IDLE processing will occur once each time a keyboard entry is executed.

Occultation test: Checks the Orbiter's position against the location of the Sun, Moon, and Earth to determine occultation (Sun, Moon, or Earth is in the Field-of-View (FOV) of the S TRK being tested).

Field-of-view test: Determines horizontal and vertical offset angles for a S TRK and the associated candidate star. The angles are then compared to a preset FOV test angle to determine if the candidate star is with the S TRK FOV.

Data filter processing: Performs first pass initialization. It checks S TRK input data for unreasonableness. As long as data is reasonable, averages of the gimbal angles and offset angles are computed. (If the data becomes unreasonable data filter processing is terminated and "last pass" processing is initiated.) During this process vehicle body rate is checked, and if it is too high, HI-RATE is displayed and again data filter processing is terminated and last pass processing is initiated. If the vehicle body rate is acceptable, then the "star confirm" logic is executed which determines if false tracking is occurring. Star confirm logic also determines if the star sighting data are acceptable for storage into the star sighting table and controls how the data is moved. If the star confirm logic stores data on two or more stars, then the two stars with a separation angle closest to 90° are selected automatically for use by IMU alignment, and star alignment is enabled.

S TRK to COAS processing: Determines the location of a star in the -Z COAS coordinates when the star is being tracked in the -Z S TRK axis. The star unit line-of-sight vector is used to compute the COAS offset angles for display.

2.1.13 Crew Optical Alignment Sight

The COAS software is controlled and processed by the S TRK SOP which contains the COAS processing module. COAS processing occurs only if there is a current MARK and/or a current ACCEPT command. The three modes of deselect, calibration, and sight are mutually exclusive. If there is a MARK command, then either sight processing or calibration processing occurs. If sight processing occurs, then either COAS target track processing occurs or non-target track processing occurs. If there is an ACCEPT command and the COAS mode is sight, then COAS sighted stars are processed for data placement in the star sighting table.

2.1.14 Microwave Scan Beam Landing System

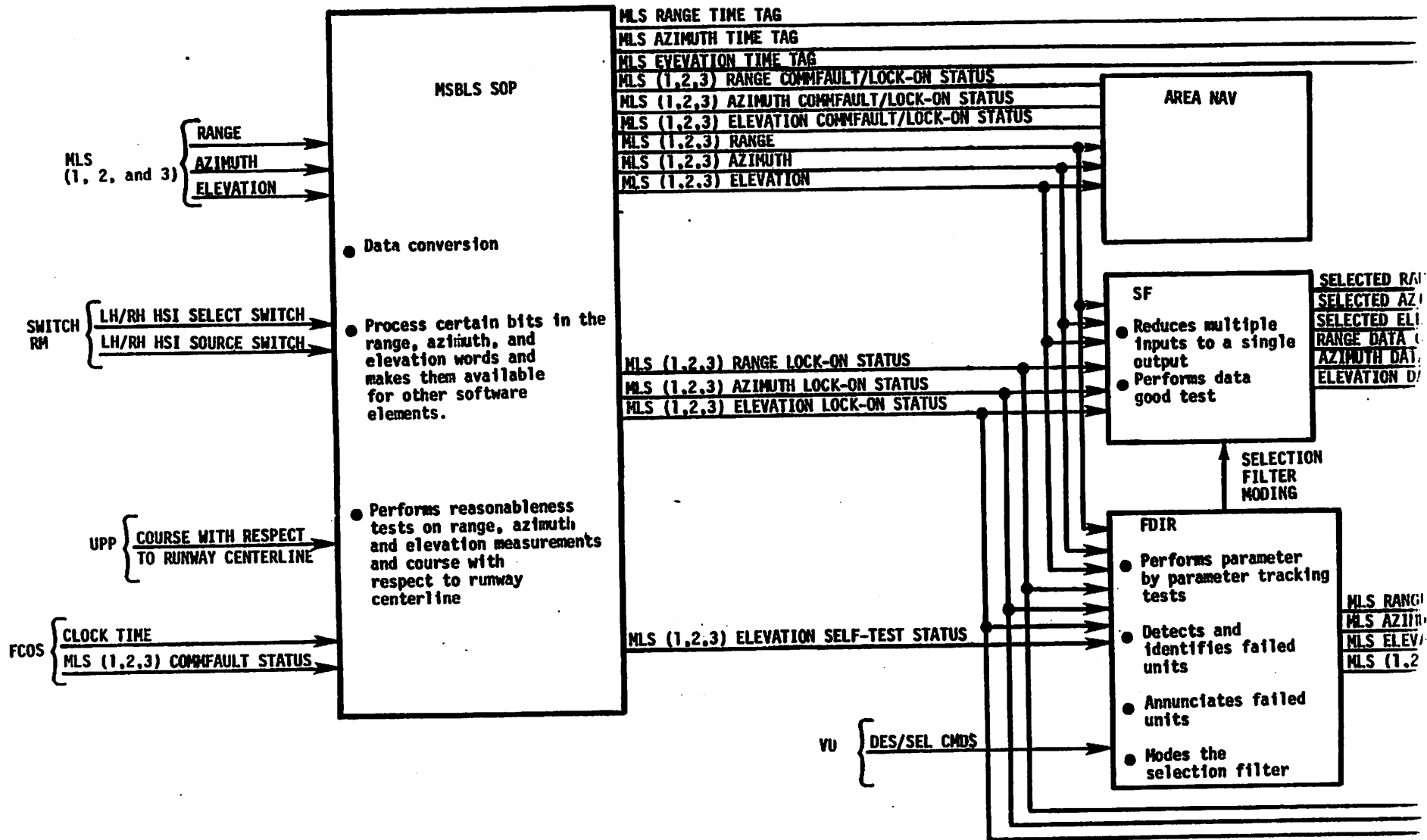
2.1.14.1 SOP.- The three Microwave Scan Beam Landing System (MSBLS) units provide range, azimuth, and elevation referenced to antennas located near the runway for navigation updates and display on the Horizontal Situation Indicator (HSI). The SOP performs two major tasks during Major Modes 305 and 603. (fig. 2.1-24).

MSBLS input processing:

- Determines if the rate of change of each measurement is reasonable when compared to a limit.
- Provides a time tag associated with each measurement for navigation.
- Monitors requests for crew display of data on the HSI.

MSBLS data processing:

- Converts the range, azimuth, and elevation words received from the MSBLS input processor to engineering units for the selection filter, FDIR, and area navigation elements.
- Generates a range, azimuth, and elevation lock-on status for the selection filter, FDIR, and GNC SYS SUMM 1 display.
- Generates a range, azimuth, and elevation commfault/lock-on status for the area navigation function.
- Generates an elevation self-test failure indication to FDIR to assist in resolving dilemmas.
- Determines if course with respect to runway centerline is within limits.



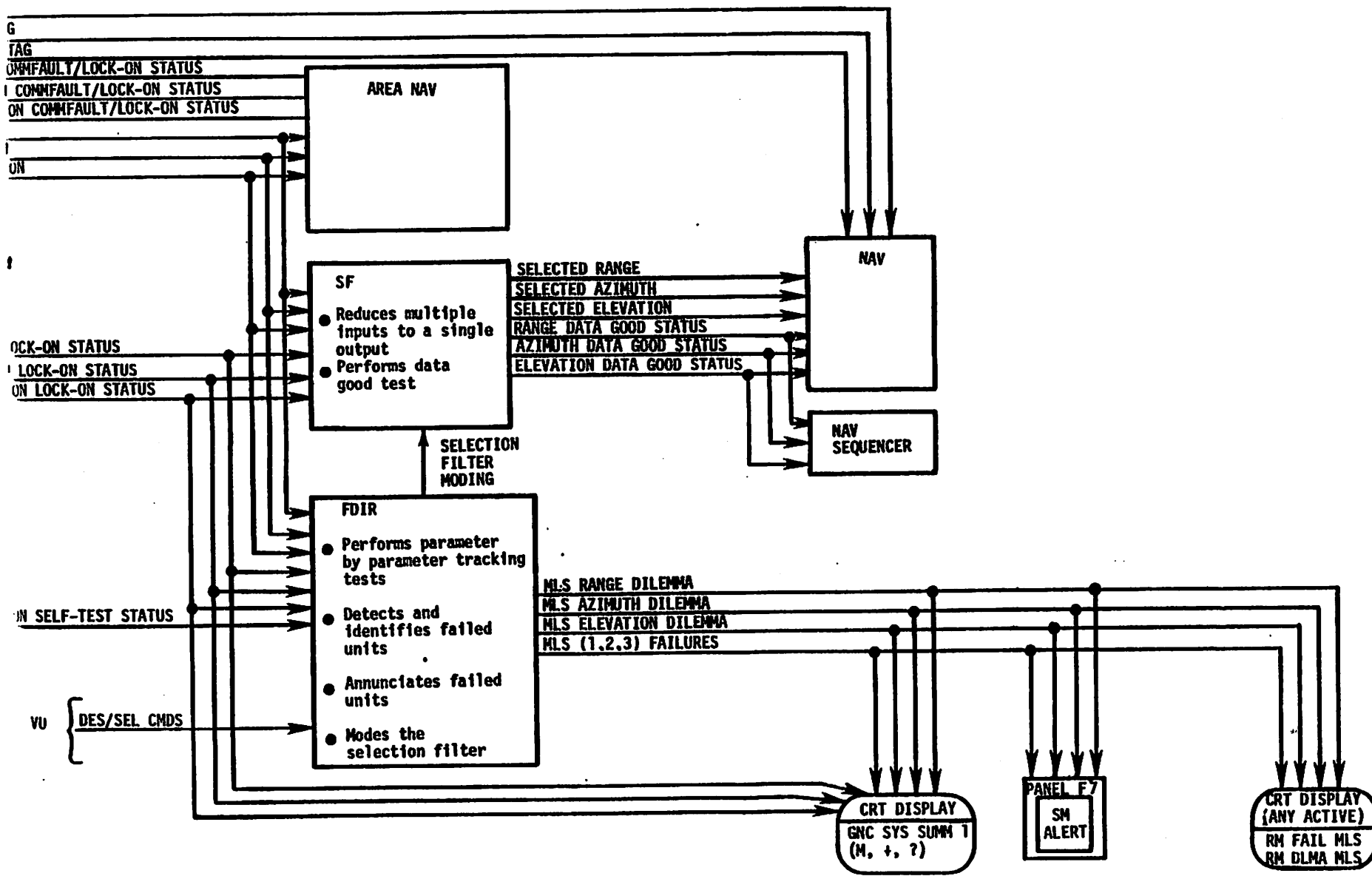


Figure 2.1-24.- The MSBLS data flow.

2.1.14.2 RM.- There are no CRT displays associated with MSBLS RM control. MSBLS RM results are displayed on GNC SYS SUMM 1 display. The only other indication of an MSBLS problem is the appearance of certain flags and anomalous data on the HSI if the affected unit is chosen for display.

FDIR: The purpose of MSBLS FDIR is to determine which parameters (range, azimuth, and/or elevation) should be declared failed and to mode the selection filter in response to these failures during major mode 305 only.

- Candidates are range, azimuth, and elevation from MSBLS 1, 2, and 3 that have not been previously declared failed.
- A parameter is temporarily withheld from FDIR (but not declared failed) for loss-of-lock or commfault.
- Declaring a parameter failed results in fault annunciation and FDIR downmoding of all parameters associated with the affected MSBLS.

Three good MSBLS's: The following comparisons are made:

$$|P_1 - P_2| \leq K, |P_1 - P_3| \leq K, |P_2 - P_3| \leq K.$$

A parameter (range, azimuth, or elevation) is declared failed if it is the common parameter in two miscomparisons three consecutive times. FDIR is downmoded for the entire MSBLS regardless of which parameter failed.

Two good MSBLS's: The following comparison is made for the remaining unfailed parameters.

$$|P_i - P_k| \leq K$$

If the tolerance is exceeded three consecutive times the presence of elevation self-test for each applicable MSBLS is checked. If the indication of a self-test failure is present, that MSBLS is declared failed and FDIR is downmoded to the case of one MSBLS. In the event that both MSBLS's or neither MSBLS indicates a self-test failure FDIR continues as in the two case and annunciates a dilemma.

One good MSBLS: No FDIR

Selection filter: The purpose of the MLS selection filter is to select single values of range, azimuth, and elevation and to provide an indication of validity (DATA GOOD) for each parameter to navigation and the navigation sequences during major modes 305 and 602.

- Candidates are range, azimuth, and elevation from MSBLS 1, 2, and 3 that have not been declared failed.
- A parameter is temporarily withheld from the selection filter for loss-of-lock or commfault.

Three good MSBLS's: Selects middle values of three ranges, three azimuths, and three elevations.

Two good MSBLS's: Averages two ranges, two azimuths, and two elevations and performs a data good test for each parameter as follows.

$$|P_i - P_k| \leq K$$

Whenever the comparison fails, navigation will not utilize the affected parameter.

One good MSBLS: Passes the range, azimuth, and elevation.

2.1.14.3 MSBLS self-test processing.- The purpose of MSBLS self-test processing is to compare the hardware generated range, azimuth, and elevation to onboard limits contained in Vehicle Utility (VU) software. Additionally, VU software receives an indication associated with each MSBLS's internal hardware status.

GNC OPS 9 (preflight): A request for any MSBLS to be in the self-test mode is made through item entries from the SENSOR SELF-TEST display. VU software performs its comparison in two steps. First, the range, elevation, and plus azimuth is compared and then the minus azimuth is compared. The VU software will not transition to the minus azimuth comparison if the plus azimuth is out-of-tolerance. This causes the display to freeze until the test is terminated by item entry. The transition to the minus azimuth will occur if range and/or elevation is out-of-tolerance. An out-of-tolerance MSBLS is determined by visual inspection of the various indicators on the display (e.g., BITE, †, ‡). There is no provision to deselect an MSBLS from software candidacy in GNC OPS 9.

GNC OPS 8 (on-orbit): Self-test operates in the manner described for GNC OPS 9, except that all MSBLS's are tested simultaneously utilizing the SENSOR TEST display. Additionally, the crew can deselect MSBLS's as FDIR and selection filter candidates.

2.1.14.4 BFS MSBLS.- There is no processing of MSBLS by the BFS.

2.1.15 Radar Altimeter

2.1.15.1 SOP.- The two RA units provide altitude from the onboard antennas to the nearest point on the ground terrain for utilization by guidance and for display on the Altitude/Vertical Velocity Indicator (AVVI). The SOP performs one task during MM 603 and 305. The RA data flow is shown in figure 2.1-25.

RA data processing:

- Converts the altitude received from the RA to engineering units for the selection filter and AVVI processor.
- Generates a lock-on status for the selection filter.

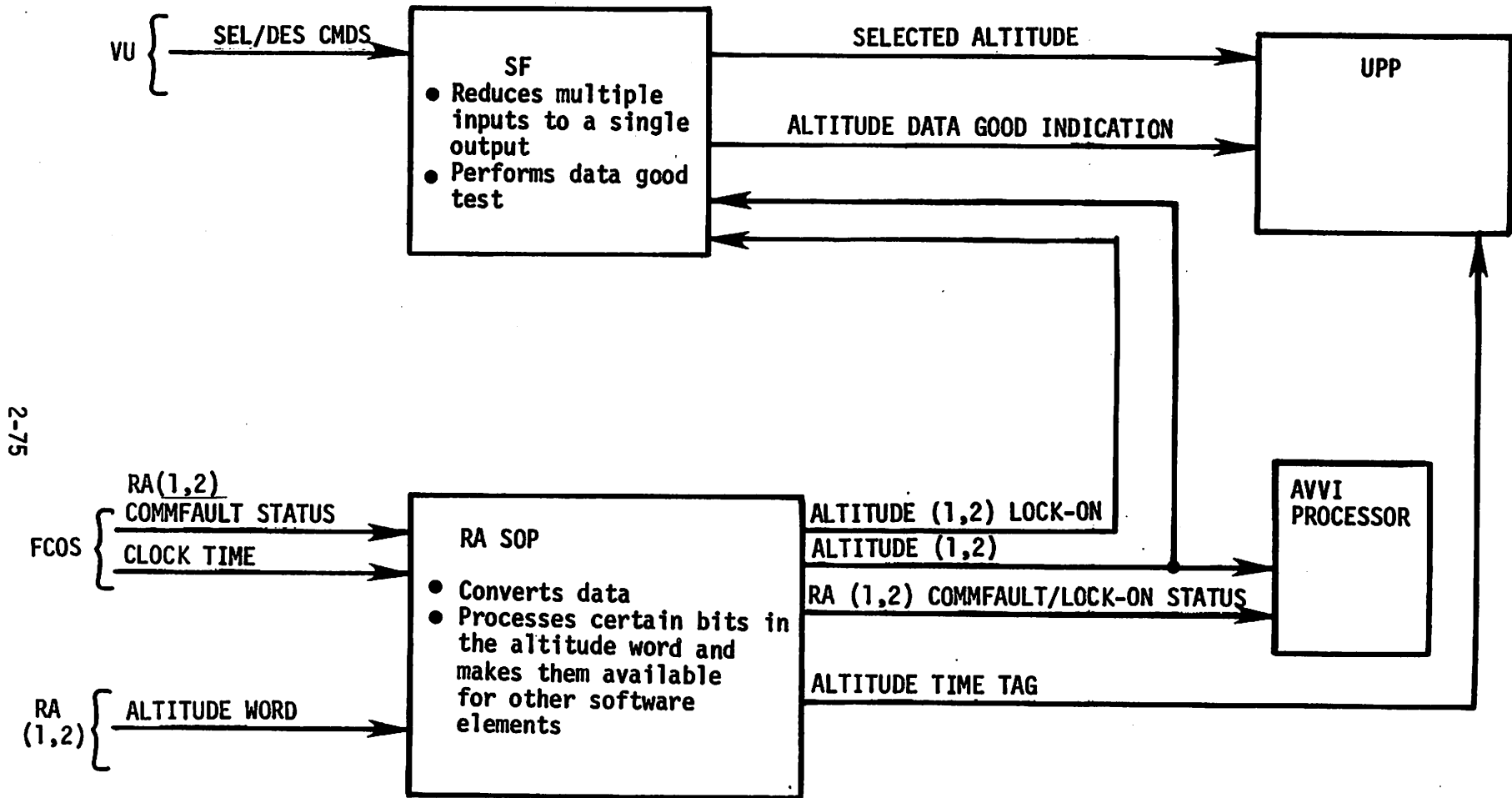


Figure 2.1-25.- Radar altimeter data flow.

- Generates a commfault/lock-on status for the AVVI processor.
- Provides a time tag associated with the altitude measurement for User Parameter Processing (UPP).

2.1.15.2 RM.- RA RM is not displayed or controlled on a CRT display. The only indication of an RA failure is the presence of an OFF flag on the AVVI display if a commfault or loss-of-lock occurs and the associated RA is chosen for display.

FDIR: There is no FDIR associated with the radar altimeter data processing.

Selection filter: The purpose of the RA selection filter is to select a single altitude and to provide an indication of altitude validity (DATA GOOD) to UPP during Major Mode 305.

- Candidates are altitude from RA1 and RA2.
- An altitude is temporarily withheld from the selection filter for loss-of-lock or commfault.

Two RA's: Average two altitudes and performs a data good test as follows:

$$|R_1 - R_2| \leq K$$

Whenever the comparison fails, guidance will not utilize RA data.

One RA: Passes the altitude.

2.1.15.3 RA self-testing processing.- The purpose of RA self-testing processing is to compare the hardware-generated altitude to an onboard limit contained in VU software. Additionally, VU software receives a validity indication associated with each RA lock-on status.

GNC OPS 9 (preflight): A request for any RA to be in the self-test mode is made through item entries from the SENSOR SELF-TEST display, VU software performs its comparison and freezes the data for display 10 seconds after a request for self-test is initiated. An out-of-tolerance RA is determined by visual inspection of the various indications on the display (e.g., BITE, †, ‡). There is no provision to deselect an RA from software in GNC OPS 9.

GNC OPS 8 (on-orbit): Self-test operates in the same manner described for GNC OPS 9 except that both RA's are tested simultaneously utilizing the SENSOR TEST display. Additionally, the crew can deselect RA's as selection filter candidates.

2.1.15.4 BFS RA.- The BFS processes RA for display on the AVVI during Major Modes 603 and 305. The BFS dedicated display processor will output RA1 if it has valid lock-on. In the event it does not, RA 2 will be utilized. 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.

2.1.16 Tactical Air Command and Navigation

2.1.16.1 SOP.- The three tactical air command and navigation (tacan) units provide slant range and magnetic bearing to the selected tacan ground station for navigation updates and for display on the HSI and the HORIZ SIT display. The tacan SOP performs four major functions during Major Modes 304, 305, 602, and 603. (fig. 2.1-26)

Self-test processing:

- Performs self-test processing in response to tacan RM.
- Limit checks the results of self-test to detect failures.
- Monitors the status of hardware internal checks to detect failures.
- Provides an indication to RM and to the HORIZ SIT display that self-test is in progress.
- Provides range and bearing self-test failure flags for TACAN RM utilization.

NOTE: Self-test processing is only valid for tacans in the GPC mode of operation. In the event that the tacans are in manual the SOP will immediately exit self-test and indicate that both tacans remain in a dilemma.

Tacan data processing:

- Provides a time tag associated with the tacan measurement for utilization by RM.
- Converts the range and bearing input words to engineering units for UPP, area navigation, and RM.
- Generates a range and bearing lock-on status for RM.
- Generates a range and bearing commfault/lock-on status for area navigation and the HORIZ SIT and GNC SYS SUMM 1 displays.
- Computes the current tacan station identification for the site hookup function.
- Monitors crew requests for data to be displayed on the HSI.

Antenna selection processing:

- Selects lower antenna whenever a new tacan channel is commanded and provides the site lookup function with the appropriate tacan number and station identification.
- Changes antennas based on range lock-on status and time.
- Sets a discrete to prevent the tacan hardware from internally commanding an antenna change.

NOTE: Normal utilization of tacan antennas are controlled by the SOP when the antenna switch is in AUTO. This auto selection logic can be inhibited if the crew selects either the UPPER or LOWER antenna switch position.

Tacan channel command processing:

- Converts the channel number selected by navigation for output to the tacan/MDM control registers.
- Formats the tacan control word to specify, channel, mode, and test conditions for output to the tacan/MDM control registers.

NOTE: During normal processing, navigation and the tacan SOP will automatically select tacan stations based on availability of stations and state vector and trajectory dependent parameters. The auto selection process is only applicable for tacans in the GPC mode of operation. The crew can override auto selection by placing the tacan(s) in the manual mode (T/R) and selecting stations utilizing the thumbwells on the tacan control panel.

2.1.16.2 Tacan RM.- The purpose of tacan RM is to select the best estimate of range and bearing from the associated tacan units for utilization by navigation. Tacan RM is displayed and controlled from the HORIZ SIT display. Additionally, tacan RM results are displayed on the GNC SYS SUMM 1 display. Tacan RM processing consists of performing eight major functions during Major Modes 304, 305, 602, and 603.

Tacan moding executive:

- Provides the overall sequencing and control for tacan RM processing.
- Determines the range and bearing usability status for the range module and bearing cone-of-confusion calculator.
- Selects a tacan number (1, 2, or 3) for utilization by the cone-of-confusion calculator and navigation.
- Generates a request to change channels based on time and range lock-on status to navigation.
- Determines tacan operational regimes.
- Generates a validity indication (DATA GOOD) associated with range and bearing.

NOTE: Range and bearing usability is a function of commfault, lock-on, power, tacan common channel considerations, and crew select/deselect commands.

Tacan operational regimes are based on altitude above the ground, elevation angle between the Orbiter and the ground station, and ground speed.

Cone-of-confusion calculator: The cone-of-confusion calculation is based on utilizing a selected tacan number in combination with the bearing usability status from the tacan moding executive to determine when the Orbiter is in the bearing cone-of-confusion. During periods when the

Orbiter is in this regime an indication is provided to navigation and the bearing module to not process bearing data.

Bearing module: The bearing module provides the moding for the bearing postselection filter. This involves FDI, selection filtering, and generation of data validity (DATA GOOD); determines failures based on self-test results; and issues various indications for the HORIZ SIT and GNC SYS SUMM 1 displays. Additionally, the bearing module adjusts for readings around 0°. This adjustment considers the fact that valid bearing data from the units could be different. (e.g., bearing 1 = 359°, bearing 2 = 1°, bearing 3 = 358°).

1. Bearing FDI: The purposes of bearing FDI is to determine which bearing parameter should be declared failed.
 - Bearing candidacy is based on the usability status from the cone-of-confusion calculator.
 - Bearing data, self-test in progress, and self-test failures are provided by the tacan SOP.

Three good bearing parameters: The following bearing comparisons are made:

$$|B_1 - B_2| \leq K, |B_1 - B_3| \leq K, |B_2 - B_3| \leq K$$

A bearing parameter is declared failed if it is the common parameter in two miscomparisons three consecutive times. FDI for the bearing parameter is downmoded to the case of two parameters.

Two good bearing parameters: The two bearing parameters are compared to a tolerance as follows:

$$|B_i - B_j| \leq K$$

If the tolerance is exceeded an indication of invalid data (DATA GOOD) is sent to the bearing postselection filter. If a miscompare occurs three consecutive times, tacan RM issues a request to the tacan SOP for the applicable tacans to enter the self-test mode. The results are returned to tacan RM for evaluation. If only one bearing parameter is indicated as failed, bearing FDI downmodes to the case of one good parameter. In the event that both bearing parameters or neither bearing parameter indicate a failure, FDI cannot isolate the failure and a dilemma is annunciated.

NOTE: Only bearing self-test results are evaluated by FDI. In the event the applicable tacans are not in GPC, FDI will receive an indication that both tacans are failed for bearing. FDI processing for range will commence at the termination of self-test regardless of the results for bearing.

One good bearing parameter: No FDI

2. Bearing selection filter: The purpose of the bearing selection filter is to provide bearing data to the bearing postselection filter.

Three good bearing parameters: Middle value selection of the three bearing parameters

Two good bearing parameters: Averages the two bearing parameters

One good bearing parameter: Passes the single bearing parameter

3. Bearing postselection filter:

- Outputs selected bearing data and an indication of validity (DATA GOOD) to navigation.
- Adjust the bearing tolerance utilized to monitor the received bearing for changes between successive cycles around 0° (e.g., one cycle = 1° and the next = 359°).
- Smooths bearing data for jumps and data dropouts.

NOTE: The bearing postselection filter tests the incoming bearing against a reference value which is the prior bearing. If the difference does not exceed a tolerance the bearing is utilized by navigation, and the reference is updated to this value of incoming bearing. Whenever the tolerance is exceeded the reference remains unchanged and the new incoming bearing is compared to it. In the event the tolerance is exceeded 10 consecutive times the reference is reinitialized to the current incoming bearing and testing continues on a cyclic basis.

Range module: The range module provides the moding for the range post-selection filter. This involves FDI, selection filtering, and generation of data validity (DATA GOOD); determines failures based on self-test processing results; and issues various indication for the HORIZ SIT and GNC SYS SUMM 1 displays.

1. FDI: The purpose of range FDI is to determine which range parameter should be declared failed.
 - Range candidacy is based on the usability status from the tacan moding executive.
 - Range data, self-test in progress, and self-test failures are provided by the tacan SOP.

Three good range parameters: The following range comparisons are made:

$$|R_1 - R_2| \leq K, |R_1 - R_3| \leq K, |R_2 - R_3| \leq K$$

A range parameter is declared failed if it is the common parameter in two mismatches three consecutive times. FDI for the range parameter is downmoded to the case of two parameters.

Two good range parameters: The two range parameters are compared to a tolerance as follows:

$$|R_i - R_j| \leq K$$

If the tolerance is exceeded, an indication of invalid data (DATA GOOD) is sent to the bearing postselection filter. If a miscompare occurs three consecutive times, tacan RM issues a request to the tacan SOP for the applicable units to enter the self-test mode. The results are returned to tacan RM for evaluation. If only one range parameter is indicated as failed, range FDI downmodes to the case of one good parameter. In the event both range parameters or neither range parameter indicates a failure, FDI cannot isolate the failure and a dilemma is annunciated.

NOTE: Only range self-test results are evaluated by FDI. In the event the applicable tacans are not in GPC, FDI will receive an indication that both tacans are failed for range. FDI processing for bearing will commence at the termination of self-test regardless of the results for range.

One good range parameter: No FDI

2. Range selection filter: The purpose of the range selection filter is to provide range data to the range postselection filter.

Three good range parameters: Middle value selection of the three range parameters.

Two good range parameters: Averages the two range parameters.

One good range parameter: Passes the single range parameter.

3. Range postselection filter:

- Outputs selected range data and an indication of validity (DATA GOOD) to navigation.
- Smooths range data for jumps and data dropouts.

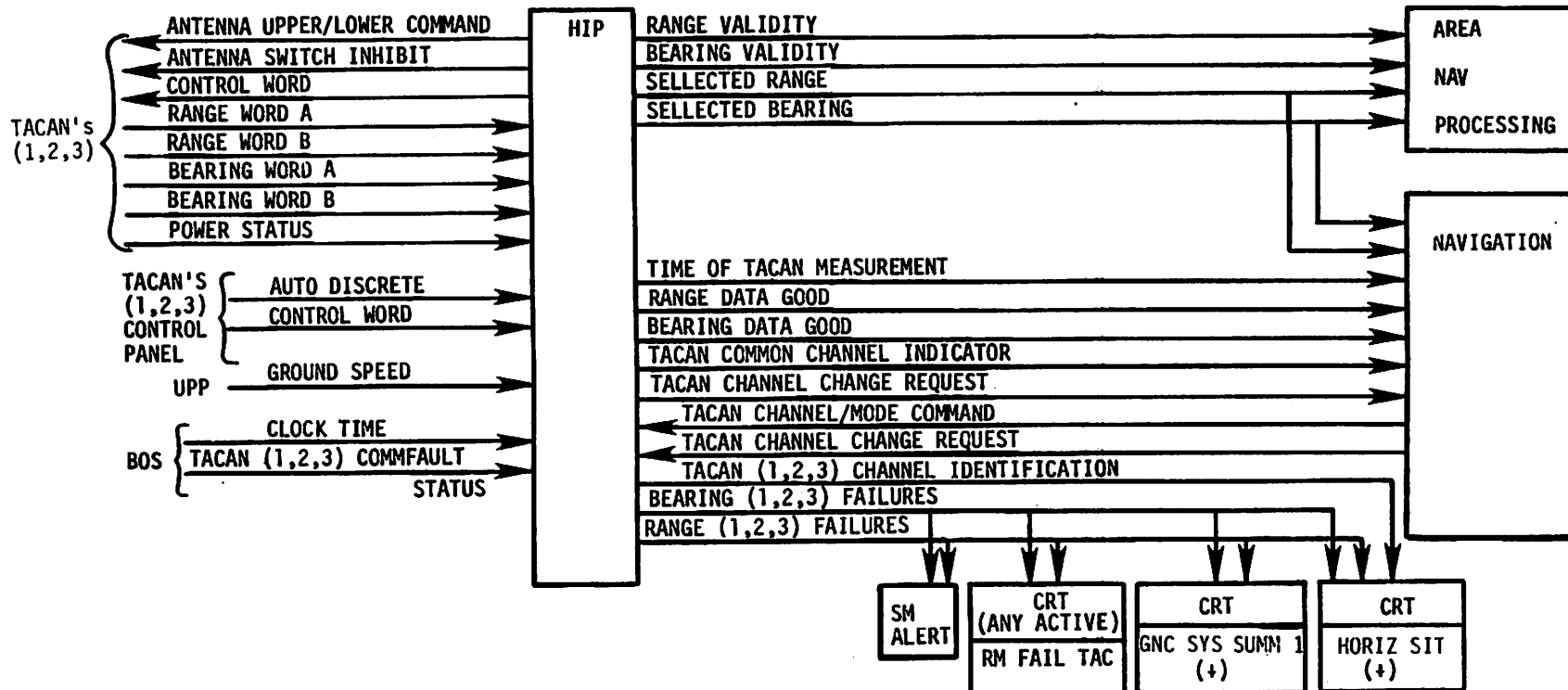
NOTE: The range postselection tests the incoming range against a reference value which is the previous range. If the difference does not exceed a tolerance, the range utilized by navigation and the reference is updated to this value of incoming range. Whenever the tolerance is exceeded, the reference remains unchanged and the new incoming range is compared to it. In the event the tolerance is exceeded five consecutive times, the reference is reinitialized to the current incoming range and testing continues on a cycle basis.

2.1.16.3 Tacan self-test processing.- The purpose of tacan self-test processing is to compare the hardware generated range and azimuth to onboard limits contained in VU software. Additionally, VU software receives an indication associated with each tacan's internal hardware status.

GNC OPS 9 (preflight): A request for any tacan to be in the self-test mode is made through item entries from the SENSOR SELF-TEST display. VU software performs its comparison and freezes the data for display 10 seconds after a request for self-test is initiated. Any out-of-tolerance tacan is determined by visual inspection of the various indications on the display (e.g., BITE, †,+). There is no provision to deselect a tacan from software candidacy in GNC OPS 9.

GNC OPS 8 (on-orbit): Self-test operates in the same manner described for GNC OPS 9 except that all tacans are tested simultaneously utilizing the SENSOR TEST display. Additionally, the crew can deselect tacans from software candidacy.

2.1.16.4 BFS tacan HIP.- The purpose of the tacan HIP is to control and monitor the operation of the three tacan systems and to process tacan data for utilization by navigation. The tacan HIP operates during both preengaged and engaged modes during Major Modes 304, 305, 602, and 603 (fig. 2.1-27).



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Figure 2.1-27.- The BFS tacan data flow.

Four major tasks are performed.

Tacan data processing:

- Time tags the input data for utilization by navigation.
- Determines range and bearing data validity for utilization by the tacan data selection and area navigation processors and the HORIZ SIT and GNC SYS SUMM 1 displays.
- Converts the range and bearing data for utilization by the tacan data selection processor.
- Computes the current channel identification for utilization by the tacan data selection processor.

NOTE: Range and bearing usability is based on lock-on, commfault, and power status. Additionally, while preengaged the BFS will not process range and bearing data if the primary flight software enters the tacan self-test mode of operation.

Antenna selection processing:

- Selects lower antenna whenever a new tacan channel is commanded.
- Changes antennas based on range lock-on status and time.

NOTE: Normal utilization of tacan antennas is controlled by the HIP when the antenna switch is in AUTO. This auto selection logic can be inhibited if the crew selects either the UPPER or LOWER antenna switch position.

Tacan channel command processing:

- Converts the channel number selected by navigation for output to the tacan/MDM control registers.
- Formats the tacan control word to specify, channel, mode, and test condition for output to the tacan/MDM control register.

NOTE: During normal processing, navigation and the tacan HIP will automatically select tacan stations based on availability of stations and on state vector and trajectory dependent parameters. This auto selection process is only applicable for tacans in the GPC mode of operation. The crew can override auto selection by placing the tacan(s) in the manual mode (T/R) and selecting stations utilizing the thumbwells on the tacan control panel.

Data selection processing: The purpose of tacan data selection is to select the best estimate of range and bearing along with associated validity indicators.

1. Tacan common channel processing: This processing determines if a common channel exists among the tacans and provides this to navigation.

In the event that a common channel does not exist, navigation will not be updated with tacan data.

2. Range acquisition filter:

- Determines range operational regime based on ground speed.
- Determines if adequate range lock-on has been achieved for tacans on a common channel.

3. Range usability status: This processing determines the usability of range data based on power, commfault, and lock-on status and common channel considerations.

4. Range selection filter: This processing encompasses the functions of both selection filtering and two-level fault detection.

- Three good ranges: Middle-value selection of the three ranges.
- Two good ranges: Range associated with the lowest numbered tacan.
- One good range: Outputs the single range.

At the two level the ranges are compared against a tolerance as follows:

$$|R_i - R_j| \leq K$$

In the event of a miscompare navigation will not be updated with range data.

5. Range postselection filter:

- Outputs selected range data and an indication of validity (DATA GOOD) to navigation.
- Smooths range data for jumps and data dropouts in the manner described for the primary system.

6. Bearing acquisition filter:

- Determines bearing operational regime based on ground speed.
- Determines if adequate bearing lock-on has been achieved for tacans on a common channel.

7. Bearing usability status: This processing determines the usability of bearing data based on power, commfault, and lock-on status and common channel considerations.

8. Bearing selection filter: This processing encompasses the functions of both selection filtering and two level fault detection. Additionally, the bearing selection filter adjusts for readings around 0° in the same manner as described for the primary system.

- Three good bearings: Middle value selection of the three bearings.
- Two good bearings: Bearing associated with the lowest numbered tacan.
- One good bearing: Outputs the single bearing.

At the two level the bearings are compared against a tolerance as follows:

$$|B_i - B_j| \leq K$$

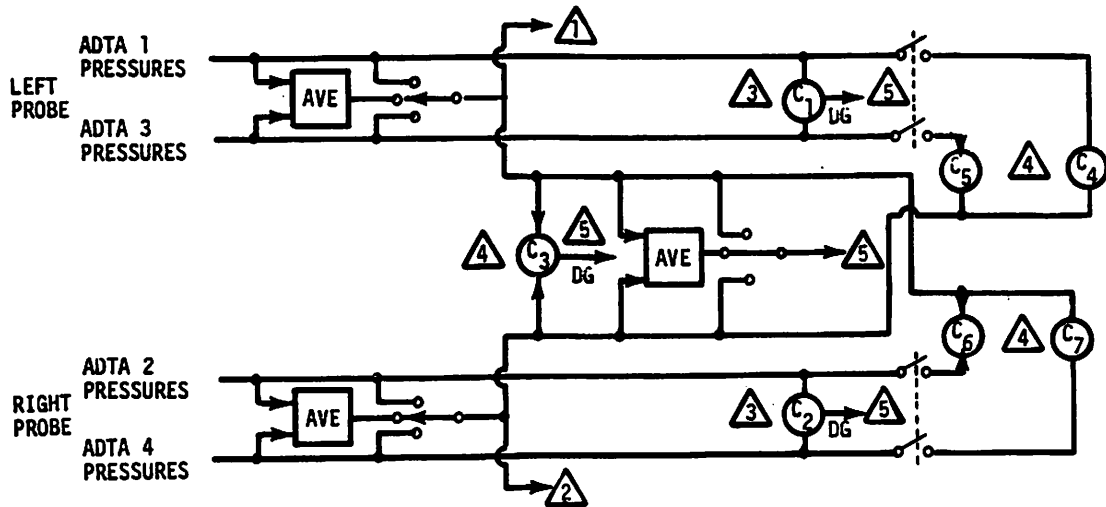
In the event of a miscompare, navigation will not be updated with bearing data.

9. Bearing postselection filter:

- Outputs selected bearing data and an indication of validity (DATA GOOD) to navigation.
- Adjusts bearing data tolerance utilized to monitor the received bearing in the same manner as described for the primary system.
- Smooths bearing data for jumps and data dropouts.

2.1.17 Air Data System

2.1.17.1 RM.- The Air Data System (ADS) provides pressure information relative to the movement of the Orbiter in the air mass (i.e., flight environment). Sensors to provide this information consists of two probes and four Air Data Transducer Assemblies (ADTA's). One probe is located on the Orbiter's left side; the other on the Orbiter's right side. Two ADTA's are connected to each probe (fig. 2.1-28). ADS RM provides processing of pressure information prior to its utilization by other software elements. This processing consists of four major tasks during Major Modes 304, 305, 602, and 603. (fig. 2.1-29)



NOTES:

- ① These pressures are utilized by the ADS SOP in response to a request by the CDR or PLT for left probe data to be displayed on the AMI/AVVI.
- ② These pressures are utilized by the ADS SOP in response to a request by the CDR or PLT for right probe data to be displayed on the AMI/AVVI.
- ③ Comparators C_1 and C_2 are unique constants different for each pressure.
- ④ Comparators C_3 , C_4 , C_5 , C_6 , and C_7 are dynamic functions of pressure, side-slip angle, and M . A unique function exists for each pressure.
- ⑤ A DATA GOOD indication is driven from either C_1 , C_2 , or C_3 depending on ADTA candidacy.

Figure 2.1-28.- The ADTA selection filter/FDIR functional flow.

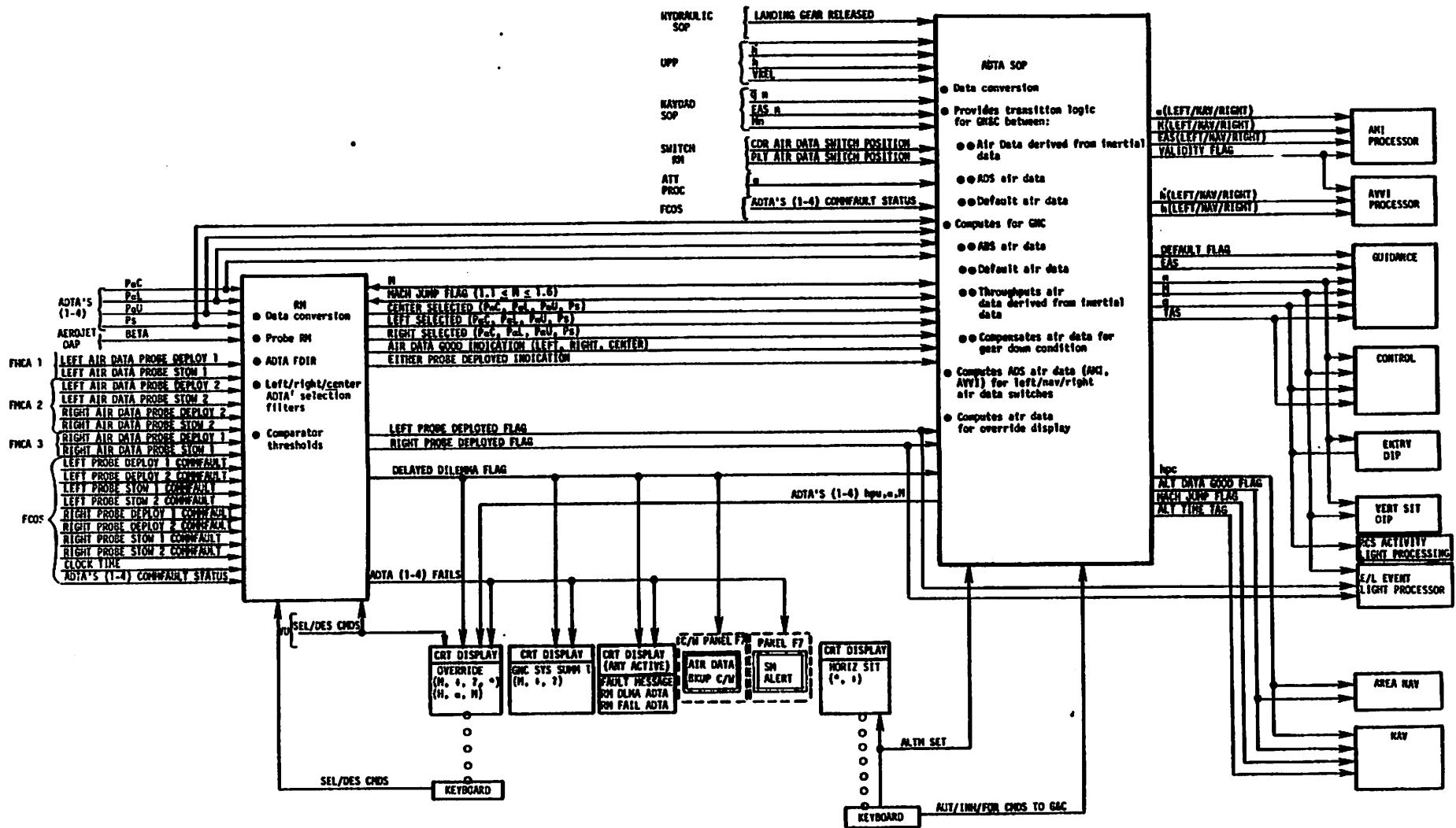


Figure 2.1-29.- Air data subsystem data flow.

Probe RM: The purpose of probe RM is to determine if one or both probes are deployed and to provide this status for other software elements.

- Performs logic to determine if a left probe is deployed by monitoring redundant deploy and stow discrettes from the Forward Motor Control Assemblies (FMCA's) and combining these with their respective commfault status from FCOS.
- Performs logic to determine if a right probe is deployed by monitoring redundant deploy and stow discrettes from the FMCA's and combining these with their respective commfault status from FCOS.
- Generates a deploy indication associated with each probe for ADS SOP and entry/landing event light processing.
- Issues an indication for ADTA FDIR, ADTA selection filter, and ADS SOP processing to commence based on at least one probe being deployed for 15 seconds.

ADTA FDIR: The purpose of ADTA FDIR is to determine which of the four pressures of alpha center pressure ($P_{\alpha C}$), alpha upper pressure ($P_{\alpha U}$), alpha lower pressure ($P_{\alpha L}$), and static pressure (P_s) from each of the four ADTA's should be declared failed and to reconfigure the selection filters accordingly. This process consists of performing a comparison of ADTA 1 and ADTA 3 pressures (C1), 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 conditionally (fig. 2.1-28).

- Candidates are ADTA's 1 through 4 that have not been declared failed or deselected by the crew.
- An ADTA commfault results in all pressures from the ADTA being withheld from FDIR (but not declared failed).
- Declaring a pressure failed ($P_{\alpha C}$, $P_{\alpha U}$, $P_{\alpha L}$, or P_s) results in fault annunciation and FDIR downmoding for the entire ADTA.
- A probe not being deployed results in removal of the ADTA's associated with that probe (e.g., ADTA's 1 and 3 for the left probe) as FDIR candidates.

ADTA FDIR comparisons:

1. Two ADTA's at C1 and C2: In this case, there are two valid pressure sets ($P_{\alpha C}$, $P_{\alpha U}$, $P_{\alpha L}$ and P_s) at each of the C1, C2, and C3 comparison check points. The comparisons of $|P_i - P_j| \leq C_k$ are made at each of the check points.

If a miscomparison occurs three consecutive times at C1 only or C2 only, comparisons C4 and C5 or C6 and C7 are performed. A C4 or C5 miscompare results in ADTA 1 or ADTA 3 (respectively) being failed, while a miscompare at C6 or C7 results in ADTA 2 or ADTA 4

(respectively) being failed. Whenever an ADTA is declared failed for any pressure (PaC, PaU, PaL, or Ps), FDIR ceases for all pressures associated with that ADTA. In the event that the results of comparison checking is inconclusive (neither ADTA fails or both ADTA's fail), the logic continues to try and identify the failed ADTA on subsequent cycles.

A single miscompare at C3 results in the pressure data being declared invalid (DATA GOOD) while three consecutive miscompares results in a dilemma. These indications are utilized by the ADS SOP to mode its computations for GN&C.

2. Two ADTA's at C1 and one ADTA at C2: Same as 1 above for C1, C3, C4, and C5 comparison checkpoints. C6 and C7 are not performed.
3. One ADTA at C1 and two ADTA's at C2: Same as 1 above for C2, C3, C6, and C7 comparison checkpoints. C4 and C5 are not performed.
4. One ADTA at C1 and one ADTA at C2: Same as 1 above for C3. No other comparisons are performed.
5. Two ADTA's at C1 and no ADTA's at C2: C1 comparison is the only check point. The first miscompare would result in a DATA GOOD failure while three consecutive miscompares would result in a dilemma.
6. No ADTA's at C1 and two ADTA's at C2: C2 comparison is the only check point. The first miscompare would result in a DATA GOOD failure, while three consecutive miscompares would result in a dilemma.
7. One ADTA at C1 or C2: No ADTA FDIR
8. No ADTA's at C1 or C2: No ADTA FDIR and a dilemma indication to the ADS SOP

NOTE: During normal processing the M utilized by ADTA FDIR is generated by the ADS SOP based on the pressures supplied by ADTA FDIR. When a dilemma occurs, NAVDAD supplies M. Secondly, during Mach jump $1.1 \leq M \leq 1.6$ a dilemma declaration will not occur for Ps at the C3 comparison checkpoint.

ADTA selection filters.- The purpose of the ADTA selection filters is to provide pressures for the ADS SOP during Major Modes 304, 305, 602, and 603.

1. Left (right) ADTA selection filters:

- Left (right) probe must be deployed.
- Candidates are ADTA's 1 and 3 for the left and ADTA's 2 and 4 for the right that have not been previously declared failed or deselected by the crew.
- An ADTA commfault results in the ADTA being withheld from the selection filter, but not declared failed.

- Two good ADTA's: Averages their output.
- One good ADTA: Passes the output.
- No ADTA's: No output.

2. Center ADTA selection filter:

- Candidates are the left (right) selection filter outputs.
- Averages the left and right selection filter outputs.
- Throughputs the left or right selection filter outputs in the event only a left or right selection filter meets candidacy requirements.
- Throughputs a single ADTA left or right selection filter output in the event only a single ADTA meets candidacy requirements.
- No output in the event that neither the left or right selection filter output meets candidacy requirements.

ADTA data conversion:

- Converts the left and right selection filter outputs to engineering units for utilization by ADTA FDIR and the air data SOP.

2.1.17.2 SOP.- The ADS SOP provides the processing which converts the various ADTA selection filter output pressures into free-stream air data related parameters for utilization by GN&C, dedicated display processing, area navigation, RCS activity light processing, entry/landing event light processing, and ENTRY TRAJ, VERT SIT, and OVERRIDE displays. Additionally, the SOP provides M and an indication of Mach jump to ADTA FDIR. The SOP performs four major tasks during Major Modes 304, 305, 602, and 603 (fig. 2.1-29).

ADTA data conversion:

- Converts the pressures from each ADTA into Mach no. (M), angle-of-attack (α), and uncorrected pressure altitude (hpu) for the OVERRIDE display.

Air data parameters for GN&C: Air data parameters are critical to the accomplishment of GN&C functions, therefore the SOP will transition between various air data sources in order to generate the following parameters.

- dynamic pressure (\bar{q})
- angle-of-attack (α)
- Mach number (M)
- Equivalent Airspeed (EAS)
- True Airspeed (TAS)
- corrected pressure altitude (hpc)

The source for air data parameter generation for GN&C is based on the interrelationships among probe RM, ADTA FDIR, free-stream M, and crew controls on the HORIZ SIT display (fig. 2.1-30).

	MACH REGION	MODING CONDITIONS			OUTPUTS TO GC (% M, \bar{q} , EAS, TAS)	
		PROBE RM STATUS	HORIZ SIT "ADTA TO G&C"	ADTA RM STATUS		
NOMINAL	$M > 2.5$	ONE OR BOTH DEPLOYED	N/A	N/A	AIR DATA DERIVED FROM NAVIGATION	
	$1.6 < M \leq 2.5$		AUT/FOR	O.K.	AIR DATA DERIVED FROM PROBE SENSED PRESSURES	
	$1.1 \leq M \leq 1.6$				AIR DATA DERIVED FROM MACH JUMP PROCESSING	
	$0.2 \leq M < 1.1$				AIR DATA DERIVED FROM PROBE SENSED PRESSURES	
FAILURE	$1.5 < M \leq 2.5$				DG FAILURE OR DILEMMA ③	AIR DATA DERIVED FROM NAVIGATION
	$1.1 \leq M \leq 1.5$			DG FAILURE ③	FREEZE AIR DATA BASED ON MACH JUMP PROCESSING	
	$0.2 \leq M \leq 1.5$			DILEMMA	DEFAULT AIR DATA AND DEFAULT FLAG TO GUIDANCE	
	$0.2 \leq M < 1.1$			DG FAILURE ③	FREEZE AIR DATA DERIVED FROM PROBE SENSED PRESSURES	
CREW	$1.5 < M \leq 2.5$				① INH	N/A
	$0.2 \leq M \leq 1.5$		DEFAULT AIR DATA AND DEFAULT FLAG TO GUIDANCE			
CREW OR RH	$1.5 < M \leq 2.5$		BOTH NOT DEPLOYED ②	N/A	N/A	AIR DATA DERIVED FROM NAVIGATION
	$0.2 \leq M \leq 1.5$					DEFAULT AIR DATA AND DEFAULT FLAG TO GUIDANCE

	MACH REGION	MODING CONDITIONS		OUTPUT TO NAVIGATION (npc.)
		PROBE RM STATUS	ADTA RM STATUS	
NOMINAL	$M > 2.5$	ONE OR BOTH DEPLOYED	O.K.	AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$1.6 < M \leq 2.5$			AIR DATA DERIVED FROM PROBE SENSED PRESSURES AND INDICATION OF MACH JUMP
	$1.1 \leq M \leq 1.6$			AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$0.2 \leq M < 1.1$			AIR DATA DERIVED FROM PROBE SENSED PRESSURES
FAILURE	N/A	DEPLOYED	DILEMMA	FREEZE AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$1.6 < M \leq 2.5$ OR $0.2 \leq M < 1.1$		DG FAILURE ③	AIR DATA DERIVED FROM PROBE SENSED PRESSURES AND SET INDICATION OF ALT DG FAILURE
CREW OR RH	N/A	BOTH NOT DEPLOYED ②	N/A	FREEZE AIR DATA DERIVED FROM PROBE SENSED PRESSURES AND SET INDICATION OF ALT DG FAILURE ④

NOTES

1. ONLY CREW CAN INVOKE INHIBIT COMMAND, SOFTWARE IS INITIALIZED TO AUT
2. IF BOTH PROBES FAIL AFTER AT LEAST ONE PROBE WAS DEPLOYED A DILEMMA IS DECLARED
3. THREE CONSECUTIVE DG FAILURES WILL RESULT IN A DILEMMA
4. IF A PROBE WAS NEVER DEPLOYED npc IS INITIALIZED TO 90,000 FT WITH AN ALT DG FAILURE INDICATION

Figure 2.1-30.- Air data output and moding.

1. Inertially derived air data parameters: Air data parameters based on inertially derived information consists of throughputting \bar{q} , EAS, and M from the Navigation Derived Air Data (NAVDAD) SOP, α from the attitude processor, and TAS from UPP. Whenever air data related parameters are generated in this manner, hpc will not be utilized to update navigation.
2. Probe sensed air data parameters: Air data parameters based on probe sensed data consists of utilizing ADTA center selection filter outputs to generate \bar{q} , EAS, M, α , TAS, and hpc.
3. Default air data parameters: Default air data parameters are comprised of throughputting EAS and M from the NAVDAD SOP (fig. 2.1-31), and an I-load profile of \bar{q} and α as a function of earth relative velocity (V_{REL}) for α and \bar{q} . UPP provides TAS. Whenever air data parameters are generated in this manner, hpc will not be utilized to update navigation.

2-100

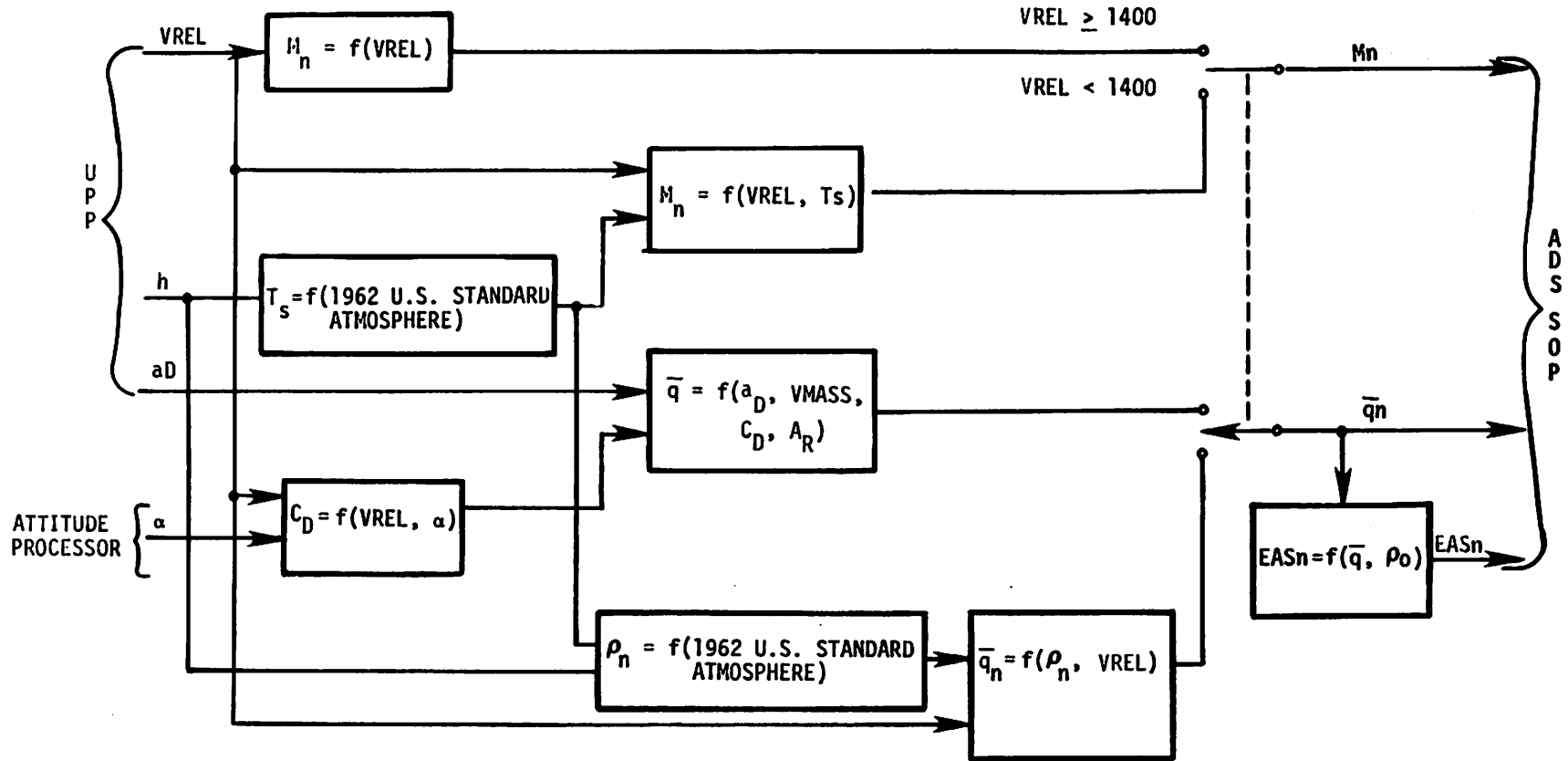


Figure 2.1-31.- The NAVDAD data flow.

4. Mach jump air data parameters: The Mach jump region is predicted to occur when $1.1 < M < 1.6$. During this period α is based on the ADTA center selection outputs. However, M and subsequent parameters such as \bar{q} , EAS, and TAS utilize a modified and limited P_s . Whenever air data parameters are generated in this manner, hpc will not be utilized to update navigation.

NOTE: The algorithm which utilizes P_s to determine altitude is based on the 1962 U.S. Standard Atmosphere. A nonstandard correction is applied to this altitude to account for expected variations in the temperature lapse rates above the landing runways (EDW, KSC). This correction is applied until 12,000 ft, at which time the crew input ALTM from the HORIZ SIT is utilized. The software determined static temperature (T_s) is utilized to compute TAS and has a similar correction scheme all the way to the ground.

Landing gear deployed air data parameters: When the hydraulic SOP indicates that the landing gear is released, the ADS SOP modifies its computations of air data related parameters to account for the disturbance to the air data probes introduced by the nose gear entering the free-stream.

Air data parameters for dedicated displays: The left (right) ADTA selection filter outputs are utilized to calculate left (right) air data parameters if the air data switches on panels F6 and F8 are in the LEFT (RIGHT) positions and the left (right) probe is deployed. Additionally, the deployed probe must have at least one non-commfaulted ADTA.

The parameters provided are hpc and pressure altitude rate (\dot{h}) for AVVI dedicated display processing and M, α , and EAS for AMI dedicated display processing.

The SOP also provides outputs to dedicated display processing in response to a crew request for the NAV position of the AIR DATA switch. Prior to probe deployment or during periods when default air data is invoked, the SOP will throughput M and EAS from the Navigationally-Derived Air Data (NAVDAD) SOP and α from the attitude processor to AMI dedicated display processing. Whenever the SOP is providing air data parameters based on ADTA center selection outputs, Alpha/Mach Indicator (AMI) dedicated display processing receives the same outputs as GN&C (M, α , EAS). AVVI dedicated display processing always receives h and \dot{h} from UPP when the AIR DATA switch is in the NAV position.

2.1.17.3 ADTA self-test processing.- The purpose of ADTA self-test processing is to compare ADTA hardware stored pressures (P_{ac} , P_{aL} , P_{aU} , and P_s) and total temperature (T) to a set of onboard limits contained in VU software. Additionally, VU software receives each ADTA's model/status word which contains the results of various tests performed within the hardware.

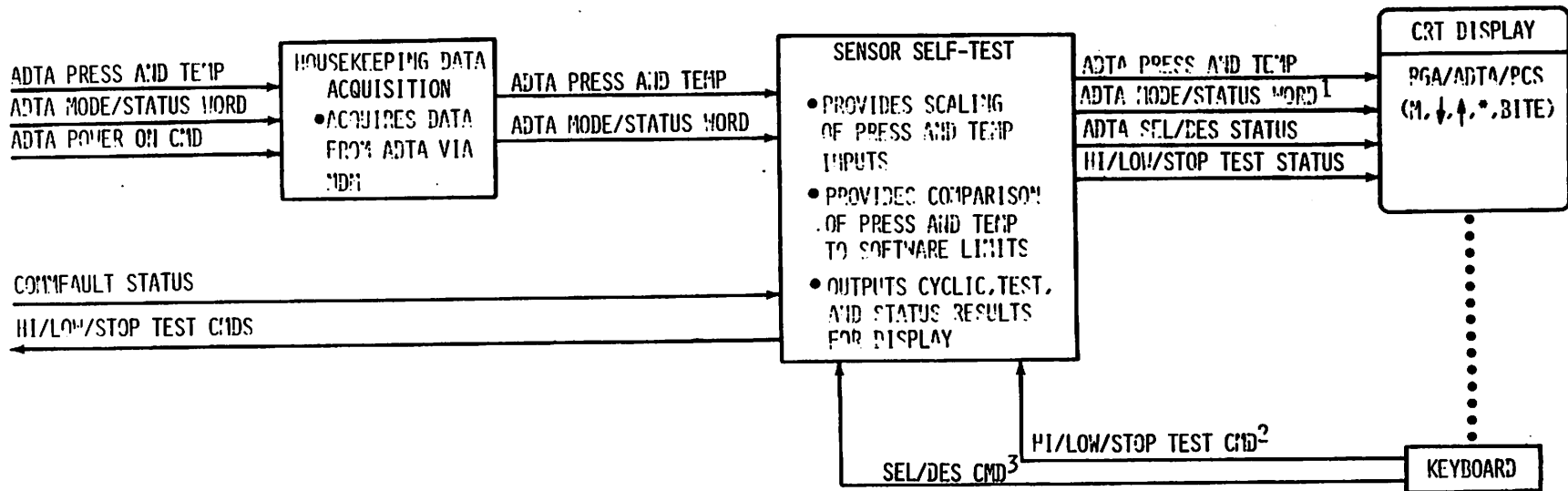
2.1.17.4 Navigation derived air data subsystem operating program.- The purpose of the NAVDAD SOP is to calculate certain air data parameters based on inertial data (fig. 2.1-31). The parameters are \bar{q} , M, and EAS. These parameters are sent to the ADS SOP. NAVDAD parameters are utilized by

guidance, control, and dedicated display processing during Major Modes 304, 305, 602, and 603.

GNC OPS 9 (preflight): A request for any ADTA's to be in either the high self-test mode or low self-test mode is through item entries from the RCS/ADTA/RGA display. VU software performs its comparisons and freezes the data for display 10 seconds after a request for either high or low self-test is initiated. An out-of-tolerance ADTA is determined by visual inspection of the various indications on the display (e.g., BITE, ↑, ↓). There is no provision to deselect an ADTA from software candidacy in GNC OPS 9.

GNC OPS 8 (on-orbit): Self-test operates in the manner described for GNC OPS 9, except that all ADTA's are tested simultaneously, utilizing the RGA/ADTA/RCS display. Additionally, the crew can deselect ADTA's from software candidacy (fig. 2.1-32).

2.1.17.5 BFS air data HIP.- The function of the air data HIP is to control and monitor the operation of the air data system and to process air data information for utilization by GN&C. The air data HIP operates both preengaged and engaged in Major Modes 304, 305, 602, 603 (fig. 2.1-33). Five major tasks are performed.

**NOTES**

1. 'BITE' IS LATCHED IF DETECTED DURING 10-SECOND TEST PERIOD
2. WILL SIMULTANEOUSLY TEST RGA'S AND ADTA'S
3. STATUS OF SEL/DES IS CARRIED ACROSS OPS TRANSITION AND IS READ BY REDUNDANCY MANAGEMENT

Figure 2.1-32.- GNC 8 air data data flow.

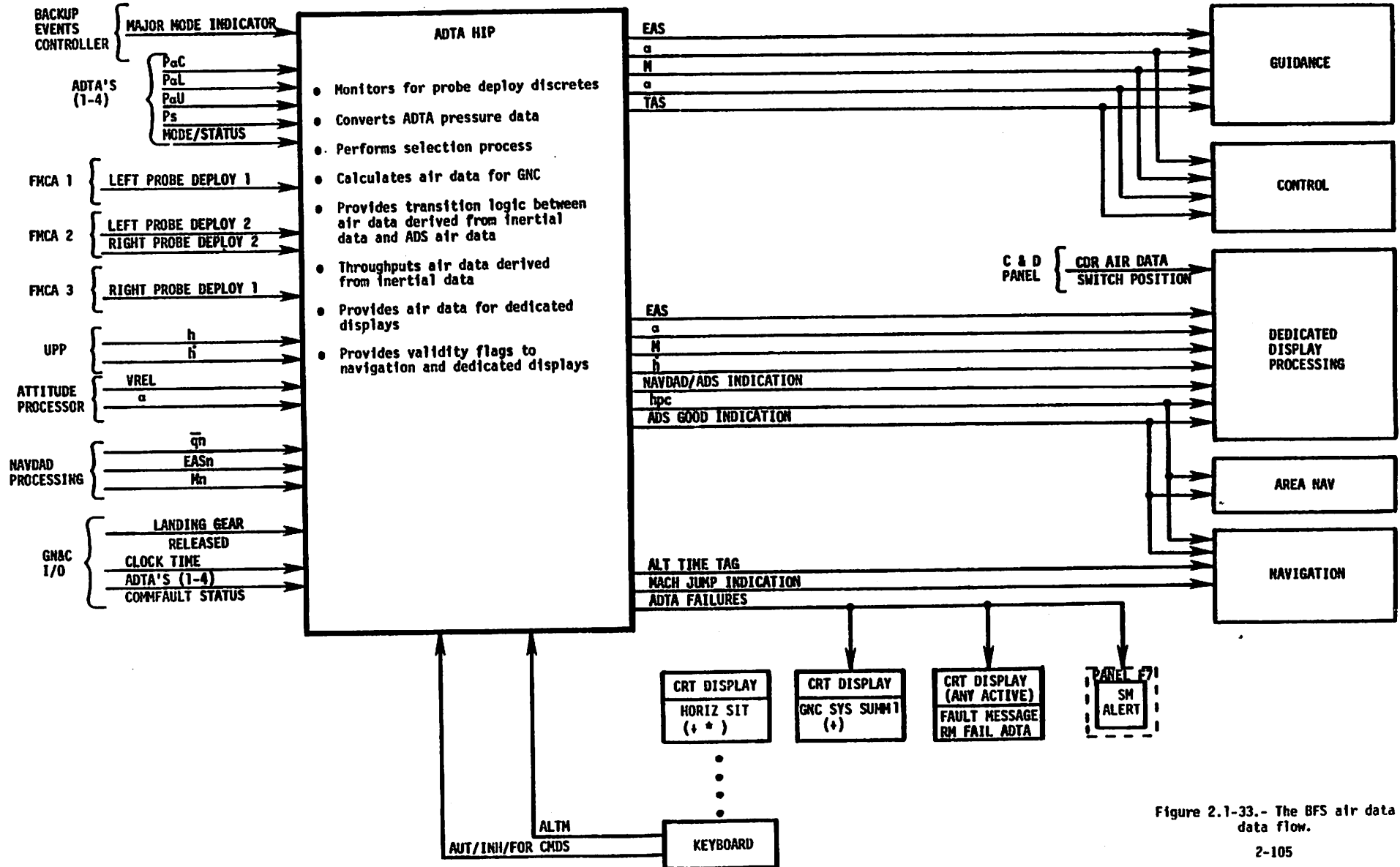


Figure 2.1-33.- The BFS air data data flow.

Probe deployment: There is no RM associated with the deploy discretes. A probe is considered deployed if one of the redundant discretes received from the FMCA's is present.

ADTA candidacy: There is no RM associated with the ADTA pressures. ADTA selection is based on commfault status and certain bits of the ADTA mode/status word.

- Candidates are ADTA's 1 through 4 that have not been declared failed by an out-of-tolerance mode/status word indication.
- Declaring an ADTA commfault results in all pressures from the ADTA being withheld (but not declared failed) from processing.
- Declaring an ADTA failed results in fault annunciation and removal of the ADTA from selection candidacy.
- A probe not deployed results in removal of its associated ADTA's (e.g., ADTA's 1 or 3 for the left probe) from selection candidacy.

ADTA selection process: The selection process for ADTA's involves prime selecting ADTA 1 for the left side and ADTA 2 for the right side. These are then averaged. In the event of a failure the standby ADTA is utilized and the averaging process continues. In the event only one ADTA has an acceptable candidacy status, its output is utilized. For cases in which no ADTA's satisfy candidacy requirements, an ADS failure indication is output to dedicated display processing, area navigation, and navigation.

Air data parameters for GN&C: Based on the interrelationships among deploy status, ADTA candidacy, free-stream M and crew controls on the HORIZ SIT display, the HIP will utilize different sources in order to generate air data parameters for GN&C (fig. 2.1-34). The ADTA HIP does not contain the provision for outputting default air data parameters. In the event that an ADS failure occurs the HIP will output air data parameters for GN&C based on NAVDAD and attitude processing.

	MACH REGION	MODING CONDITIONS			OUTPUT TO GC (α , M, \bar{q} , EAS, TAS)
		PROBE STATUS	HORIZ SIT "ADTA TO G&C"	ADTA STATUS	
N O M I N A L	$M > 2.5$	O N E O R B O T H D E P L O Y E D	N/A	N/A	AIR DATA DERIVED FROM NAVIGATION DATA
	$1.6 < M \leq 2.5$		A U T / F O R	O. K	AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$0.9 \leq M \leq 1.6$				AIR DATA DERIVED FROM MACH JUMP PROCESSING
	$0.2 \leq M < 0.9$				AIR DATA DERIVED FROM PROBE SENSED PRESSURES
F A I L U R E S	$0.2 \leq M \leq 2.5$	D E P L O Y E D	A U T / F O R	ADS FAILURE	AIR DATA DERIVED FROM NAVIGATION DATA
C R E W	$0.2 \leq M \leq 2.5$			INH ①	
F A I L U R E S	$0.2 \leq M \leq 2.5$	BOTH NOT DEPLOYED ②	N/A		AIR DATA DERIVED FROM NAVIGATION DATA

	MACH REGION	MODING CONDITIONS		OUTPUT TO NAVIGATION (hpc)
		PROBE STATUS	ADTA STATUS	
N O M I N A L	$M < 2.5$	O N E O R B O T H D E P L O Y E D	O. K.	AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$1.6 < M \leq 2.5$			AIR DATA DERIVED FROM PROBE SENSED PRESSURE AND INDICATION OF MACH JUMP
	$0.9 \leq M \leq 1.6$			AIR DATA DERIVED FROM PROBE SENSED PRESSURES
	$0.2 < M < 0.9$			AIR DATA DERIVED FROM PROBE SENSED PRESSURES
F A I L U R E S	N/A	D E P L O Y E D	ADS FAILURE	AIR DATA DERIVED FROM PROBE SENSED PRESSURES AND INDICATION OF AN ADS FAILURE
F A I L U R E S	N/A	N O T D E P L O Y E D ②	N/A	AIR DATA DERIVED FROM PROBE SENSED PRESSURES AND INDICATION OF AN ADS FAILURE ③

NOTES

1. ONLY CREW CAN INVOKE INHIBIT COMMAND, SOFTWARE IS INITIALIZED TO AUTO
2. IF NEITHER PROBE IS DEPLOYED AN ADS FAILURE WILL RESULT
3. IF A PROBE WAS NEVER DEPLOYED hpc IS INITIALIZED TO 90,000 FT WITH AN ADS FAILURE INDICATION

Figure 2.1-34.- Air data moding and outputs for GNC.

Air data parameters for dedicated displays: Request for air data parameters are output by the HIP in response to AIR DATA switch requests from panel F6 only (CDR side). Since the HIP does not perform separate calculations for each probe, a request for LEFT or RIGHT data will result in an identical output.

The HIP also provides outputs to support the NAV position of the AIR DATA switch. Prior to probe deployment or in the event of an ADS failure indication, the HIP will throughput air data parameters from NAVDAD, UPP, and attitude processing.

NOTE: In the event of an ADS failure the HIP will output air data parameters based on NAVDAD, UPP, and attitude processing regardless of the AIR DATA switch position.

2.1.17.6 BFS NAVDAD processing.- The processing and outputs provided by the BFS NAVDAD is identical to that of the NAVDAD SOP.

2.1.18 Dedicated Displays

The following tables present dedicated display information for all major modes. BFS deltas follow for each OPS (tables 2.1-9, 2.1-10, 2.1-11, 2.1-12).

TABLE 2.1-9.- DEDICATED DISPLAYS, OPS 1

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
101	HSI	Heading	A/E ATT PROC	ADI att sw	If ADI ATT sw = LVLH = angle between vehicle + X axis and the tan of insertion orbit plane, otherwise not driven (deg)
		Course	ASC UPP		Angle formed by V_{rel} vector and target orbit plane (deg)
		Course deviation			Sideslip angle discounting wind (λ between V_{rel} & X body axis)
		Glide slope indicator		Not driven	
		Primary bearing		Not driven	
		Secondary bearing		Not driven	
		Primary range		Not driven	
		Secondary range		Not driven	
ADI	Attitude	ATT PROC	Display dynamic only after IMU release	INRTL: P/R/Y body WRT arbitrary inertial referenced frame LVLH: Yaw = zero, pitch/roll = LVLH pitch and roll REF: P/R/Y body WRT arbitrary inertial reference frame (designated L/O position). Realigned to instantaneous body axes when ATT REF pb pushed	

TABLE 2.1-9.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
102	AMI	Alpha		Driven zero	Flag displayed
		ACCEL		Driven zero	Flag displayed
		M/VEL	NAV ASC UPP	---	For M < 4, reads M For M > 4, reads K ft/sec
		EAS	NAV ASC UPP	---	Nav-derived EAS (Knots)
AVVI		ALT ACCEL		Driven zero	Flag displayed
		ALT rate		Driven zero	Flag displayed
		ALT		Driven zero	Flag displayed
		RDR ALT		Driven 5000 ft	Flag displayed
ADI	Attitude	ATT PROC	ATT PROC	ADI ATT sw	Inertial/Ref P/Y/R WRT arbitrary inertial frames
		Rates R/P/Y	RGA RM		LVLH Yaw = zero P/R = LVLH P/R RM-selected RGA rate (deg/sec)
		Errors R/P/Y	G/C STEER		Total attitude errors (actual attitude minus desired attitude) and load relief (deg) Rate and error scale: Sw position ! Low ! Med ! High Full scale ! 10 ! 50 ! 100
HSI	Heading	A/E ATT PROC	A/E ATT PROC	ADI ATT sw	If ADI ATT sw = LVLH, angle between vehicle +X axis and the target insertion orbit plane, otherwise not driven (deg)
		Course	ASC UPP		Angle formed by V_{rel} vector and target orbit plane (deg)
		Course deviation			Sideslip angle discounting wind (χ between V_{rel} and +X body axis) (deg)
		Glide slope indicator			Not driven

TABLE 2.1-9.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
102 (Cont)	HSI	Primary bearing			Provides turnaround bearing in case of RTLS
		Secondary bearing	ASC UPP		Provides heading of inertial velocity vector with respect to TIO plane
103	AMI	ALPHA	ATT PROC		Angle of attack: } from +X velocity axis to negative relative wind velocity vector
		Acce1			Total load factor (vehicle) (ft/s ²)
		M/VEL	NAV		Inertial velocity (K ft/s)
		EAS	ASC UPP		Nav-derived EAS (Knots)
AVVI		ALT ACCEL		Driven zero	Flag displayed
		ALT RATE	NAV		Rate of altitude change (radial component of inertial velocity) (ft/s)
		ALT	NAV		Nav base above runway
		RDR ALT		Driven to 5000 ft	Flag displayed
HSI		Heading	ASC UPP	ADI ATT sw	Same as 102
		Course	ASC UPP		Same as 102
		Course deviation			Same as 102
		Glide slope indicator		Not driven	
		Primary bearing			Same as 102
		Secondary bearing	ASC UPP		Same as 102
		Primary range		Not driven	
Secondary range		Not driven			

TABLE 2.1-9.- Concluded

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions	
103 (Cont)	ADI	ATTITUDE	ATT PROC	ADI ATT sw	P/R/Y body WRT arbitrary inertial reference frame (INRTL) PRY body WRT LVLH frame (or biased; i.e., rotated LVLH) Defined as Y = R x V, Z along -R vector. P/R/Y body WRT arbitrary inertial reference frame (REF)	
		Rates R/P/Y	RGA RM	ADI RATE sw	RM-selected RGA rates (body axis) Sw position! Low ! Med ! High Full scale ! 10°/sec! 50°/sec! 100°/sec	
		Errors R/P/Y	G/C STEER	ADI ERROR sw	Total attitude errors (actual attitude minus desired attitude) (fly to) Sw position! Low ! Med ! High Full scale ! 10° ! 50° ! 100°	
	EVENT LIGHTS	G			Not driven	
		CONF			Not driven	
Preflare				Not driven		
Gear				Not driven		
		Nav		Not driven		
104 105 106	ADI	Attitude	ATT PROC	ADI ATT sw	(INRTL) arbitrary inertial reference frame (I-loaded) (LVLH) same as 103 (REF) arbitrary inertial reference (I-loaded)	
		Rates R/P/Y	RGA RM	ADI RM	RM-selected RGA rates (body axis) (deg/sec) Sw position! Low ! Med ! High Full Scale ! 10° sec ! 50° sec ! 100° sec	
		Errors R/P/Y	ATT PROC	ADI ERROR sw	Derived from attitude error quaternion using small angle technique Sw position! Low ! Med ! High Full scale ! 10° ! 50° ! 100°	

The ADI is the only dedicated display driven in Major Modes 104, 105, and 106.

BFS only monitors commander's attitude and ATT REF pushbutton sw. Both ADI's are driven according to this switch position. Error needles are always 50° full scale. Rate needles are always 50°/sec full scale. BFS drives the HSI only as a yaw reference in OPS 1. No other displays are driven by BFS in OPS 1. BFS defaults to middle position on scale switches.

TABLE 2.1-10.- DEDICATED DISPLAYS, OPS 3

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
301 302 303	ADI	Attitude	ATT PROC	ADI sw ADI SENSE sw ATT REF 1b	(INRTL) P/R/Y body WRT arbitrary inertial reference frame (LVLH) P/R/Y body WRT LVLH frame (or biased; i.e., rotated LVLH). Defined as Y = R x V, Z along -R. (REF) P/R/Y body WRT arbitrary inertial reference frame realigned to instantaneous body axis when ATT REF pb pushed
		Rates R/P/Y		ADI RATES sw	Sw position! Low ! Med ! High Full scale ! 10°/sec! 50°/sec ! 100°/sec
		Errors	ATT PROC	ADI ERROR sw	Total attitude errors (actual attitude minus commanded) derived from attitude error quaternion using small angle technique Sw position! Low ! Med ! High Full scale ! 10° ! 50° ! 100°
The ADI is the only dedicated display driven during Major Modes 301, 302, and 303.					
304	AMI	ALPHA	ATT PROC		Angle from + x vehicle axis to negative relative velocity vector
		ACCEL	ENT UPP		Drag acceration
		M/VEL	NAV	ADTA probe sw ADTA sw, HSD	Mach no. if $V_{rel} < \text{Mach } 4$, kft/sec if $V_{rel} > \text{Mach } 4$
		EAS	NAV	ADTA probe sw	Nay-derived EAS (knots)
	AVVI	ALT ACCEL	ENT UPP		Nav-derived altitude accel
		ALT RATE	ENT UPP		deg/sec
		ALT	ENT UPP		ft
		RDR ALT			Stowed, indicator driven to 5000 ft

TABLE 2.1-10.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
304 (Cont)	HSI	Heading	AREA NAV ATT PROC	HSI mode sw HSI source sw	Mag heading of vehicle V_{rel} vector
		Course	HSI PROC	Air data avail- ability (in tacan select)	Primary runway heading WRT MAG north
		Course deviation	AREA NAV		Valid software zero (no flag)
		Glide slope indicator			Pointer stowed and flag displayed
		Primary bearing	AREA NAV		Spherical bearing to WP1 for NEP at primary runway
		Secondary bearing	AREA NAV		Spherical bearing to SP-1 for NEP at secondary runway
		Primary range	AREA NAV		Spherical surface range to WP-2 on primary runway via WP-1 for NEP
		Secondary range	AREA NAV		Spherical surface range to WP-2 on Secondary runway via WP-1 for NEP
ADI		Attitude	ATT PROC		Y/P/R body WRT topodetic; yaw = 0 ("wings level" display)
		Rates: Pitch	RGA RM	ADI RATE sw	Selected RGA pitch rate Sw position ! Low ! Med ! High Full scale ! 10°/sec ! 50°/sec ! 100°/sec
		Roll/yaw	AEROJET DAP		Stability axis roll, yaw rates
		Errors: roll	ATT PROC	ADI ERROR sw	Total roll attitude error Sw position ! Low ! Med ! High Full scale ! 10° ! 25° ! 25°
		Pitch	AERO DAP		Angle of attack from pitch aerojet Sw position ! Low ! Med ! High Full scale ! 10° ! 20° ! 50°
		Yaw	ATT PROC		$\bar{q} < 20$, estimated sideslip from ATT PROC angle between +X body axis and relative velocity Sw position ! Low ! Med ! High Full scale ! 10° ! 50° ! 100°
			AERO DAP-yaw		$\bar{q} \geq 20$, function of N_y , yaw jet capability
EVENT LIGHTS	G	NAV			Load factor $\geq 64.8 \text{ ft/s}^2$ on steady Load factor $\geq 81.0 \text{ ft/s}^2$ on flash
	CONF			$1 \leq M \leq 2.5$ $0.9 \leq M \leq 10.0$ Alt < wheel deploy alt	Flashes, then on if ADTA probe not de- ployed Flashes, then on if speed brake out of position Flashes, then on if body flap out of position
	Preflare				N/A
	Gear				N/A
	NAV	NAV		Altitude HSD Mach no.	Navigation data missing

TABLE 2.1-10.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
305	AMI	ALPHA	ATT PROC ADTA SOP		Angle-of-attack, angle from +X vehicle axis to neg relative wind velocity vector
		ACCEL	ENT UPP		Drag acceleration (after STS-2) driven zero
		M/VEL	NAV	ADTA sw ADTA probe sw HSD	Vehicle inertial velocity magnitude/1000
		EAS	NAV	ADTA probe sw ADTA sw, HSD	(knots)
AVVI		ALT ACCEL	TAEM/AL UPP		Nav-derived altitude acceleration
		ALT RATE	NAV/ADTA SOP	ADTA probe sw ADTA sw	Back differenced baro alt with correction for vehicle pitch rate in nav parameter
		ALT	TAEM/A/L UPP ADTA SOP	ADTA probe sw ADTA data sw	TAEM and A/L UPP - alt of rear wheels above runway corrected for pitch ADTA SOP baro alt
		RDR ALT	RA SOP	RDR alt sw	Alt > 5000 ft - driven to 5000 ft Alt < 5000 ft - selected RA altitude corrects to rear wheels, assuming 10° pitch
ADI		Attitude	ATT PROC		P/Y/R body WRT topodetic; yaw=0 ("wings level" display)
		Rates Pitch	RGA SOP/RM	ADI RATE sw	Pitch = selected RGA pitch rate Sw position ! Low ! Med ! High Full scale ! 10°/sec ! 50°/sec ! 100°/sec
		Roll, yaw	AEROJET DAP		Stability axis roll, yaw rates
		Errors Roll	ATT PROC	ADI ERROR sw	Total roll attitude error Sw position ! Low ! Med ! High Full scale ! 10° ! 25° ! 25°
		Pitch	AERO DAP-PITCH		(Pre-WOW) Normal acceleration error Sw position ! Low ! Med ! High Full scale ! 0.5g's ! 1.25g's ! 1.25g's
		Yaw	AERO DAP-YAW		(q̄ 20) Function of Ny, yaw jet capability Sw position ! Low ! Med ! High Full scale ! 10° ! 50° ! 100°

TABLE 2.1-10.- Concluded

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions	
305 (Cont)	EVENT LIGHTS	G	AA		Normal acceleration ≥ 64.8 ft/s ² on steady Normal acceleration ≥ 81 ft/s ² on flashing	
		CONF		$1 \leq M \leq 2.5$ $0.9 \leq M \leq 10.0$ Alt $<$ wheel deploy alt	Flashes, then on if both air data probes not deployed Flashes, then on if speed brake out of position Flashes, then on if body flap out of position	
		Preflare			When alt wheels $<$ 1750 ft, flashes for 3 sec then on steady	
		Gear			Flashes when alt wheels $<$ 300 ft. flashes until all uplock discretes are false, the lights steady for 10 sec then off	
		NAV	NAV	NAV	Altitude HSD Mach no.	Navigation data missing
		HSI	Heading	AREA NAV/ ATT PROC	HSI mode sw HSI source sw	TAEM MAG heading of vehicle +X body axis
	Course	HSI PROC	AIR DATA avail- ability (in tacan select)	Primary runway heading WRT MAG North		
	Course	AREA NAV		Deviation from extended runway centerline (full scale $\pm 10^\circ$)	Same as TAEM (full scale $\pm 2.5^\circ$)	
	Glide slope deviation	AREA NAV		Deviation from guidance refer- ence altitude for TAEM	Deviation from guidance altitude for A/L (pointer is stowed below 1.5 kft flag = off)	
	Primary bearing	AREA NAV		Bearing to WP-2 HAC for primary runway	Bearing to WP-2 on primary runway	
	Secondary bearing	AREA NAV		Bearing to center of selected HAC for primary run- way	Bearing to WP-2 at primary runway	
	Primary range	AREA NAV		Horizontal dis- tance to WP-2 on primary run- way via WP-1 for selected HAC	Horizontal dis- tance to WP-2 on primary runway	
	Secondary range	AREA NAV		Horizontal dis- tance to center of selected HAC for primary run- way	Horizontal dis- tance to WP-2 on primary runway	

In the BFS, the AMI and AVVI are controlled only by the left air data switch. ALT rate and ALT and alpha, m/VEL, and EAS are calculated from air data (averages left and right if switch is in either left or right). RA switches are not used (defaults to 2 if 1 not locked on). HSI switches are only active on right side. MLS is not supported.

TABLE 2.1-11.- DEDICATED DISPLAYS, OPS 2

Major Mode(s)	Display	Parameters	Where computed	Qualifiers	Definitions
201 202	ADI	Attitude	ATT PROC	ADI ATT sw SENSE sw	INRTL: Pitch = Right ascension of point toward which orbiter +X axis is directed Yaw = Declination of point toward which Orbiter +X axis is directed Roll = Completes the right handed system LVLH: Local vertical, local horizontal
		Rates R/P/Y	DAP	ATT REF pb ADI RATE sw	REF: Arbitrary inertial, set via pb. Sw position !Low ! Med ! High Full scale !10°/sec ! 50°/sec ! 100°/sec IMU-derived body rates
		Errors R/P/Y	ATT PROC or on-orbit DAP	ADI errors On-orbit display	Sw position ! Low ! Med ! High Full scale ! 10°/sec ! 50°/sec ! 100°/sec Total attitude errors (actual attitude minus commanded attitude)

The ADI is the only dedicated display driven in OPS 2

TABLE 2.1-12.- DEDICATED DISPLAYS, OPS 6

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
601	AMI	Alpha	ATT PROC		Angle from +X vehicle axis to neg relative wind velocity vector ($^{\circ}$)
		ACCEL	RTLS UPP		Total load factor (ft/s ²)
		M/VEL	RTLS UPP		Vehicle inertial vel magnitude/1000 (kft/s)
		EAS	RTLS UPP		(knots)
AVVI		ALT ACCEL		Driven zero	--
		ALT rate	RTLS UPP		Estimate rate of altitude change
		ALT	RTLS UPP		Altitude above runway
		RDR ALT		Driven to 5000 ft	--
HSI		Heading	RTLS UPP	ADI ATT sw	If ADI ATT sw = LVLH, \angle between +X axis and the TIO plane otherwise not driven
		Course	RTLS UPP		Formed by V_{REL} and TIO plane
		Course deviation	A/E ATT PROC		Sideslip angle discounting wind (between V_{rel} & V_I) ($^{\circ}$)
		Glide slope indicator		Not driven	
		Primary bearing	RTLS UPP		Bearing of Shuttle position to landing site referenced to TIO plane ($^{\circ}$)
		Secondary bearing	RTLS UPP		Angle between inertial velocity vector and TIO plane
		Primary range		Not driven	
Secondary range		Not driven			
ADI		ATTITUDE	ATT PROC	ADI ATT sw	Same as 103
		Rates R/P/Y	RGA SOP/RM	ADI rate sw	Sw position ! Low ! Med ! High Full scale ! 10 ⁰ /sec ! 50 ⁰ /sec ! 100 ⁰ /sec RM-selected RGA, body axis rates
		Errors R/P/Y	G/C STEER	ADI error sw	Sw position ! Low ! Med ! High Full scale ! 1 ⁰ ! 5 ⁰ ! 10 ⁰ Total attitude errors (actual attitude minus command attitude)

EVENT LIGHTS

The event lights are not driven in Major Mode 601

TABLE 2.1-12.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
602	AMI	Alpha	ATT PROC		Angle-of-attack: \angle from +X vehicle axis to neg relative wind vector (0)
		ACCEL	RTL5 UPP		Selected AA normal, N_z
		M/VEL	NAV or ADTA SOP		Vehicle inertial vel magnitude/1000
		EAS	NAV or ADTA SOP	Air data probe sw Air data sw HSD	Nav-derived EAS (knots)
AVVI		ALT ACCEL	RTL5 UPP		NAV-derived altitude acceleration (ft/s ²)
		ALT RATE	RTL5 UPP/ ADTA SOP	Air data probe sw Air data sw	RTL5 UPP - rate of altitude change ADTA SOP - back differential baro-alt with correction for vehicle pitch rate in the RTL5 UPP parameter
		ALT	RTL5 UPP/ ADTA SOP	Air data probe Air data sw HSD	RTL5 UPP - alt of rear wheels above runway ADTA SOP - baro alt
		RDR ALT	RA SOP	RDR ALT sw Alt < or > 5000 ft	Alt > 5000 ft - driven to 5000 ft Alt < 5000 ft - selected RDR ALT corrected to rear wheels, assuming 10° pitch
HSI		Heading	AREA NAV	HSI mode sw HSI select sw	MAG heading of vehicle V_{rel} vector
		Course	HSI PROC	Air data availability (in tacan select)	Primary runway WRT MAG north
		Course deviation	AREA NAV		Valid software zero (no flag)
		Glide slope indicator	AREA NAV		N/A (pointer stowed and the flag displayed)
		Primary bearing	AREA NAV		Spherical bearing to WP-1 for NEP at primary runway
		Secondary bearing	AREA NAV		Spherical bearing to WP-1 for NEP at secondary runway
		Primary range	AREA NAV		Spherical surface range to WP-2 on primary runway via WP-1 for NEP
		Secondary range	AREA NAV		Spherical surface range to WP-2 on secondary runway via WP-1 for NEP
ADI		Attitude	ATT PROC		RPY body WRT topodetic; yaw = 0 yaw = 0 ("wings level" display)
		Rates Pitch	RGA SOP/RM		Selected RGA pitch rate Sw position!Low !Med !High Full scale !10°/sec!50°/sec!100°/sec
		Roll, yaw	AEROJET DAP		Stability axis roll, yaw rates

TABLE 2.1-12.- Continued

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions
602 (Cont)	ADI	Errors Roll	ATT PRO	ADI ERROR sw	Total roll attitude error Sw position!Low ! Med ! High Full scale !10° ! 50° ! 200°
				ADI ERROR sw	Angle-of-attack error Sw position!Low ! Med ! High (pitch & yaw) Full scale !10° ! 50° ! 100°
		Pitch	AERO DAP- pitch		
		Yaw	ATT PROC	$\dot{q} < 20$ → estimated sideslip from ATT PROC	
AERO DAP- yaw	$\dot{q} > 20$ → function of N_y , yaw jet capability				
EVENT LIGHTS	G	NAV	NAV		Total load factor ≥ 64.8 ft/s ² → on steady
					Total load factor ≥ 81 ft/s ² → flash
		CONF		$1 \leq M \leq 2.5$	Flashes, then on if both air data probes not deployed
				$0.9 \leq M \leq 10.0$	Flashes, then on if speed brake out of position
			ALT < Gear down alt	Flashes, then on if body flap out of position	
		Preflare			N/A
Gear			N/A		
NAV	NAV	NAV	Altitude HSD, Mach no.	Navigation data missing	
603	AMI	Alpha	ATT PROC/ ADTA SOP	Air data probe sw Air data sw HSD	Angle from +X vehicle axis to neg. wind velocity vector
		ACCEL	RTLS UPP		Drag acceleration
		M/VEL	NAV or ADTA SOP	Air data probe sw, air data sw, HSD	Vehicle inertial velocity magnitude/1000
		EAS	NAV/ADTA SOP	Air data probe sw ADTA sw HSD	EAS (knots)
AVVI	ALT ACCEL	RTLS UPP		NAV-derived altitude acceleration	
	ALT RATE	RTLS UPP/ ADTA SOP	Air data probe sw ADTA sw	RTLS UPP → rate of altitude change ADTA SOP → back differenced baro alt with correction for vehicle pitch rate in the RTLS UPP parameter	
	ALT	RTLS UPP/ ADTA SOP	ADTA probe sw ADTA sw	RTLS UPP - Alt. of rear wheels above runway corrected for pitch ADTA SOP - baro alt	
	RDR ALT	RA1 or 2	RDR ALT sw	Alt > 5000 ft → driven to 5000 ft Alt < 5000 ft → selected rdr alt corrected to rear wheels, assuming 10° pitch	

TABLE 2.1-12.- Concluded

Major Mode	Display	Parameters	Where computed	Qualifiers	Definitions	
603 (Cont)	HSI	Heading	AREA NAV ATT PROC	HSI mod sw HSI select sw	TAEM MAG heading of vehicle +X body axis	A/L MAG heading of vehicle +X body axis
		Course	HSI PROC	Air data avail- ability (in tacan select)		
		Course deviation	AREA NAV		Deviation from extended runway center line (full scale = ± 10°)	Same as TAEM (full scale ± 2.5°)
		Glide slope deviation	AREA NAV		Deviation from guidance reference attitude for TAEM	Deviation from guidance refer- ence altitude for A/L (pointer stowed below 1500 ft flag = off)
		Primary bearing	AREA NAV		Bearing to WP-1 on selected HAC for pri- mary runway	Bearing to WP-2 at primary runway
		Secondary bearing	AREA NAV		Bearing to center of selected HAC for pri- mary runway	Bearing to WP-2 at primary runway
		Primary range	AREA NAV		Horizontal distance to WP-2 on primary runway via WP-1 for selected EP	Horizontal distance to WP-2 on primary runway
		Secondary range	AREA NAV		Horizontal distance to center of selected HAC for primary runway	Horizontal distance to WP-2 on primary runway
ADI		Attitude	ATT PROC		RPY body WRT topodetic; yaw = 0 ("wings level" display)	
		Rates Pitch	RGA SOP/RM		Selected RGA pitch rate Sw position $\frac{Low \quad High}{Full \ scale \quad 110/sec \quad 150/sec \quad 110/sec}$	
		Roll, yaw	AEROJET DAP		Stability axis roll, yaw rates	
		Errors Roll	ATT PROC		Total roll attitude error $\frac{Low \quad High}{100/sec \quad 1250/sec \quad 1250/sec}$	
		Pitch	AERO DAP-pitch		(pre-WOW) + normal acceleration error (post-WOW) + slapdown pitch rate error $\frac{Low \quad High}{0.5g's \quad 1.25g's \quad 1.25g's}$	
		Yaw				
EVENT LIGHTS	G		Accelerometer		Normal acceleration: $\begin{matrix} > 64.8 \text{ ft/s}^2 \text{ on steady} \\ \geq 81 \text{ ft/s}^2 \text{ on flashing} \end{matrix}$	
		CONF		$1 < M < 2.5$ $0.9 \leq \bar{M} \leq 10.0$ Alt \leq wheel deploy alt	Flashes, then on if both ADTA probes not deployed Flashes, then on if speed brake out of position Flashes, if body flap out of position	
		Preflare			When ALT wheels < 1750 ft, flashes for 3 sec, then on steady	
		Gear			Flashes when h (wheels) is < gear on alt then on steady for 10 sec when all 3 uplock discretes become false	
		NAV	NAV	Altitude HSD Mach no.	Navigation data missing	

The BFS does not drive ACCEL on the AMI or VERT ACCEL on the AVVI in 603.

2.2 SOFTWARE SEQUENCING

2.2.1 Redundant Set Launch Sequence

The RS launch sequence is used during the launch countdown in conjunction with the LPS to perform the onboard automatic functions required in the last 25 sec before SRB ignition. In addition, the RS launch sequence controls the onboard countdown clock from T-20 min until SRB ignition. The ability is also provided to call countdown HOLD's, recognize HOLD requests from the LPS, and recognize RESUME COUNT or RECYCLE commands from LPS. The SSME ignition commands are issued; and, after the required thrust level is reached, the SRB's are ignited. Failure of the SSME's to reach the required thrust level will result in an inhibit of the SRB ignition and a controlled SSME shutdown. The RS launch sequence terminates immediately after issuing SRB ignition.

REDUNDANT SET LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
T - 20:00	<p>1. <u>RS LAUNCH SEQUENCE INITIATE</u></p> <p>RS launch sequence initiated by crew with OPS 101 PRO. Onboard countdown clock started.</p> <p>At sequence initiation, issue RS COUNTDOWN HOLD flag.</p> <p>Go to countdown hold procedure (p. 2-130).</p> <p>Throughout the execution of the sequence, the SSME SOP is monitored for indications of the following in each main engine:</p> <ul style="list-style-type: none">● Pad data path failure● Pad command path failure● SSME controller channel failure● Electronic lockup● Hydraulic lockup● Major component failure <p>If any of these problems are detected for two consecutive passes, the RS launch sequence will either call a countdown hold or initiate shutdown of the ME's.</p>
T - 00:35.02	2. Configure vent doors for launch.
T - 00:25.0	<p>3. <u>LPS MONITOR</u></p> <p>Monitor LPS for:</p> <p>LPS GO FOR AUTO SEQUENCE flag</p> <p>If false, order COUNTDOWN HOLD (go to countdown hold procedure).</p>
T - 00:20.0	<p>4. <u>HOLD RECYCLE</u></p> <p>Holds occurring beyond this time will automatically cause the countdown clock and sequences to reset at T - 09:00. This may or may not occur for holds before this time.</p>

RS LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
T - 00:16.0	5. <u>CLOSE OUTBOARD FILL VALVES</u> Send commands to: A. Close LO2 outboard fill valves B. Close LH2 outboard fill valves
T - 00:15.0	6. <u>ARM PIC's</u> Arm: A. SRB ignition Pyro Initiator Controllers (PIC's) B. T-0 umbilical release PIC's
T - 00:12.0	7. <u>PIC VOLTAGE MONITOR/IMU INERTIAL</u> A. Check SRM ignition PIC voltages. If any of the voltages are low or commfaulted for two consecutive passes, issue either a HOLD (go to hold procedure) or a SEQUENTIAL ME SHUTDOWN (go to engine shutdown procedure). B. Command IMU to inertial. C. Begin monitoring for flight critical MDM channel fails. If any occurs, issue hold or initiate pad shutdown.
T - 00:09.5	8. <u>OPEN LO2 ACCUMULATOR RECIRC VALVES 1 & 2</u>
T - 00:08.0	9. <u>INITIALIZE NAV SYSTEM</u> Issue: A. NAV INITIALIZATION flag B. 100 PERCENT THROTTLE command
T - 00:06.5	10. <u>VERIFY MPS SYSTEM READY</u> A. If any of the following valves are comm-faulted for two cycles, issue a hold.

RS LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
	<ul style="list-style-type: none"> B. Confirm LH2 and LO2 outboard fill valves are closed. If either valve is open and there is no LPS VALVE POSITION BYPASS flag, issue a HOLD (go to countdown hold procedures). C. Confirm LO2 accumulator recirculation valves 1 & 2 are open. If either valve is closed and there is no LPS VALVE POSITION BYPASS flag, issue a HOLD (go to countdown hold procedure). D. Verify that ENG READY flag from SSME SOP for each SSME is set. If not, issue a hold. If all flags are set: <ul style="list-style-type: none"> (1) Command ME's 1, 2, & 3 LH2 prevalues to open. (2) Send MPS START ENABLE CMD flag to SSME SOP.
T - 00:05.0	11. <u>COMMAND MPS LO2 OVERBOARD BLEED VALVE TO CLOSE</u>
T - 00:04.0	12. <u>STATUS MONITOR</u> <ul style="list-style-type: none"> A. Verify LO2 overboard is closed; if not, issue a HOLD (go to countdown hold procedure). B. Verify ME's 1, 2, & 3 LH2 prevalues are open; if any of the valves are not open, issue a HOLD (go to countdown hold procedure). C. Verify each vent door in launch position; if any are not and there is no LPS override for each failure, issue a HOLD (go to countdown hold procedure). D. Monitor SSME SOP for ME's 1, 2, & 3 controllers' engine ready indications. If any are not received, order a HOLD (go to countdown hold procedure). E. Monitor LPS for GO FOR ENGINE START flag; if flag if false, issue a HOLD (go to countdown hold procedure).

RS LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
T - 00:03.8	13. <u>START MAIN ENGINES</u> <ul style="list-style-type: none">A. Issue MPS TVC SERVO OVRD cmdB. Issue ENG START CMDS ISSUED flagC. Issue MPS E-3 START CMD flagD. Start 4.44 sec ENG TIMER FOR THRUST OK. (The engines now have 4.44 sec to build up to 90 percent Pc.)E. Start four FRF associated timers.F. After 120 ms, issue MPS E-2 START CMD flagG. 102 ms later:<ul style="list-style-type: none">(1) Issue MPS E-1 START cmd(2) Set E-1 START CMD ISSUED flag
P _C > 90% for all ME's	14. <u>PREPARE SSME's</u> <ul style="list-style-type: none">A. Start ALL ENG OK CHECK timer (2.64 sec)B. Terminate MPS TVC SERVO OVRD cmdC. Issue PREP SSME's FOR LIFTOFF flagD. Check status if ME's 1, 2, & 3 actuator ports for 2.64 sec:<ul style="list-style-type: none">(1) If any port status is commfaulted for two successive cycles, start sequential engine shutdown (go to engine shutdown procedure).(2) If any port is declared failed for two successive cycles, start sequential engine shutdown (go to engine shutdown procedure).

RS LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
15.	<u>CHECK FOR FRF</u>
A.	Wait until ALL ENG OK CHECK timer expires
B.	If no FRF, go to 16.
C.	If FRF TEST FLAG SET is true:
(1)	Check ME's for FASCOS LIMIT EXCEEDED flag. If it is true for three successive cycles, initiate engine shutdown (go to engine shutdown procedure).
(2)	On first pass, issue SRM IGN FIRE 1 & 2 flags
(3)	On next pass: <ul style="list-style-type: none">● Terminate SRM IGN FIRE 1 & 2 flags● Issue UMB RELEASE FIRE 1 & 2 flags
(4)	When FLT CNTL TVC INIT TIME DELAY (8 sec) elapses, initiate the FRF MPS TVC gimbal profile.
(5)	When FRF THROTTLE TO 92 PERCENT TIME DELAY (18.4 sec.) elapses, throttle the ME's to 92 percent.
(6)	When FRF THROTTLE TO 100 PERCENT TIME DELAY (20.2 sec) elapses, throttle the ME's to 100 percent.
(7)	When FRF CUTOFF TIME DELAY (22.0 sec.), initiate engine shutdown (go to engine shutdown procedure).

RS LAUNCH SEQUENCE

<u>TIME</u>	<u>EVENT</u>
-------------	--------------

16. MODE FOR LIFT-OFF

A. Issue commands to:

- Fire SRB's
- Terminate LPS polling
- Reset mode control Mission Elapsed Time (MET)
- Read and store Greenwich Mean Time (GMT)
- Start event timer
- Release T-0 umbilical
- Master reset Master Event Controllers (MEC's) 1 & 2

B. Terminate RS launch sequence

RS LAUNCH SEQUENCE

Countdown Hold Procedure

1. If $T > T_0 - 20$ sec, go to step 4A.
2. Stop countdown clock.
3. Monitor LPS for RESUME COUNT COMMAND flag.
If false, go to step 4.
If true:
 - A. Clear all RS launch sequence downlist items.
 - B. Reset:
 - LPS and RS COUNTDOWN HOLD flags
 - Resume COUNT COMMAND flag
 - SRB RSS SAFE flag
 - C. Reset countdown time if necessary due to new launch GMT.
 - D. Resume count at countdown time.
4. Monitor LPS for RECYCLE COUNT COMMAND flag
If false, go to step 1.
If true:
 - A. Terminate:
 - SRM IGNITION ARM flag
 - T-O UMBILICAL RELEASE ARM flag
 - B. Issue:
 - MEC 1 & 2 master reset
 - SRB RSS SAFE flag
 - C. Reset countdown clock to $T_0 - 09:00$ and stop clock.
 - D. Reinitialize the RS launch sequence.
 - E. Go to step 3.

RS LAUNCH SEQUENCE

Engine Shutdown Procedure

When a SHUTDOWN is commanded, a single engine is initiated to shutdown first (either the earliest detected malfunctioning engine or, in the case of no engine failures, ME 1). After a 1.8 sec delay, shutdown of the remaining engines is initiated simultaneously.

1. Terminate:
 - A. Prepare SSME's FOR LIFT-OFF flag
 - B. Solid Rocket Motor (SRM) IGNITION ARM flag
 - C. T-O UMB RELEASE ARM flag
2. Issue:
 - A. CMD SSME's TO PRESTART POS flag
 - B. MPS SLEW COMPLETE flag
 - C. LAUNCH SEQUENCE ABORT flag
3. For the first engine to be shutdown, E-X:
 - A. Issue E-X SHUTDOWN ENABLE flag to SSME SOP.
Wait 4.5 sec.
Close E-X LO2 prevalve.
Wait 1.4 sec.
Close E-X LH2 prevalve.
4. 1.8 sec after step 3A has been initiated for the first engine, start step 3A simultaneously for the remaining two SSME's.
5. 8 sec after engine shutdown initiation:
 - A. If both prevalues for ME's 1, 2, & 3 have been commanded to close, terminate sequence.
 - B. If any engine is not in shutdown or postshutdown phase, issue countdown HOLD and terminate sequence.

There is no RS launch sequence in the BFS since the BFS cannot launch the vehicle.

2.2.2 MPS Dedicated Display Driver

MAIN PROPULSION SYSTEM DEDICATED DISPLAY DRIVER SEQUENCE

1. MONITOR EXTERNAL TANK (ET) SEPARATION

If the ET sep commands have been issued:

- Turn off all MAIN ENGINE STATUS lts
- Terminate MPS DD DRIVER SEQ.

If the ET SEP commands have not been issued:

- Go to step 2.

2. CHECK MAIN ENGINE STATUS LIGHT PARAMETERS

A. Complete steps 2 B and 2 C for the C (L,R) engine, then go to step 3.

B. Red MAIN ENGINE STATUS light

Monitor SSME SOP for:

- SSME redline limit exceeded
- SSME shutdown phase
- SSME postdown phase

If any of the above occurs:

- Turn on red MAIN ENGINE STATUS lt
- Go to step 2C.

If none of the above occurs:

- Turn off red MAIN ENGINE STATUS lt
- Go to step 2C.

C. Amber MAIN ENGINE STATUS light

Monitor SSME SOP for:

- electronic hold
- hydraulic lockup
- data path fail
- command path fail

If any of the above occurs:

- Turn on amber MAIN ENGINE STATUS lt
- Go to step 2A

If none of the above occurs:

- Turn off red MAIN ENGINE STATUS light
- Go to step 2A

3. MPS PREVALVE INHIBIT/Pc METER PROCESSING

A. Do steps 3B, 3C, and 3D for the C(L,R) engine, then go to step 4.

B. Check for flight data path fail

Monitor SSME SOP for flight data path fail.

If detected:

- Remove MPS PREVLV CLOSE INHIBIT commands
- Drive Pc meter to zero
- Go to step 3A.

If not detected:

- Go to step 3C

C. PREVALVE INHIBIT

Monitor SSME SOP for percent Pc.

- If percent Pc \geq 30 percent, issue MPS PREVLV CLOSE INHIBIT commands
- If percent Pc $<$ 30 percent, remove MPS PREVLV CLOSE INHIBIT commands

Go to step 3D.

D. Pc METER DRIVE

- Drive MPS Pc meter to percent Pc being output by SSME SOP.
- Go to step 3A

4. MECO COMMAND MONITOR

Monitor R/S LCH SEQ for launch sequence abort command.
Monitor SSME OPS SEQ for MECO command (shutdown phase).

If any of the above occurs:

- Wait 4.5 seconds (I-loaded)
- Remove MPS PREVLV CLOSE INHIBIT commands for the C, L, and R engines
- Go to step 1

Note: This sequence is active for GNC OPS 101, 102, and 103. Sequence begins at transition to 101 and ends at issuance of ET SEP commands.

2.2.3 SRB Data Acquisition

SRB MDM data must be obtained from the SRB's during prelaunch operations and during SRB powered flight portions of the mission. The GPC's contain provisions to acquire data from all four (two each SRB) MDM's. The acquisition of SRB MDM data will be initiated at transition to OPS 101. (Data are used only in downlist.)

2.2.4 Space Shuttle Main Engine Operations

Since the SSME OPS sequence is extremely complicated, it would not be practical to present it in the standard sequence format. Therefore, the following description was extracted from the sequence requirements book, SD 76-SH-0026B (1 May 1979) and will serve to define the sequence.

2.2.4.1 Introduction.- The SSME operations sequence is initiated at T-0 SRB IGNITION and is used during the ascent phase to:

- Monitor the operating phase of each main engine.
- Issue inhibit commands to prevent a second engine from automatically shutting down if one shutdown has already occurred.
- Monitor the state of cockpit switches via flags from the switch processor and issue appropriate commands.
- Monitor a flag from GNC software for proper time to check the LO₂ and LH₂ low level sensors.
- Monitor a cutoff timing request flag from GNC software for proper time for MECO, to meet the desired time for start of thrust tailoff with accuracy requirements of ±40 ms.
- Issue main engine shutdown commands when required.
- Close LO₂ and LH₂ prevalues for each engine after shutdown has occurred.

In addition, an SSME data path fail flag from the SSME SOP is checked and, if set and shutdown commands have been issued for that engine, then its prevalues are closed after a time delay. A MECO CONFIRMED flag is set by the SSME operations sequence after it has been confirmed that all engines have shut down. Also, a flag is set for the External Tank (ET) separation sequence after the prevalues for all engines are commanded closed. This is necessary since shutdown times may differ.

2.2.4.2 Overview.- The SSME operations sequence is initiated when the RS launch sequence fires the SRB's and sets the T-0 flag. At this point the main engines are at or above the required thrust, but the engine controllers will not enter the mainstage phase until 5.0 seconds after receipt of the start commands. The SSME OPS sequence operates cyclically from initiation at T-0 until MECO is verified and a few seconds after the prevalues are commanded closed. Under normal operation throughout ascent there are

no commands to issue until the end of second stage when the main engines must be shutdown and certain Main Propulsion System (MPS) valves are closed.

Continuous monitoring of certain inputs is required in the event that: an automatic shutdown of one engine occurs; the crew operates any of the manual switches; data from any of the main engines is lost; or, near the end of boost, either the LH₂ or LO₂ low level sensors indicate fuel or oxidizer depletion.

If shutdown of an engine occurs, for any reason, the LO₂ and LH₂ prevalves for that engine must be closed after appropriate time delays, and the remaining two engines are inhibited from performing an automatic shutdown.

The switch processor software monitors the position of several crew station switches for the MPS and sets flags which are monitored by SSME OPS. The MPS switches being monitored are the three shutdown pushbutton switches and the limit shutdown switch, which can override the automatic inhibit logic and inhibit or enable engine automatic shutdown.

The SSME OPS sequence monitors the main engine chamber pressure and manual shutdown switches. If shutdown is confirmed, the appropriate MPS valves are closed. The operating phase and mode within a phase are determined from the engine status word by the SSME SOP, which gathers the engine data from the EIU, decodes it, and sets applicable flags for the various user software packages. If valid data is not available from the engine, SSME SOP sets a data path fail flag. If this flag is set and shutdown commands have been issued, SSME OPS proceeds, after a time delay, to close the engine prevalves.

SSME OPS, upon receiving a flag from guidance, begins monitoring the fuel and oxidizer low level sensors. A check is made to see whether any sensors have been disabled during pre-launch checkout because of faulty indications. If any two LO₂ low level sensors, which have not been disabled, indicate a dry condition, the logic will issue the MECO commands. Likewise, two LH₂ low-level sensors which indicate dry and have not been previously disabled, will cause the issuance of MECO commands.

Normal engine shutdown, MECO, is triggered by the vehicle achieving the desired velocity and a flag being set by guidance software. The MECO commands issued include shutdown enable and shutdown commands through the EIU's to each of the three main engine controllers. The MECO commands are issued until it is determined that each engine has shut down. At this point a MECO CONFIRMED flag is set for GNC applications and to initiate the external tank separation sequence.

SSME OPS continues to operate until the required time after prevalves for each engine are commanded closed and the close commands are removed. Nominal and RTLS shutdown conditions require different time delays for prevalve closure. When LO₂ and LH₂ prevalves for all three engines have been commanded closed, a flag is set for the ET separation sequence. This is required to prevent initiation of ET disconnect valve closure prior to initiation of all prevalve closures. When the prevalve's close commands have been removed and the flag set, the SSME OPS sequence is terminated.

Each main engine is monitored for FASCOS vibration limits exceeded. If one main engine shows that vibration limits have been exceeded, that engine will be commanded to shut down. The remaining two engines will be inhibited from shutting down. If two or all three engines show vibration limits exceeded, these inputs will not result in any engine shutdown. These engine shutdowns will be due to actual engine vibration rather than vehicle vibration.

2.2.5 SRB Separation

SRB SEPARATION SEQUENCE

<u>TIME</u> (Sec)	<u>EVENT</u>
105	1. <u>MONITOR SEPARATION CUES</u>
122	Proceed to step 2 if: A. $P_c < 50$ psia on both SRB's and Δt between cues is < 4.0 sec. (If > 4.0 sec, then go to step 1B.) B. When MET > 131 sec continue
	2. <u>PREPARE FOR SEPARATION</u>
122 (131)	A. Arm separation PIC's B. Safe range safety system (Safe and Arm (S&A) device to SAFE) C. Remove range safety system battery power Wait 2.3 sec
124.3 (131.3)	D. Null SRB TVC actuators E. Transition to stage II Flight Control System (FCS) configuration Wait 1.7 sec
126 (135)	F. Terminate SRB Ascent Thrust Vector Control (ATVC) power G. Terminate SRB ATVC fault detection circuitry
	3. <u>CHECK SEPARATION INHIBITS</u>
	A. Monitor the following crew control positions: ● SRB SEP sw - MAN/AUTO ● SRB SEP sw - AUTO ● SRB SEP pb - pushed Go to step 4 if: ● SRB SEP sw - MAN/AUTO ● SRB SEP pb - pushed Otherwise, continue step 3.

TIME (Sec)

EVENT

B. Monitor the following:
Pitch, yaw, and roll rates and dynamic
pressure (Q)

If rates and Q within limits, go to step 4.

If any rate or Q out of limits, then:

SM ALERT lt - on
SM ALERT tone - on
SEP INH (CRT)
Return to step 3A

4. COMMAND SEPARATION

A. Issue SRB SEP FIRE 1 & 2 commands

SRB SEPARATION OCCURS

126*
(135*)

5. COMPLETE SEPARATION SEQUENCE

A. Turn off SRB power buses

B. Remove all MEC commands

Wait 0.04 sec

C. Reset MEC 1 & 2

D. Terminate SRB SEP SEQ

*Plus Δ TIME between sep inhibit and manual
override of inhibits or restoration of inhibit
within limits.

BFS SRB separation sequence: The BFS does not perform the 4 second Δt reasonableness check on the $P_c < 50$ cues. If both $P_c < 50$ cues are present, the SRB separation sequence begins immediately (if the sequence is scheduled and not already begun).

2.2.6 External Tank Separation

NOMINAL ET SEPARATION SEQUENCE

<u>TIME</u> (Sec)	<u>EVENT</u>
522.2	<p><u>MECO COMMANDED</u></p> <ol style="list-style-type: none">1. <u>MONITOR SEPARATION CUES</u><p>Proceed to step 2 if:</p><ol style="list-style-type: none">A. All SSME's below 30 percent power level (MECO CONFIRMED)orB. Two SSME's below 30 percent power level and the remaining SSME with a data path failure2. <u>DETERMINE SEPARATION MODE (AUTO/MANUAL)</u><ol style="list-style-type: none">A. Monitor the following crew control positions:<ul style="list-style-type: none">● ET SEP sw - MAN● ET SEP sw - AUTO● ET SEP pb - pushedB. Proceed to step 3 if:<ul style="list-style-type: none">● ET SEP sw - AUTOor● ET SEP sw - MAN● ET SEP pb - pushedC. Return to step 2A if:<ul style="list-style-type: none">● ET SEP sw - MAN● ET SEP pb - not pushed3. <u>PREPARE FOR SEPARATION</u><ol style="list-style-type: none">A. Open LH2 and LO2 feedline relief shutoff valves.B. Enable External Tank (ET) umbilical door and latch lock to be opened.C. Check if all prevalves (LH2 & LO2) commanded closed (flag set from SSME OPS sequence). If not, return to step 1.D. Arm ET tumble system.<p>Wait 1 second (from step 3C).</p>

<u>TIME</u> (Sec)	<u>EVENT</u>
523.2	<p>E. Close LH2 & LO2 feedline disconnect valves.</p> <p>F. Turn on ET/ORB SEP camera.</p> <p>G. Fire ET tumble system pyro.</p> <p>H. Engines gimballed to MPS dump position (from MPS TVC CMD SOP).</p> <p>Wait 1.5 seconds (from step 3E).</p>
524.7	<p>I. Arm ET umbilical unlatch PIC's.</p> <p>Wait 1.5 seconds.</p>
526.2	<p>J. Deadface power across ET/Orbiter interface.</p> <p>K. Issue ET UMBILICAL UNLATCH FIRE 1 & 2 CMD's.</p> <p>***** *ET UMBILICALS UNLATCHED* *****</p> <p>L. Begin retraction of ET umbilical plates.</p> <p>Wait 5.5 seconds.</p>
531.7	<p>M. Arm structural separation PIC's</p> <p>Wait 1.5 seconds, proceed to step 4.</p>
533.2	<p>4. <u>CHECK SEPARATION INHIBITS</u></p> <p>A. Monitor the following:</p> <ul style="list-style-type: none"> ● LH2 & LO2 feedline disconnect valve positions ● Pitch, yaw, roll rates ● Presence of commfault in regard to disconnect valve positions <p>B. Proceed to step 4D if:</p> <p>All feedline disconnect valves closed and no commfault present and pitch, yaw, and roll rates ≤ 0.2 deg/sec; otherwise, continue to step 4C.</p>

TIME (Sec)

EVENT

- C. If any feedline disconnect valve open
or
pitch, yaw, or roll rate > 0.2 deg/sec,
then
SM ALERT lt - on
SM ALERT tone - on
SEP INH message (on CRT message line)

Turn off ET/ORB SEP camera.

Proceed to step 4E.

- D. Monitor the following:
- ET SEP sw - MAN
 - ET SEP sw - AUTO

Proceed to step 5 if ET SEP sw - AUTO.

Otherwise, continue to step 4E.

- E. Monitor the following:
- ET SEP sw - MAN
 - ET SEP sw - AUTO
 - ET SEP pb - pushed

Turn off ET/ORB SEP camera and return to step 4A if:

- ET SEP sw - MAN
- ET SEP pb - not pushed

Otherwise, continue to step 5.

5. COMMAND SEPARATION

- A. Inhibit RCS jet firings.
- B. Turn on ET/ORB SEP camera (if turned off previously).
- C. Issue ET SEP FIRE 1 & 2 cmd's.
- D. MPS D/D SEQ turns off all red and amber engine status lights.

TIME (Sec)

EVENT

6. COMPLETE SEPARATION SEQUENCE

- A. Remove all MEC cmds.
- B. Enable RCS jet firings.
- C. Terminate all feedline disconnect valve CLOSE cmds.

Wait 5 seconds.

538.2

- D. Reset MEC 1 & 2.
-

BFS ET separation: For nominal ET separation, the BFS sequence is identical to the PASS sequence. There is no ET FAST SEP capability presently planned for the BFS.

One exception: In step 4A, BFS does not check for presence of commfault.

OPS 6 RTLS ET SEP

The following are delta's to the nominal sequence for the RTLS ET SEP sequence:

1. Post ET SEP, sequence closes ET umbilical doors
2. ET SEP INH limits are different*
3. ET SEP INH is automatically overridden after 6 seconds on RTLS (i.e., maximum time from MECO that ET SEP will occur is 18 sec (12 sec + 6 sec)).

$$* \alpha = -40 \begin{matrix} +20 \\ -850 \end{matrix}$$

$$\beta = 00 \pm 20$$

$$\text{Pitch rate} = -.250/s \begin{matrix} +.50/s \\ -4.750/s \end{matrix}$$

$$\text{Yaw rate} = 00/s \pm 0.50/s$$

$$\text{Roll rate} = 00/s \pm 50/s$$

BFS OPS 6 ET SEP is same as PASS

2.2.7 Main Propulsion System Dump

NOMINAL MPS PROPELLANT DUMP SEQUENCE

<u>TIME</u> (Sec)	<u>EVENT</u>
520	1. <u>MECO CONFIRMED</u> Wait 10 seconds, proceed to step 2.
530	2. <u>INTERCONNECT He SUPPLY BOTTLES</u> A. Issue OPEN cmds to L, C, & R engine helium OUT interconnect valves. B. Issue OPEN cmd to L engine helium crossover valve.
	3. <u>DETERMINE DUMP MODE</u> A. Monitor the following: <ul style="list-style-type: none">● RTLS abort flag - set● LO2 dump sw - start● LO2 dump sw - GPC● LO2 dump sw - stop B. If LO2 dump sw - stop, return to step 3A. C. If LO2 dump sw - start, proceed to step 4. D. If LO2 dump sw - GPC and OMS 1 burn not initiated, proceed to step 5. Otherwise, return to step 3A.
Variable	4. <u>MANUAL LO2 DUMP</u> (NOTE: Concurrent with step 4, monitor OMS burn flag. When flag = true, start LO2 dump timer.) A. Issue L, C, & R, LO2 PREVALVE CLOSE cmds. Wait 1 second. B. Issue L, C, & R, LO2 PREVALVE OPEN cmds. C. Issue LO2 DUMP START cmds to L, C, & R engine controllers (causes main oxidizer valves to open).

<u>TIME</u> (Sec)	<u>EVENT</u>
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D. Issue L02 manifold pressurization valves (2) OPEN cmds.

 BEGIN L02 DUMP

E. Proceed to step 6.

5. AUTO L02 DUMP

A. Monitor OMS burn flag. When flag is true, start L02 dump timer (140 seconds) and continue to step 5B; otherwise, return to step 3A.

640

B. Issue L, C, & R L02 prevalue CLOSE cmds.

Wait 1 second.

C. Issue L, C, & R L02 prevalue OPEN cmds.

D. Issue L02 DUMP START cmds to L, C, & R engine controllers (causes main oxidizer valves to open).

E. Issue L02 manifold pressurization valves (2) OPEN cmds.

BEGIN L02 DUMP

F. Proceed to step 6.

Variable

6. L02 DUMP TERMINATION

A. Monitor the following:

- L02 dump sw - GPC
- L02 dump sw - stop
- L02 dump timer - timed out

B. If L02 dump sw - GPC, and L02 dump timer timed out, then proceed to step 6D. Otherwise, return to step 6A.

C. If L02 dump sw - stop, continue to step 6D. Otherwise, return to step 6A.

TIME (Sec)

EVENT

D. Issue L02 manifold pressurization valves (2)
CLOSE cmds.

Wait 10 seconds.

E. Issue L, C, & R L02 prevalues CLOSE cmds.

STOP L02 DUMP

Variable

7. LH 2 DUMP (AUTO/MANUAL)

A. Monitor the following crew control positions:

- LH2 dump sw - start
- LH2 dump sw - GPC
- LH2 dump sw - stop

B. If LH2 dump sw - start
or

LH2 dump sw - GPC,
then proceed to step 7D. Otherwise, continue
to step 7C.

C. If LH2 dump sw - stop, return to step 7A.

D. Start LH2 dump timer (40 seconds).

E. Issue LH2 dump START cmds to L, C, & R
engine controllers (causes main oxidizer
valves to close, starts 10-second helium
purge of engine fuel lines and opens main
fuel valves).

F. Issue L, C, & R LH2 prevalues CLOSE cmds.

Wait 1 second.

G. Issue L, C, & R LH2 prevalues OPEN cmds.

Wait 2 seconds (from step 6E).

H. Remove L02 L, C, & R prevalue CLOSE cmds.

Wait 10 seconds (from event 7E).

TIME (Sec)

EVENT

I. Issue LH2 manifold pressurization valves (2)
OPEN cmd.

BEGIN LH2 DUMP

Variable

8. LH2 DUMP TERMINATION

- A. Monitor the following:
- LH2 dump sw - GPC
 - LH2 dump sw - stop
 - LH2 dump timer - timed out
- B. If LH2 dump sw - GPC and LH2 dump timer timed out, then proceed to step 8D. Otherwise, return to step 8A.
- C. If LH2 dump sw - stop, continue to step 8D. Otherwise, return to step 6A.

D. Issue LH2 manifold pressurization valves (2)
CLOSE cmds.

Wait 10 seconds.

E. Issue TERMINATE SEQUENCE cmds to L, C, & R engine controllers (causes main fuel valves to close).

STOP LH2 DUMP

F. Remove L, C, & R LH2 pre valve OPEN cmds (leaves LH2 pre valves open).

Variable

9. COMPLETE MPS DUMP SEQUENCE

A. Request engines gimbal to stow position (MPS TVC CMD SOP).

TIME (Sec)

EVENT

B. Remove the following cmds:

- All LH2 prevalue OPEN cmds
- All L02 and LH2 feedline disconnect valve CLOSE cmds
- LH2 recirc disconnect valve CLOSE cmd
- All L02 and LH2 fill/drain valve CLOSE cmds
- Replace LH2 ullage press 1, 2 & 3 xdcr cmds
- Replace L02 ullage press 1, 2 & 3 xdcr cmds
- ET/ORB sep cameras ON cmd
- ET/ORB sep cameras htrs ON cmd
- All L02 overboard bleed valve CLOSE cmds
- All LH2 prevalue CLOSE cmds
- L, C & R engine helium OUT interconnect valves OPEN cmds
- L engine helium crossover valve OPEN cmd
- FASCOS power cmds

Wait 2 seconds.

C. Remove the following cmds:

- All L02 prevalue CLOSE cmds
- All L02 prevalue OPEN cmds

BFS MPS dump: The BFS sequence for a nominal MPS propellant dump is always initiated and terminated automatically (i.e., the software never looks at the L02 and LH2 propellant dump switch positions). The PASS sequence for a nominal MPS propellant dump can be initiated and terminated manually by the crew.

RTLS MPS PROPELLANT DUMP SEQUENCE

<u>TIME (Sec)</u>	<u>EVENT</u>
Variable	<ol style="list-style-type: none">1. <u>MM 602 INITIATED</u> (unpowered RTLS)2. <u>INTERCONNECT He SUPPLY BOTTLES</u><ol style="list-style-type: none">A. Issue OPEN cmds to L, C, R engine helium OUT interconnect valves.B. Issue OPEN and to L engine helium cross-over valve.3. <u>INITIATE LH2 DUMP</u><ol style="list-style-type: none">A. Issue LH2 RTLS IB/OB dump valve OPEN cmds.B. Issue LH2 RTLS manifold repress valve OPEN cmds. <div style="border: 1px solid black; padding: 5px; text-align: center;">! BEGIN LH2 DUMP !</div>4. <u>INITIATE LO2 DUMP</u><ol style="list-style-type: none">A. Issue L, C, R LO2 prevalues CLOSE cmds. Wait 1 second.B. Issue L, C, R LO2 prevalues OPEN cmds.C. Issue LO2 dump START cmds to L, C, R engine controllers (causes main oxidizer valves to open). <div style="border: 1px solid black; padding: 5px; text-align: center;">! BEGIN LO2 DUMP (VENT) !</div>5. <u>DUMP TERMINATION CHECK</u><ol style="list-style-type: none">A. Monitor the ground relative velocity (V) magnitude in the M50 coordinate system.B. If $V < 4500$ ft/s, continue to step 6; otherwise, return to step 5A.

TIME (Sec)

EVENT

6. LH2 DUMP TERMINATION

- A. Remove LH2 RTLS IB/OB dump valve OPEN cmds.
- B. Remove LH2 RTLS manifold repress valve OPEN cmds.

! !
! STOP LH2 DUMP !
! !

7. L02 DUMP TERMINATION

- A. Remove L02 manifold repress valve OPEN cmds.
- B. Issue L, C, R L02 prevalues CLOSE cmds.
- C. Issue TERMINATE SEQUENCE cmds to L, C, R engine controllers (causes main oxidizer valves to close).

! !
! STOP L02 DUMP (VENT) !
! !

Wait 2 seconds (after step 7B)

- D. Remove L, C, R L02 prevalue CLOSE cmds.

8. COMPLETE MPS DUMP SEQUENCE

- A. Remove the following commands:

- All LH2 prevalue OPEN cmds
- All L02 & LH2 feedline disconnect valve CLOSE cmds
- LH2 recirc disconnect valve CLOSE cmd
- All L02 & LH2 fill/drain valve CLOSE cmds
- Replace LH2 ullage press 1, 2, 3 xdcr cmds
- Replace L02 ullage press 1, 2, 3 xdcr cmds
- ET/ORB sep cameras ON cmd
- ET/ORB sep cameras htrs ON cmd
- All L02 overboard bleed valve CLOSE cmds
- All LH2 prevalue CLOSE cmds
- FASCOS power cmds

Wait 2 seconds.

TIME (Sec)

EVENT

B. Remove the following commands:

- All LO2 prevalve CLOSE cmds

C. Monitor the ground relative velocity (V) magnitude in the M50 coordinate system

D. If $V \leq$ TBD ft/s, continue to step 8E; otherwise return to step 8C.

E. Issue He supply blowdown valves 1 & 2 OPEN cmds

F. Start He purge timer

Wait TBD seconds

G. Remove the following commands:

- He supply blowdown valves 1 & 2 OPEN cmds (valves close)
- L, C, R engine He OUT interconnect valves OPEN cmds
- L engine He crossover valve OPEN cmd.

BFS RTLS MPS propellant dump sequence: BFS sequence is identical to the PASS sequence.

2.2.8 Abort Control

ABORT CONTROL SEQUENCE (STS-1 ONLY)

PASS ABORT CONTROL SEQUENCE

1. COMMAND ALL VENT DOORS TO CLOSE
2. CHECK FOR CONTINGENCY ABORT BY MONITORING
 - Second SSME FAIL CONFIRM
 - CONT DUMP ENABLE
 - CONT DUMP INHIBIT

If any is true, go to step 9.
3. IF ABORT ONCE AROUND (AOA) IS DECLARED, GO TO STEP 6
4. IF ABORT TO ORBIT (ATO) IS DECLARED, GO TO STEP 7
5. RTLS ABORT - PRE-MECO
 - A. Select RTLS I-loaded table values for OMS/RCS burn control.
 - B. Wait until Powered Pitch Around (PPA) maneuver is started, then fire OMS.
 - C. Command OMS/RCS interconnect (open OMS and RCS crossfeed valves, close RCS tank isolation valves). This takes 4.5 sec.
 - D. If OMS PROP RCS +X ΔT COMP and OMS PROP NULL ΔT COMP equal 0 (AOA or ATO), go to step 5I.
 - E. If RCS + X - ON OMS-ON TIME < I-loaded OMS PROP RCS +X ΔT COMPUTED (255 sec) fire four RCS +X jets. Otherwise, terminate RCS +X fire command and set RCS +X COMPLETE flag.
 - F. If RCS NULL JET-ON OMS-ON TIME < OMS PROP NULL ΔT COMPUTED (51.7 sec):
 - If RTLS, fire 20 RCS null jets.Otherwise, terminate RCS NULL JETS FIRE cmd and set RCS NULL COM-LETE flag.
 - G. If OMS ON TIME < OMS T COMPUTED (260 sec), fire OMS. Otherwise, issue OMS CUTOFF cmd.
 - H. If RCS +X COMPLETE flag and RCS NULL COMPLETE flag are true, continue to step 5I. If not, go to step 5E.

I. Terminate OMS/RCS interconnect.

J. Monitor

- MECO PREPARATION discrete
- MECO CONFIRMED flag

If either is true, go to step 8A. Otherwise, go to step 5G.

6. AOA ABORT

A. Select AOA OMS/RCS burn control values from one of two I-loaded tables, depending on V_{REL} at SSME failure and/or mission phase at abort declaration.

NOTE: There will be no pre-MECO OMS or RCS dump for AOA for STS-1.

B. Set TABLE SEL COMPLETE flag and go to step 5D.

7. ATO ABORT

A. Select ATO OMS/RCS burn control values by modifying initial ATO I-loaded tables as a function of V_{REL} at SSME failure and/or mission phase at abort declaration.

NOTE: There will be no pre-MECO OMS or RCS dump for ATO for STS-1.

B. Set TABLE SEL COMPLETE flag and go to step 5D.

8. MECO AND POST-MECO

A. If MANUAL TAKEOVER is declared, set MODE 2 INDICATOR (controls OMS/RCS disconnect sequence).

B. Inhibit OMS and RCS firings in preparation of ET separation.

ET SEP OCCURS

C. Wait until CONT DUMP ENABLE is true (crew manually starts dump via CONT ABORT, SPEC 52, display), then go to step 10A.

9. CONTINGENCY ABORT - PRE-MECO

A. Terminate all RCS JET FIRE cmds.

B. Set MODE 2 INDICATOR (used by OMS/RCS seq).

C. Issue OMS to RCS INTERCONNECT cmd.

- D. Wait until CONT DUMP ENABLE signal is true, then continue to step 9E.
- E. If OMS L or R POD FU QTY > CONT FU QTY LEVEL (8.6 percent), go to step 9F. Otherwise, go to step 9H.
- F. Issue ABORT OMS IGN CMD, and after an RCS IGN DELAY timer has expired, fire RCS +X jets and 20 RCS null jets.
- G. If MECO PREPARATION DISCRETE or MECO CONFIRMED flag is true, go to step 9H; otherwise, go to step 9E.
- H. Terminate all OMS and RCS fire cmds and CG TRIM cmd. Issue OMS CUTOFF cmd.

 MECO OCCURS

10. CONTINGENCY ABORT - POST-MECO

- A. When zero thrust occurs
 - Cancel OMS/RCS interconnect
 - Set mode 2 indicator
- B. Wait until MM 602 is entered, then continue to step 10C.
- C. Wait until CONT DUMP ENABLE signal is true, then continue to step 10D. If the signal were true pre-MECO, it will still be true now.
- D. If L or R OMS FU QTY < CONT FU QTY LEVEL (8.6 percent), go to step 10H. Otherwise continue to 10E.
- E. First pass only, command OMS/RCS interconnect.
- F. If a 3 sec CG TRIM DELAY has expired, issue ABORT OMS IGN cmd.
- G. If a 2 sec IGN PRESS DELAY has expired and Nz < CONT Nz LIMIT (.05g), fire 4 RCS +X jets and 20 RCS null jets and go to step 10D.
 - If Nz > CONT Nz LIMIT (.05g):
 - Terminate RCS +X and 20 NULL jets fire cmds.
 - Cancel OMS/RCS interconnect cmd.
 - Fire 4 RCS +X jets using RCS propellants, start timer, and go to step 10D.
- H. Terminate RCS null jets fire cmd.
 - If the OMS have been firing, terminate OMS fire cmd.
 - Continue to step 10I.

I. Cancel OMS/RCS interconnect cmd.

Set CONT LOX DUMP STOP IND true.

Fire four RCS +X jets

J. If RCS +X JETS ON TIME > 125 sec or CONT DUMP INH is true, terminate RCS +X burn and go to step 10C. Otherwise go to 10I.

BFS ABORT CONTROL SEQUENCE

The functions of the pre-MECO BFS abort control sequence are identical to those of the PASS. Twenty seconds after RTLS MECO, however, the BFS performs an automatic RCS dump (as given below) where the PASS has no AUTO dump capability.

8. MECO AND POST-MECO

A. If MANUAL TAKEOVER is declared, set MODE 2 INDICATOR (controls OMS/RCS disconnect sequence).

B. Inhibit OMS and RCS firings in preparation of ET separation.

ET SEP OCCURS

C. Wait until MM 602 + 20 sec, then fire four +X RCS jets.

D. If RTLS RCS DUMP INHIBIT is true (crew stops dump via DUMP STOP on BFS VERT SIT 1), terminate RCS +X ON cmd. Otherwise, continue to step 8E.

E. If 75 sec have elapsed since the RCS +X were ignited, terminate RCS +X ON cmd. Otherwise, go to step 8C.

2.2.9 Vent Doors

The Orbiter's vent and purge system is made up of 18 active ports and is divided into the following six groups:

Left and right ports 1 and 2

Left and right port 3

Left and right port 5

Left and right ports 4 and 7

Left and right port 6

Left and right ports 8 and 9

All vent ports have a purge position with the exception of left and right vent ports 3, 4, 5, and 7. The purge position is required to maintain a

Vent door sequencing can be changed from mission to mission by means of I-loaded commands and times in the sequencing program. This may be done to protect payloads from acoustic damage at launch and maintain sufficient positive ΔP to prevent contamination of compartments.

Vent doors are opened or closed in groups. The delay time between group commands vary per mission stage.

ASCENT

T-35 sec open all doors

LH and RH vents 4 and 7	T-35 to T-30
LH and RH vents 3	T-30 to T-25
LH and RH vents 6	T-25 to T-20
LH and RH vents 5	T-20 to T-15
LH and RH vents 1 and 2	T-15 to T-10
LH and RH vents 8 and 9	T-10 to T-5

T-4 software checks motor

Travel times - if not <5 seconds, issues HOLD flag.
Can be overridden by LPS if problem anticipated previous to countdown.

ON-ORBIT

EET = 7:39

Close all doors for entry.

LH and RH vents 1 and 2	Execution (CRT)
LH and RH vents 3	Execution + 5 second
LH and RH vents 5	Execution + 10 second
LH and RH vents 4 and 7	Execution + 15 second
LH and RH vents 6	Execution + 20 second
LH and RH vents 8 and 9	Execution + 25 second

ENTRY

VREL = 2400 ft/s

Open all doors

LH and RH vents 4 and 7	2400 ft/s
LH and RH vents 3	2400 + 480 ms
LH and RH vents 6	2400 + 960 ms
LH and RH vents 5	2400 + 1.44 sec
LH and RH vents 1 and 2	2400 + 1.92 sec
LH and RH vents 8 and 9	2400 + 2.4 sec

RTLS

MM 602

Close all doors

LH and RH vents 1 and 2	MM 602
LH and RH vents 3	602 + 480 ms
LH and RH vents 5	602 + 960 ms
LH and RH vents 4 and 7	602 + 1.44 sec
LH and RH vents 6	602 + 1.92 sec
LH and RH vents 8 and 9	602 + 2.4 sec

RTLS

VREL = 2400 ft/s

Open all doors

LH and RH vents 4 and 7	2400 ft/s
LH and RH vents 3	2400 + 480 ms
LH and RH vents 6	2400 + 960 ms
LH and RH vents 5	2400 + 1.44 sec
LH and RH vents 1 and 2	2400 + 1.92 sec
LH and RH vents 8 and 9	2400 + 2.4 sec

BFS vent doors: The BFS vent door control sequence is similar to the PASS sequence. Major differences are:

- The BFS does not provide vent door control for the launch configuration.
- The BFS does not have on-orbit capability to manually command the vent doors. The vent doors can be manually opened or closed only in OPS 3 and MM 602 and 603.

2.2.10 Hydraulic System Landing Gear Isolation Valves

This sequence is executed at 1 Hz during entry or RTLS to provide automatic control of the three hydraulic system landing gear isolation valves such that proper heating of the hydraulic fluids and landing gear hardware components is insured prior to landing gear extension. When relative velocity of the Orbiter in an M50 system drops to 19,000 ft/s, hydraulic system 3 landing gear isolation valve is opened. At 5-min intervals, hydraulic systems 2 and 1, respectively, are then opened. Two seconds after each valve is automatically opened, the sequence terminates the OPEN command and the valves are latched open.

A momentary switch is provided for manual operation of the valves by the crew. The center position is nonfunctional. However, this position is labeled GPC to inform the crew that the isolation valves will be opened automatically during entry.

For an RTLS abort, the 5-min timing sequence is bypassed and all three valves are opened simultaneously when major mode flag 602 is set.

The BFS sequence for the hydraulic system landing gear isolation valve is the same as that in PASS except that during entry, the BFS software must be in major mode 303 or 304 before the isolation valve logic is entered.

The landing gear isolation valve sequence interfaces are illustrated in figure 2.2.1.

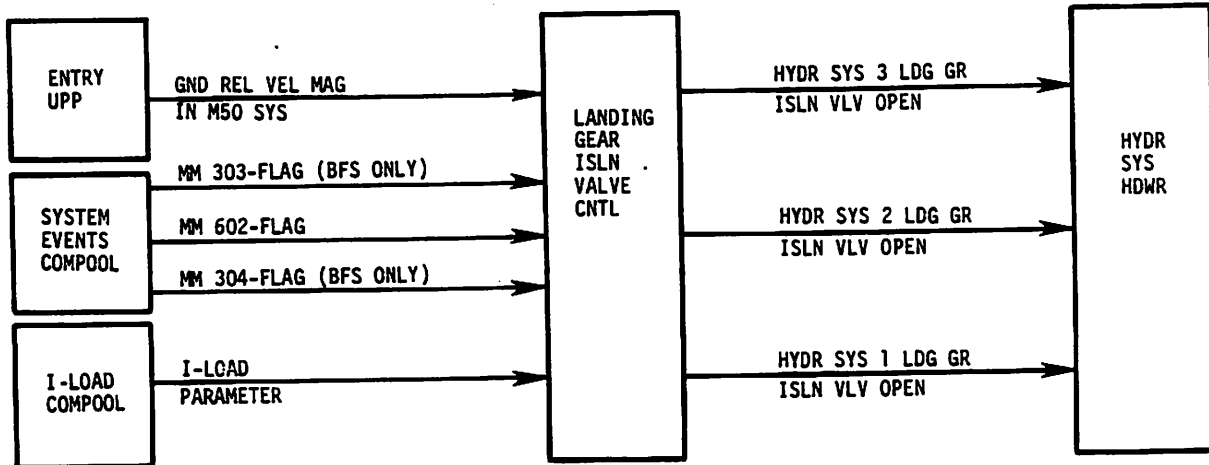


Figure 2.2-1.- Landing gear isolation valve sequence interfaces.

2.2.11 RCS/RCS Crossfeed and Reconfiguration

The RCS propellant crossfeed sequence is available during ascent and entry for contingencies where the crossfeed sequence enables propellants from one aft pod to feed the thrusters of both aft pods. The RCS/RCS crossfeed sequence is initiated by placing the master RCS crossfeed switch (panel 07) in the FEED FROM LEFT or FEED FROM RIGHT position. The RCS propellant crossfeed reconfiguration sequence is used to reconfigure the RCS propellant feed to normal after an RCS crossfeed sequence has been employed. The reconfiguration sequence is initiated by placing the master RCS crossfeed switch in the OFF position. These sequences are available in OPS 1, 3, and 6 for the PASS.

If the master crossfeed switch is placed in the FEED FROM LEFT position, the RCS propellant crossfeed sequence issues the following commands in the order listed to accomplish a crossfeed:

- Close OMS crossfeed valves
- Open RCS crossfeed valves
- Open left aft RCS tank isolation valves
- Close right aft RCS tank isolation valves

The commands are similar for a crossfeed from right.

When the switch is placed in the OFF position, the RCS propellant configuration sequence first issues commands to open the aft RCS tank isolation valves and then issues commands to close the RCS crossfeed valves.

In order to establish a known configuration of the valve commands, a change in the master crossfeed switch position will terminate the open and close GPC commands for the following valves before and after the above crossfeed or reconfigure commands are issued:

- Aft RCS helium pressure valves
- Aft RCS tank isolation valves
- RCS crossfeed valves
- OMS crossfeed valves

All sequences of commands allow time (1.5 sec) between commands to each group of valves so that the commanded positions can be reached before the next command is issued. All aft RCS valve switches should be placed in the GPC position before initiation of this sequence.

The RCS/RCS crossfeed and reconfiguration sequence is not available in the BFS.

OMS/RCS interconnect: The OMS/RCS interconnect sequence automatically maintains the OMS tank ullage pressure during manual OMS to RCS interconnects and enables the OMS to RCS gaging sequence. The interconnect sequence is available during OPS 2 and in OPS 8 if the sequence is active during the transition from OPS 2 to OPS 8.

2.2.12 RCS Quantity Monitor

The RCS quantity monitor sequence uses the GPC to calculate the usable percent of fuel and oxidizer remaining in each RCS module (forward, aft left, and aft right). The sequence also uses the calculated quantities to determine and output the lowest usable quantity (oxidizer or fuel) for each RCS module, and to determine if the difference between each pair of tanks exceeds a preset tolerance (leakage detection). Additionally, the sequence provides automatic closure of the high pressure helium isolation valves on-orbit when the C&W high limit on tank ullage pressure is exceeded. The RCS quantity monitor sequence is available in OPS 2, 8, and 3 in the PASS and in OPS 1, 3, and 6 in the BFS. The initial helium weight and the total usable propellant weight I-loads are to be uplinked after entry into OPS 2 for the PASS sequence, and will be based on propellant load data obtained prior to launch in OPS 1 for the BFS sequence. The I-loads must be protected across OPS transitions between on-orbit and entry.

RCS quantities are computed based on the pressure, volume, and temperature method, which requires that pressure and temperature measurements be combined with a unique set of constants to calculate the percent remaining in each of the six propellant tanks. Correction factors are included for residual tank propellant at depletion, gaging inaccuracy, and trapped line propellant, so that the computed quantity should represent usable (rather than total) quantity. The sequence also includes the following effects:

- Helium gas compressibility and helium bottle expansion at high pressure
- Oxidizer vapor pressure as a function of temperature
- Oxidizer and fuel density as a function of temperature and pressure

The sequence assumes that helium which leaves the helium bottle flows to the propellant tank to replace propellant. As a consequence, computed quantity for a propellant tank will decrease for normal usage, propellant leaks, or helium leaks. Transducer failures or comm faults are accounted for by substitution of alternate measurements and the corresponding quantity on CRT displays will have an M in its status field to indicate missing data. If no substitute parameter is available, the quantity calculation for that propellant tank is suspended and the crew is alerted by a class 3 alert, a fault message on the CRT fault message line, and a down arrow beside the quantity.

The digital display on panel O3 displays the usable quantity of fuel and oxidizer for each tank pair, and a separate output of the lowest usable quantity (fuel or oxidizer) for each tank pair.

Gross leak detection, based on a comparison between fuel and oxidizer, will result in a class 2 alarm when a preset absolute difference of 12.6 percent is exceeded. A bias of 12.6 percent is added when a leak is detected, so that subsequent leaks in that same module may be detected. The crew is alerted by a class 2 alarm, the backup C&W light, and the forward, left, or right RCS C&W light on the C&W matrix on panel F7, a fault message on the CRT, and a down arrow beside the quantity on the display.

For each helium tank, a propellant tank ullage pressure over 312 psia will cause the helium isolation valves to be commanded closed in OPS 2. Should

the tank ullage pressure return within normal limits, the close commands will be removed. The crew is alerted by a class 2 alarm, the backup C&W light and the forward, left, or right RCS C&W light on the matrix on panel F7, and a fault message on the CRT fault message line.

The BFS performs the same calculations as the PASS, but does not use substitute parameters, does not annunciate missing parameters, and does not add in the 12.6 percent bias for subsequent tank leaks in a pod.

OPS 1 quantity monitor: The RCS quantity monitor sequence uses the GPC to calculate the usable percent of fuel and oxidizer remaining in each RCS module (forward, aft left, and aft right). The sequence also uses the calculated quantities to determine and output the lowest usable quantity (oxidizer or fuel) for each RCS module, and to determine if the difference between each pair of tanks exceeds a preset tolerance (leakage detection).

The RCS quantity monitor is available in OPS 1 BFS. If a parameter is not available, substitute parameters will not be used or annunciated, and the quantity calculation will be suspended for that propellant tank. BFS will perform gross leak detection but will not add a bias to the percentage in order to detect subsequent or continuing leaks.

The initial helium weight and the total usable propellant weight I-loads will be based on propellant load data obtained prior to launch.

RCS quantities are computed based on the pressure, volume, and temperature method, which requires that pressure and temperature measurements be combined with a unique set of constants to calculate the percent remaining in each of the six propellant tanks. Correction factors are included for residual tank propellant at depletion, gaging inaccuracy, and trapped line propellant, so that the computed quantity should represent usable (rather than total) quantity. The sequence also includes the following effects:

- Helium gas compressibility and helium bottle expansion at high pressure
- Oxidizer vapor pressure as a function of temperature
- Oxidizer and fuel density as a function of temperature and pressure.

The sequence assumes that helium which leaves the helium bottle flows to the propellant tank to replace propellant. As a consequence, computed quantity for a propellant tank will decrease for normal usage, propellant leaks, or helium leaks.

Gross leak detection is based upon a comparison between fuel and oxidizer and a crew alert will be output when the difference is greater than 12.6 percent.

OPS 2 quantity monitor: The RCS quantity monitor is available in the PASS in OPS 2 and 8. The quantity monitor in the PASS works as previously described in OPS 1 in BFS with the following added features.

Transducer or comm faults are accounted for by substitution of alternate measurements and the corresponding quantity on CRT displays will have an M in its status field to indicate missing data. If no substitute parameter is available, the quantity calculation for that propellant tank is

ON FIRE SEQUENCE (Continued)

- C. If PURGE INHIBIT not present (from maneuver display), purge engines according to the OMS ENG switch (panel C3) position (i.e., purge only engines whose switch is in ARM/PRESS).

If purge inhibit present, go to step E.

Wait 360 ms.

- D. Issue purge commands to proper engine(s).

Wait 2 secs.

- E. Terminate purge commands. Reinitialize timers and first pass flag.

2.2.14 On-Orbit OMS/RCS Interconnect Sequence

Even though this sequence is called OMS/RCS interconnect, it does not perform the automatic OMS-to-RCS software gaging sequence and monitors OMS propellant tank pressure to automatically pressurize the tanks if the pressure is lower than an I-loaded value.

This sequence is executed during GNC on-orbit OPS only (OPS 2 and 8). The BFS does not contain this capability.

The OMS/RCS interconnect sequence is initialized by L-OMS or R-OMS item entry on the RCS SPEC CRT display. Selection of the OFF item entry terminates the sequence.

The function of configuring the OMS and the RCS values to achieve propellant interconnect is performed manually by the crew.

ON-ORBIT OMS/RCS INTERCONNECT SEQUENCE

1. DETERMINE IF RCS FEED FROM LEFT OR RCS FEED FROM RIGHT OMS

- A. If either RCS from left or RCS from right flags are true, go to step 1B; if both flags are false, go to step 3.
- B. Determine if interconnect should be initiated (CONN INIT flag = true); if flag true, go to step 1C; if false, go to step 2.
- C. Set GAGING flag to true;
Set CONN INIT flag to false.
- D. Terminate OPEN and CLOSE commands to the left and right RCS XFEED and TANK ISOL valves and to the left and right OMS XFEED valves.

- E. Do not continue if prop tank ullage pressure commfault is present.
2. DETERMINE IF OMS PROPELLANT TANKS SHOULD BE REPRESSURIZED
- A. If any OMS propellant tanks (4) ullage pressure is less than 230 psi.
 - B. Issue commands to open He and VAPOR ISOL valves of pod (left or right) that press is low and start timer A counting.
 - C. When timer A (240 sec) expired, terminate open commands to He and VAPOR ISOL valves.
 - D. If any OMS propellant tank pressure is less than 230 psi, set OMS/RCS VLV MISCOMP = TRUE.
 - E. If any OMS HE or VAPOR ISOL is open, set OMS/RCS VLV MISCOMP = TRUE.
3. TERMINATE SEQUENCE IF RCS FEED FROM LEFT AND RCS FEED FROM RIGHT ARE FALSE
- A. Set CONN INIT flag to true so that the next time the feed from left or right flags is set, the sequence is initialized.
 - B. Set GAGING flag to false.

NOTE: This sequence sets the OMS/RCS VLV MISCOMP flags, but the message and alarm annunciation has been deleted; i.e., there will be no annunciation to the crew of this condition existing.

2.2.15 OMS-to-RCS Gaging Sequence

The OMS-to-RCS gaging sequence is executed during on-orbit GNC OPS (PASS only). This sequence is initiated by the OMS to RCS interconnected sequence whenever the crewman performs the L-OMS or R-OMS item entry in the RCS SPEC display. The sequence is terminated when the off-item entry is selected. The OMS-to-RCS gaging sequence computes the amount of left or right OMS propellant used by the AFT RCS jets and displays this quantity in terms of percent on the RCS SPEC CRT display. This sequence also checks for and annunciates excessive use of OMS propellant by the RCS.

OMS TO RCS GAGING SEQUENCE

1. CALCULATE OMS PROPELLANT USED
- A. Calculate propellant used by main thrusters by adding the weight of propellant used during aft jets' starts and aft jets' cycles.
 - B. Calculate propellant used by the vernier jets based on the number of aft vernier jets' cycles.

C. Add propellant used by main jets and propellant used by verniers.

2. TOTAL OMS PROPELLANT USED

- A. If left OMS to RCS flag is true, add propellant used to the total propellant from left used by RCS.
- B. If right OMS to RCS flag is true, add propellant used to the total propellant from right OMS used by RCS.
- C. Compute total left (and right) OMS quantity used by RCS in percentage terms. (This value is used for display in the RCS SPEC display.)

3. CALCULATE IF OMS USED EXCEEDS LIMIT

- A. If left (right) OMS to RCS flag is true, check if left (right) OMS QTY used > L-OMS (R-OMS) limit (8.37 percent I-LOAD value which is equivalent to 1000 lb of propellant).
- B. If limit is exceeded, set flag to be used for outputting a class 3 message OMS/RCS QTY and an ↑ by the quantity used on the RCS SPEC display.

2.2.16 Abort OMS-to-RCS Interconnect

The abort OMS-to-RCS interconnect sequence provides the commands necessary to feed OMS propellant to the RCS jets when required by the abort control sequence. In addition, controls are provided to reconfigure to the normal RCS and OMS propellant feed from their respective tanks.

If a mode 2 (contingency mode) indicator is set, the order of valve sequencing is modified to allow continuous propellant flow to the RCS jets. This sequence is available during OPS 1 and 6.

The BFS OMS/RCS interconnect sequence differs from the PASS in that it does not provide a contingency mode of abort OMS-to-RCS interconnect.

ABORT OMS-TO-INTERCONNECT SEQUENCE

1. OMS-TO-RCS INTERCONNECT COMMAND

- A. Terminate all OPEN and CLOSE commands to OMS and RCS XFEED valves and to RCS TANK ISOL valves.
- B. Check if mode 2 indicator is true. If true, go to contingency intercon step 2; if not true, continue.
- C. Command L & R RCS TANK ISOL valves closed.

Wait 1.5 sec.

D. Command L & R OMS XFEED valves open.

E. Command L & R RCS XFEED valves open.

Wait 1.5 sec.

F. Set OMS/RCS INTERCON COMPLETE flag to true.

2. CONTINGENCY INTERCONNECT (MODE 2 INDICATOR SET)

A. Command L & R XFEED valves and L OMS A XFEED valves and R OMS B XFEED valves open.

B. Command L & R RCS TANK ISOL valves closed.

Wait 1.5 sec.

C. Command L OMS B XFEED valves and R OMS A XFEED valves open.

D. Set OMS/RCS INTERCON COMPLETE flag to true.

3. OMS-TO-RCS RETURN TO NORM CONFIG

A. Terminate all OPEN and CLOSE commands to OMS and RCS XFEED valves and to the RCS TANK ISOL valves.

B. Check if mode 2 indicator is true. If true, go to contingency reconfig step 4; if not true, continue.

C. Command L & R RCS XFEED valves closed.

D. Command L & R OMS XFEED valves closed.

Wait 1.5 sec.

E. Command the L & R RCS TANK ISOL valves open.

Wait 1.5 sec.

F. Set OMS/RCS INTERCON COMPLETE flag to false.

4. CONTINGENCY RECONFIGURATION

A. Command L & R RCS TANK ISOL valves open.

Wait 1.5 sec.

B. Command L & R RCS XFEED valves closed.

Wait 1.5 sec.

C. Command L & R OMS XFEED valves closed.

D. Set OMS/RCS INTERCON COMPLETE flag to false.

2.3 NAVIGATION SOFTWARE

The ascent navigation principle function uses selected IMU data and a model of the earth's gravitational acceleration to maintain a current estimate of the shuttle state vector.

2.3.1 Navigation Initialization

At T-8 sec, the launch pad location (latitude, longitude, and altitude) of the nav base is transformed to an M50 position vector. An M50 velocity vector is computed from the earth's angular velocity vector. These values provide an initial state vector which will be propagated using the basic gravity model and sensed velocity changes from the IMU's.

2.3.2 Mainstage Navigation

The navigation scheme used is that known as SUPER-G. A state vector is propagated using IMU velocities and a gravity model. The state is corrected for the change in gravity over the integration step (fig. 2.3-1).

High rate user demands are fulfilled by UPP. UPP integrates the state at a higher rate than SUPER-G and provides the state data to users. At the end of each nav cycle, UPP is updated using the more accurate SUPER-G state.

2.3.3 Orbit Insertion Navigation

The SUPER-G scheme is used during orbit insertion as it was during mainstage ascent; however, there is a modification. In this stage of the flight, if the IMU sensed acceleration is above a noise threshold ($3.2 \times 10^{-2} \text{ ft/s}^2$), then the same nav model is used that was used in mainstage (i.e., IMU ΔV 's and a simple model of the earth's gravitational acceleration). If the IMU sensed acceleration is below the threshold, IMU ΔV 's are ignored and an atmospheric drag acceleration model is used instead. Also, a more accurate model of the earth's gravity is used by including higher order terms in the existing gravity model.

OPS 1 BFS navigation: This is an outline of the major differences between the PASS and the BFS for navigation in OPS 1.

At the transition to MM 101, BFS navigation receives a "snapshot" of the REFSMAT from PASS transfer data. At T-8 seconds, navigation is initialized and started.

The BFS transitions to MM 102 at SRB ignition. From this point on, the BFS is capable of assuming control of the vehicle and can be engaged.

First and second stage navigation maintains a state vector based on IMU data and a gravity model. The BFS uses a mid-value select method for IMU

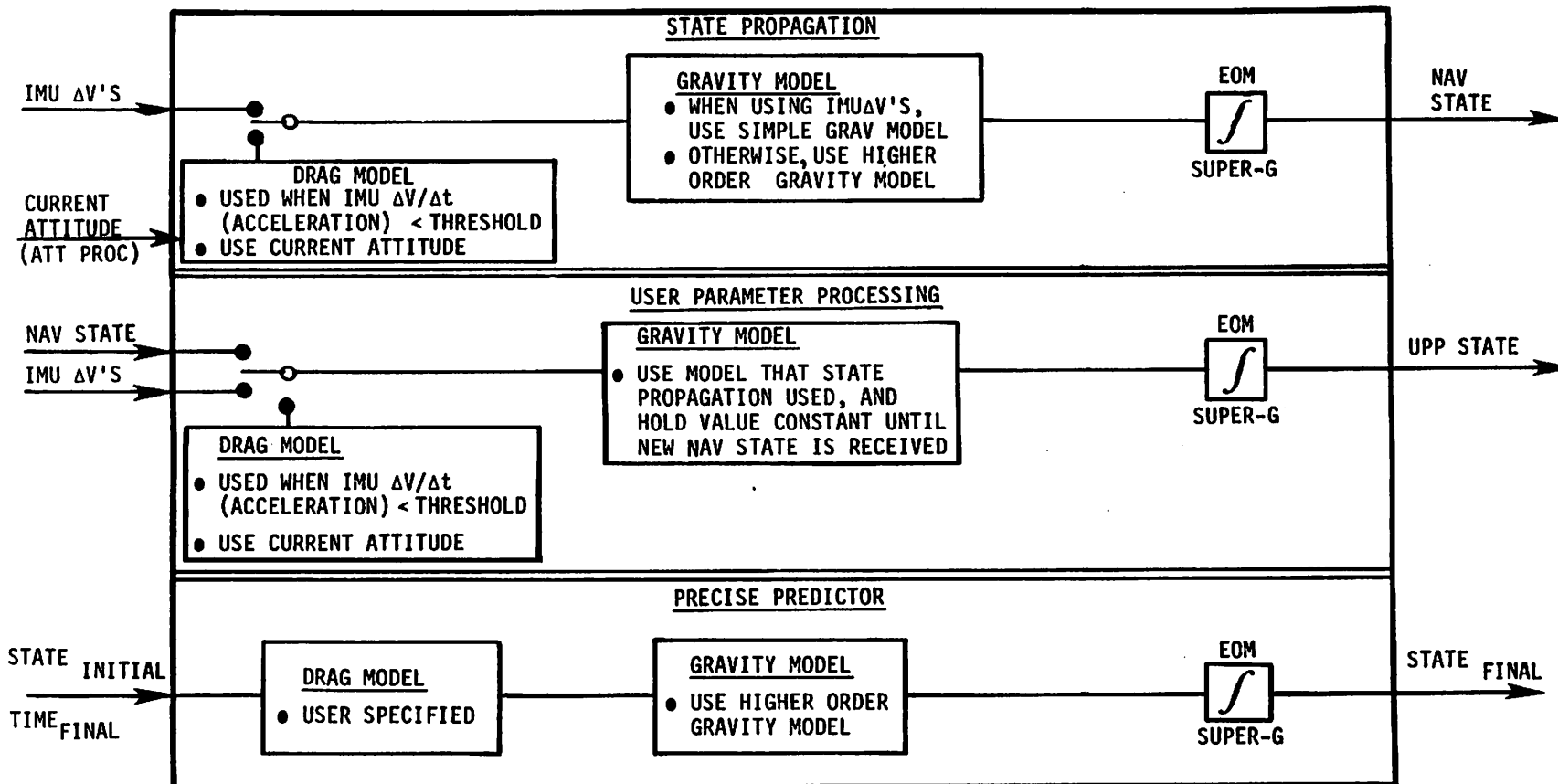


Figure 2.3-1.- Mainstage navigation.

data rather than a more complex redundancy management system as in the PASS. The BFS gravity model is the same as in the PASS.

BFS navigation for OMS 1 and OMS 2 maneuvers is similar to mainstage navigation. State vector data can be updated via uplink or keyboard entry.

The BFS can remain in MM 106 indefinitely and navigation simply stores the last state vector it receives. The planned procedure, however, is to transition the BFS to OPS 0 while on-orbit. The BFS can later be engaged in OPS 0 and transitioned to OPS 3, but if the PASS is not active, uplinks will be required to update the state vector and to supply other data.

2.3.4 On-Orbit Navigation

The on-orbit/RNDZ nav sequencer, on-orbit precise predictor, precision nav, the venting and uncoupled thrusting models, and covariance matrix propagation have been deferred until TBD.

State propagation in OPS 2 is functionally the same as OPS 1 orbit insertion, figure 2.3-2. The ascent/entry precise predictor is used on-orbit for state vector updates.

Because of the deferral of the precise integrator and some of the models, state vector uplinks must be made about once per revolution to maintain accuracy of the state. The mission time of the last update is displayed on the ON-ORBIT OPS display, figure 2.3-3.

OPS 8 navigation is identical (part of the major function base for this transition) to OPS 2 navigation.

2-171

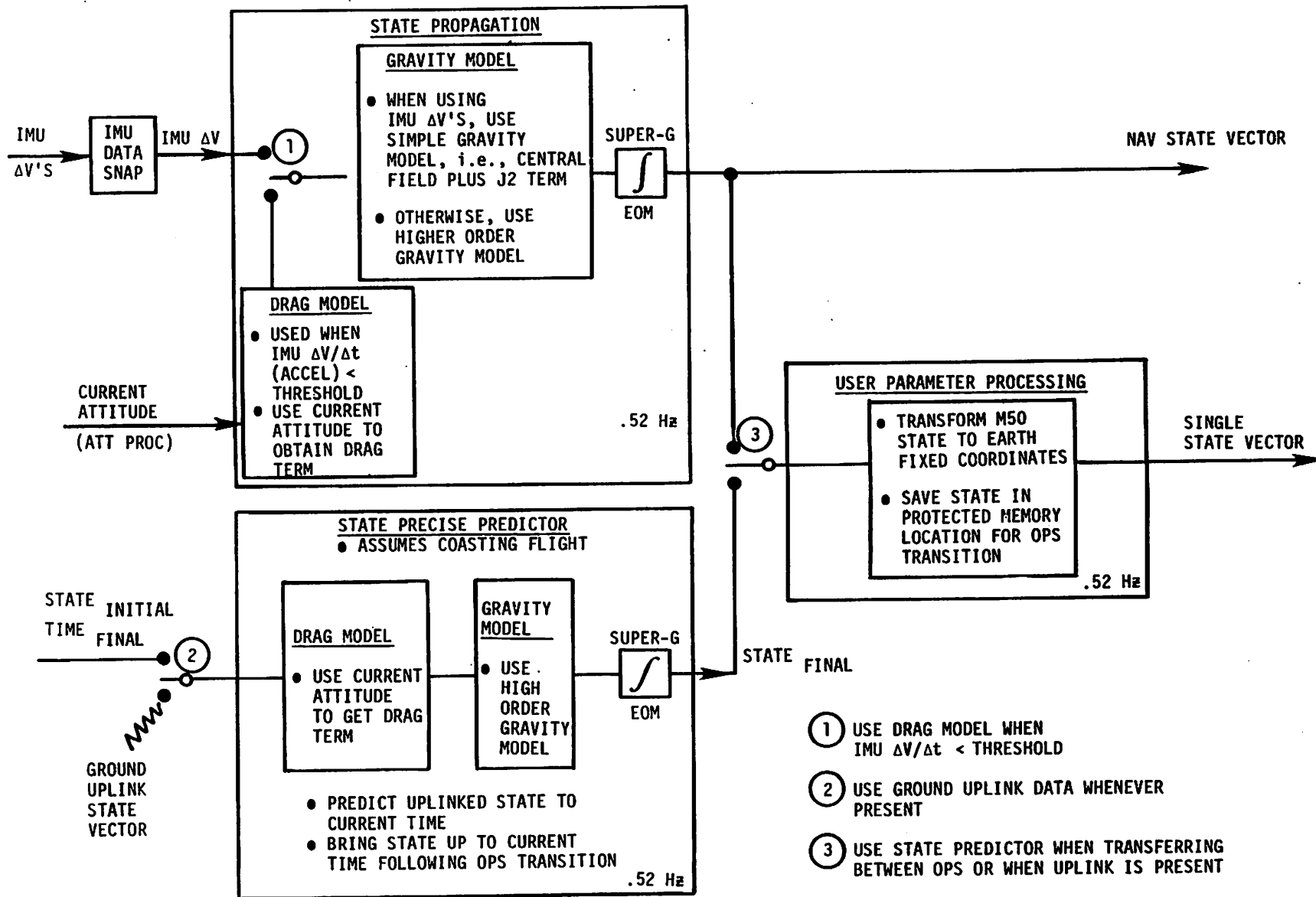


Figure 2.3-2.- On-orbit navigation.

2011 / /

ORBIT

2 000/08:38:15

000/01:21:45

AUTO ATTITUDE MNVR		LVLH/ROT OPT	
INRTL	ATT	BODY	VECTOR
1 R	<u>0.00</u>	5 P	<u>270.00</u>
2 P	180.00	6 Y	0.00
3 Y	0.00	7 0	0.00
MNVR	4*	LVLH	8
		ROT	9
STOP/ATT		HOLD	10

STATE UPDATE
0/ 6:14:37

LAND SITE UPDATE
13 PRI 19 EDW17
14 SEC 20 EDW20
15 ALT 21 EDW22

ATT ERR TOT 11*
DAP 12

VENT DOOR CNTL
OPEN 16*
CLOSE 17
RTC ENA 18
INH 19*

ATT	ROLL	PITCH	YAW
CUR	76.35	0.78	0.00
REQD	0.00	180.00	0.00
ERR	- 76.35	+179.22	+ 0.00
RATE	- 0.200	+ 0.300	+ 0.000

THRUST MONITOR
ΔVX + 0.10
ΔVY - 0.22
ΔVZ + 0.14
ΔVTOT 0.28
ΔV RESET COMP 20
TOT 21

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Figure 2.3-3.- ORBIT display.

2.3.5 Deorbit Navigation

Deorbit/entry navigation maintains current vehicle position and velocity data using SUPER-G with a gravity model and either IMU data or a drag model (similar to OPS 1 and 2), but in three independent state vectors. Entry UPP (similar to ascent UPP) provides this state to users (guidance, etc.). The ascent/entry precise predictor provides prediction capability to users and for updates. Entry navigation has two unique functions: (1) the vector selector, which selects the best components from the three state vectors and combines them into one state, and (2) the covariance matrix propagator which propagates a covariance matrix to be used by OPS 3 navigation when external sensor measurements are incorporated.

Entry navigation and UPP sequencers and control: The sequencers are called once in MM 301 and MM 303 and at 6.25 Hz in MM 302 until deorbit execute navigation subphase is begun.

The navigation sequencer sequences the navigation subphases. The first is deorbit preparation (MM 301 and part of MM 302). At the OPS transition the sequencers initialize the three state vectors and the covariance matrix based on one state vector from previous OPS and an IMU data snap.

The state propagator is run at 0.26 Hz and UPP at 0.52 Hz. The second navigation subphase is deorbit execute (part of MM 302). This subphase begins at 15 seconds before time of ignition. The propagator is run at 0.52 Hz. The UPP rate had been increased to 1.04 Hz a few seconds previously at guidance initiate (TIG-15). The third is entry (MM 303 to preland). The propagator is run at 0.26 Hz. UPP is still at 1.04 Hz.

Vector selector: Entry navigation propagates three independent state vectors to prevent pollution and thereby maintain a good state vector. In the three IMU case, mid-value selection ensures selection of the good IMU. However, in the two IMU case when the IMU ΔV 's are averaged, a borderline IMU would pollute the state. Therefore, separate states are maintained for each IMU so that in the two IMU case when the borderline IMU is finally declared failed, navigation can revert back to the single good IMU (and state) that was preserved.

The vector selector must provide a single state to the user parameter processor for use by guidance and other functions. In the three state case the selector mid-value selects the state component by component (fig. 2.3-4). In the two state case it averages component by component. It should be mentioned that IMU data is incorporated into all the states if the mid-value selected IMU derived acceleration is above the threshold.

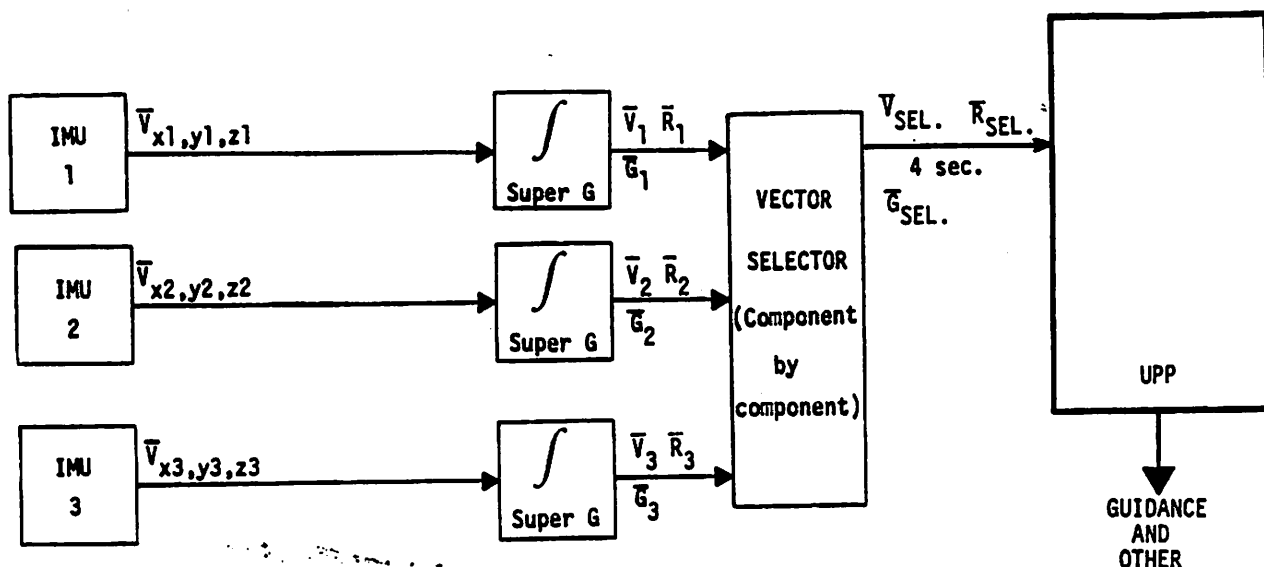


Figure 2.3-4.- Vector selector.

Covariance matrix propagation: The covariance matrix is a statistical estimate of the error in the state vector. It will be used during MM 304 and MM 305 when external measurements are incorporated into navigation. However, it must be propagated throughout OPS 3 to maintain the estimate of the error. It is initialized when a state vector update is made. One matrix is propagated based on the selected state. A 6 x 6 state transition matrix, PHI, consisting of partials of the current state with respect to the previous state, is used to propagate the covariance matrix as in the following equation.

$$\text{Cov Matrix} = (\text{PHI}) (\text{Cov Matrix} (\text{PHI})^T + \text{State Noise})$$

Manual update: Manual update of onboard state may be made by inputting position and velocity errors in runway coordinates through the "HORIZ SIT" display (SPEC 50) (fig. 2.3-5) available in OPS 3 just as in OPS 1. State vector updates, both M50 whole vector (during free fall only) from ground low speed filter and runway deltas from high speed filter, are available via uplink just as in OPS 1.

3051/050/

HORIZ SIT

2 002/54:00:51
000/17:45:51

ALTM

1 29.85

RWY

EDW 17 2*

EDW 22 3

MAN 35 4

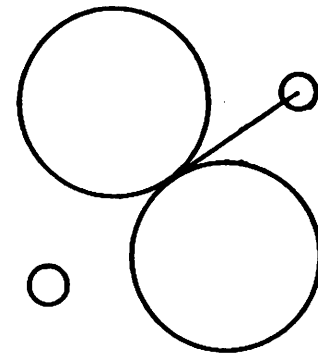
GLD SLOPE

LO WND 5*

HI WND 6

NEP SW 7

NAV DELTA	
A	B
[]	[]
AZ 9	[]
AZ 10	[]
AZ 11	[]
AZ 12	[]
AZ 13	[]
LOAD 14	[]
16 AZ	[]



NAV	RESID	RATIO	AUT	INH	FOR	TAC1 112	TAC2 112	TAC3 112
TAC AZ	+ 2.05	0.3	17*	18	19	+ 96.85	+ 97.23	+ 96.24
RNG	+ 0.04	0.1				+ 71.53	+ 71.43	+ 71.74
						DES 29	DES 30	DES 31
DRAG H	+ 2050	1.1	20*	21	22			
ADTA H			23*	24	25			ABS 32*
ADTA TO G&C			26*	27	28			DELTA 33

2-175

Figure 3.2-5.- HORIZ SIT display.

Runway redesignate: Runway redesignate is the capability to redesignate the target runway for entry from the minitable. The minitable runways are displayed on the HORIZ SIT display and the target runway is initialized as the primary runway (item 2). The tacans of the minitable are associated with the primary runway. If the secondary or alternate runway is redesignated as the target, the tacans will cover the primary geographical location.

2.3.6 Atmospheric Navigation (MM 304 and MM 305)

Overview: Navigation utilizes information from external sensors to estimate the current Orbiter state vector (vehicle position and velocity). The nominal STS-1 groundtrack is depicted in figure 2.3-6. Important events during the entry sequence are indicated on this groundtrack.

2-177

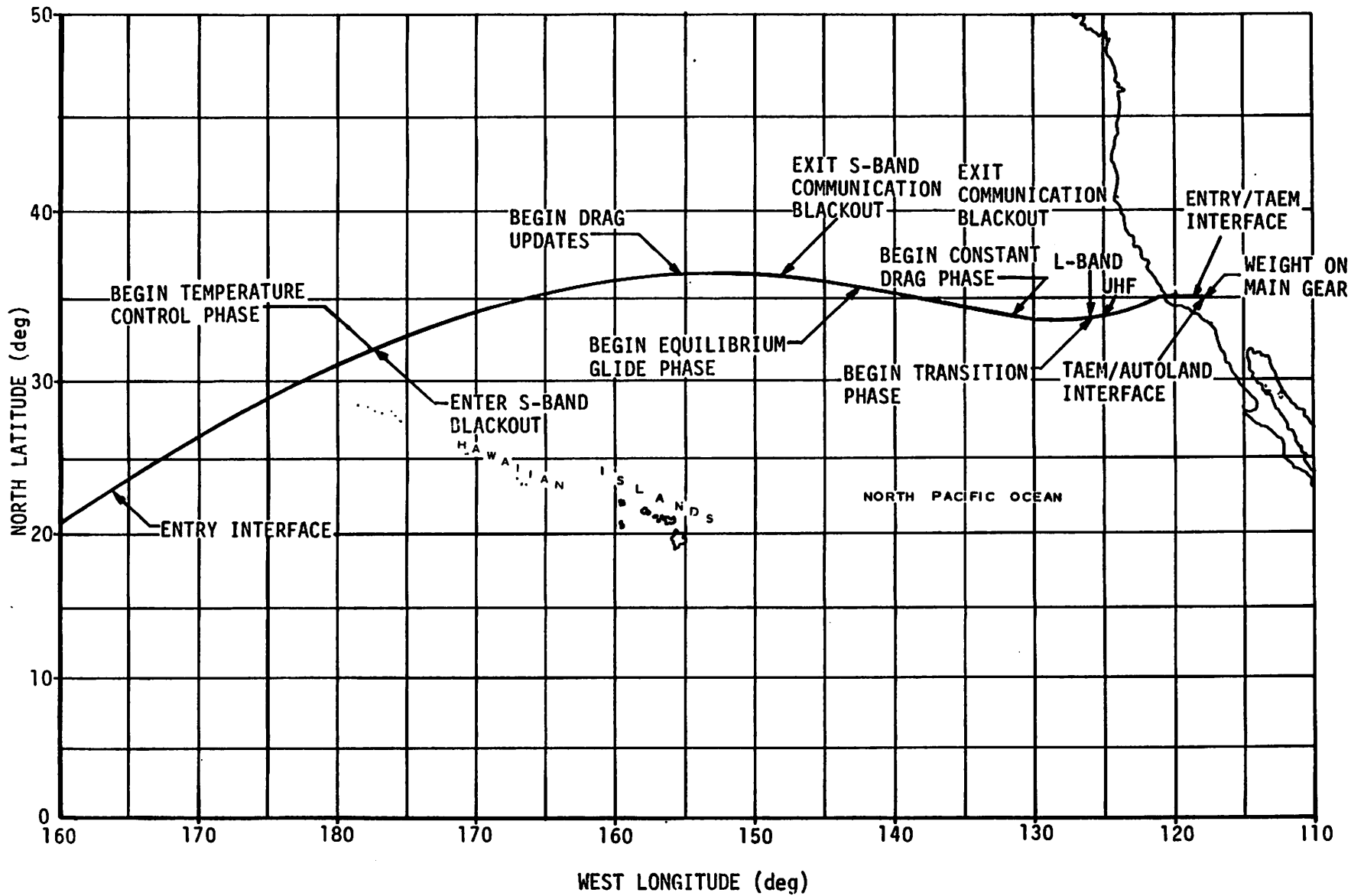


Figure 2.3-6.- STS-1 entry interface through landing groundtrack.

Entry navigation stages: The navigation software from transition to MM 304 to landing consists of two stages.

- The entry navigation state is initiated at OPS transition to OPS 3 and is still active at transition from MM 303 to MM 304. Entry nav calculations continue until acquisition of MSBLS data at approximately 20,000 ft.
- Preland nav is active from MSBLS data acquisition until rollout on the runway. If the Orbiter lands on a runway without MSBLS capability or good MSBLS data does not become available, entry nav will be used until the Orbiter has stopped on the runway.

Characteristics common to both entry and preland nav will be discussed, followed with information unique to each of the two nav stages.

External sensors: External sensors send measurements of the Orbiter's position and velocity to the nav software. The sensors available during the entry and preland phases, the quantities each sensor measures, and the nominal altitude regions where each sensor provides data are listed in table 2.3-1.

The drag altitude is a software device which uses IMU data. The 280,000 ft upper altitude limit nominally corresponds to a sensed drag acceleration level of 11 ft/s^2 . The lower altitude limit is a software limit since the drag altimeter's expected accuracy is poor below 85,000 ft.

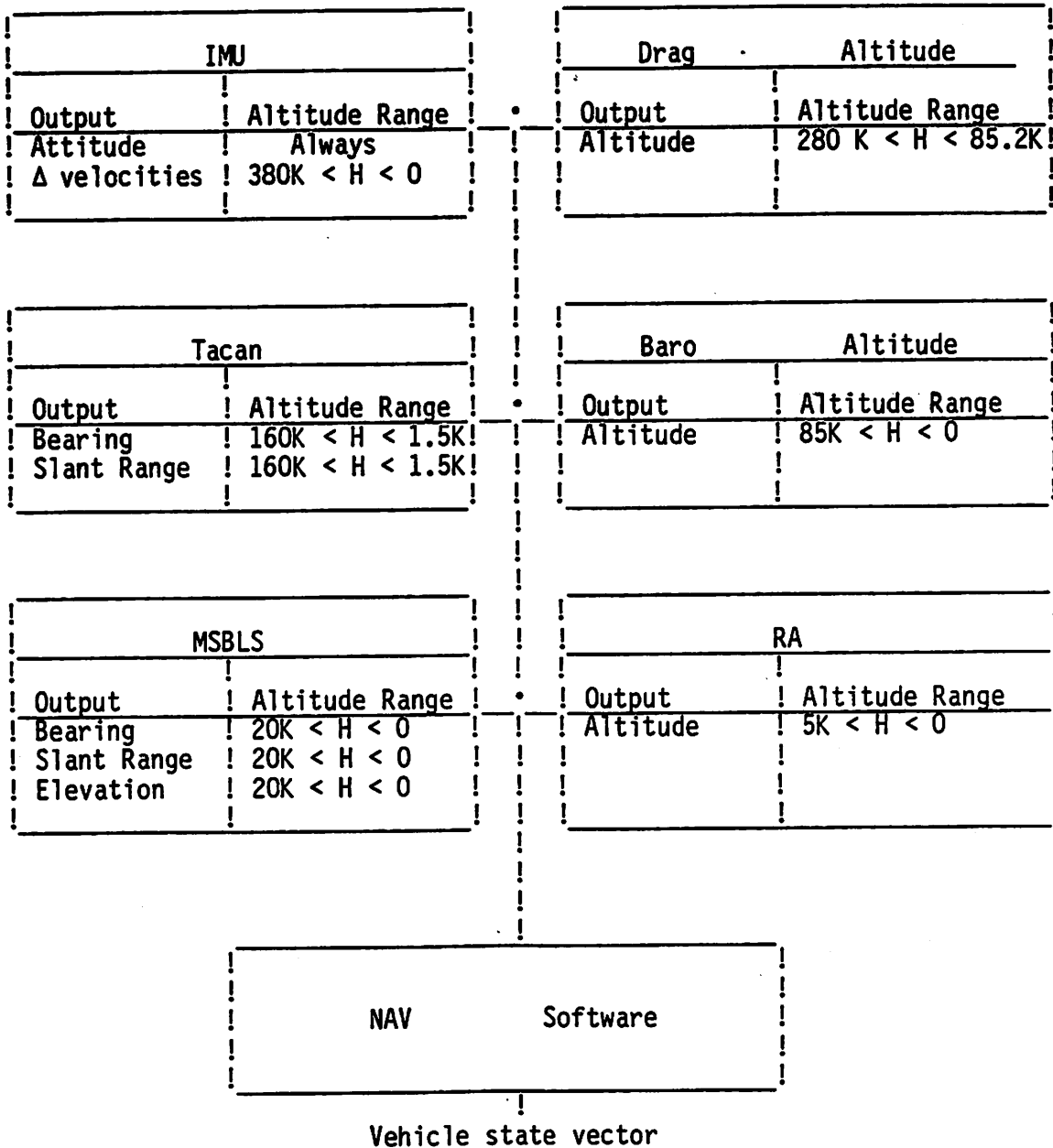
The drag altitude uses the relationship between drag and air density to estimate altitude. This type of measurement contains errors because the atmosphere is difficult to model, especially at lower altitudes, but it provides information available during blackout ($265,000 \text{ ft} < H < 162,000 \text{ ft}$). Without drag altitude updates, errors grow exponentially with time. With updates, altitude and downrange errors are bounded, but crossrange errors still grow. However, crossrange errors are not critical in the entry flight phase. The 3 sigma altitude error for drag altimeter measurements is approximately two miles.

The upper tacan altitude limit of 160,000 ft is an approximate limit corresponding to two tacan LRU's locked on and a data good flag from tacan RM. The lower altitude limit of 1.56 ft. is a software limit to prevent incorporation of data which could be contaminated with ground signal reflections.

The upper MSBLS limit is an approximate limit corresponding to the first good data from MSBLS RM.

The radar altimeter upper altitude limit is the approximate altitude where data will begin appearing on the AVVI. Radar altitude data is never used in the navigation software. However, below an altitude of about 180 ft, corresponding to a horizontal distance of approximately 4000 ft from the runway threshold, the radar altimeter data is incorporated into UPP.

TABLE 2.3-1.- ENTRY NAVIGATION EXTERNAL SENSORS



*All altitudes are feet

Navigation overview summary: Entry navigation software performs the following calculations.

- Stores current sensed velocity from the IMU's and current sensor data from all available sensors.
- Propagates the state vector to current time with SUPER-G integration scheme.
- Propagates the covariance matrix to current time.
- Sets flags indicating which data will be processed during current cycle.
- Computes the residual for each measurement processed.
- Calculates the residual edit test threshold and compares this to the residual for each measurement processed.
- Updates the state vector and covariance matrix for each measurement processed.
- Stores the state vector and gravitational vector for UPP.
- Calculates measurement processing statistics for the Horizontal Situation Display (HSD).

Entry Navigation

- **Entry Nav Characteristics** - Entry nav is initialized at EI-5 minutes when MM 304 is called by the crew and remains active through Microwave Landing System (MLS) lock-on in MM 305. Three state vectors and one covariance matrix are calculated during the entry nav phase. External sensors active during this phase are IMU's, tacans, drag altitude, and baro altimeter.
- **Entry Nav Data Flow** - The covariance matrix is a statistical estimate of the "cloud of uncertainty" about the state vector. In simple terms, the covariance matrix is a band of uncertainty around the state vector that is used to determine the quality of the external sensor data. If the external sensor data falls within this band, the data is declared good and is used to update the state vector. However, if the sensor data falls outside the band of uncertainty, the data is declared bad, edited out, and not used to update the state vector. Each time external sensor data is used, navigation assumes that the state vector is becoming more accurate; therefore, it begins to shrink the band of uncertainty around the state vector. The nav software utilizes a premission calculated estimate of the IMU and Equations of Motion (EOM) to calculate the covariance matrix each cycle. The covariance matrix for a six element state vector is a 6 x 6 matrix.

Three sensed velocity measurements, one from each IMU, are sent to the nav software during the entry nav phase for fail operational/fail safe redundancy. These sensed velocities do not account for gravitational acceleration. The three sensed velocities should be similar, with slight differences due to instrument and alignment errors.

A SUPER-G scheme is utilized to integrate the EOM forward. This SUPER-G scheme accounts for the difference in acceleration over the measurement interval for a given computation cycle.

Before incorporating external sensor data, a vector selector creates a single state vector from the three available IMU's. This single state vector is mid-value selected on a component basis. Mid-value selecting is employed instead of averaging because IMU failures tend to be axially dependent.

Position, velocity, and gravity in M50 coordinates are the only parameters output by the nav software. UPP provides the interface between navigation and guidance. UPP takes the state vector and gravitational acceleration output by nav and drives the input parameters required by guidance, displays, and other software users. Because entry nav software runs once every 4 seconds, UPP calculates a new state vector every 0.16 seconds using a simpler integrator than the SUPER-G algorithm, which is used in the entry nav software. The UPP state vector integrator is reset each time the nav software outputs a nav state vector, eliminating any divergences between the UPP and nav state vector. To improve the accuracy of the UPP integrator, sensed velocities are sent from the IMU's every 0.16 seconds.

If the FDIR software detects an IMU failure, it sets a flag which is sent in both the vector selector and velocity selection filter.

The interaction between the IMU's, nav software, UPP, and FDIR is shown in figure 2.3-7.

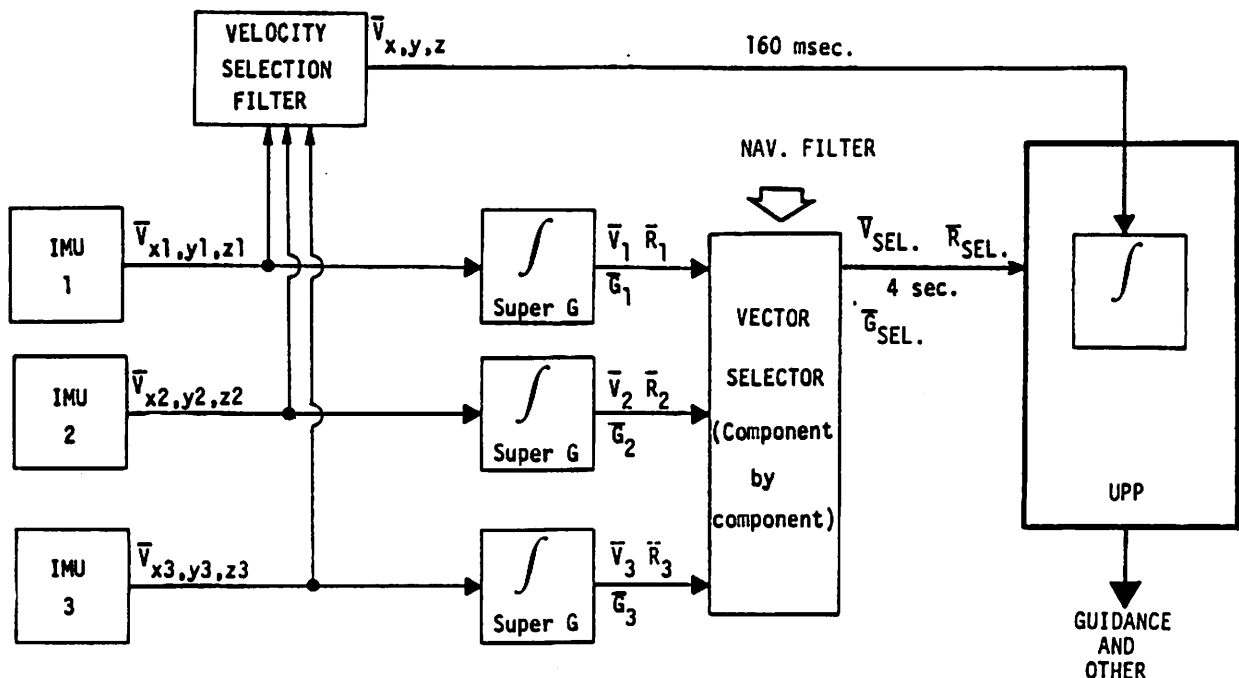


Figure 2.3-7.- Entry navigation data flow.

- State Vector Update - When external sensors are available, the block labelled "Vector Selector" in figure 2.3-4 becomes a Kalman filter. The Kalman filter combines the state vector calculated with the SUPER-G algorithm with the external sensor measurements of what the state vector should be to get an updated state vector. The updated state vector is more accurate than either the SUPER-G state vector (commonly called the propagated state vector) or the measurement alone.

The operation of the nav software when data from external sensors are incorporated is depicted schematically in figure 2.3-8.

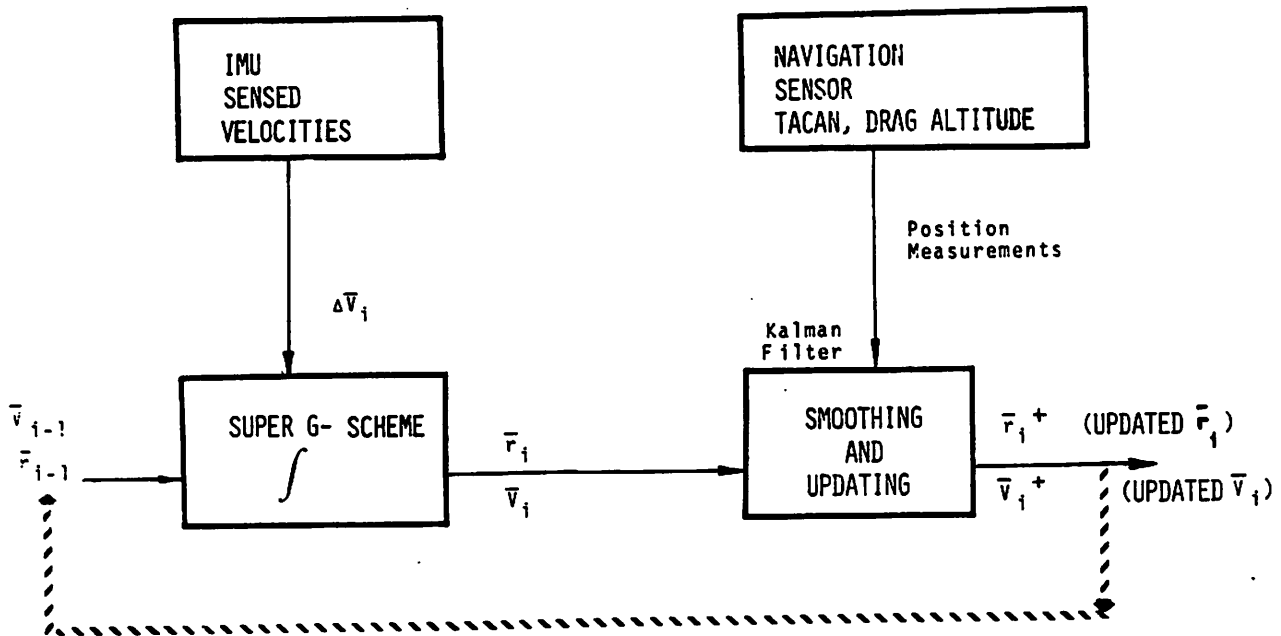


Figure 2.3-8.- State vector update.

- HSD Measurement Statistics - Information on the Kalman filter software is shown on the HSD. Refer to figure 2.3-5.

The Residual (RESID) column provides the difference between the sensor measurement and what the sensor measurement would be if the Orbiter were located exactly at the propagated state vector position and velocity. This column is full of zeros when the model EOM represents the Orbiter exactly and the external sensors contain no instrument errors.

The RATIO column displays the ratio of the residual to the residual test threshold. This threshold is computed each nav cycle and is used to determine if the residual is reasonable. Data labelled good by redundancy management yield a reasonable residual less than one. When the ratio is greater than one, the measurement data has failed the edit test and is assumed to be bad.

A "↓" appears below the "0" in RATIO next to the number in the RATIO column when three out of four of the last measurements from that sensor fail the edit test.

- AUTO/INHIBIT/FORCE (AIF) Switch - The AIF switch controls the usage of the residual edit test in deciding whether or not to use the measurement data to update the state vector. This software switch is controlled by the execution of the proper item number for the various external sensors. The AIF switch has the capability to override the residual edit test, as shown in the following table.

TABLE 2.3-2.- AUTO/INHIBIT/FORCE SWITCH

AIF switch position	Data passed edit test?	Use data to Update state vector ?*
AUTO	Yes	Yes
	No	No
INHIBIT	Yes	No
	No	No
FORCE	Yes	Yes
	No	Yes

* Assuming the data is labelled good by RM.

With the AIF switch in the INHIBIT position, the state vector sent to UPP is the propagated state vector, regardless of whether or not the residual passed fails the edit test. When the AIF switch is in the FORCE position, the state vector will always be updated, even if the residual fails the edit test.

The state vector update scheme for the cases of INHIBIT and FORCE AIF switch positions are depicted in figures 2.3-9 and 2.3-10. The auto scheme has already been discussed and is shown in figure 2.3-8.

- HSD Tacan Data - Raw tacan data is also shown on the HSD. When the absolute (ABS) is selected, the three columns of numbers on the lower right-hand corner of the HSD give the raw azimuth and range measurements in nautical miles and degrees for each of the three tacan LRU's. With the delta mode selected, these columns display the difference between the residual and each of the LRU's.
- HSD Manual State Vector Update - Mission Control Center (MCC) has the capability to calculate the Orbiter state vector independently from the Orbiter. The crew can enter a manual update to the onboard state vector, as voiced from MCC and based on ground calculations.

Items 8 through 13 on the HSD are utilized to enter deltas to the onboard state vector. These deltas, in runway coordinates, are added to the onboard state vector without a time tag since state vector error changes slowly. Execution of item 14 sends the loaded corrections

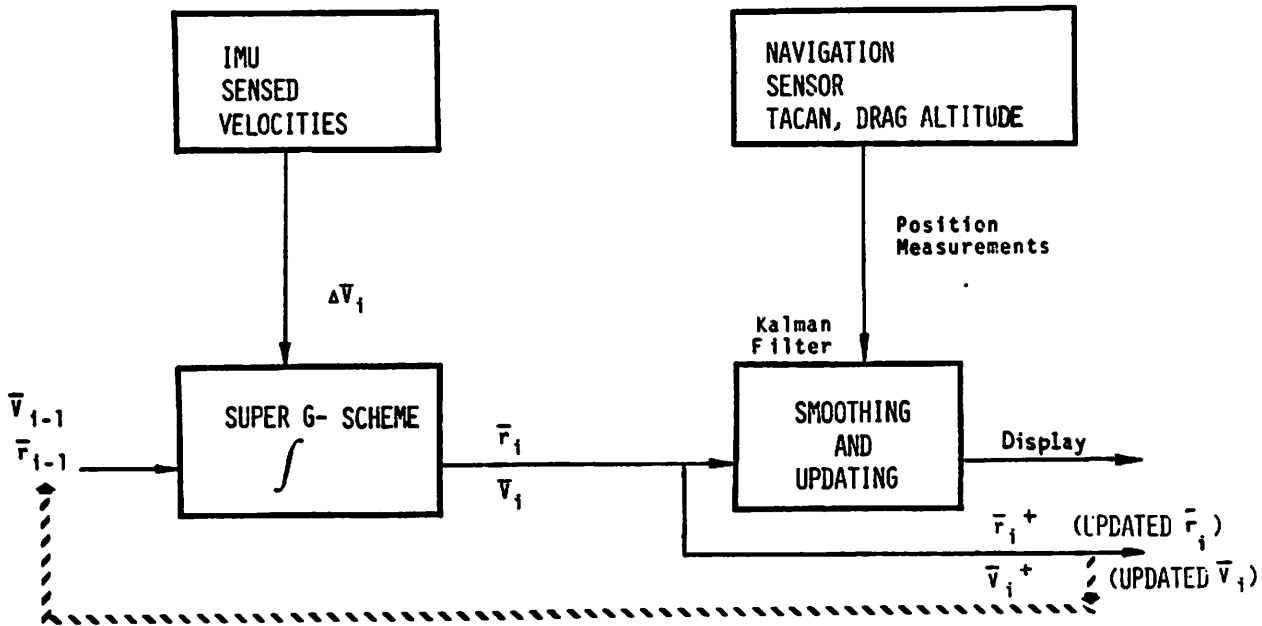


Figure 2.3-9.- State vector update (inhibit).

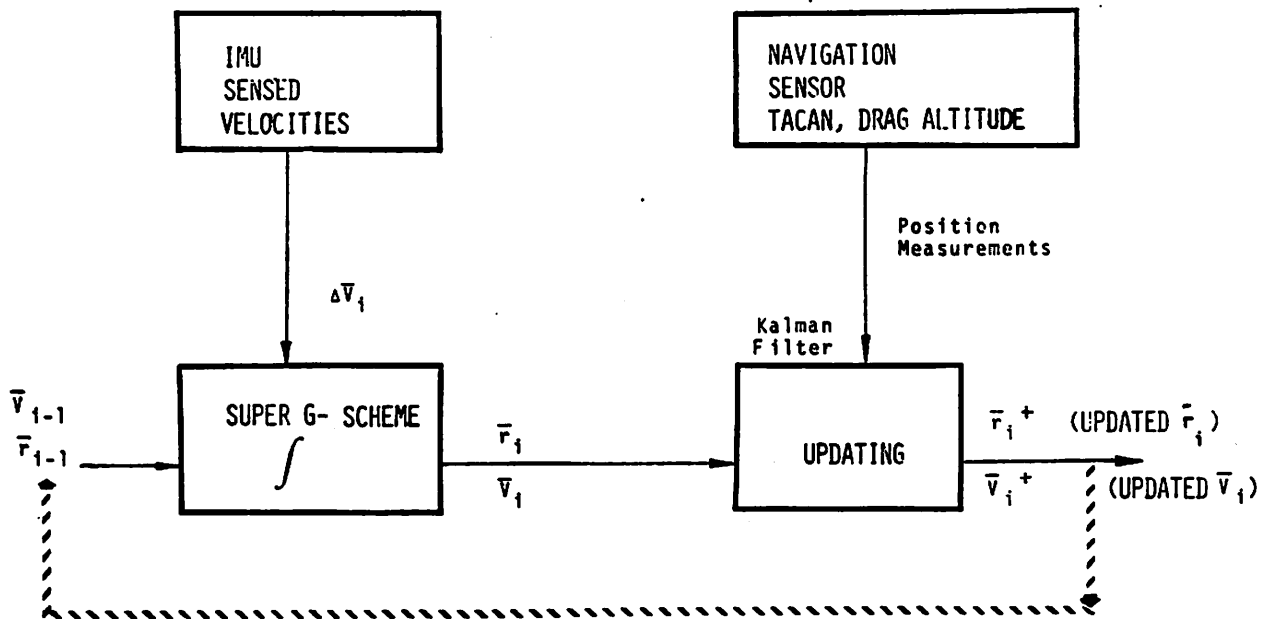


Figure 2.3-10.- State vector update (force).

to the nav software and reinitializes the covariance matrix to an I-loaded set of numbers corresponding to the accuracy of the ground calculations.

Since reinitializing the covariance matrix usually makes the edit test threshold larger, a zero delta update can be performed if MCC thinks the threshold is failing good data.

The HSD outputs the raw tacan data during the preland nav phase identically as during the entry nav phase.

- Nav Event Light - The rightmost of the five event lights on the eyebrow panels flashes when there is an unexpected absence of altitude, cross-range, or downrange data on two consecutive computation cycles. The lights will flash until the missing data are restored. Detailed criteria for nav event light flashing are listed below:
 - The light is off above an altitude of 140,000 ft when good tacan data may be unavailable.
 - Between 140,000 and 120,000 ft, the nav light will flash if any of the following conditions are met:
 - No tacan range update has occurred.
 - No tacan bearing update has occurred, unless the Orbiter is in the cone-of-confusion.
 - Below 12,000 ft the nav light flashes if there is no MSBLS range, azimuth, or elevation, unless there is no MSBLS capability available at the selected runway, in which case the light logic reverts to tacan case.
 - Below 500 ft, the nav event light will flash if radar altimeter data are missing for two consecutive cycles.

Preland Navigation

- Preland Nav Characteristics - Preland nav is automatically initialized when entry nav receives a data good flag from MSBLS RM. This nominally occurs at an altitude of approximately 20,000 ft during MM 305. Once preland is initiated, it remains active through Orbiter rollout on the runway. If the Orbiter lands at a contingency site with no MSBLS capability, entry nav remains active through rollout on the runway.
- Preland Nav Data Flow - Since accurate data is very important near the ground, preland nav cycles every 2 seconds instead of every 4 seconds as in entry nav. With the shorter cycle time, fewer calculations can be performed in preland nav. Also, MSBLS data is so accurate that a multi-state vector concept is unnecessary.

Preland nav uses the same basic equations as entry nav. The output of the three IMU's are combined to provide a single state vector instead of three separate state vectors. The velocity selection filter mid-value selects the sensed velocity readings on a component basis. Otherwise, the preland nav data flow is essentially the same as the entry nav data flow.

The preland nav (single state vector) data flow concept is presented in figure 2.3-11.

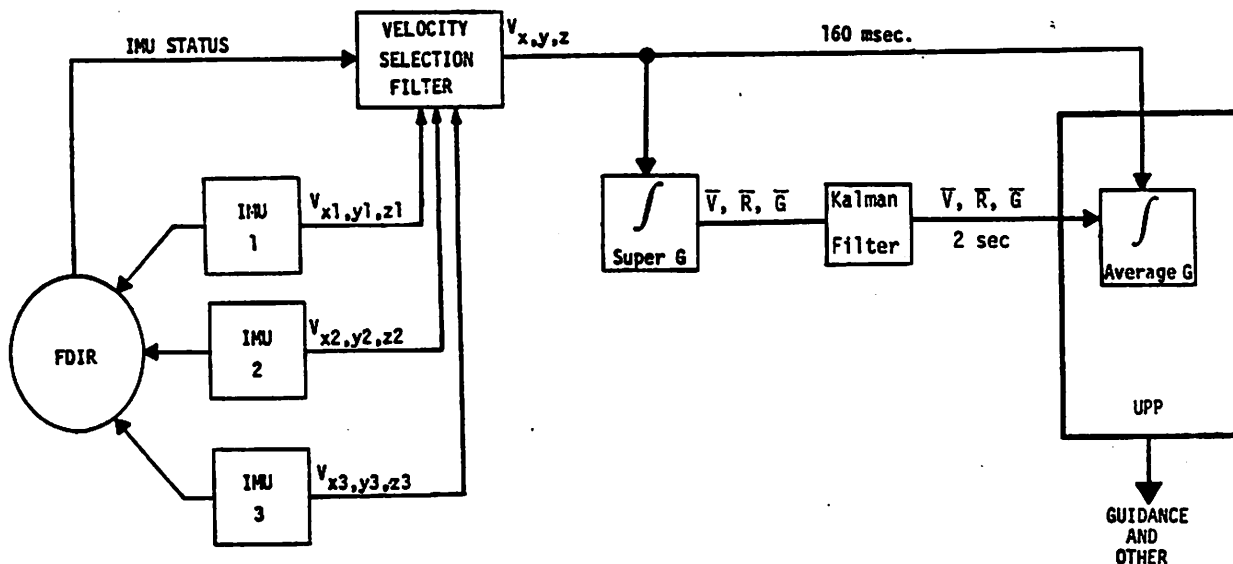


Figure 2.3-11.- Preland navigation (one state vector concept).

BFS Variations From the Primary Navigation System

The capability of the BFS entry navigation function will not include processing for automatic landing. Thus, the MSBLS will not be employed by the BFS entry navigation function, nor will radar altimeter data be utilized by the BFS. Other variations are the redundancy management related tasks (vector selection, three state to one state, and mid-select).

2.3.7 RTLS Navigation

Navigation during the glide phase of an RTLS is called "RTLS Navigation" and is not the same as the navigation used during the powered portion of an RTLS abort. Powered RTLS nav is the same as mainstage nav. RTLS nav is initiated on transition to MM 602. Significant differences are that along with IMU data, external sensor data is now also processed to update the state vector. A Kalman filtering technique compares external data to propagated data and a new, more refined state vector is created. Drag acceleration is used to calculate pseudo-altitude measurements. The only difference between RTLS nav and entry nav is that only one state vector is propagated as opposed to three during nominal entry. Preland navigation is the same as RTLS nav and no transition to preland nav takes place. External sensor availability differs slightly from the nominal case. Tacan is available at about 225,000 ft and drag altitude at 11

ft/sec at $\approx 180,000$ ft. Baro alt is available at about 80,000 ft but is inhibited during the mach jump region 1.6 to 0.9 due to uncertainties of air flow. MSBLS is available at 20,000 ft. The edit controls work identically to nominal entry navigation.

RTLS BFS navigation: There is no difference between PASS and BFS navigation during the powered portion of an RTLS. Up until MM 602, the ascent navigation is utilized whether in PASS or BFS. Starting at MM 602, "descent navigation" is used. Significant differences here are that descent navigation cannot process MSBLS which means no autoland capability exists. Also radar altimeter cannot be displayed, although software still uses it. One state vector is still propagated which is no different from RTLS navigation in PASS. Updating the state vector can come from one of three places: uplink, HORIZ SIT display, or by transfer from PASS. Aside from the above mentioned differences, navigation in the BFS or PASS mode are the same. During the A/L, descent nav continues propagating one state vector all the way to the ground. Also BFS nav during Terminal Area Energy Management (TAEM) no longer uses the drag model.

2.4 GUIDANCE SOFTWARE

The ascent guidance principle function is subdivided into the phases of first stage, second stage, and orbit insertion. The target for mainstage guidance is an I-loaded set of MECO conditions. The target for orbit insertion is an I-loaded set of apogee and perigee conditions.

2.4.1 First Stage Guidance

First stage guidance is active from SRB ignition until SRB SEP command +4.0 sec. Guidance in this stage consists of an I-loaded table of pre-determined roll, pitch, and yaw attitudes referenced to relative velocity. (There are I-loaded tables for the cases of three SSME's firing, left SSME or right SSME out, and center SSME out.) In the event of a main engine failure, the attitude profile is such that the trajectory is lofted.

Examples of the I-loaded attitude profile are shown in figures 2.4-1, 2.4-2, and 2.4-3. The attitude remains constant for approximately the first 10 sec so that the tower is cleared.

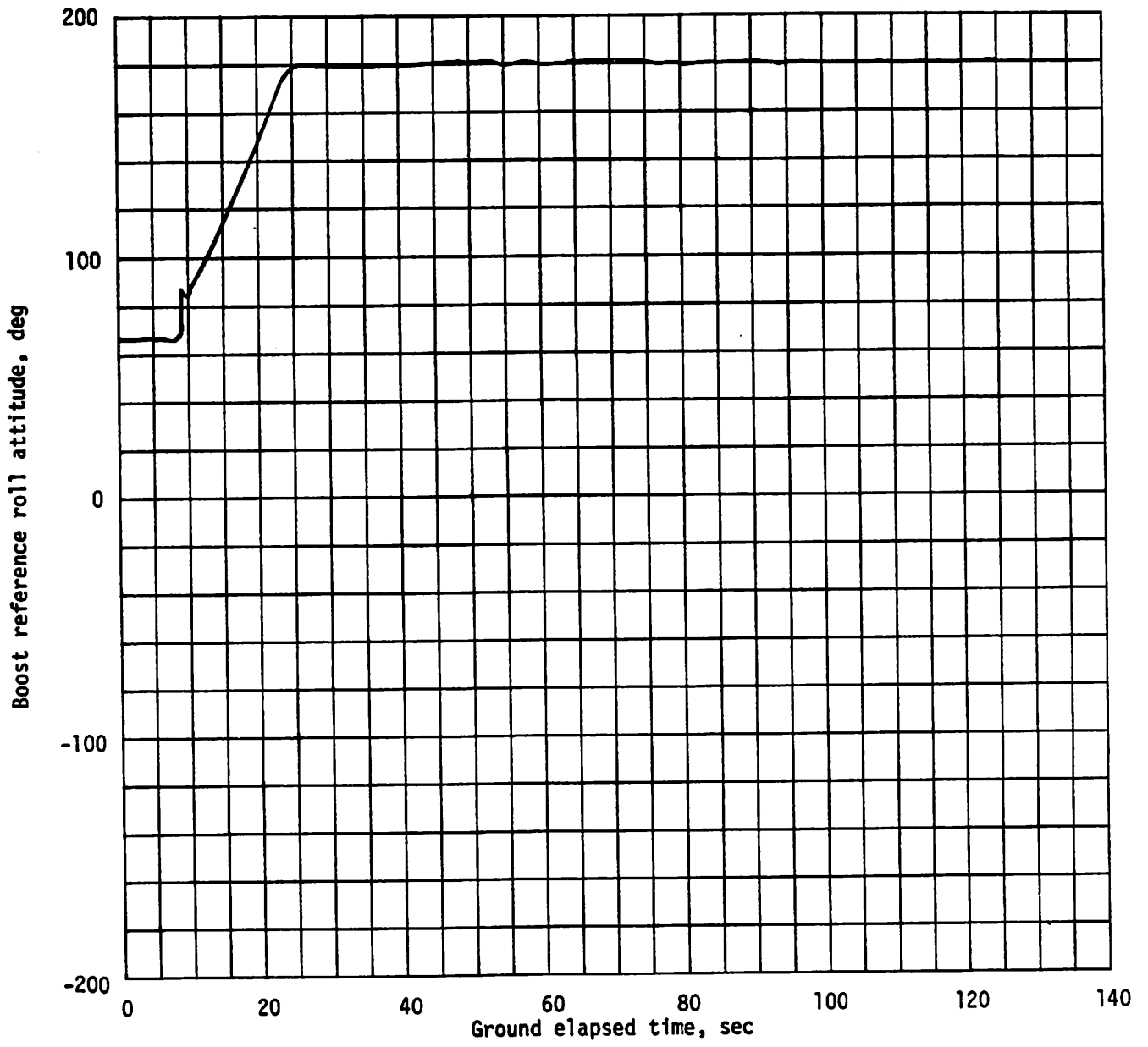


Figure 2.4-1.- Roll profile (first stage).

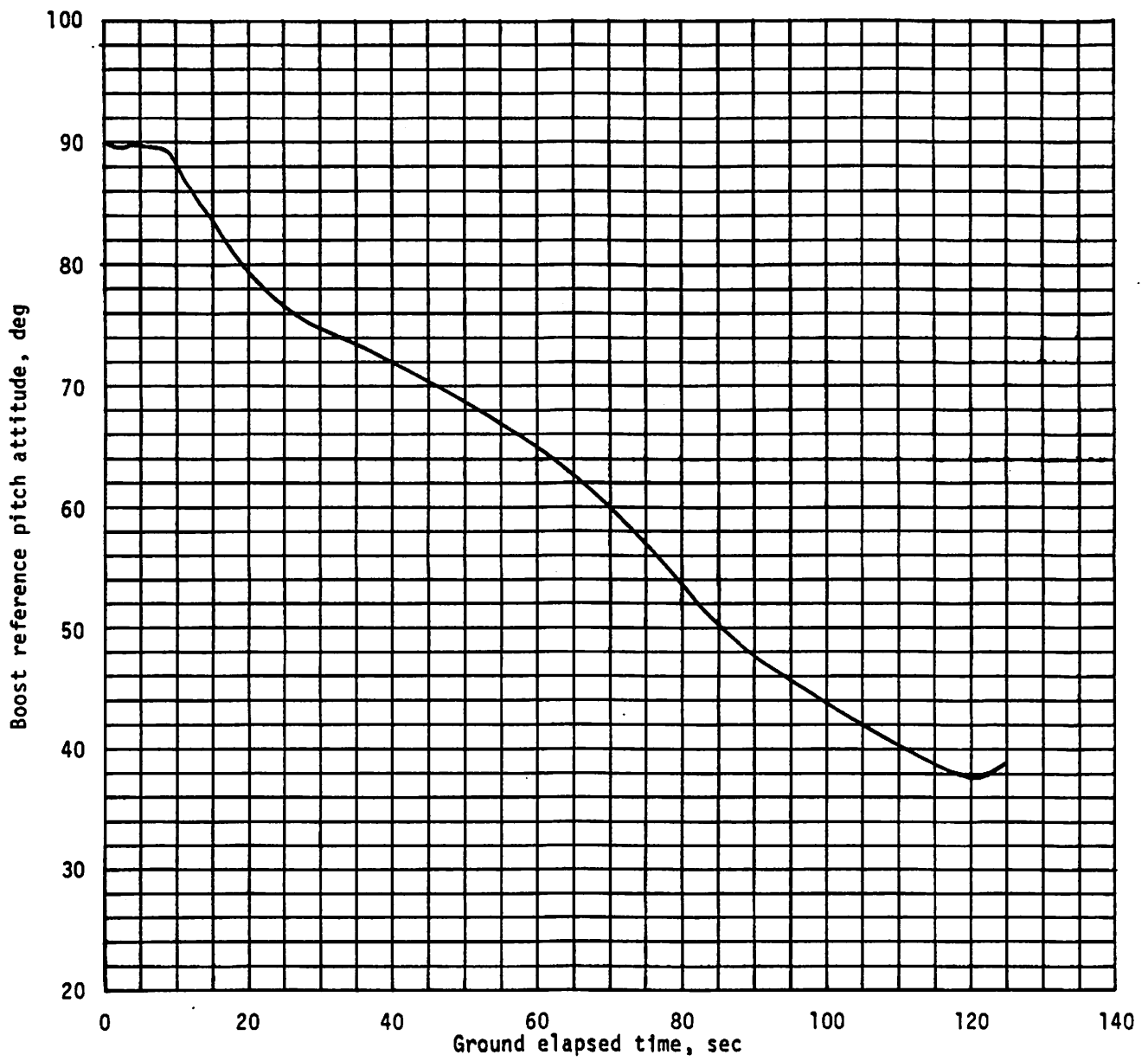


Figure 2.4-2.- Pitch profile (first stage).

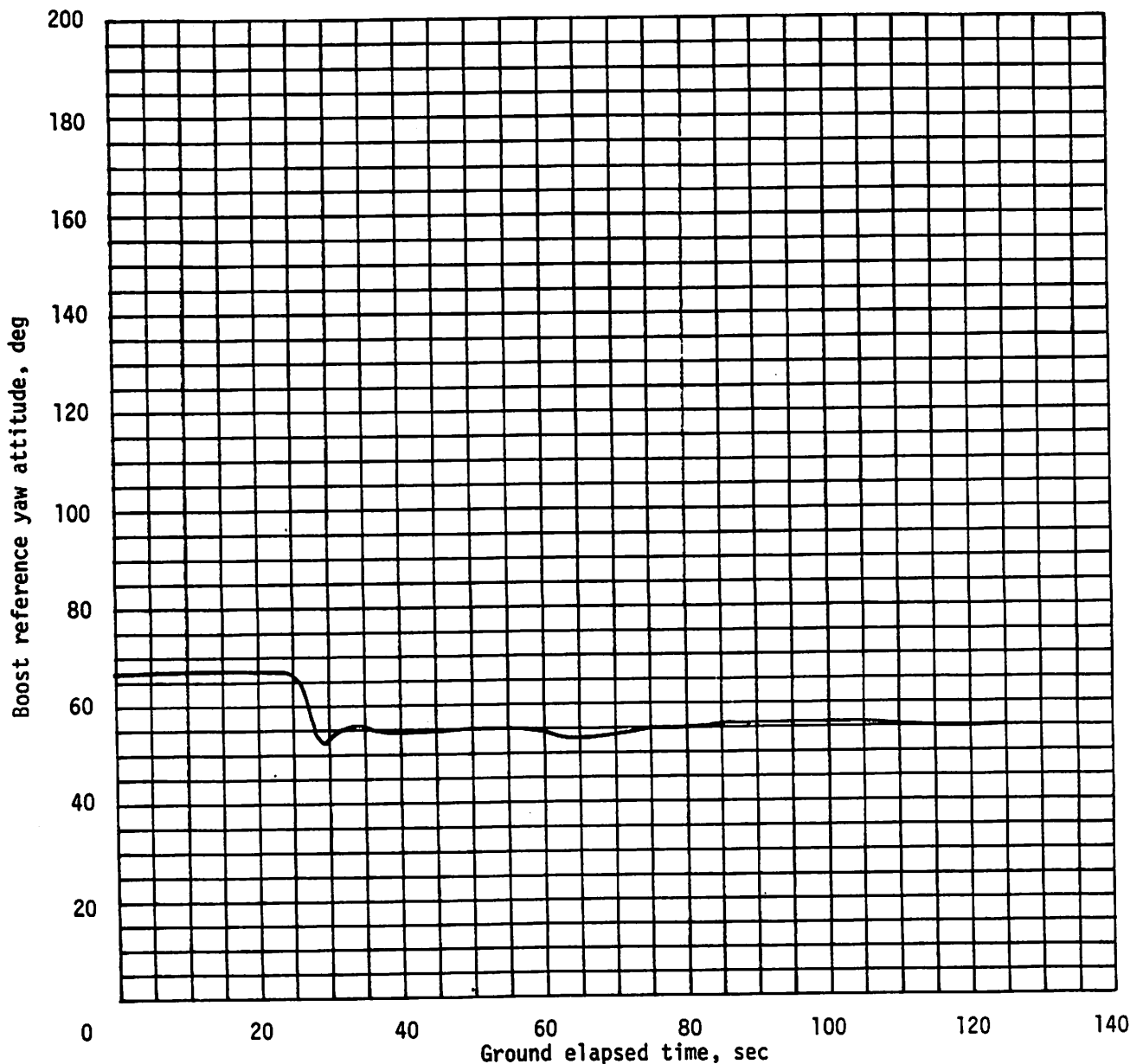


Figure 2.4-3.- Yaw profile (first stage).

When the SRB SEP command is received, a three-axis attitude hold is commanded for 4 sec. Second stage guidance is initiated for convergence at SRB SEP command. Following the 4 sec attitude hold, second stage guidance is engaged.

Guidance also commands the MPS throttle. A typical throttle profile is shown in figure 2.4-4.

Only the PLT may assume manual throttle via the SBTC. In the event of manual throttle takeover, the auto throttle commands are inhibited.

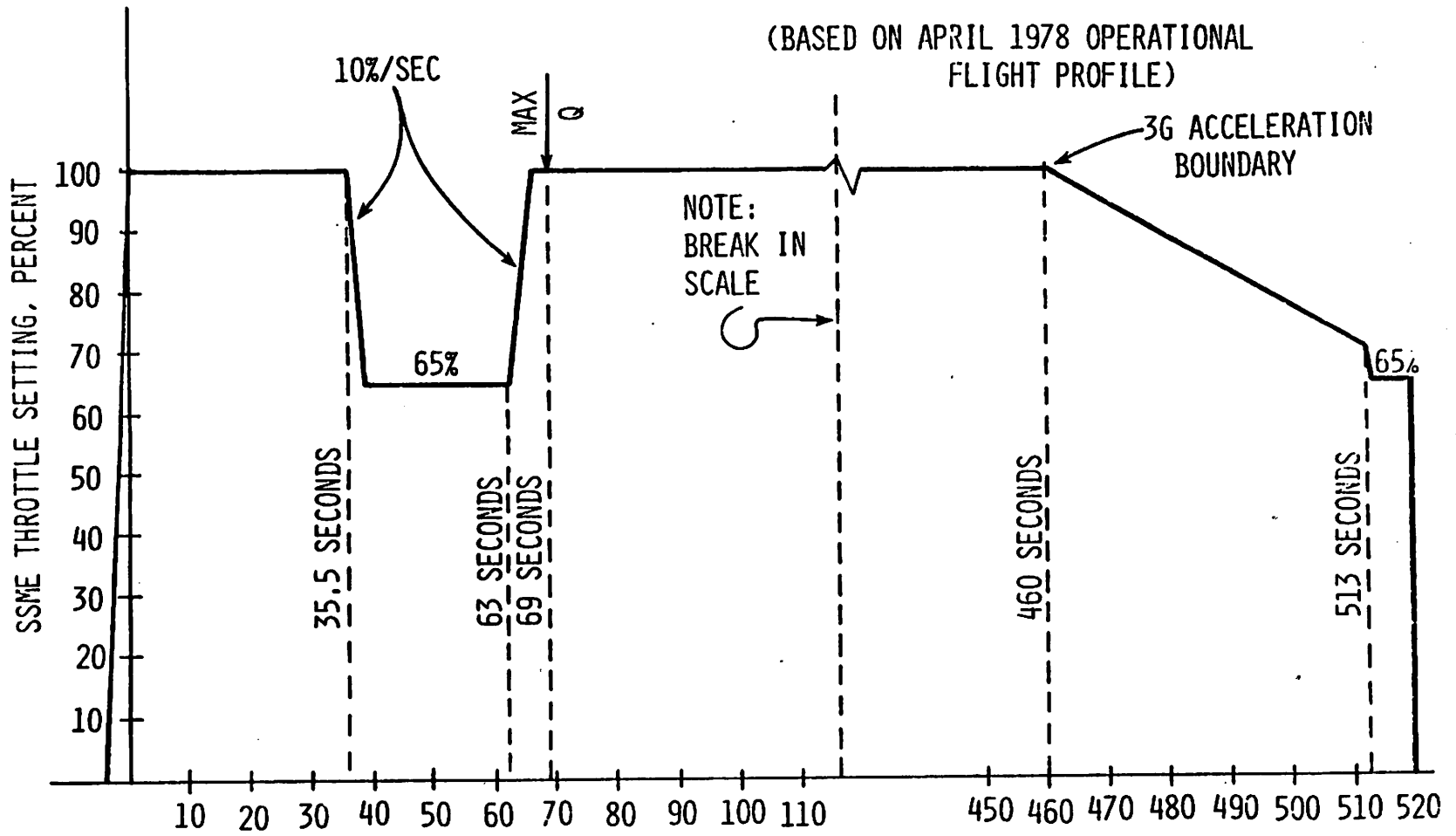


Figure 2.4-4.- Throttle profile.

2.4.2 Second-Stage Guidance

The second-stage guidance routine known as Powered Explicit Guidance (PEG 1) is engaged 4 sec after SRB SEP command in order to guide the mated vehicle to the desired MECO conditions.

The MECO conditions are:

V \simeq velocity

R \simeq radius from earth center

γ \simeq flightpath angle

i \simeq desired orbital plane (inclination)

These conditions define the target for which guidance aims.

A representation of these parameters is shown in figure 2.4-5.

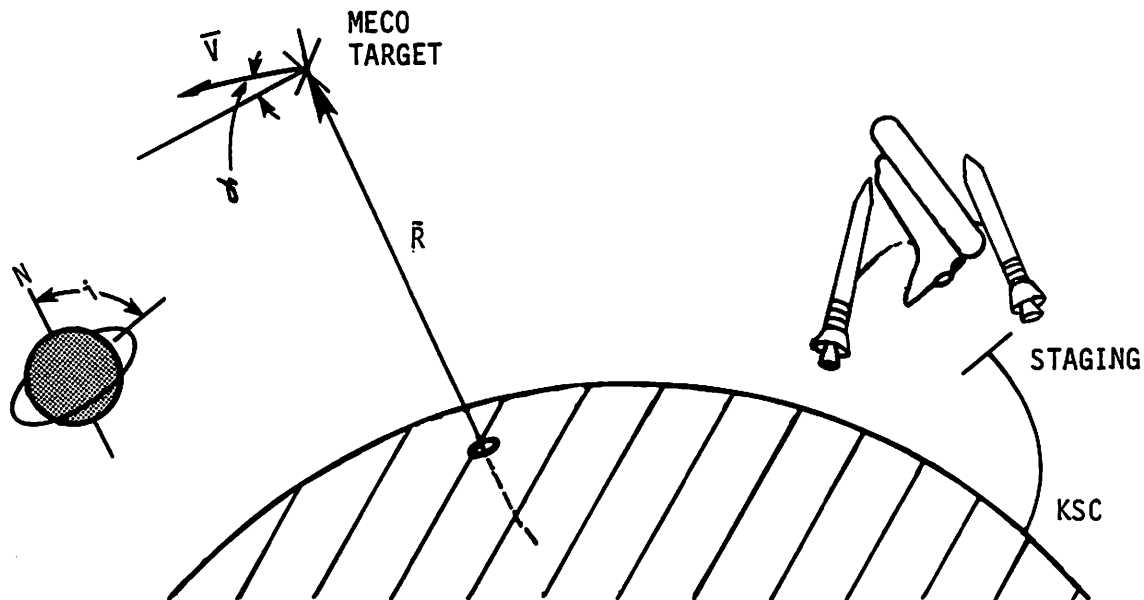


Figure 2.4-5.- The MECO target.

Second stage guidance is terminated at MECO command. Attitude profiles for second stage are shown in figures 2.4-6, 2.4-7, and 2.4-8.

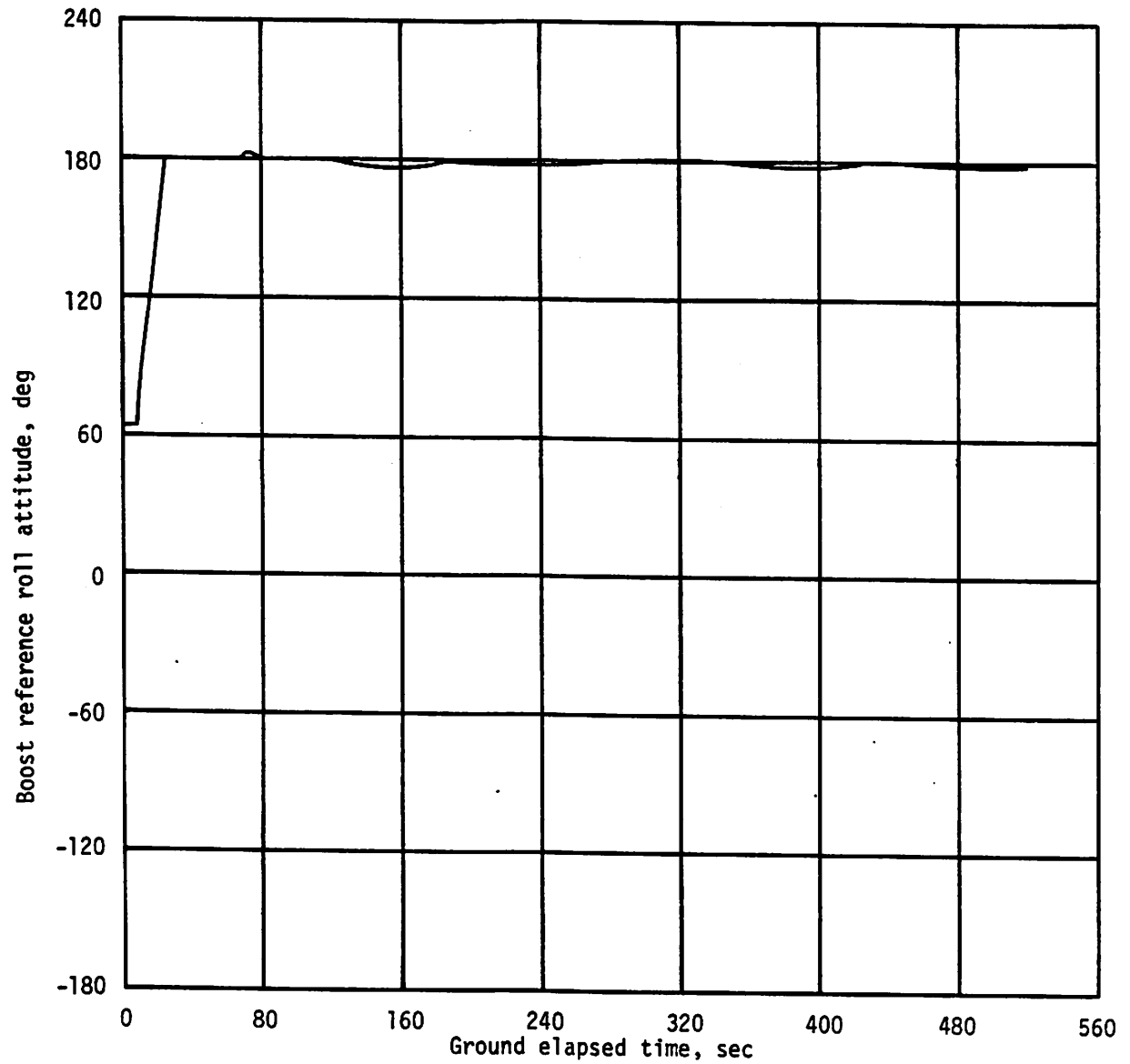


Figure 2.4-6.- Roll profile (second stage).

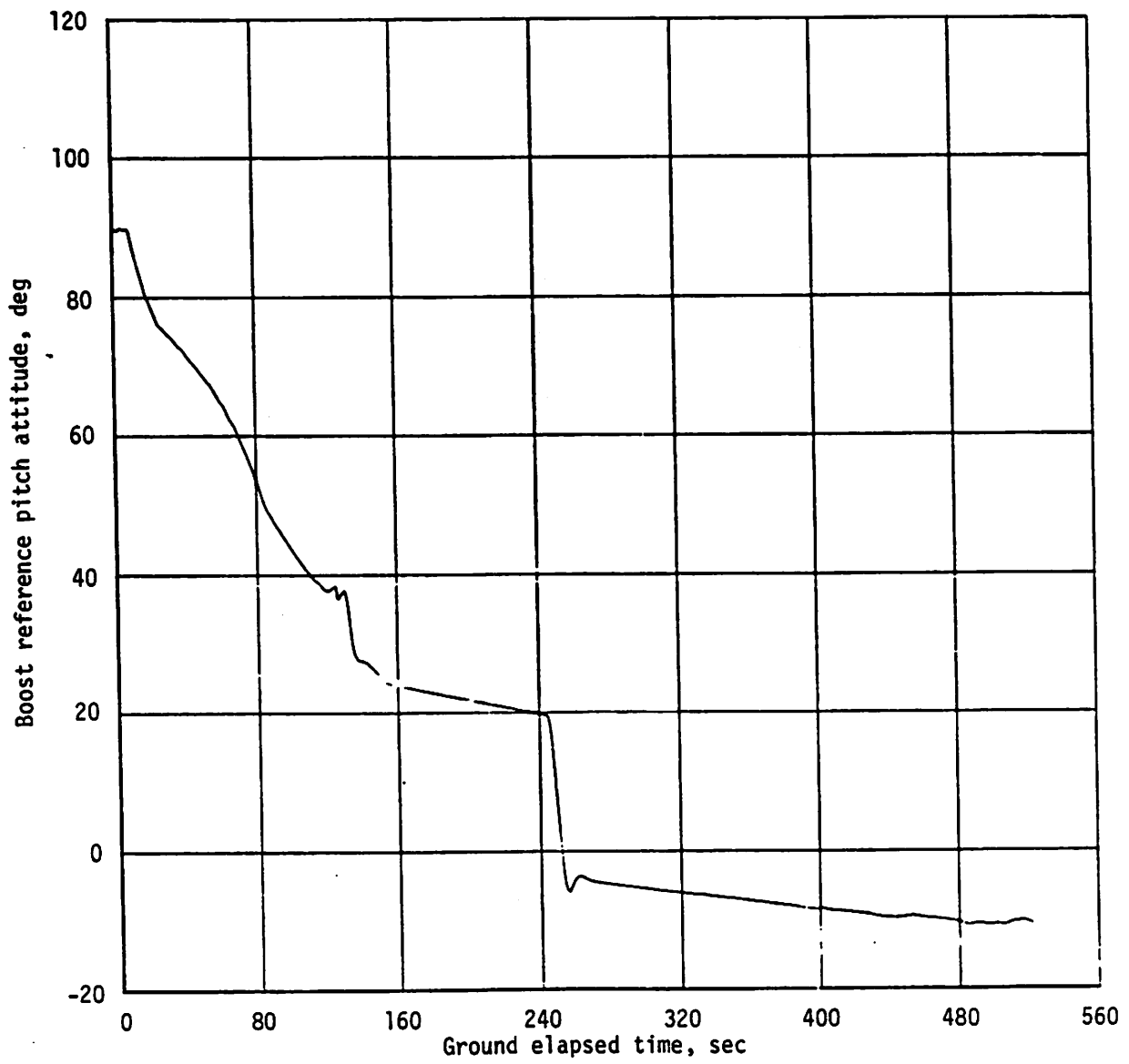


Figure 2.4-7.- Pitch profile (second stage).

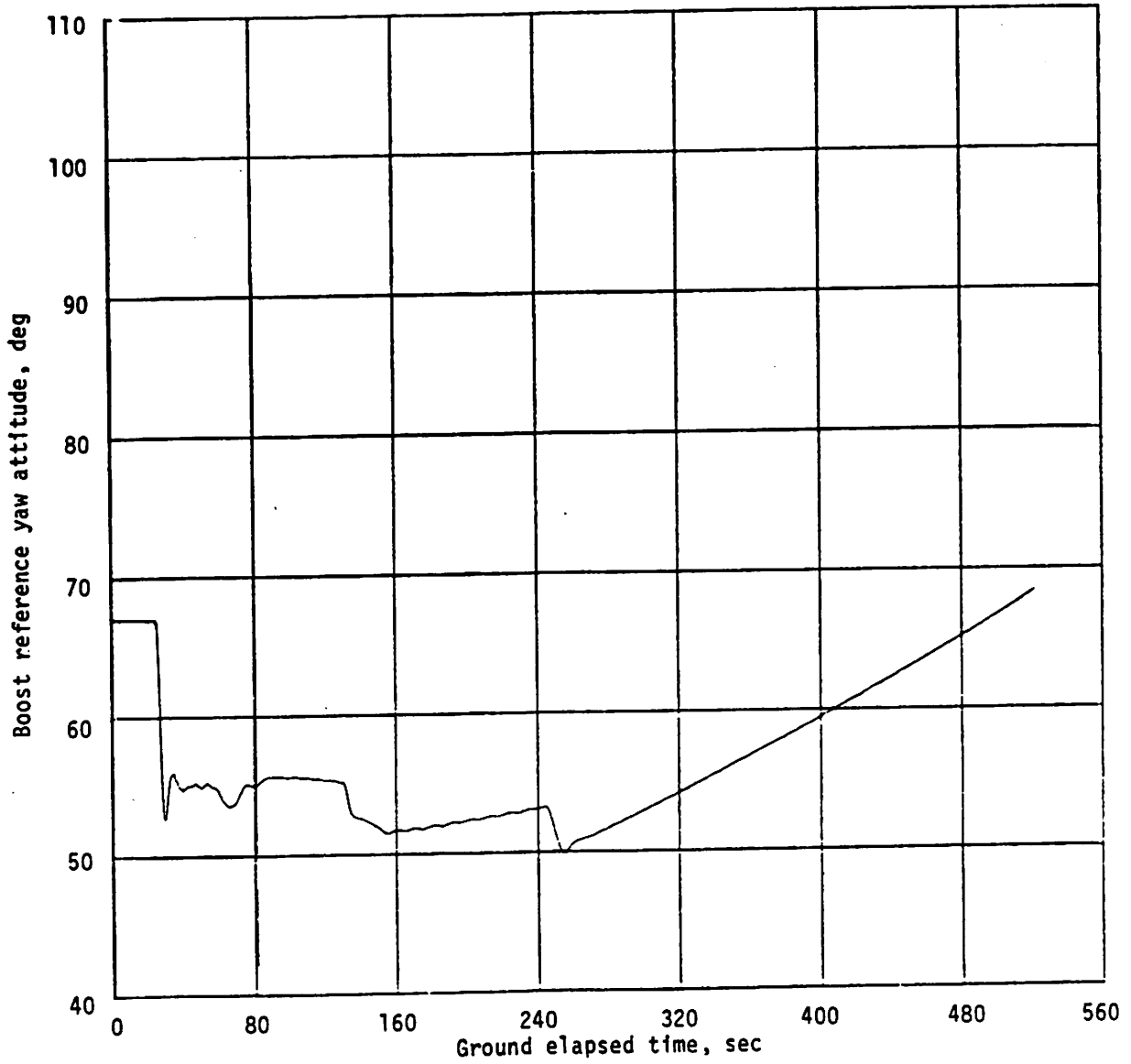


Figure 2.4-8.- Yaw profile (second stage).

Powered explicit guidance: PEG 1 consists of a number of equations that cyclically compute the difference between the targeted state (MECO target) and the current state (computed by navigation). The result of the computations is a velocity error and a range error. The velocity error is termed "velocity-to-go" (V_{GO}) since it represents the velocity needed to be gained in order to achieve the targeted velocity. Likewise with the range error, it is termed "range-to-go" (R_{GO}). Refer to figure 2.4-9.

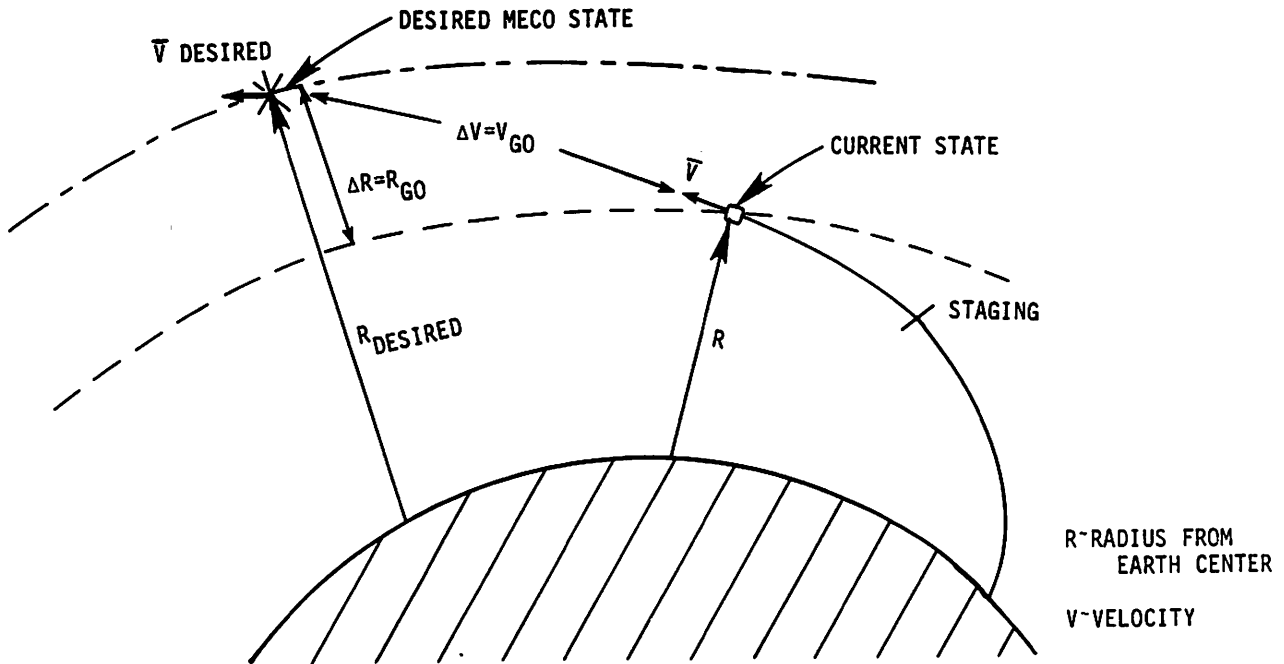


Figure 2.4-9.- Range-to-go computation.

The PEG 1 guidance computations are made assuming three hypothetical thrust phases occur throughout second stage. Refer to figure 2.4-10.

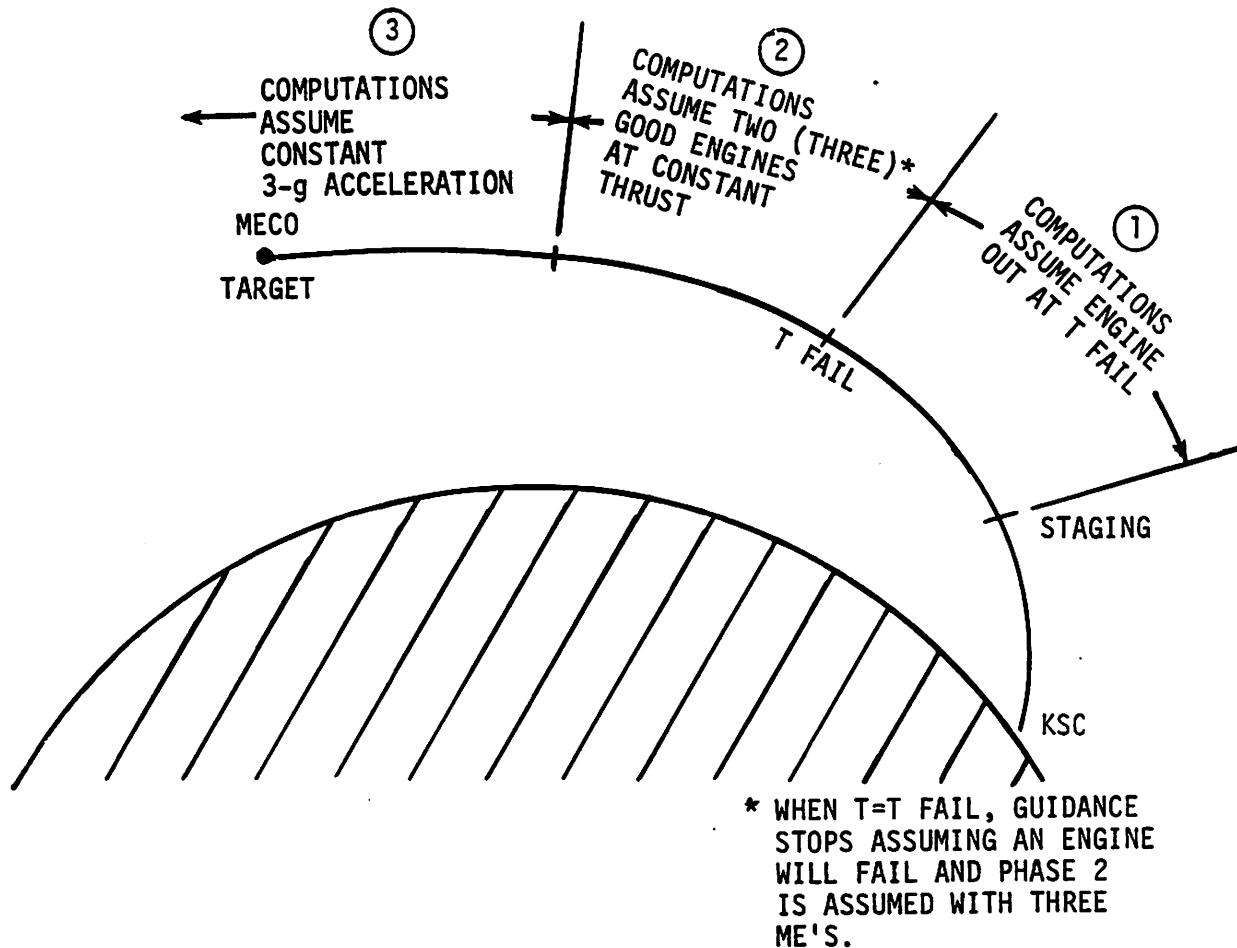


Figure 2.4-10.- PEG 1 hypothetical phases.

In the first phase (1), guidance computations are made assuming that an engine will fail at time = T FAIL (I-load). T FAIL is defined as the earliest time at which a two engine AOA is possible (≈ 240 sec). Therefore, for time prior to T FAIL, the guidance equations reflect that an SSME failure occurs at T FAIL and the remainder of second stage is made with two SSME's. The second phase (2) is defined from T FAIL until 3g acceleration is reached. The guidance computations assume constant thrust throughout this phase. If MET < T FAIL, guidance assumes only two SSME's in this phase since it assumes an SSME failure will occur at T FAIL. As soon as MET = T FAIL and an SSME failure does not occur, guidance assumes three SSME's at constant thrust during the second phase. Finally, the third phase (3) is defined from the beginning of 3g acceleration to guidance cutoff. Guidance assumes 3g constant acceleration through this phase.

Since guidance assumes that an engine failure will occur at T FAIL, it commands a lofted trajectory while $MET < T FAIL$ so as to counteract the effects of the reduced thrust it expects at T FAIL. Therefore, when $MET = T FAIL$ and guidance begins assuming three good SSME's, it will find that the predicted trajectory of the mated vehicle overshoots the desired MECO target and consequently corrects for this by commanding a large (~ 25 deg) pitch (nose toward earth) attitude change. Refer to figure 2.4-11.

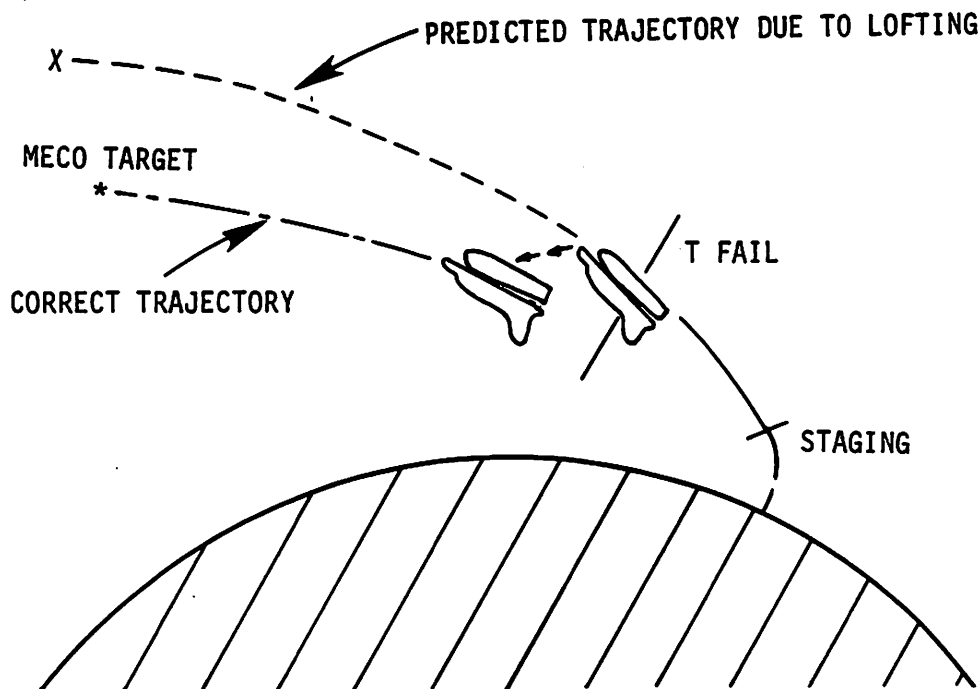


Figure 2.4-11.- T FAIL maneuver.

Guidance cycle: The guidance cycle consists of three major subtasks of predictor, corrector, and steering parameter computation. The predictor logic actually predicts the burnout state of the vehicle assuming constant steering parameters. The corrector logic compares the predicted and desired states and corrects for any discrepancies. The steering parameters are the output of the guidance routine and are sent to flight control. During each cycle, PEG computes new values for the steering parameters based upon current conditions. This allows for the effects of propulsion system failures to be compensated for real time. It also reduces guidance errors since the PEG equations become more accurate as time to go to MECO decreases.

The guidance cycle is entered at the steering parameter computation step in order to minimize transport lag. The predictor and corrector steps are preparation for the next guidance cycle.

In the first guidance cycle following initiation of PEG 1, a first estimate of V_{G0} is required to begin the guidance convergence process (it may take more than one cycle through PEG in one guidance cycle to obtain convergence). This is accomplished by initializing the predicted MECO state and V_{G0} as follows:

$R_p = R_{NAV}$ predicted position equals current position
 $V_p = V_{NAV}$ predicted velocity equals current velocity
 $V_{GO} = 0$ velocity to be gained equals zero

The PEG corrector step is then entered. The corrector step computes the difference between the predicted state and the desired state (MECO target). This correction is called V_{MISS} where

$$V_{MISS} = V_{DESIRED} - V_{PREDICTED}$$

The next step is to correct the value of V_{GO} :

$$V_{GO} (NEW) = V_{GO} (CURRENT) + V_{MISS}$$

Refer to fig. 2.4-12.

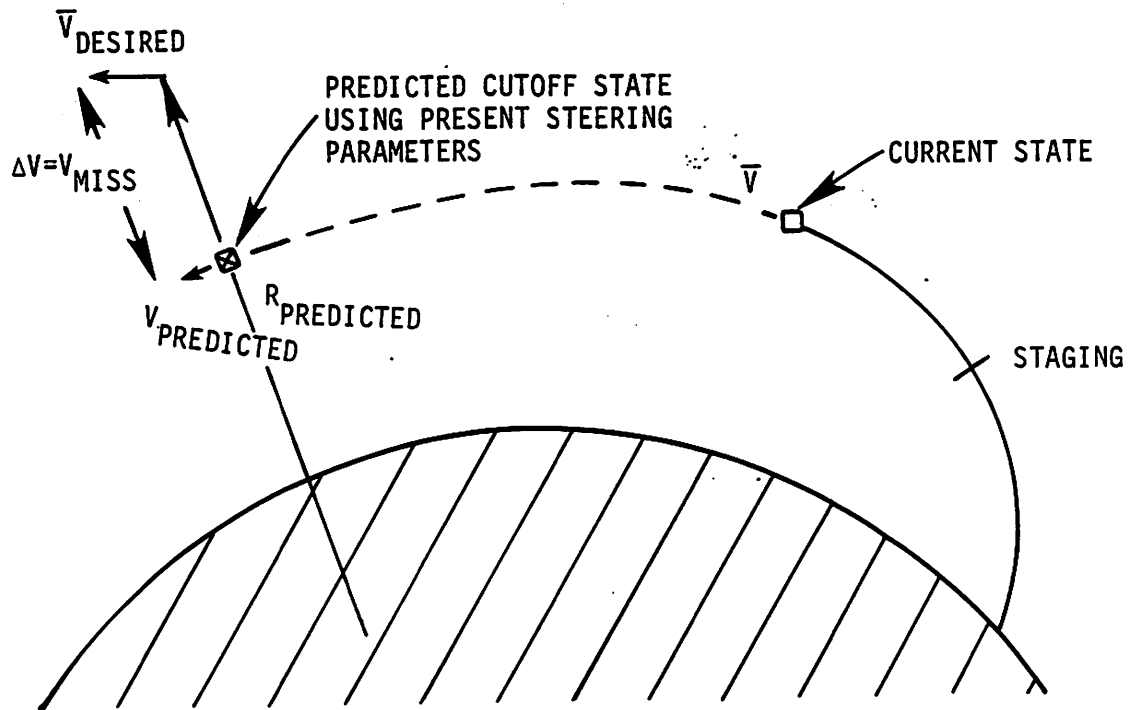


Figure 2.4-12.- V_{MISS} computation.

This completes the initialization cycle for the computation of V_{GO} . V_{GO} will then be decremented each cycle by the amount of measured ΔV over the cycle.

Time until MECO: The time remaining until MECO, T_{GO} , is computed and updated each cycle. T_{GO} is also displayed on the ascent trajectory display.

When T_{GO} reaches 40 sec, the position constraints are released on the MECO target.

When T_{GO} reaches 10 sec, second stage guidance no longer updates T_{GO} . Instead, the MPS guidance cutoff task is initiated and computes desired MPS cutoff time based solely on desired cutoff velocity magnitude (fine count-down).

3g limiting: Guidance controls the g loading on the vehicle by monitoring the acceleration and commanding a reduced throttle if 3 g's is exceeded. The acceleration is calculated using IMU velocity delta's and a delta time.

Manual throttling: Manual throttling is available via the pilot's SBTC. The SBTC must be matched to the auto system command before the auto system will relinquish control of the throttle. SSME shutdown must be commanded manually when manual throttling is engaged.

2.4.3 Orbit Insertion Guidance

Powered Explicit Guidance (PEG 4): Linear Terminal Velocity Constraint (LTVC) or PEG 4 is the primary guidance during the MANEUVER EXECUTE MM's 104, 105, and 302. The goal of PEG 4 is to achieve a desired relationship between horizontal and vertical velocity components at the target position. PEG 4 is called once before the burn to display ignition attitude, time-to-go, and velocity-to-go, on the XXX MNVR YYY display. It is then called cyclically starting 15 sec before ignition to provide flight control with steering parameters and to update the display. The shaded areas of figure 2.4-13 are the target parameters pertaining to PEG 4.

2-201

```
1041/ / OMS 1 MNVR EXEC 1 000/00:09:06
GMBL CK 1 000/00:01:34
      L      R      TRIM      L      R
P -0.6 -0.6 P 12 [-]10.7
Y -6.3 +6.6 Y 13 [-]6.4 14 [+]6.6
SEL
PRI 2* 5* ENG SEL OMS
SEC 3 6 OMS BOTH 15* PURGE ENA 19*
OFF 4 7 L 16
BURN ATT 8 R 180 SURF DRIVE
          9 P 359 RCS +X ACC 18 ON 22
          10 Y 5 21 WT 203192 OFF 23*
      HA      HP      F RCS      ARM 24
TGT 149 + 52 DUMP 25
CUR 149 + 14 OFF 26*
TFF 15:41
REI
ΔVTOT 201.0
TGO 1:44
VGO X + 193.15
      Y + .13
      Z + 55.88
TARGET
27 TIB / : 10:39.9
31 C1 0 36 ΔVX [ ]
32 C2 [+]0000 37 ΔVY [ ]
33 HT 149.201 38 ΔVZ [ ]
34 OT 160.915
35 PRPLT [ ]
LOAD 39 ST CRT TMR 40
```

Figure 2.4-13.- MNVR EXEC display (PEG 4 parameters).

The targets for the OMS 1 burn appear automatically on the display when MM 104 is active. They are loaded automatically into guidance (in MM 104 only). They appear automatically for the OMS 2 burn following OPS 105 PRO. However, the crew must load them into guidance by way of the item entry (item 39).

The PEG 4 target parameters are:

- TIG - Time of ignition
- H - Target altitude
- θ - Target earth central angle from launch site
- C_1 - y intercept of the linear plot (target line) of vertical velocity (y) versus horizontal velocity (x)
- C_2 - Slope of the vertical velocity versus horizontal velocity target line

Note: θ and TIG are biased to enable the same ΔV for either two OMS or one OMS engines burning.

A pictorial view of the OMS target parameters is shown in figure 2.4-14.

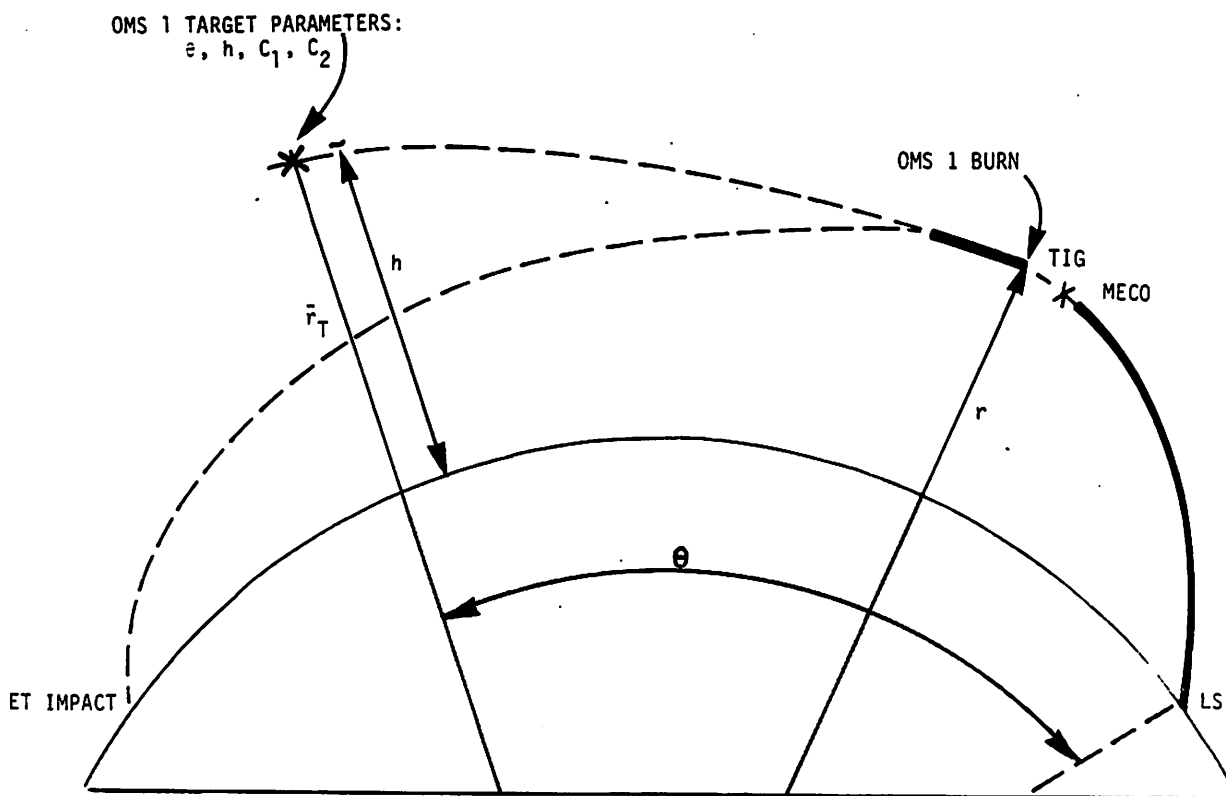


Figure 2.4-14.- OMS 1 PEG 4 target.

The PEG 4 algorithm finds a desired velocity (V_D) at a given position (predicted burnout position, R_p) that will transfer the vehicle to the target position (R_T) over a coasting trajectory while satisfying C_1 and C_2 (fig. 2.4-15).

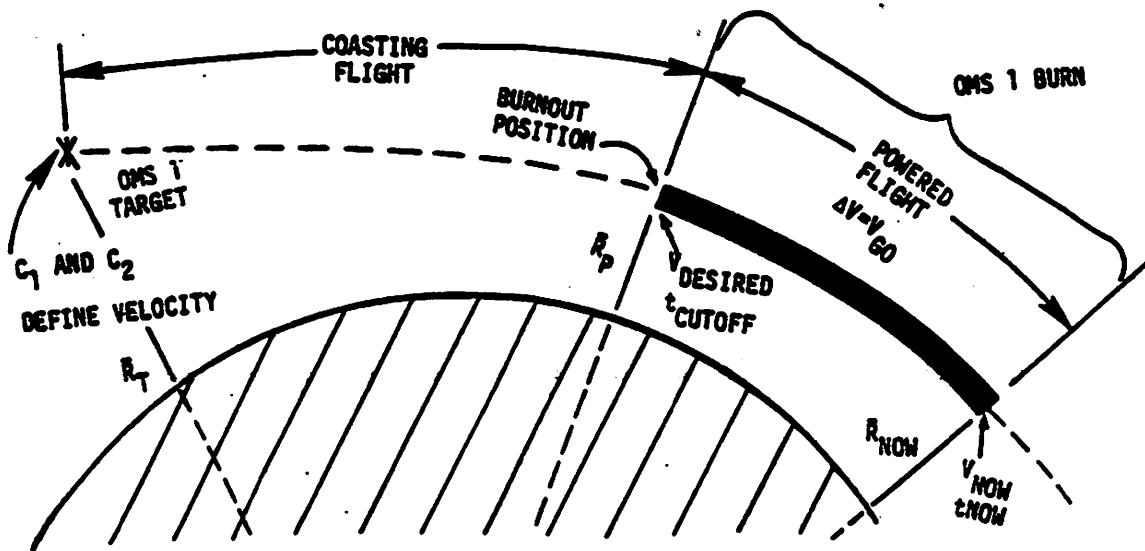


Figure 2.4-15.- OMS 1 trajectory.

PEG 4 cyclically computes a velocity-to-go (V_{GO}) vector which is the velocity remaining to be gained in the burn. A predicted burnout state is compared with the desired state in order to generate the velocity correction factor, V_{MISS} . V_{MISS} is used to update V_{GO} and the process starts over again (fig. 2-4-16).

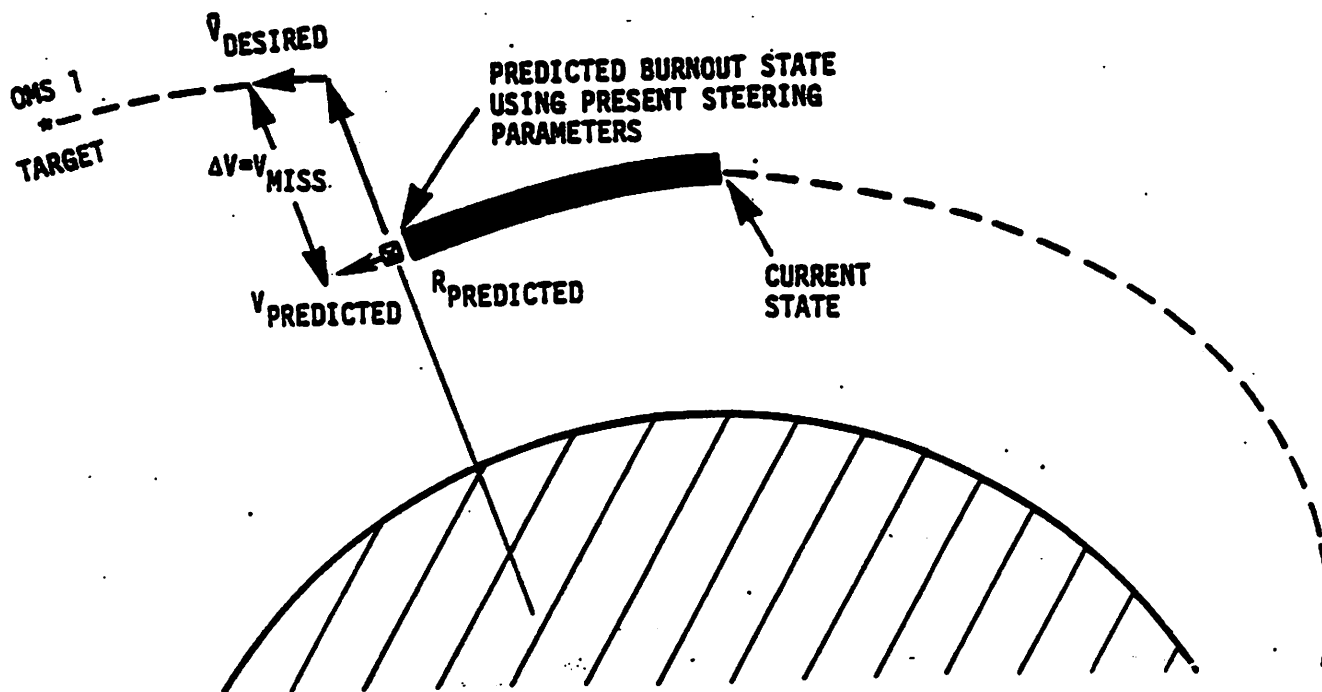


Figure 2.4-16.- V_{GO} update.

If $|V_{MISS}| \leq .01 |V_{GO}|$, guidance is converged and V_{GO} and T_{GO} are updated on the MNVR display and the following parameters are sent to flight control:

$\bar{\lambda}$: reference thrust vector

λ : thrust vector turning increment

t_λ : time constant

At TIG - 15 sec, the guidance function is cycled at 1.04 Hz and uses current state, not TIG state in its computations. At TIG - 10 sec, "EXEC" will flash on the CRT to notify the crew to depress "EXEC" on the keyboard to enable OMS ignition at TIG. If "EXEC" is depressed after TIG, the OMS will ignite immediately up until 5 min. After TIG, the guidance function decrements velocity-to-go (" $\Delta VTOT$ " and " V_{GO} ") using IMU sensed ΔV (V_{GO} displayed in body coordinates) and decrements time-to-go (T_{GO}).

Since an OMS burn cutoff is based on time rather than velocity-to-go, a velocity residual may exist following OMS cutoff. Therefore, the residual (V_{GO}) must be zeroed with the RCS via the THC before exiting OMS 1 MNVR EXECUTE (MM 104).

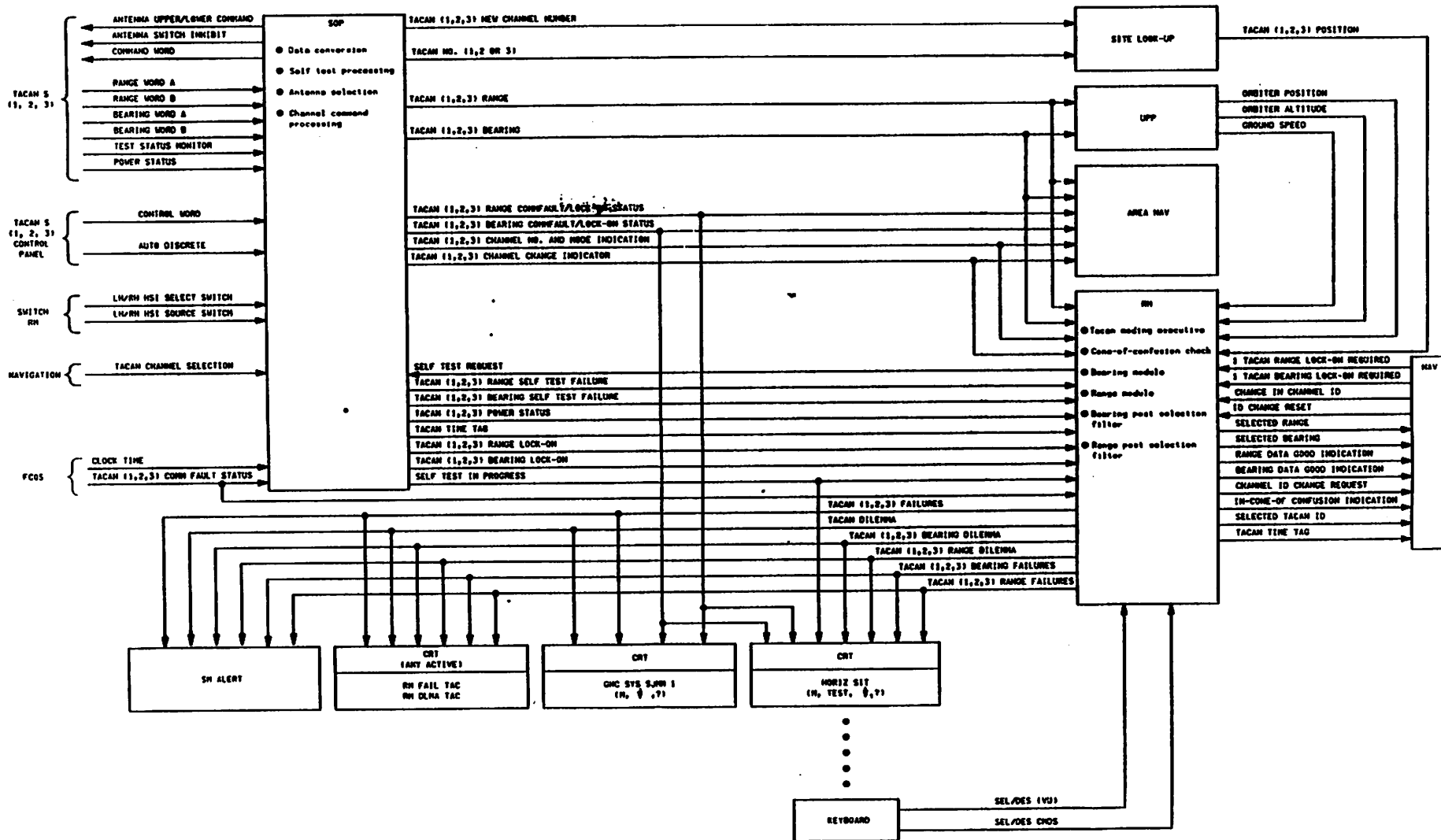


Figure 2.1-26.- Tacan data flow.

The only differences between the OMS 1 and OMS 2 burns are:

- target parameter values
- for OMS 2, "LOAD" must be done by the crew ("LOAD" flashes when MM 105 is entered)

If a displayed target is different from the software recognized target (i.e., a correction to the display that was not "LOAD"ed), "LOAD" will flash until the inconsistency is eliminated by re"LOAD"ing or by returning the parameters to their original values.

Ignition attitude: The ignition attitude is calculated by deriving the thrust vector from OMS trim angles (or RCS I-load) and lining it up with the commanded TIG thrust vector and working in the Local Vertical Local Horizontal (LVLH) roll angle.

OMS engine status: A down arrow in the status column for either the left or right OMS would indicate one of the following: (1) an OMS failed to ignite, (2) an OMS failed during the burn, or (3) an OMS failed to cut off as determined.

OMS engine failure: In the event of a single OMS engine failure (if the burn was initiated with the two OMS engines), the guidance automatically transitions across the failure (given knowledge of the failure) and completes the burn with the remaining OMS engine. Additionally, the guidance transitions across the failure of both OMS engines and continues to solve the guidance equations by assuming the thrust level of the four +X RCS jets. In the event of a propulsion system failure, the guidance targets remain unchanged throughout the burn.

OPS 1 BFS guidance: The following section outlines the major difference between PASS and BFS guidance for OPS 1.

Guidance initialization data are received by a PASS/BFS data transfer at T-23 minutes. The BFS uses a single powered flight guidance module for all powered phases of the flight. First stage BFS guidance is started at SRB ignition and must be running prior to engagement so that it is ready to take over when required. If an SSME fails during ascent, the velocity at the engine failure time is obtained by data transfer from the PASS and is saved by BFS guidance for abort targeting. The mass of the Orbiter is calculated by first stage guidance for use by second stage guidance. The PASS computes the mass using a measured throttle command based on SSME chamber pressure, but after the BFS is engaged the mass is calculated using BFS throttle commands. The BFS does not acknowledge manual throttle inputs to the SSME's.

Burn targets for OMS maneuvers can be changed with keyboard entries to either a PASS or BFS keyboard. The BFS can be engaged at any time during an OMS burn. During the OPS 1 coast mode, the only function of BFS guidance is to calculate current apogee and perigee altitudes.

2.4.4 Powered Explicit Guidance

PEG 7 is an alternate guidance scheme for OMS 1, OMS 2, and deorbit. PEG 7, also known as external delta velocity (ΔV), directs the thrust along the velocity-to-go (V_{GO}) vector until a specified velocity change has been achieved. There is no target position/velocity feedback. The velocity values are keyed in manually.

The PEG 7 target parameters are:

TIG - time of ignition
 ΔV_x } external delta velocity-to-
 ΔV_y } be-gained in LVLH
 ΔV_z } coordinates at TIG

The location of the PEG 7 parameters on the MNVR display is shown in figure 2.4-17.

```

1041 / / OMS 1 MNVR EXEC 1 000/00:09:06
GMBL CK 1 000/00:01:34
      L      R
P  -0.6    -0.6
Y  -6.3    +6.6
SEL
PRI  2*      5*
SEC  3      6
OFF  4      7
BURN ATT  8  R 180
          9  P 359
          10 Y  5

      HA      HP
TGT  149    +  52
CUR  149    +  14
TFF      15:41
REI

ΔVTOT      201.0
TGO          1:44
VGO X + 193.15
      Y +   .13
      Z +  55.88

TRIM      L      R
P          12 [-] 10.7
Y  13 [-] 6.4  14 [+] 6.6

ENG SEL
OMS BOTH  15*
          L  16
          R  17
RCS +X ACC 18
21 WT 203192

SURF DRIVE
ON  22
OFF 23*

F RCS      ARM  24
DUMP  25
OFF  26*

TARGET
27 TIG  --- / --- : 10:39.9
31 C1      --- 0
32 C2      [+] .0000
33 HT      149.201
34 θT      160.915
35 PRPLT  [ ] ---

LOAD  39

ST CRT TMR  40

```

Figure 2.4-17.- PEG 7 parameters on MNVR EXEC.

2.4.5 Orbit Guidance

The ORBIT OPS display (fig. 2.3-3) provides the crew with four mutually exclusive options for auto attitude control:

- Performing a three-axis auto maneuver
- Tracking LVLH about any body vector
- Rotating about the body vector at any DAP discrete rate
- Stopping any of the above options and commanding attitude hold

These functions are available only in OPS 2 (8).

MNVR OPT: The use of item entry for INRTL ATT, items 1, 2, and 3, allows the crew to select an inertial roll/pitch/yaw attitude that they wish the Orbiter to assume. These items will be initialized zero the first time the display is called and thereafter will reflect crew inputs. Then, whenever item 4 is selected (if the DAP is in auto), the Orbiter will initiate a simultaneous three-axis rotation to the desired attitude whereupon attitude hold will be commanded automatically. If the DAP was not in auto, the maneuver will be initiated immediately after auto is selected.

LVLH/ROT OPT: The desired body vector, to be used for either LVLH or the ROT option, is entered via items 5 and 6 with limits of 0° - 359.99° of pitch and 0° - 90° or 270° - 359.99° in yaw. These items are initialized as zero and thereafter reflect crew input. Execution of item 9, Rotation (ROT), initiates the rotation in a clockwise sense about the body vector at the selected DAP discrete rate (if the DAP is in auto). If the DAP is in manual, the rotation will begin as soon as auto is selected. A typical use of this option would be for passive thermal conditioning.

When item 8 (LVLH option) is selected and the DAP is in auto, the DAP will immediately initiate a maneuver to point the desired body vector at the center of the earth with the desired Orbiter orientation about the body vector. This desired orientation is defined by the omicron angle entered in item 7. This item is initialized as zero and thereafter reflects crew input. More precisely, omicron is the angle formed by two planes where one of the planes is defined by the negative position vector and the negative angular momentum vector. The other plane is defined by the vehicle +Y axis and the desired pointing vector. If the DAP was in manual at maneuver selection, the maneuver will start when auto is selected on the DAP. Use of this option will allow such things as continued observation of the Earth's surface through the cockpit windows without continued maneuvering.

The fourth option is the STOP/ATT HOLD option (item 10). It allows the crew to deselect any other option and hold attitude automatically about the existing attitude at the time the stop option is selected. STOP/ATT HOLD is also automatically selected when: (a) items 1, 2, or 3, are changed when automatic maneuver (item 4) is selected; (b) items 5 or 6 are changed when rotation (item 9) is selected; (c) or items 5, 6, or 7 are changed when LVLH (item 8) is selected.

Attitude monitor: The current body attitude (CUR) is the current orientation in degrees of the Orbiter in the ADI inertial frame. The required body attitude (REQD) is the attitude computed by the LVLH option or entered by the MNVR option with respect to the ADI inertial frame. Although the columns are in an order of roll/pitch/yaw, the information displayed is in a pitch/yaw/roll sequence. This means that to get from current attitude to required attitude, the errors as calculated must be executed in a pitch/yaw/roll sequence. The same rotations, performed in a different sequence, may not lead to the desired attitude.

There are two types of attitude error (ERR) that can be displayed, the origin of which is controlled by the ERR ATT function. Selecting item 11, TOT, sends the vehicle's total attitude errors to the ERR line and to the ADI. Item 12, DAP, sends the DAP errors to the ERR line and the ADI.

The rate line describes the vehicle's current rotational rates about the appropriate axes.

Thrust monitor: Orbiter total and body component accumulated delta velocities (ft/sec) are calculated using IMU-sensed velocity and M50-to-body quaternion and are displayed on the orbit display. The thrust monitor can be used when executing an RCS translation. Anytime the IMU acceleration threshold (used to switch from drag model in nav) is exceeded, the accumulators will increment accordingly. Either the total (item 21) or component (item 20) accumulators can be reset to zero via the orbit display.

There is no orbit guidance in the BFS.

2.4.6 Landing Site Update

The following is a description of the landing site update capability which begins with information about the minitable and the maxitable. Landing site update is an OPS 2 function; however, it is closely related to OPS 3 topics and cannot easily be treated separately.

Minitable: The list of three runways on the HORIZ SIT display (fig. 2.3-15) represents a set of data known as the minitable. The arrangement of the data in the minitable is shown in the list below.

- Runway data for: Primary runway
Secondary runway
Alternate runway
- MSBLS data for: Primary runway (if it has MSBLS)
Secondary runway (if it has MSBLS)
- Data for 10 tacan stations

It should be noted that the minitable will not accommodate MSBLS data for the alternate runway. Also, not every runway that might appear as the primary and secondary runway in the minitable will necessarily have MSBLS data.

The minitable is I-loaded with the appropriate runways for each GNC memory configuration.

<u>Memory configuration</u>	<u>Mission phase</u>	<u>Minitable runways</u>
GNC 1	Ascent	RTLS
GNC 2	On-orbit	Nominal end-of-mission
GNC 3	Entry	AOA

The minitables for STS-1, cycle 3, are shown in table 2.4-1.

TABLE 2.4-1.- MINITABLES FOR STS-1, CYCLE 3

Mission	RTLS	AOA	Nominal
Memory configuration	GNC 1	GNC 3	GNC 2
Primary runway	KSC 15	NOR 17	EDW 23
Secondary runway	KSC 33	EDW 23	NOR 17
Alternate runway	None	NOR 35	EDW 17
MSBLS for primary runway	MSBLS for KSC 15	MSBLS for NOR 17	MSBLS for EDW 23
MSBLS for secondary runway	MSBLS for KSC 33	MSBLS for EDW 23	MSBLS for NOR 17
Tacan stations	SAV 74 SSI 35 JAX 92 OMN 73 ORL 59 VRB 120 PBI 102 COF 96 MBL 11 MBL 11	TUS 118 CIE 105 DUG 25 EDW 111 SVC 55 CUS 49 DMN 23 TCS 74 HMN 85 MBL 69	SBP 71 PRB 90 GVO 112 AVE 118 FLW 122 BFL 101 GMN 108 MBL 11 MBL 69 EDW 111

Mission sequences: In case of RTLS abort selection, the GPC's are in the ascent memory configuration and the I-loaded minitable is for RTLS at the KSC landing site.

In an AOA case, the GPC's would transition directly from OPS 1 to OPS 3 (ascent to entry) and so the AOA minitable is I-loaded in the entry memory configuration. The AOA runway set contains a primary and alternate runway at the Northrup Strip which is the AOA landing site. The secondary runway in the AOA set is at Edwards to cover an off nominal deorbit burn situation. It is also there for another purpose. During the on-orbit phase, one of the five GPC's maintains the entry memory configuration as a precaution. This GPC is called the freeze-dried computer. If both mass memory units failed, the freeze-dried computer would be the only source of entry software (other than the BFS). The AOA minitable would be the only set of landing site data available, which is why it contains one runway for the nominal landing site at Edwards AFB.

In a nominal mission sequence, the GPC's would transition from ascent to on-orbit and finally to the entry memory configuration. The I-loaded minitable in the on-orbit memory configuration contains the nominal end-of-mission runway set. When the transition from OPS 2 to OPS 3 is made, the OPS 2 minitable data is automatically transferred to the OPS 3 minitable.

The nominal end-of-mission runway set includes two runways for Edwards and one for Northrup to cover a deorbit underburn case. The sequences for AOA and nominal end-of-mission are summarized below.

<u>OPS transition sequence</u>	<u>Resulting OPS 3 minitable</u>
OPS 1 → OPS 2 → OPS 3 - - - - -	Nominal end-of-mission
OPS 1 → OPS 3 - - - - -	AOA

Maxitable and landing site update: When the primary GPC's are in the ascent or entry memory configuration, the only landing site data available is what is in the minitable. However, when the GPC's are in the on-orbit memory configuration, there is a minitable and a maxitable. The maxitable contains runway, MSBLS, and tacan data for all contingency landing sites. The maxitable arrangement for STS-1, cycle 3, is shown in table 2.4-2.

TABLE 2.4-2.- MAXITABLE FOR STS-1, CYCLE 3

Columns	(1)	(2)	(3)	(4)	(5)
Runways	1! EDW 23	4! KSC 15	7! KAD 05	11! AOG 10	15! NOR 17
	2! NOR 17	5! KSC 33	8! KAD 23	12! AOG 28	16! NOR 35
	3! EDW 17	6! None	9! HNL 08	13! EDW 15	17! NOR 23
	!	!	10! HNL 26	14! EDW 04	18! NOR 05
MSBLS	MSBLS for EDW 23	MSBLS for KSC 15	No MSBLS	No MSBLS	No MSBLS
	MSBLS for NOR 17	MSBLS for KSC 33			
Tacan stations	SBP 71	CTY 57	CCK 50	MTR 33	PHX 103
	GVO 112	SRQ 99	TYN 105	BEJ 105	GUP 98
	NSI 39	PIE 111	MYC 122	CNL 53	TUS 118
	FLW 122	MCF 47	LYC 98	GBR 83	ONM 115
	FIM 72	GNN 109	KAD 57	AOG 23	TCS 74
	GMN 108	OCF 84	NBS 73	MTR 33	SJN 70
	MBL 69	LAL 107	SOK 58	BEJ 105	SVC 55
	MBL 11	ORL 59	LIH 82	CNL 53	DMN 23
	PMD 92	COF 96	CKH 86	GBR 83	HMN 85
	EDW 111	MBL 11	HNL 100	AOG 23	MBL 69

When in OPS 2, the crew can transfer runways and associated data from the maxitable to the minitable. This is done via Landing Site Update (LAND SITE UPDATE) on the ORBIT display (fig. 2.3-3). Each runway in the maxitable has an identifying number from 1 to 18. These identifiers can be entered with item numbers 13, 14, and 15 to change the PRI, SEC, and ALT runways. The identifiers shown in the ORBIT display are 19, 20, and 21 which are the identifiers for the minitable runways.

When the minitable is changed, the new runway names will appear under LAND SITE UPDATE; and when the transition from OPS 2 to OPS 3 is made, it is the updated minitable which is automatically transferred.

Updating the primary runway in the minitable with a runway from the maxitable does three things:

- Runway data for the new runway are transferred to the minitable.
- If the runway has MSBLS data, then those data are transferred.
- Tacan station data are transferred for the set of tacans in the column below the selected runway as listed in table 2.4-2.

Updating the secondary runway in the minitable does two things:

- Runway data for the new runway are transferred to the minitable.
- If the runway has MSBLS data, then those data are transferred.

Updating the alternate runway in the minitable transfers only runway data.

It is important to note that the choice of the primary runway completely controls the selection of tacan stations for the minitable. Since the minitable holds three runways but only one set of 10 tacan stations, the runways should be from the same geographic area to provide tacan coverage. The maxitable has the runways arranged in columns by geographic area for this purpose. In the case of columns 3 and 4, there are runways from two different geographic areas grouped together. In column 3 there are five tacans for Kadena and five for Hawaii. Selection of any runway in that column as the primary runway will result in all 10 tacans in that column being transferred to the minitable. In column 4, two contingency runways at Edwards AFB are grouped with the runways at Rota, Spain; however, all tacans are for Rota.

In comparing the minitable and maxitable it can be seen that there is some repetition of minitable runways in the maxitable. This repetition does provide some redundancy. However, when the nominal or abort runways appear in the maxitable, they are there to be used primarily as contingency landing sites and the tacan sets associated with them are different for that reason.

Data in the minitable and maxitable can be changed by uplink command from the ground during the mission.

Landing site update in the backup flight system: The setup of the maxitable and minitable in the BFS is slightly different from the PASS. In the BFS there is only one memory configuration for all mission phases and there is no OPS 2 software. For this reason the BFS contains both a mini-table and maxitable at all times. The arrangement of landing site data in the BFS is shown.

TABLE 2.4-3.- BFS MAXITABLE AND MINITABLE FOR STS-1, CYCLE 3

(a) BFS maxitable

1!EDW 23!	4!KSC 15!	7!KAD 05!	11!AOG 10!	15!NOR 17!	19!EDW 23!	22!NOR 17!
2!NOR 17!	5!KSC 33!	8!KAD 23!	12!AOG 28!	16!NOR 35!	20!NOR 17!	23!EDW 23!
3!EDW 17!	6!None	9!HNL 08!	13!EDW 15!	17!NOR 23!	21!EDW 17!	24!NOR 35!
!	!	10!HNL 26!	14!EDW 04!	18!NOR 05!	!	!
MSBLS for EDW 23 and NOR 17	MSBLS for KSC 15 and KSC 33	No MSBLS	No MSBLS	No MSBLS	MSBLS FOR EDW 23 and NOR 17	MSBLS FOR NOR 17 and EDW 23
10 Tacans	10 Tacans	10 Tacans	10 Tacans	10 Tacans	10 Tacans	10 Tacans
_____ Contingency sites _____					!OPS 2 !minitable! !(nominal !end-of- !mission)	!OPS 3 !minitable! !(AOA)

(b) Initial minitable

KSC 15	PRI
KSC 33	SEC
None	ALT
MSBLS for KSC 15 and KSC 33	
10 Tacans	

OPS 6
Minitable
(RTLS)

NOTE: All tacan sets are the same as in primary tables.

The maxitable in the BFS is enlarged to include the contingency sites as in the PASS plus AOA and nominal end-of-mission sites. During ascent the minitable contains RTLS runways. At the beginning of OPS 3, one of two sets of runways is transferred automatically from the maxitable to the minitable. If AOA has been selected on the abort rotary switch then the AOA runway set (number 22, 23, and 24 in the BFS maxitable) is transferred to the minitable. If an AOA has not been selected, then the nominal end-of-mission runway set (number 19, 20, and 21 in the BFS maxitable) is transferred to the minitable. If an AOA is performed without moving the abort rotary switch, then the BFS minitable must be manually changed via LAND SITE UPDATE.

The BFS HORIZ SIT display shown in figure 2.4-18 has both the RWY list as in the PASS display and also a LAND SITE UPDATE capability. Both runway selection (item 2, 3, and 4) and landing site update (item 29, 30, and 31) work the same as in the PASS. BFS landing site update, however, is available at all times during ascent and entry. It is important to note that if the minitable is updated with the PASS in OPS 2, the BFS cannot monitor the changes and so the procedure must be repeated with the BFS in order to keep the minitables consistent. The BFS will monitor runway selection (item 2, 3, and 4) and will keep consistent with the PASS automatically. The chart in figure 2.4-19 summarizes the differences between the BFS and PASS for landing site update and minitable selection.

3051/050/

HORIZ SIT

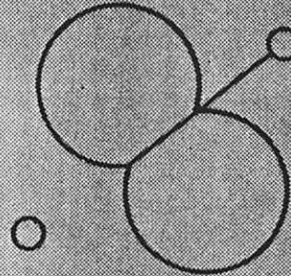
5 002/54:00:51
BFS 000/17:45:51

ALTM
1029.85

RWY
EDW 23 2*
NOR 17 3
FDW 17 4

GLD SLOPE
LO WND 5*
HI WND 6
NEP SW 7

NAV DELTA
ΔX 8
[]
ΔY 9
[]
ΔZ 10
[]
Δ1 11
[]
Δ2 12
[]
LOAD 14

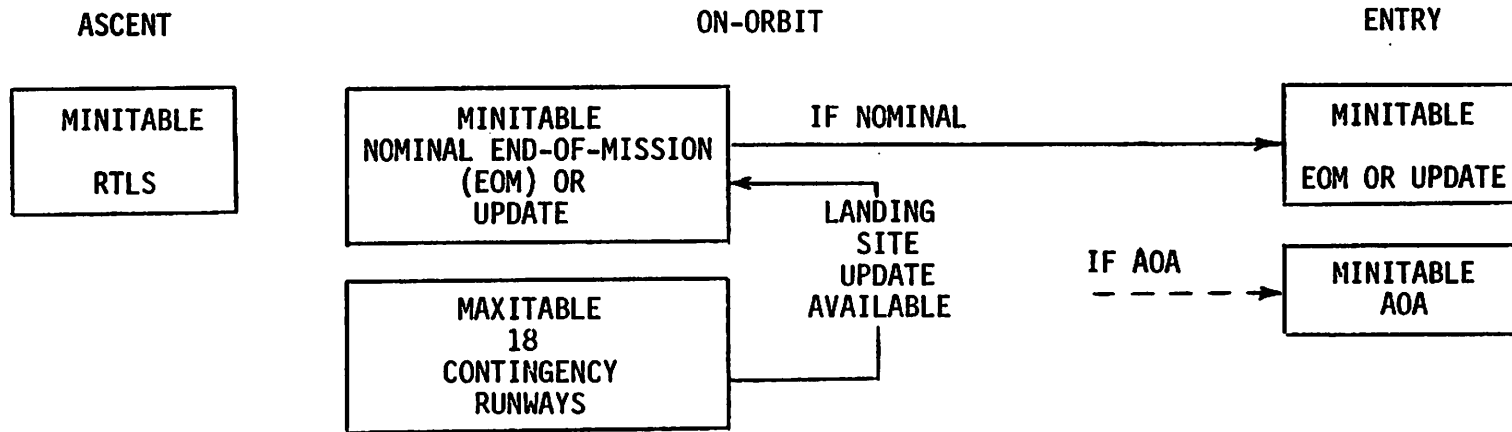


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ELEV 9		PASS/BFS SV XFER 15				16 AT []		
NAV	RESID	RATIO	AUT	INH	FOR	TAC1 112	TAC2 112	TAC3 112
TAC AZ+	2.05	0.3	17*	18	19	+ 96.85	+ 97.23	+ 96.24
RNG+	0.04	0.1				+ 71.53	+ 71.43	+ 71.74
DRAG H+	2050	1.1	20	21*	22	LAND SITE 29 PRI 19 EDW23		
ADTA H			23	24	25	UPDATE 30 SEC 20 NOR17		
ADTA TO G&C			26*	27	28	31 ALT 21 EDW17		
						VENT DOOR	OP 32	CL 33

Figure 2.4-18.- BFS HORIZ SIT display.

(a) Primary system



(b) BFS

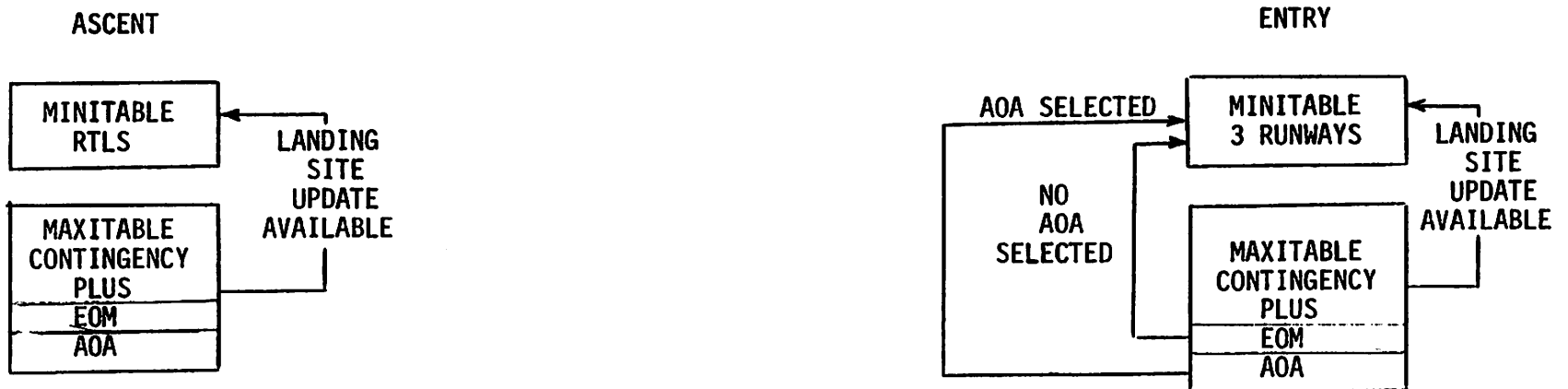


Figure 2.4-19.- Primary and BFS comparison.

2.4.7 Deorbit/Powered Explicit Guidance

The primary active guidance scheme for the deorbit burn is PEG 4. This is the same scheme as orbit insertion guidance and is discussed in section 2.4.3. The targeting, however, is different because for deorbit a narrow trajectory corridor must be obtained to satisfy heating constraints.

The target for deorbit guidance is an acceptable set of vehicle conditions at Entry Interface (EI). The conditions are:

Target altitude \checkmark h (referenced above Fischer ellipsoid)
Angle from TIG to EI \checkmark θ
Velocity constraints \checkmark C_1, C_2

These parameters are computed on the ground and are loaded by the crew via the MNVR display.

Propellant may be wasted in OPS 3. The total propellant to be burned and the direction (positive or negative angular momentum direction) or zero for no wasting must be input to item 35 on the display. Wasting affects burn attitude, increases velocity-to-go and time-to-go, and will be reflected in a larger θ , hence an earlier TIG.

Entry attitude: Before entry interface, the proper entry attitude must be set up. Upon transition to preentry monitor (MM 303), an uplinked inertial attitude is displayed in the BURN ATT slot of DEORB MNVR COAST. This attitude may be changed via item entry.

2.4.8 Atmospheric Guidance (MM 304 and MM 305)

Entry guidance defines a steering profile, based on the current vehicle state vector, to bring the Orbiter to a desired landing within prescribed limits.

Entry guidance stages: The guidance software, from transition to MM 304 to landing, consists of three stages:

- Entry is active from transition to MM 304 until TAEM interface. This nominally corresponds to the interval between EI - 5 minutes (557,000 ft) and a relative velocity of 2500 ft/s (83,000 ft).
- TAEM is active from entry/TAEM interface until A/L interface. This nominally corresponds to the interval between 83,000 ft ($M \approx 2.5$) and 10,000 ft ($M \approx 0.9$).
- Several requirements are necessary for the initiation of A/L guidance. Nominally initiation occurs at an altitude of approximately 10,000 ft. A/L guidance remains active until Orbiter rollout on the runway.

Entry trajectory profile: The nominal STS-1 entry altitude as a function of range is shown in figure 2.4-20. The large arrows in this figure indicate the regions in trajectory where each of the OPS CRT displays are used.

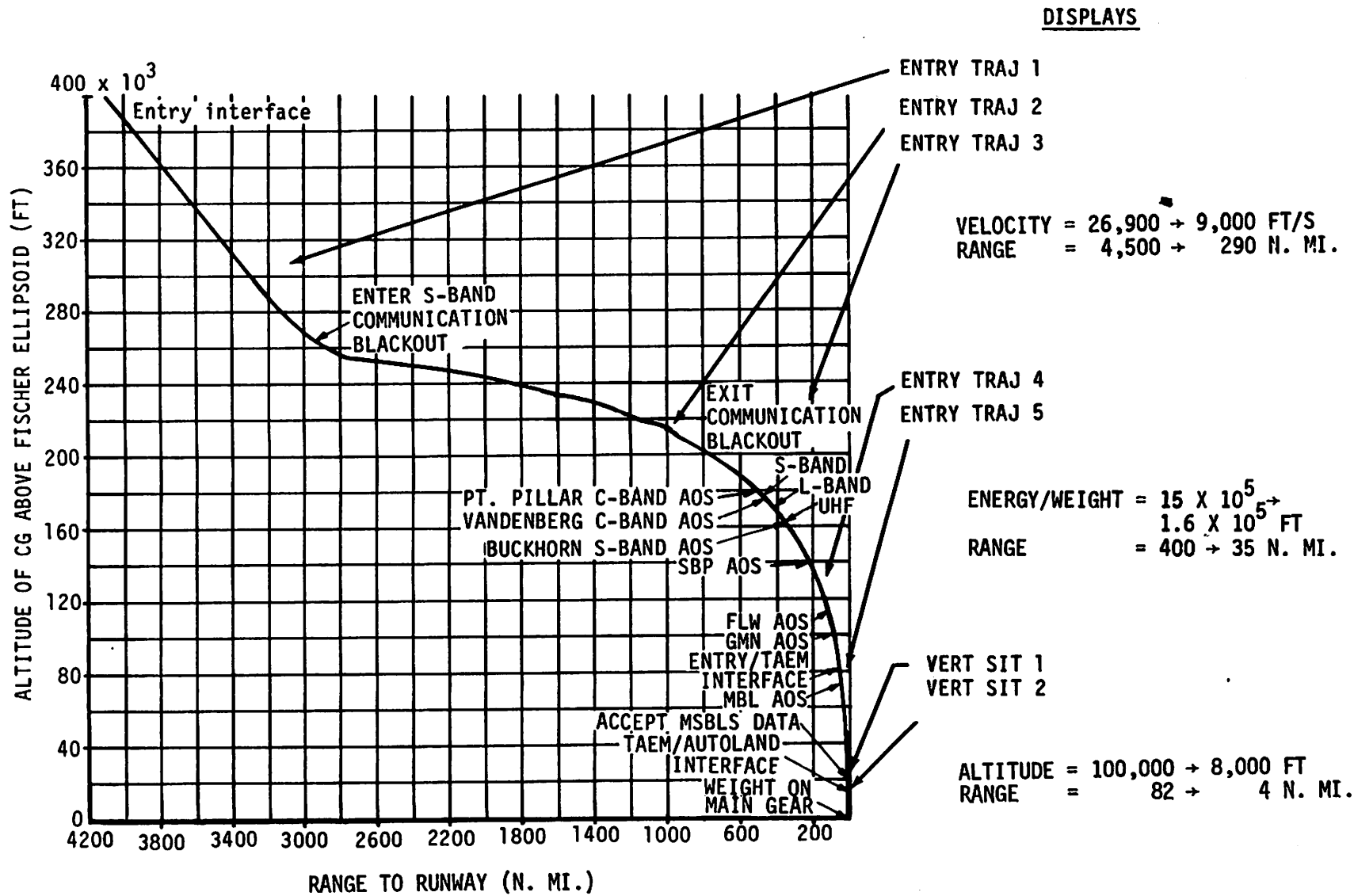


Figure 2.4-20.- STS-1 entry range profile.

Entry guidance characteristics: Entry guidance is active from transition to MM 304 until TAEM interface (velocity = 2500 ft/s) is reached. This flight phase nominally corresponds to an altitude range between 557,000 ft and 83,000 ft and a time interval between 34:45 min and 5:32 min before landing.

Entry guidance must dissipate the tremendous amount of energy the Orbiter possesses when it enters the earth's atmosphere ensuring the following conditions are met:

- The Orbiter does not burn up (entry angle too steep) or skip out of the atmosphere (entry angle too shallow).
- At TAEM interface, the Orbiter is capable of reaching the desired touchdown point.

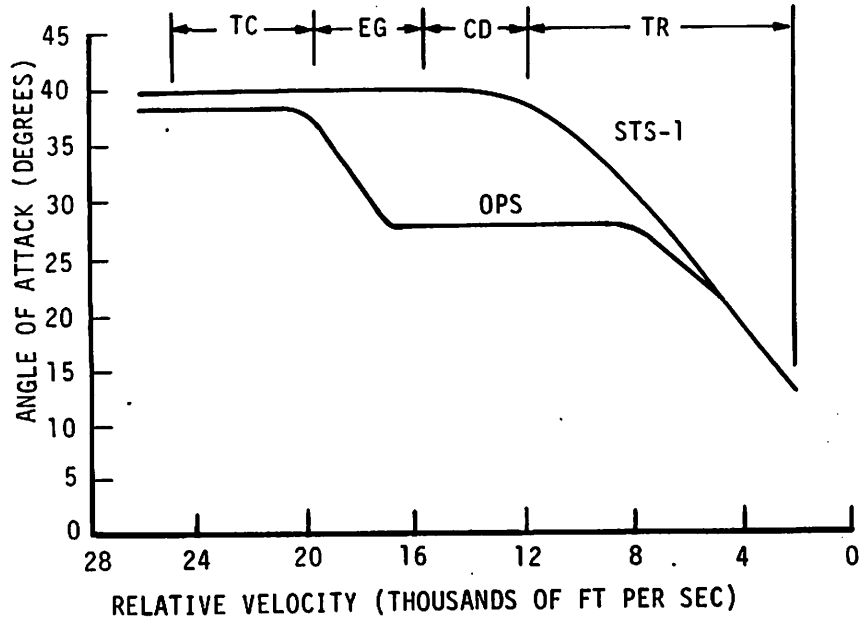
The Orbiter's energy during entry is controlled by the atmospheric drag on the vehicle's surface. Higher atmospheric drag levels enable faster energy dissipation with a steeper trajectory. Normally, angle-of-attack (α) and roll angle (ϕ) enable the atmospheric drag of any flight vehicle to be controlled. However, for the Orbiter, α was rejected as the primary control parameter because surface temperatures are very sensitive to changes in α . The α schedule used during entry is I-loaded as a function of relative velocity (V_{REL}), leaving ϕ as the primary energy control. Increasing ϕ decreases the vertical component of lift, causing a higher sink rate and energy dissipation rate. Increasing ϕ does raise the surface temperature of the Orbiter, but not nearly as drastically as an equal α command. Figure 2.4-21 shows α and ϕ function of V_{REL} throughout the five entry phases.

Guidance calculates a reference vertical Lift-to-Drag ratio (L/D) which would keep the Orbiter on the reference drag profile if it were already on it. In general, the Orbiter will not be on the reference drag profile. Two error feedback terms are therefore added to the reference L/D, one for the difference between the actual and desired drag and one for the difference between the actual altitude rate and altitude rate corresponding to the reference profile.

The downrange footprint capability at entry interface (EI) is about -525 and +875 n. mi. from the target runway. The crossrange capability for STS-1 is ± 550 n. mi. from the target runway, but this limit will be increased to ± 780 n. mi. on later flights. The lower STS-1 crossrange limit places lower temperatures on the Orbiter during entry. The maneuver capability at EI is illustrated in figure 2.4-22.

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ENTRY ANGLE OF ATTACK PROFILE



GUIDANCE PHASES:

- TC = TEMPERATURE CONTROL
- EG = EQUILIBRIUM GLIDE
- CD = CONSTANT DRAG
- TR = TRANSITION

STS-1 BANK ANGLE PROFILE

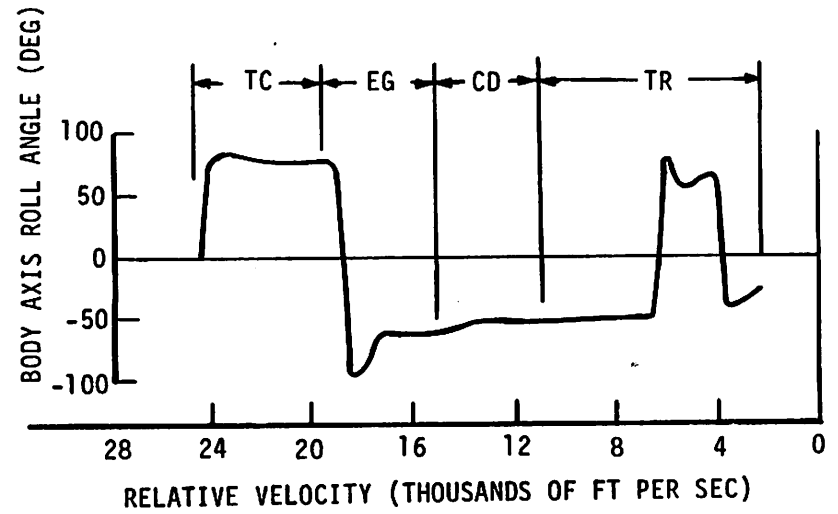


Figure 2.4-21.- STS-1 entry guidance commands.

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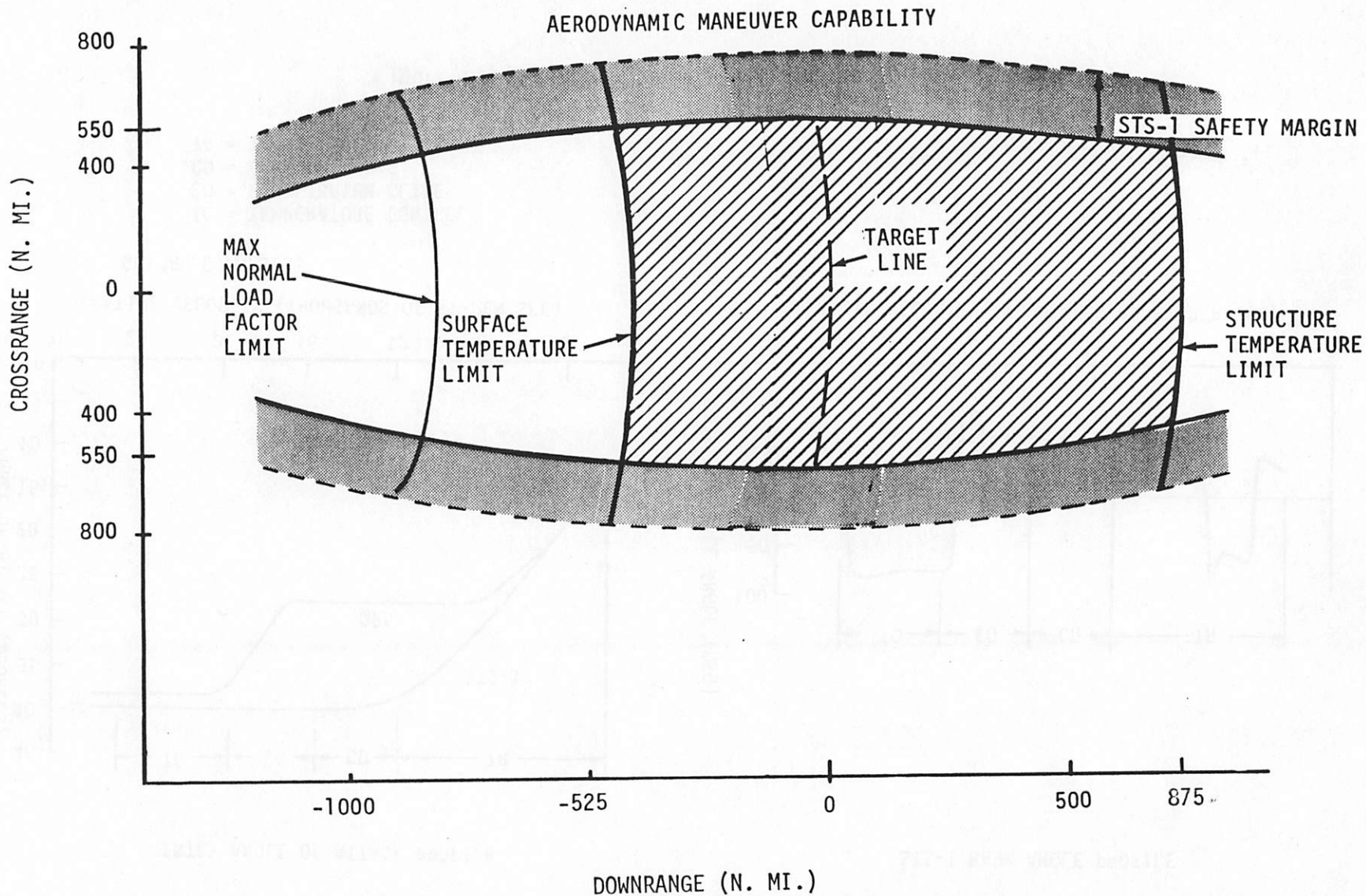


Figure 2.4-22.- Maneuver capability at entry interface.

The sign of the roll angle is used to control crossrange. Azimuth error is the angle between the plane containing the Orbiter's position vector in M50 coordinates and the HAC tangency point, and the plane containing the Orbiter's position vector and velocity vector. This angle is shown as $\Delta\psi$ in figure 2.4-23.

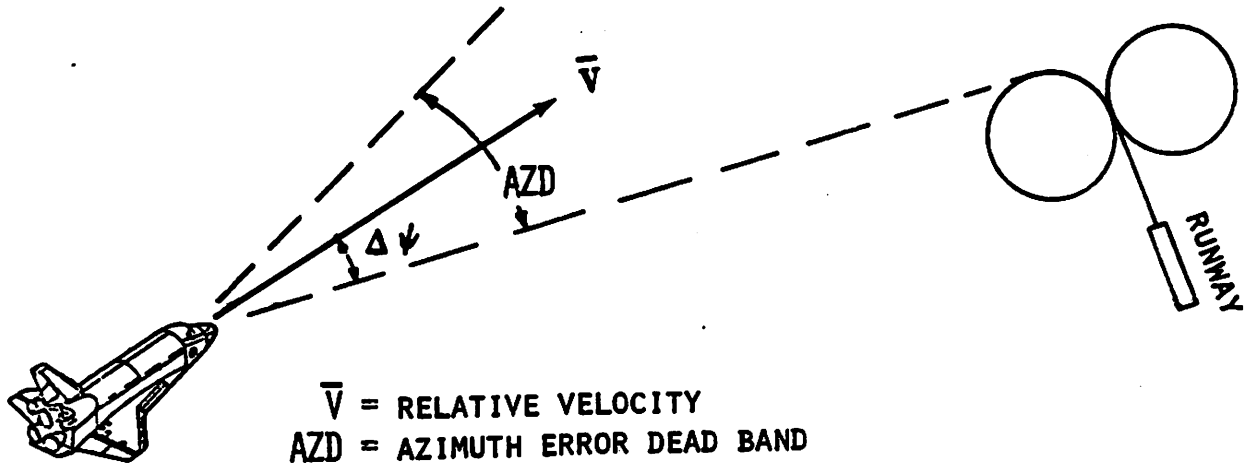


Figure 2.4-23.- Crossrange control.

When the azimuth error exceeds an I-loaded number, the sign of the roll angle is reversed. The current I-load azimuth error deadband is shown in Figure 2.4-24.

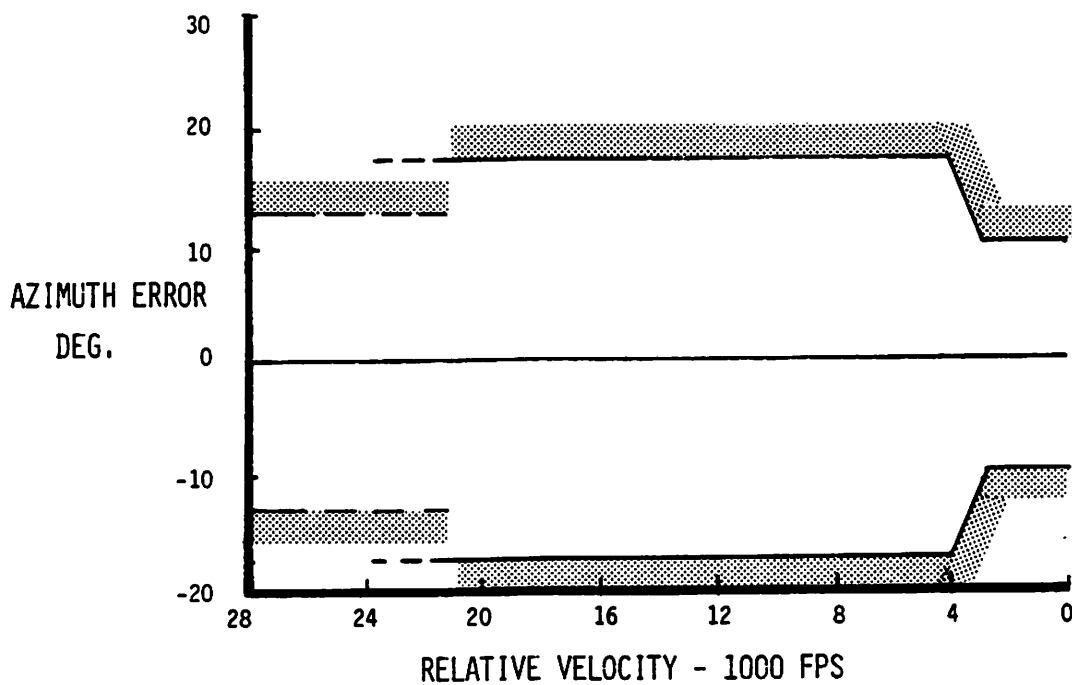


Figure 2.4-24.- Heading error deadband.

Error sources can cause the actual drag profile flown to parallel the reference profile. To correct for these nontransient error sources, the difference between measured drag and reference drag is integrated, converted to an altitude rate bias, and added to the vertical L/D command.

The $\Delta \dot{H}$ bias feedback term should stabilize at the value which will just correct the bias error. The term is only included in the calculation of $(L/D \text{ VERT})_{\text{CMD}}$ when the difference between measured drag and reference drag is increasing.

The magnitude of the $\Delta \dot{H}$ bias feedback term is shown in the lower right hand corner of the five ENTRY TRAJ displays. If the crew has been flying the Orbiter manually and decides to go back to auto, the $\Delta \dot{H}$ bias feedback term should be zeroed if the \dot{H} bias is greater than 30 to prevent unnecessary transients. This is accomplished by executing item 2 on the appropriate ENTRY TRAJ display.

Runway redesignation: To minimize transients due to runway reselection, UPP calculates range-to-go for both the prime and secondary runways and outputs the average to guidance. This averaging process terminates automatically at an I-loaded velocity, which is currently 8000 ft/s. The range-to-go is then calculated only to the prime runway.

The crew has the capability to change the prime runway by executing the item number next to the desired runway on the HSD display. Upon runway reselection, the asterisk moves to the new runway and range-to-go is calculated with respect to that runway. The runway reconfiguration is accomplished by executing items 2, 3, or 4 on the upper left-hand portion of the display.

Entry guidance phases: Entry guidance is divided into the following five phases (fig. 2.4-25).

- **Preentry Phase** - This phase orients the Orbiter for entry into the earth's atmosphere with $\phi = 0^\circ$ and $\alpha = 40^\circ$. No active guidance calculations are performed during this phase. The preentry phase terminates when the sensed acceleration reaches $0.176g$ (5.5 ft/s^2). This point corresponds to a dynamic pressure of 10 psf, approximately the point where the aerosurfaces become effective. The ENTRY TRAJ 1 display, is used to monitor the guidance during this phase.
- **Temperature Control Phase** - This phase is designed to keep the TPS surface and backface temperatures within the design limits. A constant heating rate is established by flying a parabolic drag profile as a function of V_{REL} . Two parabolas are used in the nominal drag reference profile shaping, the first covering temperature control phase initialization to $V_{\text{REL}} \geq 19,000 \text{ ft/s}$. The drag reference profile is calculated each guidance cycle and the proper roll command output to FCS to keep the actual drag or drag references. Small amplitude modulation is also used for short term control of drag transients. ENTRY TRAJ 1 is still employed to monitor the guidance commands.

- Equilibrium Glide Phase - This phase transitions from the rapidly increasing drag levels of the temperature control phase to the constant drag level of the constant drag phase. Equilibrium glide flight is defined as flight in which the flight path angle (γ), the angle between the local horizontal and the velocity vector, remains constant. Equilibrium flight provides the maximum downrange capability. Nominally, the equilibrium glide phase starts at $V_R = 19,000$ ft/s and lasts until the drag acceleration reaches 33 ft/s². During this phase, the roll commands are calculated to keep the Orbiter on the drag reference profile and within the established crossrange boundaries. ENTRY TRAJ 2 is used to monitor guidance during the equilibrium glide phase.
- Constant Drag Phase - This phase covers entry in the region between a drag acceleration value of 33 ft/s² and $V_{REL} = 10,500$ ft/s. The roll angle commands are calculated to keep the Orbiter on the drag reference profile and within crossrange limits. Commanded α is initially 40° , but it begins to ramp down in this phase, reaching approximately 36° by the end of the phase. ENTRY TRAJ 2 and 3 are used to monitor guidance functions during the constant drag phase.
- Transition Phase - This phase transitions the Orbiter from the high drag levels of the constant drag phase to the TAEM interface point. In the transition phase, the drag reference profile rotates about a specific endpoint rather than shifting up or down to adjust for trajectory dispersions. This scheme attempts to always hit the nominal TAEM interface point even when the flight has not been nominal up to that point.

The flight path angle is assumed to be negligible in the first four phases, but the flight path angle increases to significant nonzero values in transition phase. Drag vs. velocity calculations have proven to be too complex when (γ) is significantly non-zero. During the transition phase, this problem is solved by employing reference profile and range equations derived in a drag vs. energy/weight form. A linear drag vs. energy/weight profile fits best between the vehicle constraints in this region, and it also provides a smooth transition to TAEM guidance.

During this phase, roll commands are calculated to keep the Orbiter on the drag reference profile and control crossrange. Commanded α continues to ramp down, reaching a value of approximately 14° by TAEM interface.

This phase requires two displays, ENTRY TRAJ 4 and ENTRY TRAJ 5, to monitor the guidance functions.

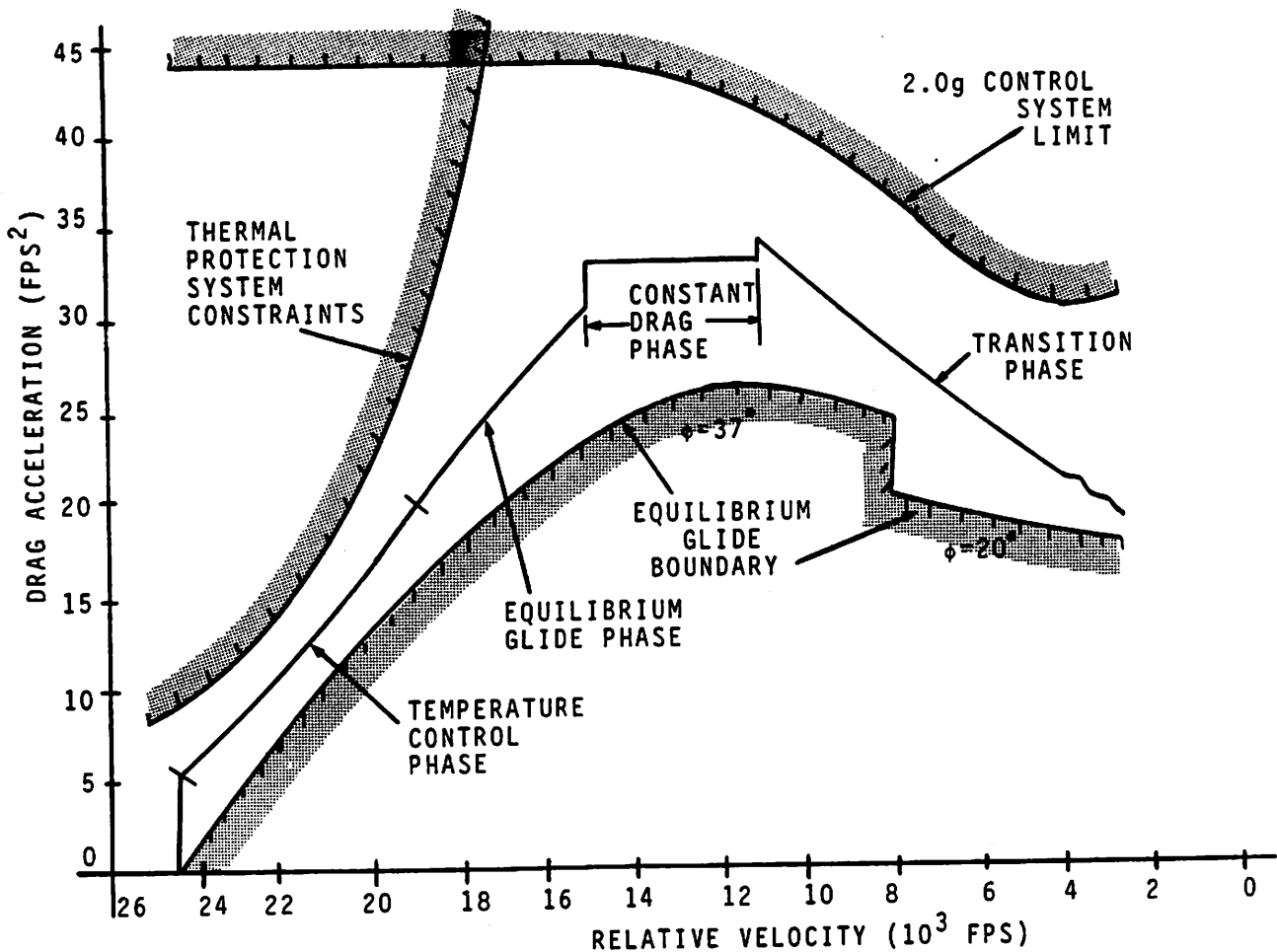


Figure 2.4-25.- Entry guidance nominal drag reference profile.

Entry guidance phase transition logic: The nominal transition logic for each of the five entry guidance phases is summarized in table 2.4-4.

The entry guidance phase transition sequence described on the preceding pages is only used when the Orbiter's energy state is close to nominal. Some of these phases are bypassed if the entry state of the Orbiter is much less or greater than nominal.

If the Orbiter is low on energy (current range-to-go much greater than nominal at the current velocity), entry guidance will command lower than nominal drag levels to decrease the energy lost as heat. The equilibrium glide phase will be longer than normal, and guidance will transition directly from equilibrium glide to the transition phase.

If the Orbiter has too much energy (current range-to-go much less than nominal at the current velocity), entry guidance will command higher

than nominal drag levels to dissipate the extra energy. Guidance bypasses the equilibrium glide phase and transitions directly from the temperature control phase to the constant drag phase. Entry guidance can transition from the preentry phase to the constant drag phase if the Orbiter's energy state is extremely high. This sequencing scheme does violate the Thermal Protection Subsystem (TPS) limits.

The off-nominal entry guidance phase transition schemes are summarized in figure 2.4-26.

TABLE 2.4-4.- ENTRY GUIDANCE PHASE TRANSITION LOGIC

GUIDANCE PHASE	1 PREENTRY	2 TEMPERATURE CONTROL	3 EQUILIBRIUM GLIDE	4 CONSTANT DRAG	5 TRANSITION
DESCRIPTION	ATTITUDE HOLD PRIOR TO 0.176G	DRAG PROFILE WITHIN TPS CONSTRAINTS	EQUILIBRIUM GLIDE TYPE TRAJECTORY	CONSTANT DRAG	LINEAR DRAG-ENERGY PROFILE, MINIMIZE ROLL COMMANDS
NOMINAL SPEED REGION	26,000 - 25,000	25,000 - 19,000	19,000 - 14,000	14,000 - 10,500	10,500 - 2,500
POSSIBLE PREVIOUS PHASES	CREW SELECTS ENTRY PROGRAM	PREENTRY	TEMPERATURE CONTROL	PREENTRY TEMPERATURE CONTROL EQUILIBRIUM GLIDE	EQUILIBRIUM GLIDE CONSTANT DRAG
POSSIBLE SUBSEQUENT PHASES	TEMPERATURE CONTROL CONSTANT DRAG	EQUILIBRIUM GLIDE CONSTANT DRAG	CONSTANT DRAG TRANSITION	TRANSITION	GUIDANCE (TAEM)
LOGIC TO TERMINATE THIS PHASE NOTE: (X) MEANS NEXT PHASE IS NUMBER X	ACCEL > 0.176G(2) ----- CONSTANT DRAG TO SATISFY RANGE > 0.33 FT/S ² AND ACCEL > 0.176G(4)	V < 19,000(3) ----- V < 27,637 AND DRAG REF < DRAG REF FOR PHASE 3 (3) (ROLL SMOOTHING LOGIC) ----- CONSTANT DRAG TO SATISFY RANGE > 33 FT/S ² (4) ----- V < 16,300 AND DRAG REF > DRAG REF FOR PHASE 4 (4) (ROLL SMOOTHING LOGIC)	V < 12,800 AND DRAG REF > DRAG REF FOR PHASE 5 (5)		V ≤ 2,500

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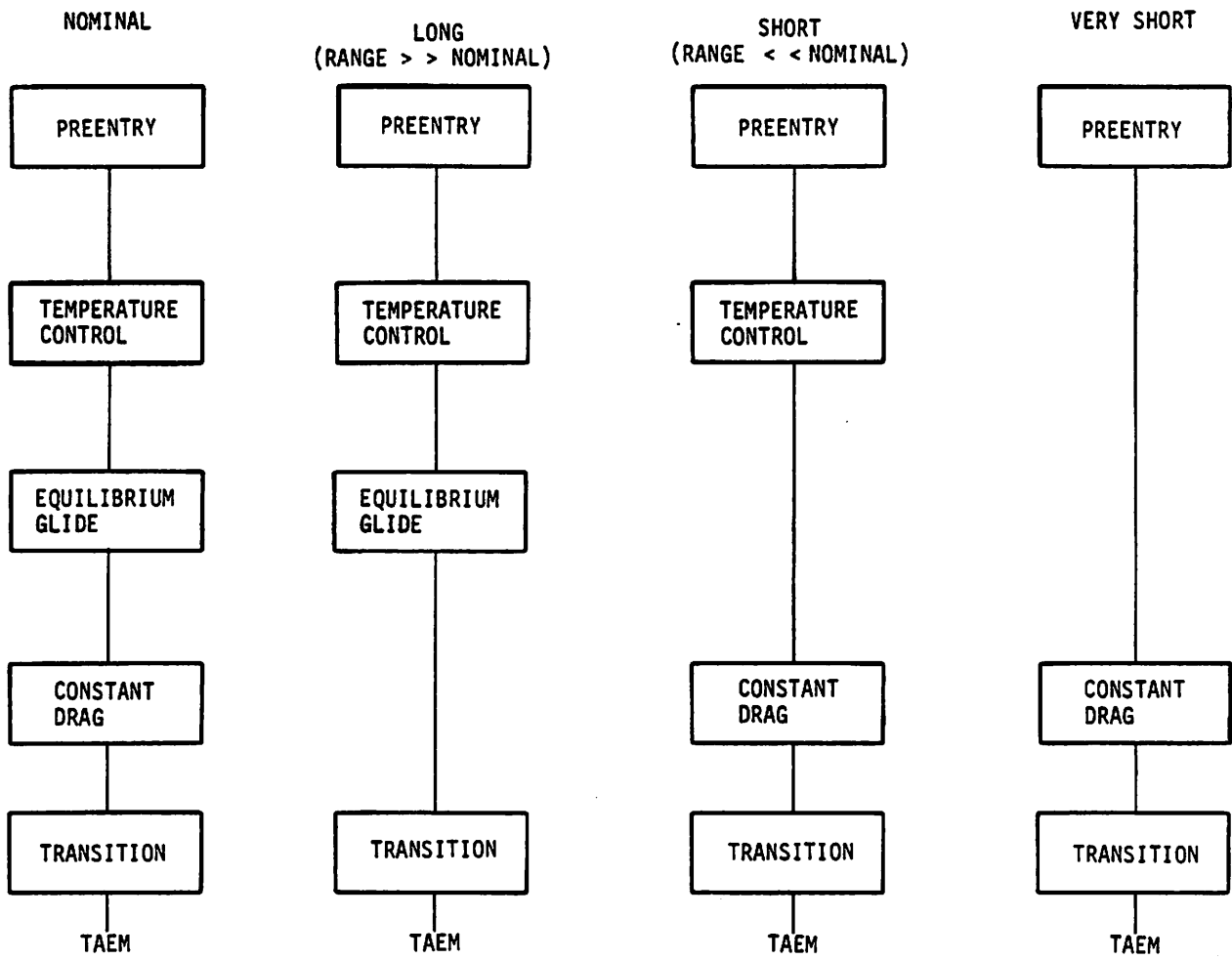


Figure 2.4-26.- Phase transition cases.

Reference drag profile: The equations for determining the range obtained by flying a given drag profile are simplest if drag vs. relative velocity is used for the first four phases and drag vs. energy/mass is used for the fifth phase. By using a reference profile expressed in these variables, the range obtained by flying the profile is determined by a set of five algebraic equations, one for each phase, rather than requiring a numerical integration scheme.

The first step in defining the reference drag profile is the determination of the various vehicle constraints during entry. Thermal constraints limit the upper drag range in the higher velocity region, and structural loads limit the upper drag range in the lower velocity region. The lower drag range is limited by the aerodynamic capability of the Orbiter. For any given ϕ , the lowest drag levels are obtained with equilibrium flight. In the higher velocity regions, commanded ϕ is limited to a minimum of 37° . As the Orbiter approaches TAEM interface, the commanded value of ϕ drops to 20° .

With the vehicle constraints established, a nominal drag profile can be defined. The nominal drag reference profile for STS-1 was selected as the profile midway between the vehicle constraints. The nominal drag reference profile and vehicle constraints are plotted in figure 2.4-25.

Constraint and nominal drag reference lines appear as solid lines on the ENTRY OPS CRT displays (fig. 2.4-27). These entry OPS CRT display lines correspond exactly to the lines shown on the drag vs. velocity plot shown in figure 2.4-25. These lines look different because the CRT lines are plotted as velocity vs. range rather than drag vs. velocity. The dashed lines on the five ENTRY TRAJ displays are constant drag lines, the equivalent of horizontal lines on a drag vs. velocity plot.

A new drag reference profile is calculated each guidance cycle to adjust for off-nominal situations. The desired drag level is increased or decreased to make the range obtained by flying the new profile equal to the current range-to-go sent from NAV. The reference drag profile is shifted to higher than normal drag levels if the range-to-go at the current velocity is less than nominal, and it is shifted to lower than normal drag levels if the range-to-go is greater than nominal. Guidance alters the drag profile for one phase at a time.

The drag reference profile calculated as discussed to this point contains two sharp corners. These corners occur at the transition from equilibrium glide to constant drag phase, and at the transition from constant drag to transition phase. This sudden change in the slope of the reference drag profile causes a discontinuity in the commanded roll angle, followed by some oscillation until the commanded roll angle settles down. Roll smoothing logic is utilized to lessen the effect of the two sharp corners. This involves guidance solving ϕ command for both phases simultaneously. The first phase ϕ command is followed until ϕ command from the first phase is approximately equal to the ϕ command generated in the second phase. At this point the second phase ϕ command is followed so that a large jump in ϕ command does not occur when transitioning.

The only effect of the roll-smoothing logic visible to the crew is a sudden jump in the drag reference shown on the ENTRY TRAJ display at the beginning and end of the constant drag phase.

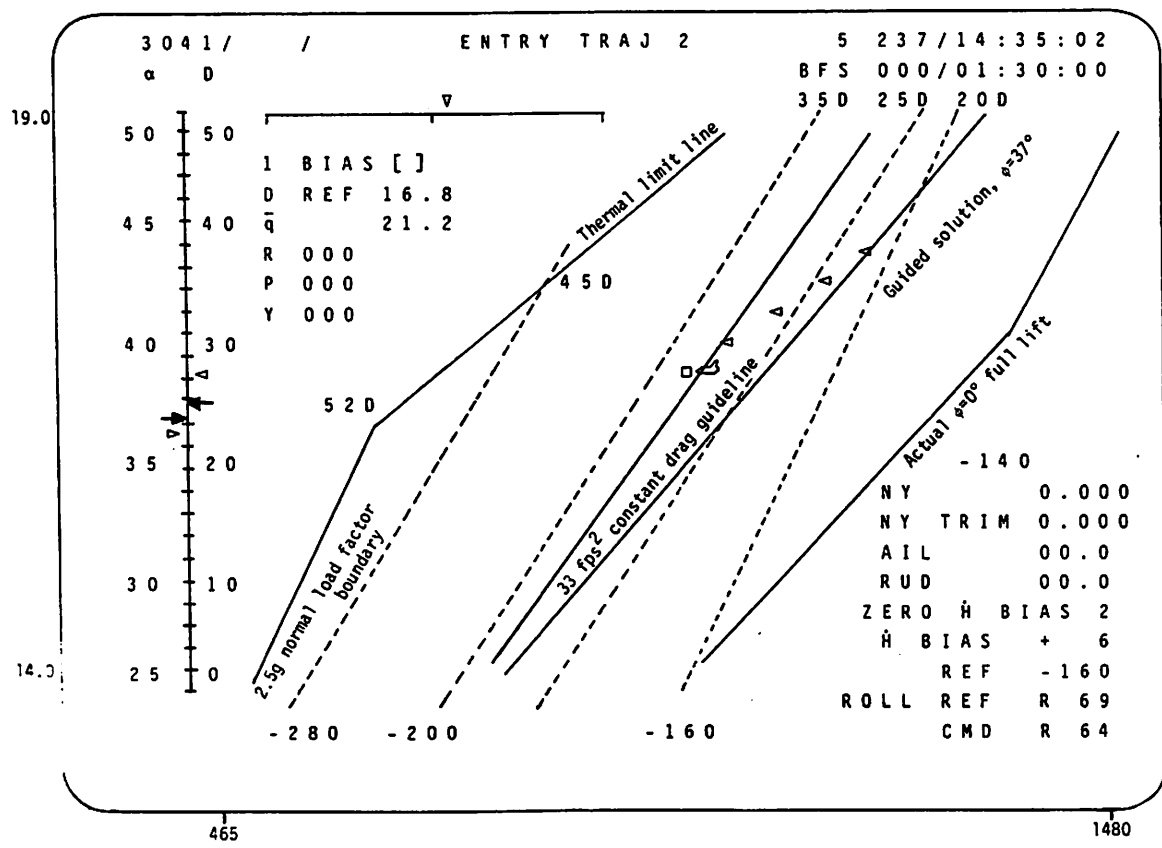
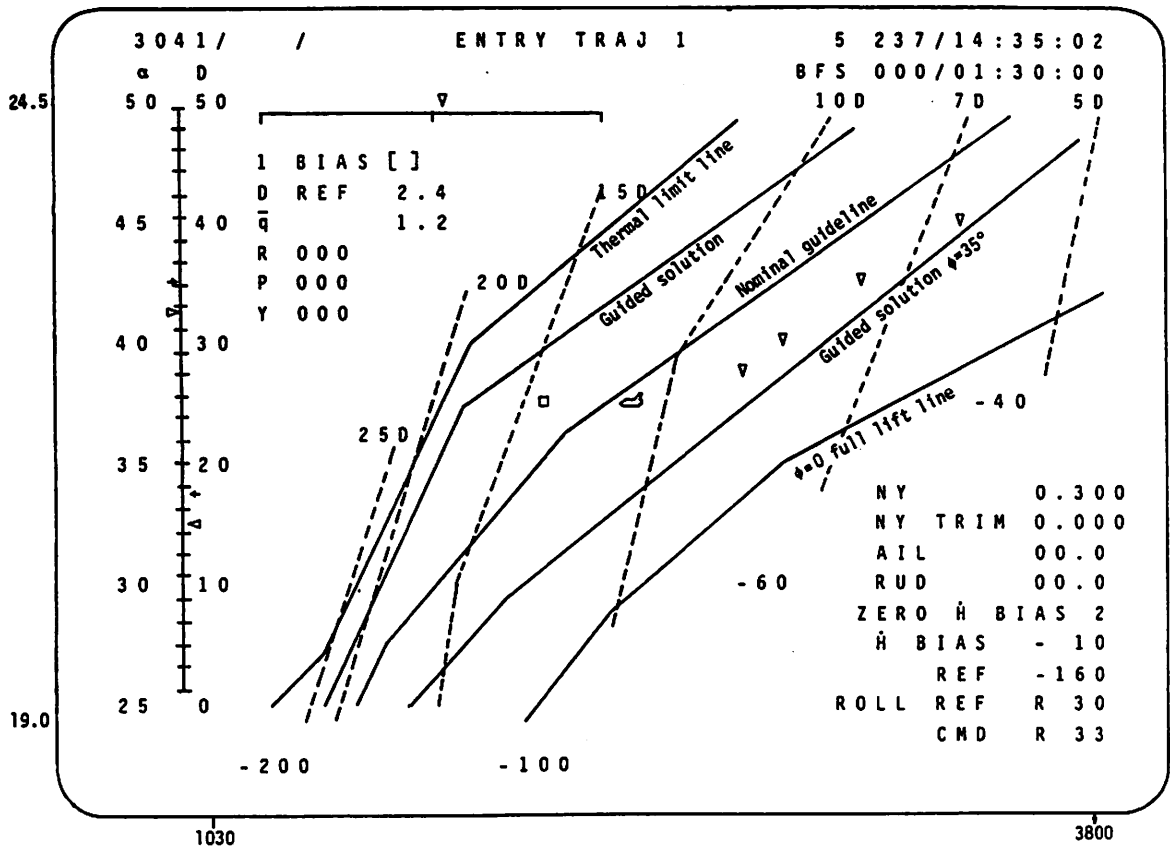


Figure 2.4-27.- ENTRY TRAJ displays (page 1 of 3).
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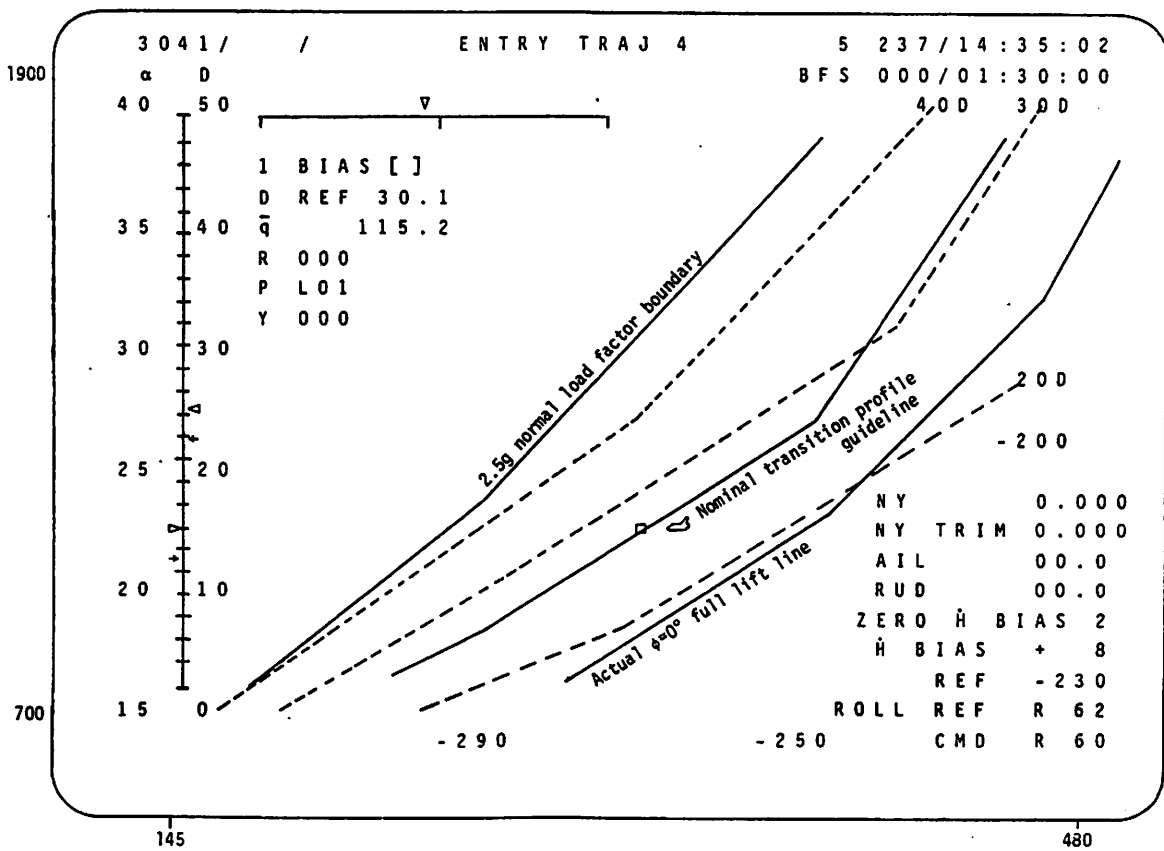
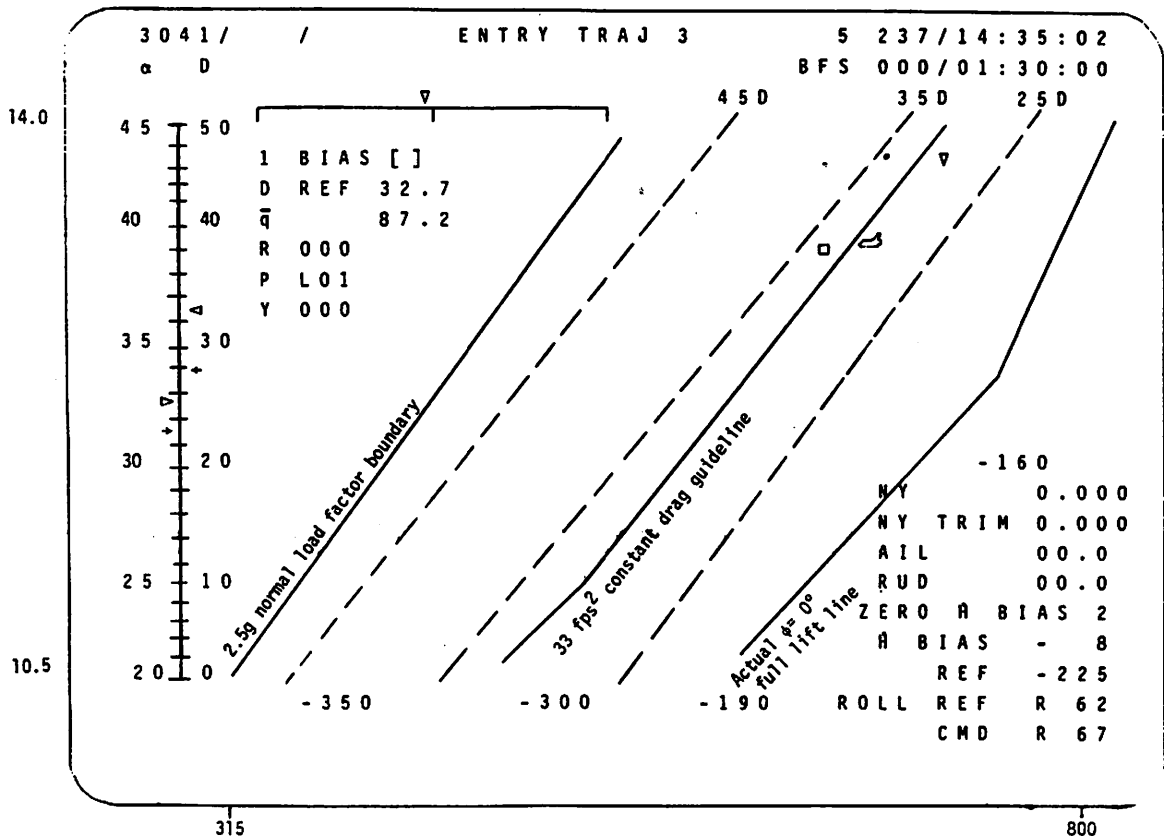


Figure 2.4-27.- ENTRY TRAJ displays (page 2 of 3).

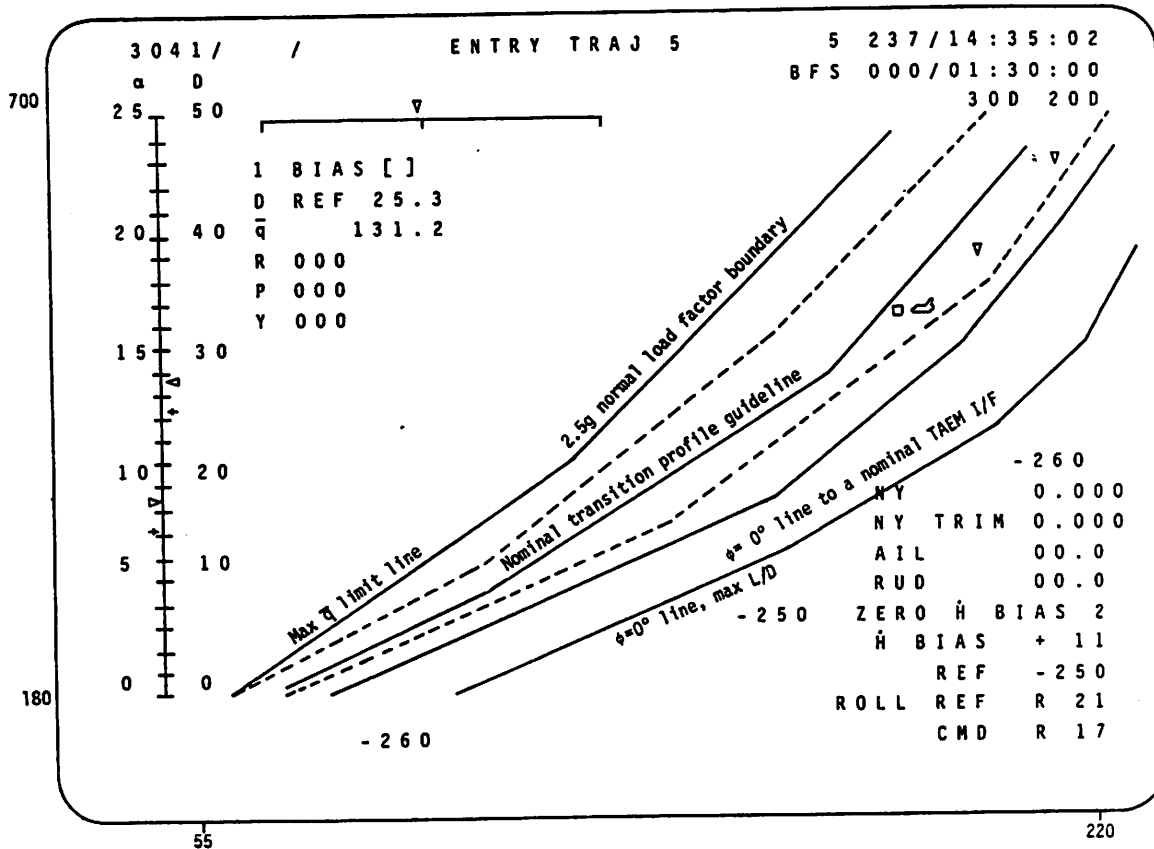


Figure 2.4-27.- ENTRY TRAJ displays (page 3 of 3).

TAEM guidance characteristics: TAEM guidance is initialized at a relative velocity of 2500 ft/s, corresponding to an approximate altitude of 83,000 ft. This condition nominally corresponds to a range of 52 n. mi. from the landing runway. The TAEM groundtrack geometry is shown in figure 2.4-28.

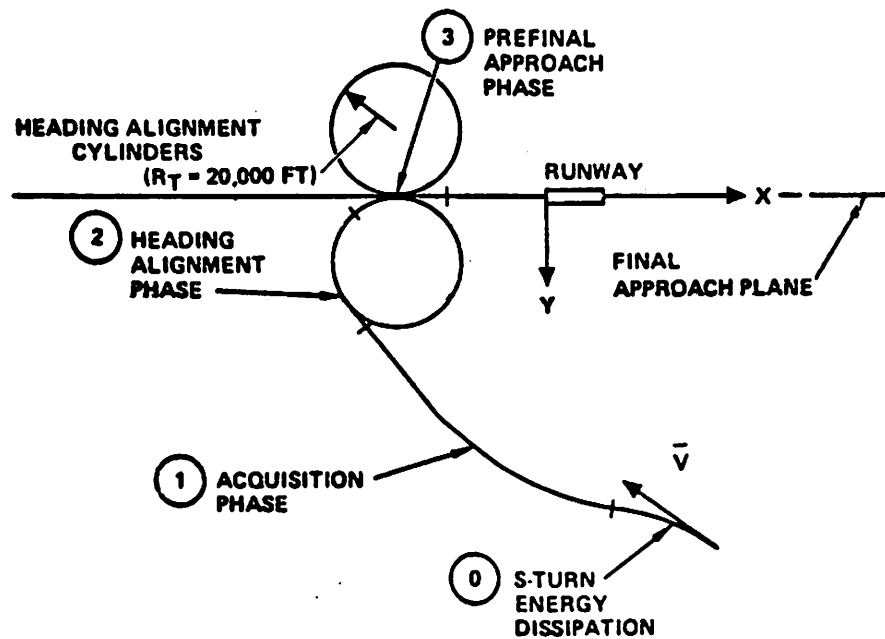


Figure 2.4-28.- TAEM guidance phases.

The central region of the HSD provides a groundtrack display of current and predicted Orbiter positions with respect to the HAC's and runway touchdown points. The Orbiter symbol in the lower center of the HSD remains stationary, and the other symbols move in relation to this Orbiter symbol. The three small circles in front of the Orbiter represent the predicted Orbiter position 20, 40, and 60 seconds from the current time.

The functional flow scheme utilized by TAEM guidance is depicted in figure 2.4-29.

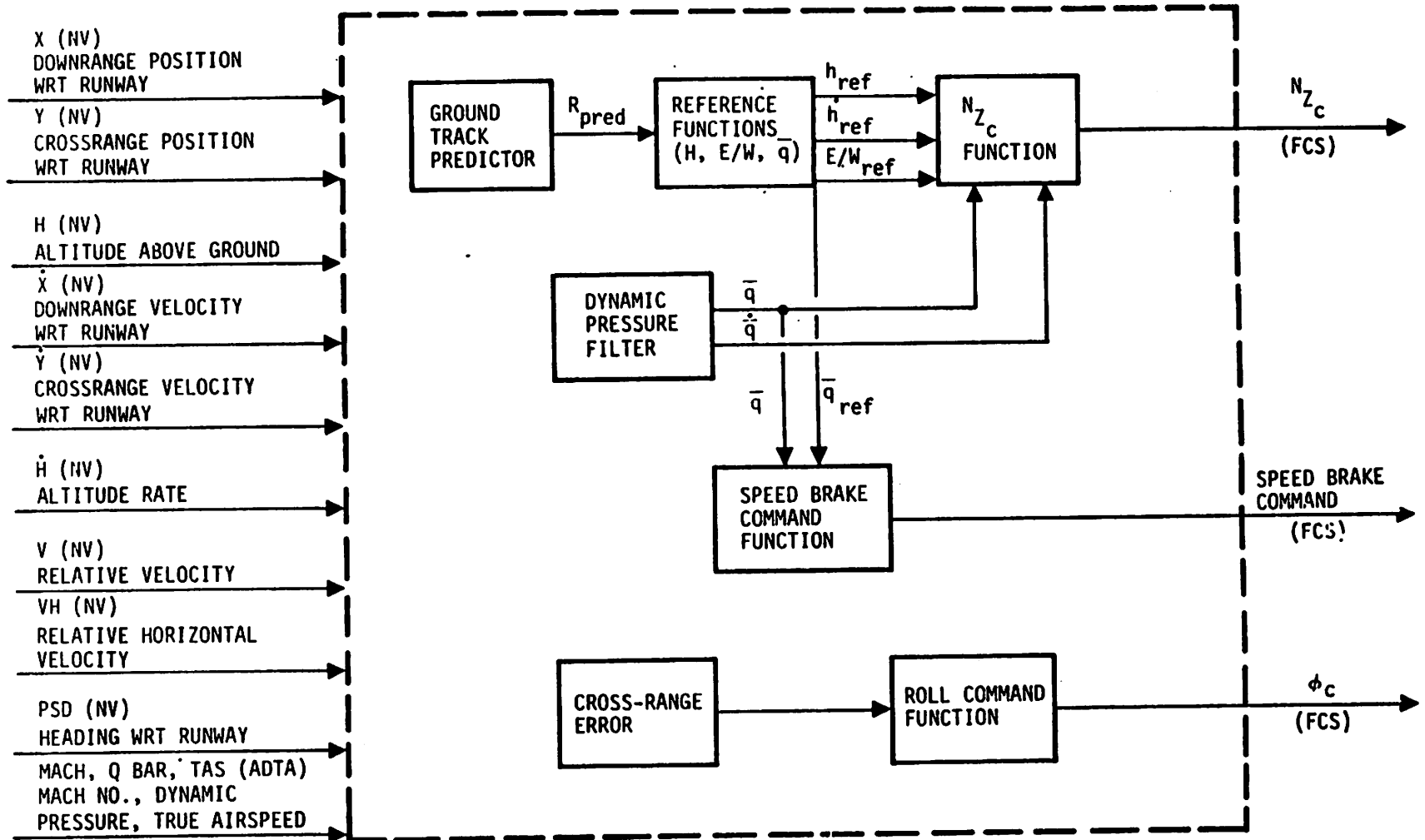


Figure 2.4-29.- TAEM guidance.

Glideslope/nominal entry point redesignation: Items 5 and 6 on the HSD enable the ground intercept point to be shifted. When the HSD is first called by the crew, the asterisk is beside item 5 for the nominal low headwind condition. MCC monitors the wind conditions and advises the crew on the headwinds.

Item 7 for STS-1 allows the crew to switch from a left-hand approach (Minimum Entry Point (MEP)) to the closest HAC approach to the runway (Normal Entry Point (NEP)). MEP comes up automatically so if the crew desires NEP, they execute an item 7 to throw the software toggle switch. MEP no longer means minimum entry point. NEP still functions the same as it did as the nominal entry point.

TAEM guidance phases:

- S-turn Phase - The S-turn phase is only used in a high energy state when the A/L interface criteria cannot otherwise be met. The Orbiter turns away from the HAC tangency point and executes an S-turn until sufficient energy has been dissipated to allow a normal approach. Incremental normal acceleration is limited to $\pm 0.5g$ and roll to 30° supersonic or 50° subsonic during the S-turn phase. In addition to S-turns, the speed brake can be utilized to modify drag, L/D ratio, and the flight-path angle in high energy conditions.
- Acquisition Phase - During the acquisition phase, the Orbiter turns until the flight direction becomes tangent to the nearest HAC. This phase continues until the HAC has been acquired. Incremental normal acceleration is limited to $\pm 0.5g$ and roll to 50° during the acquisition phase.
- Heading Alignment Phase - The heading alignment phase following the HAC until landing runway alignment $\pm 20^\circ$ has been achieved. Incremental normal acceleration is limited to $\pm 0.5g$ and roll to 60° during the heading alignment phase.
- Prefinal Phase - During the prefinal phase, the Orbiter pitches down to acquire the steep glide slope and increase airspeed. The autoland interface airspeed requirement at 10,000 ft is 290 ± 12 KEAS. Incremental normal acceleration is limited to the range of $+1.5g$ to $-0.75g$ and roll is limited to 30° .

TAEM guidance phase transition logic: The nominal transition logic for each of the four TAEM guidance phases is summarized in table 2.4-5.

TABLE 2.4-5.- TAEM PHASE TRANSITION CRITERIA

Event	Criteria for occurrence of event
Initiate S-turns	Range > HAC center MEP X coordinate + 153,000 ft E/W > 0.699 (range - HAC center MEP X coordinate) -27,000 ft
Transition from S-turn to acquisition	E/W < reference E/W
Transition from acquisition to heading alignment	Distance to HAC center < 1.01 radius of HAC

TABLE 2.4-5.- Concluded.

Event	Criteria for occurrence of event
Transition to pre-final	Range < HAC center X coordinate + 4000 ft ($\pm 12^\circ$)

TAEM guidance energy profile: TAEM guidance flies an energy over weight profile by using attitude to control potential energy and drag to control kinetic energy. TAEM guidance loop gains get higher for tighter control as the Approach and Landing (A/L) interface is approached. Orbiter energy over weight as a function of range to the landing runway during the TAEM guidance region is presented in figure 2.4-30.

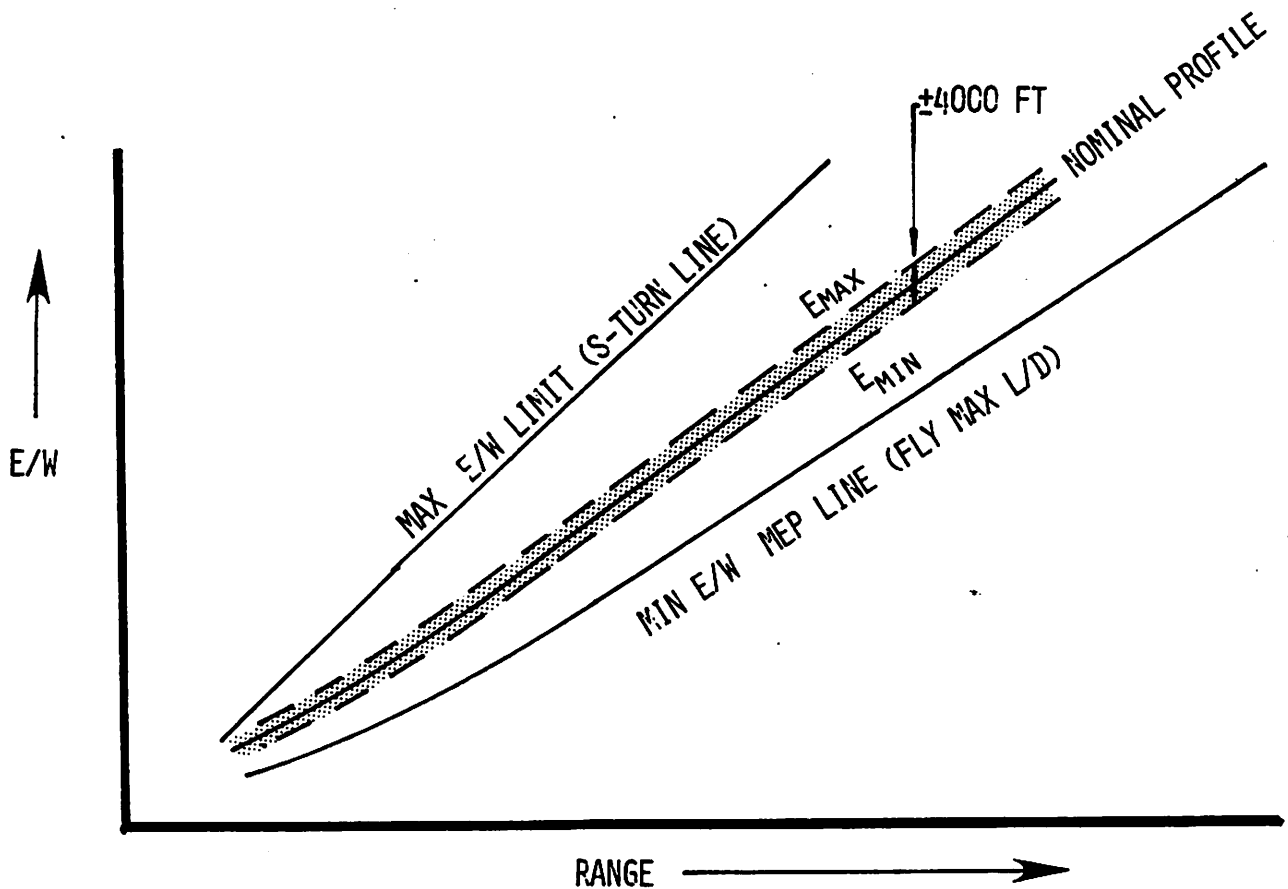
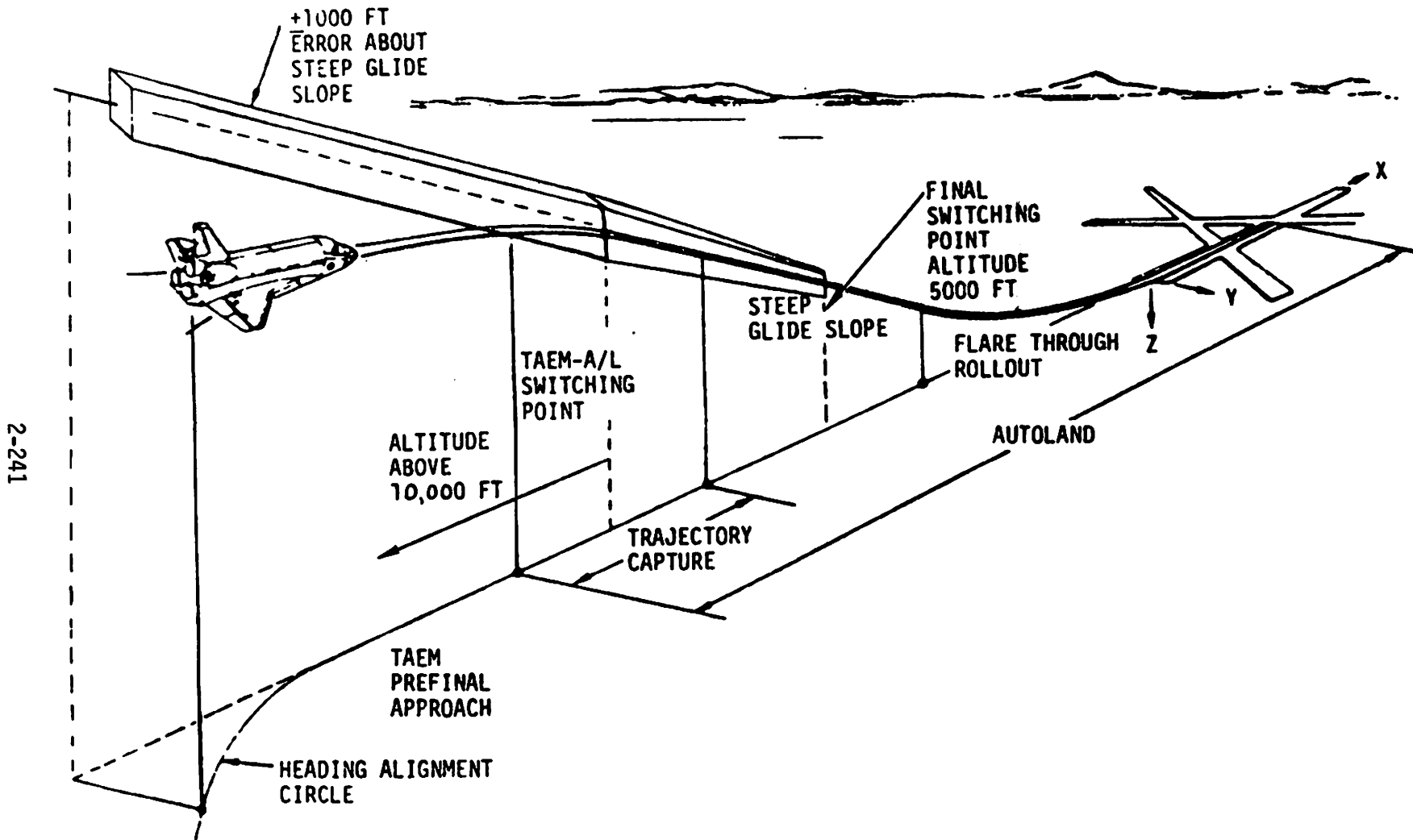


Figure 2.4-30.- Energy over weight range profile.

A/L guidance characteristics: The transition to A/L guidance nominally occurs at a 10,000 ft altitude and 20° flight path angle. The functions of A/L guidance are summarized in table 2.4-6. A/L guidance geometry is depicted in figure 2.4-31.

TABLE 2.4-6.- APPROACH AND LANDING GUIDANCE

Phase	Name	Description
1	Trajectory capture	From the TAEM interface to guidance lock on to the steep glide slope.
2	Steep glide slope	The steep glide slope is tracked in elevation and azimuth.
3	Flare and shallow glide slope	The glide slope angle is reduced in preparation for landing.
4	Final flare	Final flare reduces the sink rate to near zero for touchdown.
5	Touchdown	From main wheel impact on the runway to nose gear contact.



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Figure 2.4-31.- TAEM and autoland interface.

A/L guidance phases:

- Trajectory Capture Phase - The trajectory capture phase exists from A/L guidance initialization to steep glide slope lock-on. Normal acceleration, roll, and speed brake commands are computed to place the Orbiter on the steep glide slope trajectory.
- Steep Glide Slope - Orbiter elevation and azimuth are tracked by nav while the Orbiter is on the 20° steep glide slope and A/L guidance computes normal acceleration, roll, and speed brake commands to FCS.
- Flare and Shallow Glide Slope Phase - The Orbiter initiates the flare to reduce glide slope angle for landing at an altitude of 1850 ft. Normal acceleration is modulated to accomplish the flare and transition to the shallow glide slope angle of 1.5°. This normal acceleration modulation as a function of time is illustrated in figure 2.4-32. The pullup maneuver transitions to a constant 0.5g normal acceleration circularization at an altitude of 1494 ft, and then transitions to an exponential capture at a range of 5091 ft for the predicted touchdown point. During this phase, a discrete signal is sent to FCS to retract the body flap to the trail position, and the speed brake is in the close position.

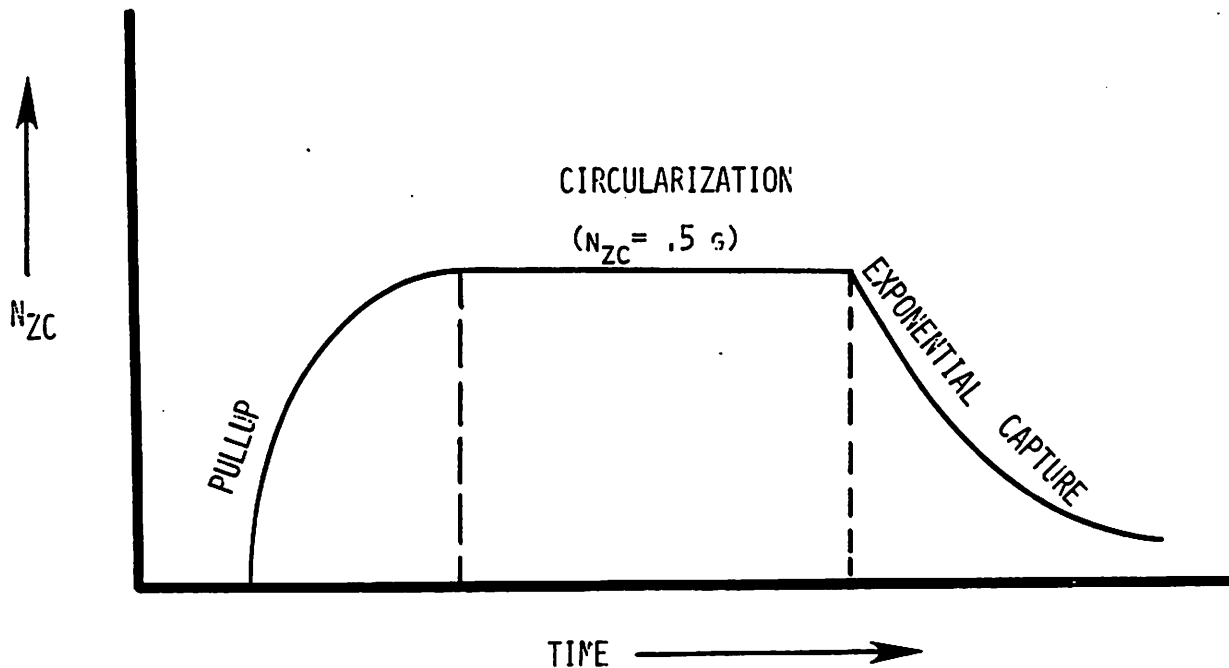


Figure 2.4-32.- Normal acceleration profile.

- Final Flare - Below an altitude of 100 ft, A/L guidance solves for the altitude and \dot{A} which will produce an $\dot{A} = 2$ ft/s at $H = 2$ ft with a 6-second flare maneuver. For altitudes less than 5 ft, a flat-turn-only capability exists for making azimuth corrections. The autoland touchdown requirements are listed in table 2.4-7.

TABLE 2.4-7.- AUTOLAND TOUCHDOWN REQUIREMENTS

Condition	Units	Ref	Tolerance (1 sigma)	Not less-- than	Not to exceed
Downrange position	ft	3000	(±400)	(1500)	(3900)
Lateral position	ft	0	(±20)	(-60)	(+60)
Touchdown velocity	KEAS	185	(±4)	174	210
Lateral velocity	ft/s	0	(±10)	(-50)	(+50)
Sink rate	ft/s	(3)	(±0.66)	1	9
Roll angle	deg	0	(±2)	-6	+6
Roll rate	deg/sec	0	(±2)	(-8)	(+8)
Crab angle	deg	0	(±3)	(-10)	(+10)
Angle of attack	deg	12	TBD	TBD	(+15)
Ground relative velocity	kn				225

OPS 3 BFS Variations From the Primary Guidance System

The BFS entry through landing guidance differs from the primary system in several areas. First and most important, the BFS does not perform any active guidance. It is essentially a display driver. This comes about from the requirement that the control mode is totally manual, although there still exists an interface with flight control for display purposes.

Since there is no autoland requirement for the BFS nor any MSBLS, BFS guidance is essentially removed even as a display driver after TAEM. Upon receiving the approach and land discrete, the ADI error needles are stowed.

The internal structure of BFS guidance differs from the primary in order to accommodate backup operation constants constraints.

2.4.9 Powered Return to Launch Site

The Powered Return to Launch Site (PRTL) is that portion of Return-to-Launch Site (RTLS) during which the SSME's are burning and typically begins when an abort is initiated and terminates at MECO. A typical RTLS profile is shown in figure 2.4-33.

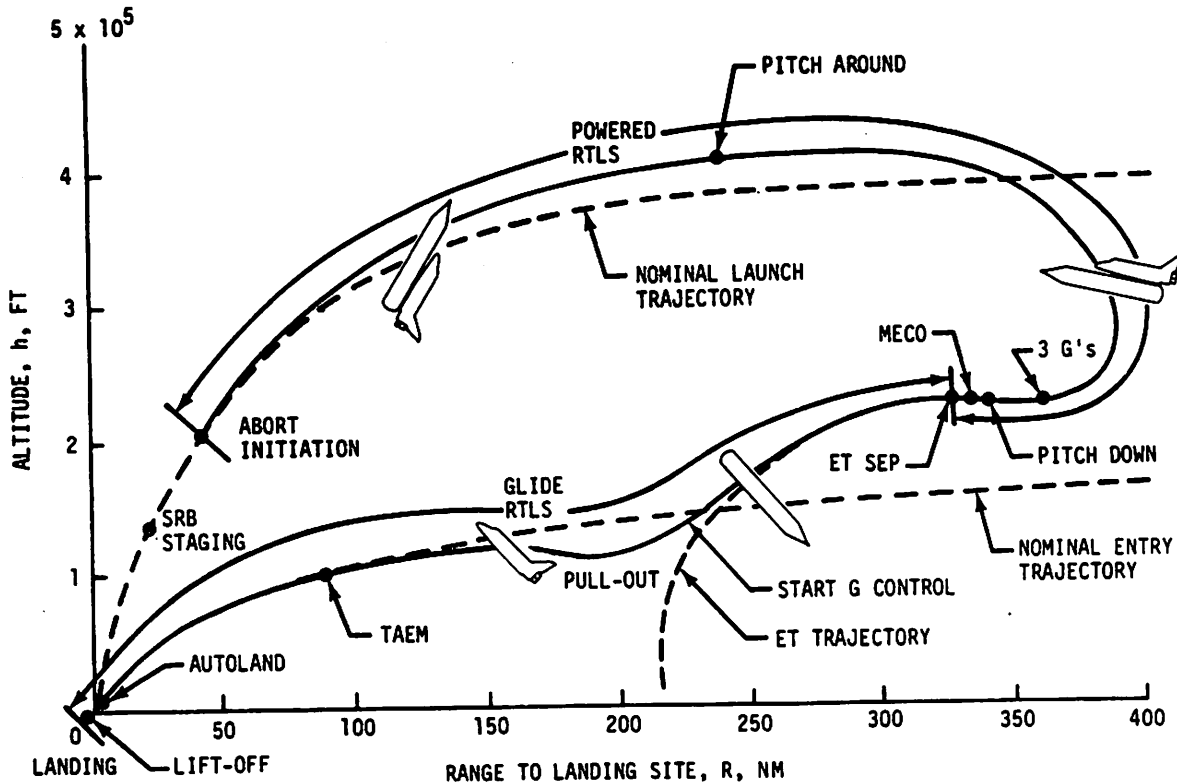


Figure 2.4-33.- Typical RTLS profile.

The PRTL guidance scheme, PEG 5, is discussed below. It is followed by a more detailed discussion of each guidance phase, fuel dissipation, pitch around, and Powered Pitch Down (PPD).

- Fuel Dissipation Phase - When an RTLS is initiated, the fuel dissipation phase begins. This phase is designed to continue powered flight downrange without violating poststaging heating constraints. This phase will also loft the Orbiter/External Tank (ET) configuration at an I-loaded initial attitude to a suitable altitude ($h \sim 400,000$ ft). Fuel dissipation is not performed for late RTLS aborts where pitch around would be performed immediately.

The Orbiter must be configured for entry during the PRTL phases. Therefore, at the beginning of powered pitch around, an automatic propellant dump is initiated using the OMS and RCS. The OMS/RCS propellant

dump will be completed prior to the PPD maneuver to minimize dynamic effects near MECO and ET separation. The Orbiter has a relatively narrow range of acceptable c.g. positions. Therefore, failure to burn the OMS/RCS propellants can result in reduced stability.

- Pitch Around Phase - The pitch around maneuver is an in-plane plus pitch maneuver (excluding yaw steering effects) performed at approximately $10^{\circ}/s$. This places the Orbiter/ET in a heads-up attitude pointing back toward the launch site (retrograde). A quick pitch around is desirable because the guidance algorithm is "fooled" by potential energy gained during the turn. A slow turn exaggerates this problem and the excess potential energy causes throttling to become erratic as guidance tries to reach the MECO condition. Once the pitch around has occurred, fly-back guidance will be initiated to null the downrange velocity and add sufficient uprange velocity for Orbiter return to launch site. The vehicle will be explicitly targeted to the altitude, flight path angle, and range-velocity line. Active SSME throttling will be performed to make final adjustments to the remaining MPS propellant to allow for a satisfactory ET separation and for 3g limiting. The pitch around maneuver of an RTLS is delayed until the MPS propellant remaining is just sufficient to achieve the required RTLS MECO targets. Therefore, for late RTLS aborts (i.e., selected late in the mode boundary), pitch around will be performed immediately.

Coarse weight control is achieved by selection of the time at which pitch around will begin. Fine weight control is achieved by MPS throttling after pitch around is complete.

- Powered Pitch Down - The final PRTLs phase is the PPD phase. This phase starts approximately 20 seconds prior to MECO where active MECO targeting is terminated. The targets used for guidance up to the PPD initiation point are biased to place the mated vehicle in a condition from which an open-loop PPD maneuver produces the desired MECO conditions. PPD takes the mated vehicle from a relatively high angle of attack to the required separation attitude ($\alpha = -40^{\circ}$) and pitch rate ($Q = 0^{\circ}/s$). MECO will then be performed when the desired cut-off velocity has been achieved based on the MECO range-velocity line. The auto guidance will stabilize the vehicle at α and Q at least 5 seconds prior to MECO.

If manual guidance is selected during PPD, the PPD task and MPS guidance cut-off task will still be completed as in auto guidance. However, if manual guidance is active during PPD, the engines will not be commanded to minimum power level unless the crew returns to auto guidance or manually throttles the SSME's.

- RTLS MECO - Active targeting for MECO begins when RTLS abort is activated and terminates when the PPD phase occurs. Specific MECO targets are range, velocity, and angle of attack. Targets provide a set of desirable energy conditions for MECO and subsequent glide to the runway. A target altitude consistent with velocity is selected to keep dynamic pressure within limits. Prior to MECO, engine throttling to 65 percent MPL should occur. The GPC's signal MECO when the MPS countdown timer reaches zero seconds or an MPS propellant depletion signal is received.

If an auto shutdown does not occur, shutdown must be accomplished manually. If the engines are not shutdown at MECO time, a ΔR error will be generated due to the vehicle possessing excessive energy. With a normal RTLS MECO, some excessive energy is planned. With an MPS overburn, the additional energy will require a much larger S-turn.

- MPS DUMP - Failure to dump MPS propellants after ET separation will result in a c.g. position further aft than desired. MPS dumping is automatic during RTLS and should be completed prior to 100,000 ft. LH2 is dumped overboard through a special line on the port aft fuselage side. Approximately 242 lb of LH2 is dumped. LO2 is vented through the SSME and approximately 1350 lb is vented.

Powered Explicit Guidance (PEG 5): Although the RTLS abort might be required for a failure in first stage, the actual RTLS sequence and guidance is not initiated until after SRB separation. The PEG 5 is specifically intended for the powered portion of an RTLS abort. The target constraints include altitude, flight path angle, and a relative velocity which is a linear function of the range to the landing site.

The fuel dissipation phase is performed in the plane in which the abort initiation occurs. A new plane is recomputed each cycle for the RTLS as the vehicle reverses direction. This new desired plane is defined as one containing the current vehicle position and the Kennedy Space Center (KSC) position vector (target point to prime heading alignment circle) (fig. 2.4-34).

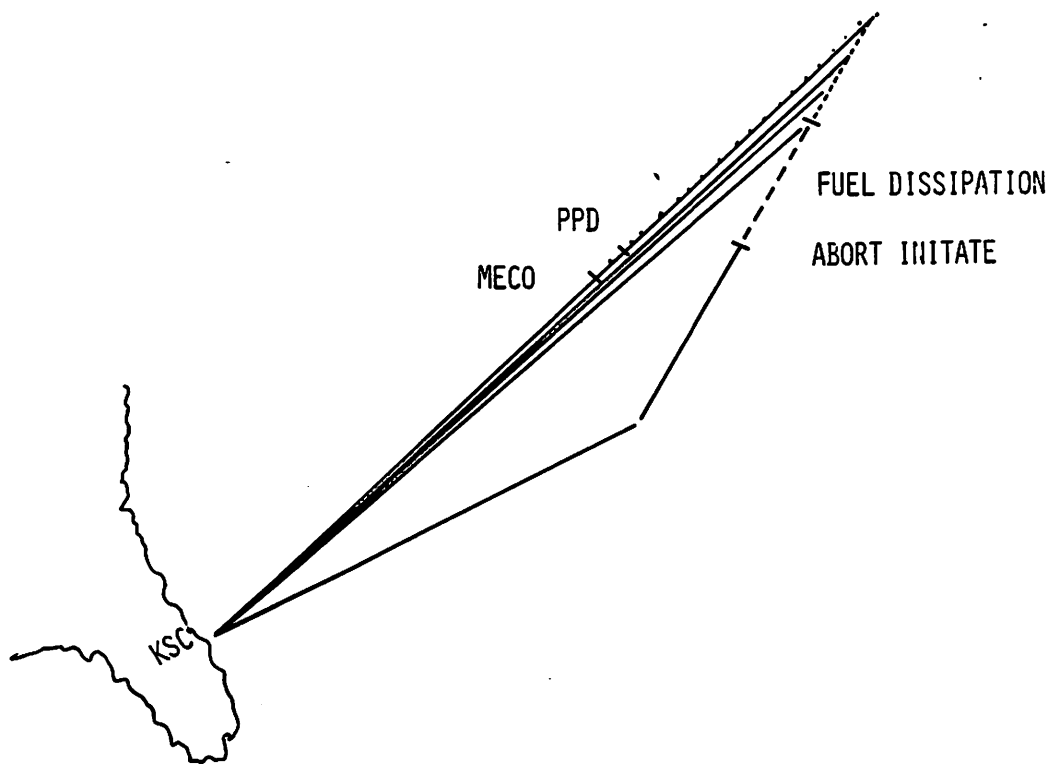


Figure 2.4-34.- Fuel dissipation phase.

On each PEG 5 cycle, a velocity-to-go (V_{GO}) is calculated representing a vector difference between the target velocity and the current velocity. Even though the vehicle is thrusting away from the desired target, no turn is initiated because fuel dissipation is still being done. PEG 5 will predict the RTLS MECO target position and velocity cyclically, and it also knows the desired MECO target (fig. 2.4-35). A vector difference is then calculated (V_{MISS}) which represents the current guidance error (fig. 2.4-36). A correction routine is then performed using V_{MISS} which will reorient the guidance to achieve the desired cut-off conditions. The V_{MISS} calculated each cycle is tested against an I-loaded threshold. If the magnitude of V_{MISS} is within the threshold, the PEG 5 solution is considered converged. If not, the guidance cycles to try and minimize V_{MISS} until it converges. After approximately three cycles of guidance with no convergence, guidance will recycle to the beginning and start over. However, the last valid set of guidance calculated data will be used to eliminate any transients. PEG computations stop at the initiation of the PPD phase. The targets used for guidance to the PPD point are biased to place the Orbiter in a condition from which an open-loop PPD maneuver produces the desired MECO conditions.

Manual throttle: Transferring back to AUTO throttle during RTLS flyback leaves the throttle at the last commanded manual setting. In other words, there is no auto throttling for fuel depletion if AUTO throttle has been reselected after powered pitcharound.

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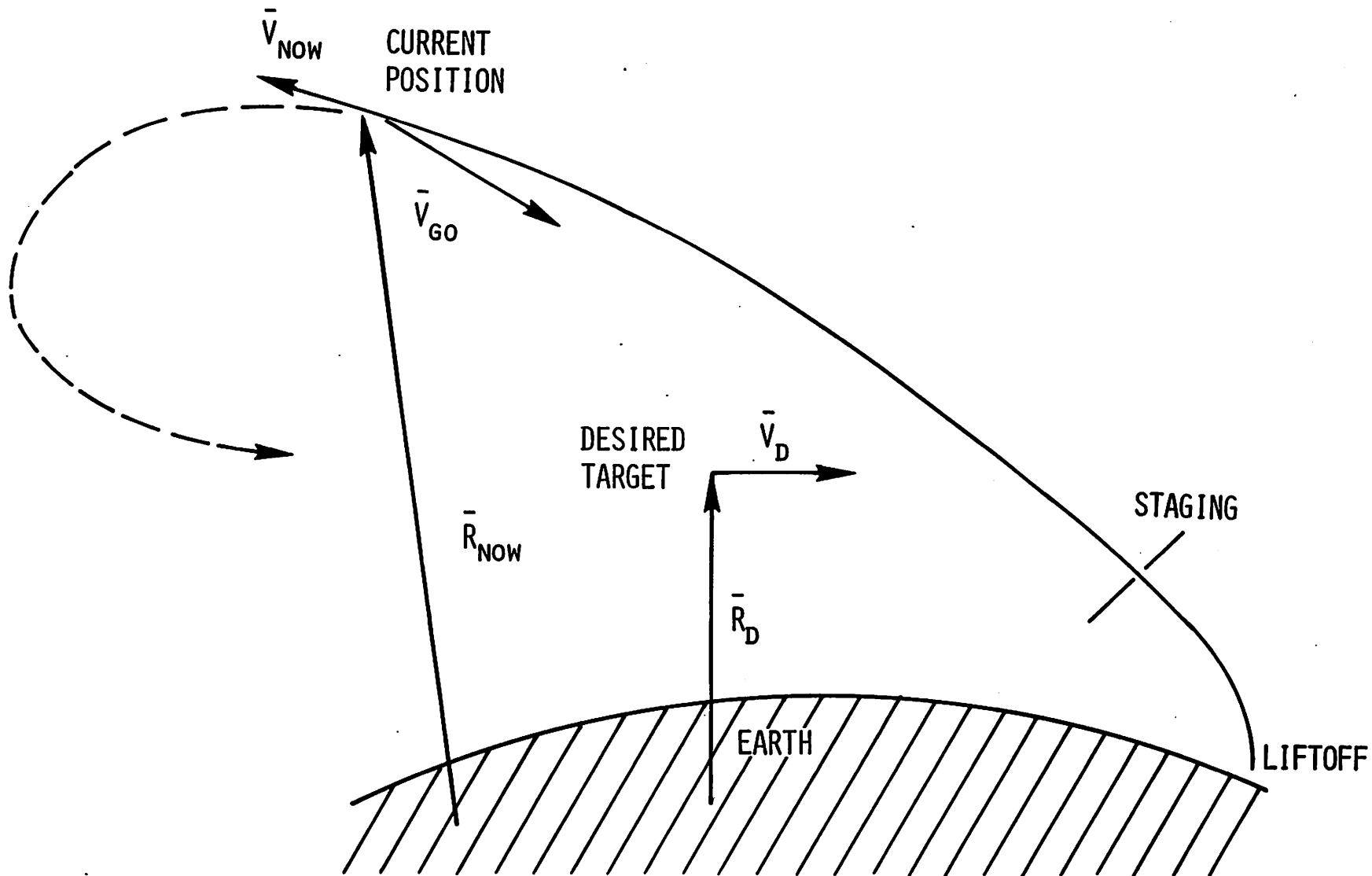
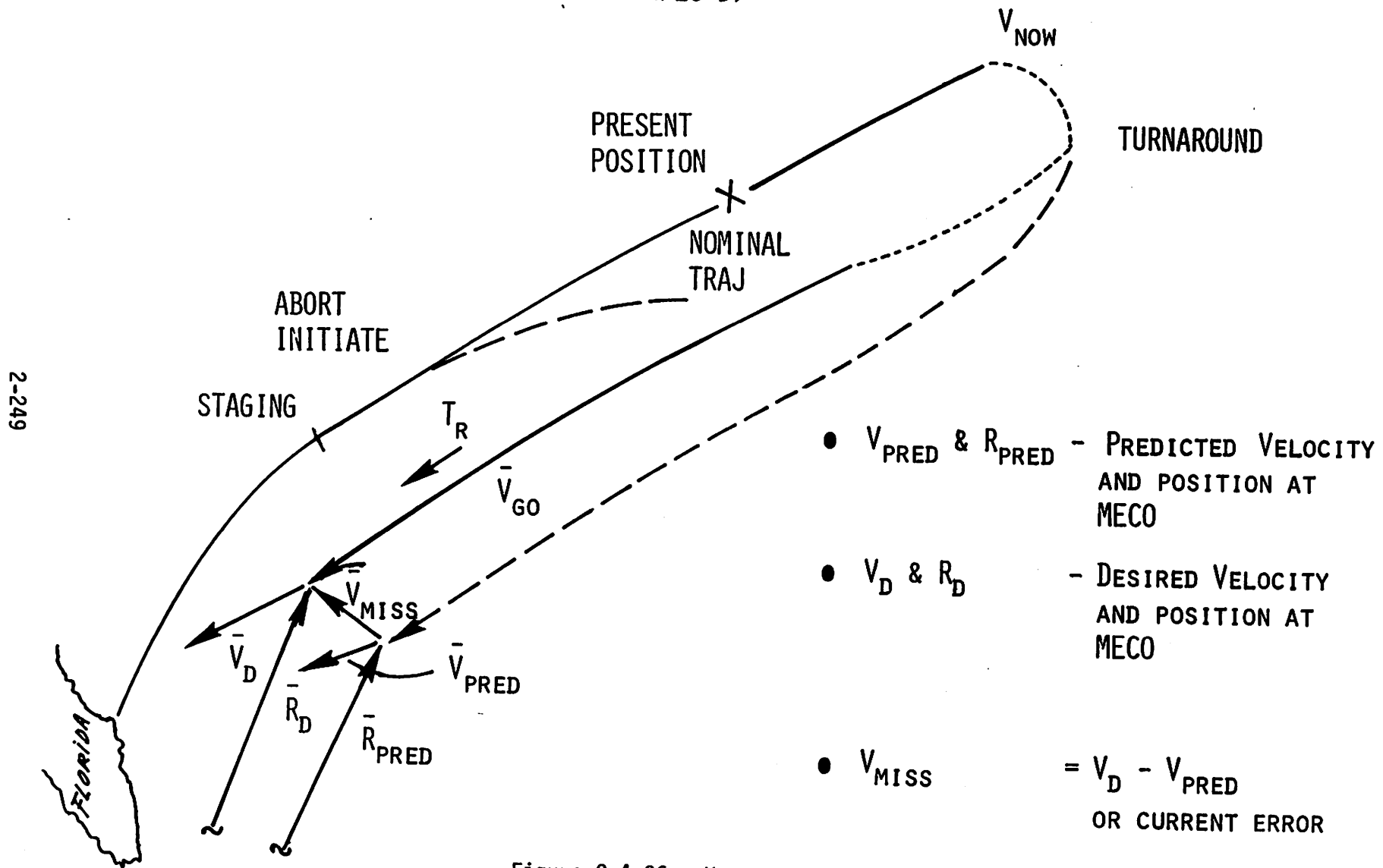


Figure 2.4-35.- The MECO target.

RTLS PHASE
GUIDANCE
(PEG 5)



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- V_{PRED} & R_{PRED} - PREDICTED VELOCITY AND POSITION AT MECO
- V_D & R_D - DESIRED VELOCITY AND POSITION AT MECO
- $V_{MISS} = V_D - V_{PRED}$ OR CURRENT ERROR

Figure 2.4-36.- V_{MISS} .

2.4.10 GRTLS Guidance

GRTLS TAEM: With the termination of PEG 5 guidance, nominal guidance during entry begins. This guidance includes TAEM and autoland and is responsible for the ranging and heading of the Orbiter to be near the predetermined TAEM interface target.

The TAEM phase brings the Orbiter to a final state near the runway centerline. GRTLS consists of eight flight phases which are by necessity slightly different from those utilized during a normal entry. The eight phases are:

- Alpha recovery
- Nz hold
- Alpha transistion
- S-turn
- Acquistion
- Heading alignment
- Pre-final
- A/L

The first seven of these phases are contained within the GRTLS TAEM guidance and of these, the first three phases are unique to the GRTLS abort mode. The next four phases differ very little from nominal TAEM phases and will be discussed later. The approach and landing phase, which is not considered as a part of TAEM, is identical to the nominal case.

GRTLS TAEM phases: The first three phases of GRTLS TAEM are required to stabilize the trajectory following Orbiter/ET separation and to introduce ranging. The last four phases accomplish finer ranging as a function of energy per unit weight and differ from nominal TAEM phases in the following ways:

- Energy per unit weight, altitude, and dynamic pressure references are extended to higher speeds to include all possible conditions existing where the trajectory is stabilized following the load relief maneuver.
- To provide greater energy control, the bank angle limit during the S-turn is increased from 30 to 50 degrees.

Other than these differences, these four phases of TAEM are identical to normal entry TAEM (section 2.4.8, TAEM guidance phases).

It is the additional three phases that make GRTLS TAEM different from nominal TAEM.

Alpha recovery: The Alpha recovery phase begins at the end of the Orbiter/ET -Z translational separation burn and maneuvers the Orbiter attitude from the low angle of attack required for separation to the high angle of attack and lift-vector-up bank attitude, in preparation for atmospheric flight. This attitude is maintained until transistion to the Nz hold phase. For STS-1 the commanded RTLS ET SEP angle of attack will be 90°. This will ensure that G/C STEER will issue a RATE command for a sufficient length of time to allow the Orbiter to reach MM 602 ($\alpha = 100^\circ$).

The constant angle of attack, however, will be 50° which minimizes the peak dynamic pressures. Following the separation burn the Orbiter will pitch up at $2^\circ/\text{sec}$ until the transition to MM 602. The criteria for MM 602 is $\alpha = 10^\circ$ and 10 sec have passed since ET SEP. It is now that the fuel and OX dumps begin and end prior to $V_{REL} = 4500 \text{ ft/s}$. Attitude control during the alpha recovery phase is accomplished using both aft RCS thrusters and aerodynamic surface effectors. The elevons and body flap are activated on transition to MM 602.

The recovery angle of attack (50°) is maintained until the normal force (N_z) increases to within 0.35 g's of the desired 2g limit. Switching to the N_z hold phase within 0.35 g's provides a smooth transition into the N_z hold phase with minimum N_z overshoot.

N_z hold phase: The N_z hold phase maintains the normal load factor at a predetermined value by reducing the angle of attack at a rate to achieve this flight condition. To minimize aerodynamic loads, the Orbiter is maintained in a lift vector up attitude during this phase. For STS-1, the commanded N_z is a constant 2g allowing a 0.5g margin for the structural design limit. Also during this phase, alpha must be greater than 19° to allow sufficient lateral control ability margins, particularly in the presence of aerodynamic variations. To prevent execution of maneuvers that could induce subsequent high aero loads and unfavorable flight conditions, all ranging techniques including S-turns are inhibited during this phase. After arresting the descent rate $\dot{h} = 250 \text{ ft/s}$, the angle-of-attack profile is smoothly merged with the optimum profile for this flight regime. If alpha is too large when the descent rate is arrested, a $1^\circ/\text{sec}$ pitch turn rate is incorporated until the optimum angle of attack is reached. If the angle of attack is too low, the opposite occurs, when the optimum angle of attack is reached, guidance transitions to the alpha transition phase.

Alpha transition phase: The alpha transition phase is used to further stabilize the vehicle to a desired alpha and also introduces range control. During this phase, the guidance commands a predetermined open loop angle-of-attack profile and a bank angle designed to direct the velocity vector toward the nearest HAC and execute S-turns if necessary. Because the alpha recovery and N_z hold phases are both open loop with regard to energy management, and sufficient energy must be available to cope with open-loop energy losses (headwinds, L/D variations). GRTLs is intentionally targeted to produce an excess energy state at the initiation of the alpha transition phase. Therefore, an S-turn is a normal part of the alpha transition phase. Additional S-turning is accomplished during the S-turn phase if it is required. Guidance switches to the near-nominal TAEM phases when $M = 3.2$.

BFS RTLS guidance: BFS RTLS guidance utilizes PEG 5 guidance just as in PASS. However, during ascent (powered RTLS), only the auto mode is possible. Error needles are displayed, but CSS is not engagable. Also, since MLS is not available, the approach and landing is CSS only. There is no autoland capability. It is for this reason that TAEM guidance is not terminated until 2000 ft. Normally, autoland (PASS) guidance is initiated at 5000 ft but BFS doesn't have this capability. Below 2000 ft there is no guidance available, not even needles. All phases of GRTLs TAEM in BFS are identical to PASS GRTLs TAEM.

2.5 FLIGHT CONTROL SOFTWARE

The flight control section is divided into three major sections: first stage, second stage, and orbit insertion. Since second stage control is nearly identical to first stage, only differences are discussed:

2.5.1 First Stage - Automatic Control System

In the automatic mode, the control system processes steering parameters from guidance along with feedback signals from the body mounted sensors (RGA, AA). This processing results in the generation of engine bell deflection commands both for the SSME's and the SRB's (aerosurface are not used for control during ascent).

A summary of the inputs and outputs of the major subfunctions within flight control software is shown in table 2.5-1.

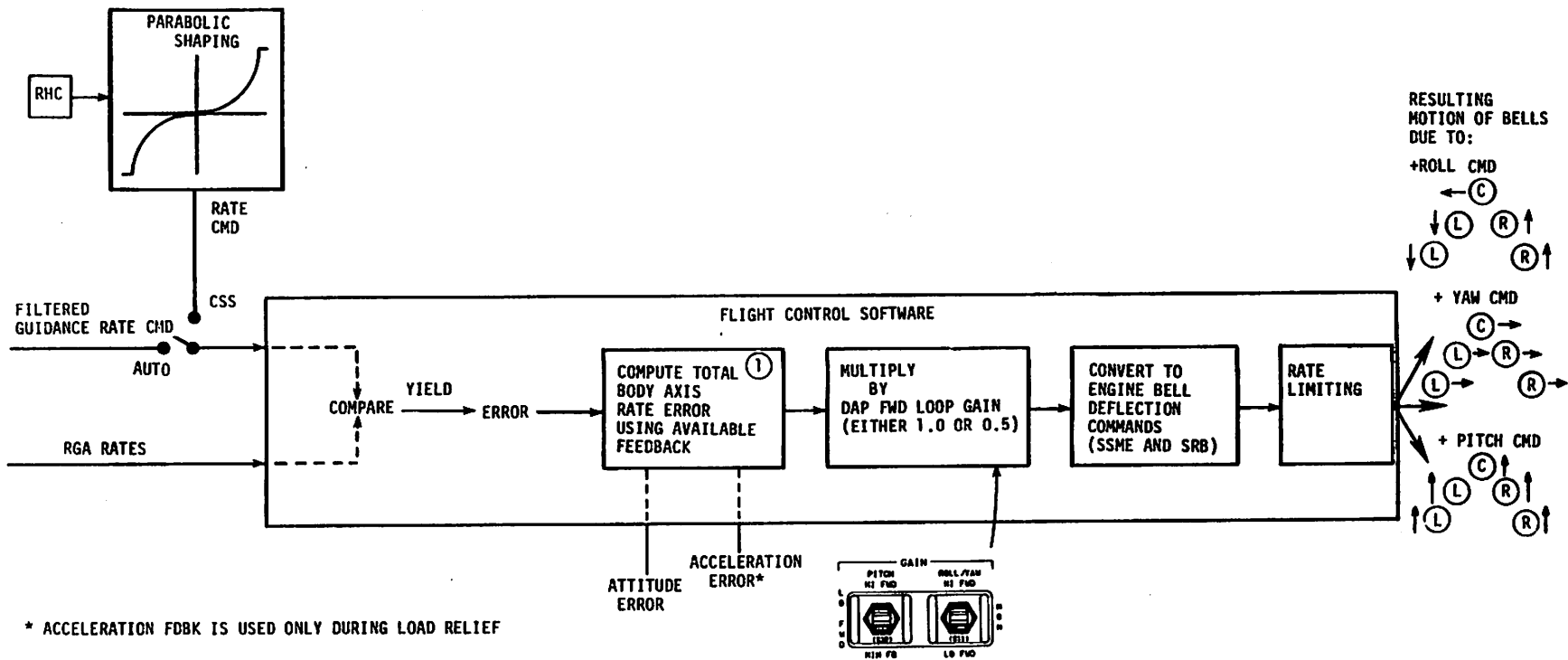
TABLE 2.5-1.- FLIGHT CONTROL SOFTWARE SUMMARY, MM 102

Subfunction	Inputs/source	Outputs/destination	Purpose
Guidance/Control Steering (G/C STEER)	<ul style="list-style-type: none"> Attitude quaternion/attitude processor Commanded attitude quaternion/guidance 	<ul style="list-style-type: none"> Attitude error/digital autopilot Attitude error/ADI processor (for display) Rate command/digital autopilot 	<ul style="list-style-type: none"> Filter (smooth) guidance commands Compute attitude error (error between current attitude and filtered commanded attitude) Generate rate commands from commanded attitude Convert the above to body coordinates
Digital Autopilot (DAP) Thrust Vector Control (TVC)	<ul style="list-style-type: none"> Attitude errors/G/C STEER Rate commands/G/C STEER ORB RGA roll rate/ORB RGA RM SRB RGA pitch, yaw rates/SRB RGA RM Normal and lateral acceleration/I-load Normal and lateral accelerations/accelerometers RM 	<ul style="list-style-type: none"> SSME bell deflection commands/MPS TVC CMD SOP SRB bell deflection commands/SRB TVC CMD SOP 	<ul style="list-style-type: none"> Generate engine bell deflection commands required to zero out attitude error Using acceleration error-as primary feedback, perform load relief in high q region
Aerosurface control	<ul style="list-style-type: none"> Elevon actuator primary ΔP's/ΔP feedback SOP Elevon schedule/I-load Body flap position/body flap position feedback SOP 	<ul style="list-style-type: none"> Elevon commands (slaved)/AERO ACT SOP Body flap rate drive command/AERO ACT SOP Rudder command/AERO ACT SOP Speed brake command/AERO ACT SOP 	<ul style="list-style-type: none"> For hinge moment load relief, drive elevons to I-load position (bias I-loaded value in event of large ΔP) Command rudder and speed brake to zero deflection. Drive body flap to zero if it exceeds deadband about zero degrees
RCS JET roll control (used only in event of dual SSME failure)	<ul style="list-style-type: none"> Body axis roll error/TVC 	<ul style="list-style-type: none"> RCS JET ON/OFF command/ASC RCS CMD SOP 	<ul style="list-style-type: none"> Performs roll control of vehicle in event of dual SSME failure RCS not available until MET > 90 sec and both Pc's < 400 or MET > 120 sec

Guidance/control steering interface: The function of Guidance/Control Steering Interface (G/C STEER) is to filter or smooth the steering parameters received from guidance. The steering parameters, in the form of desired thrust direction, are converted to body coordinates and limited in rate and acceleration. Body axis attitude commands and attitude rate commands are extracted from the steering parameters. Roll, pitch, and yaw attitude errors are computed by comparing command to actual (attitude data from IMU). Attitude errors and rate commands are sent to the DAP. Attitude errors are also displayed on the ADI.

Digital auto pilot: The DAP processes attitude errors and rate commands from G/C STEER along with sensor input in order to generate engine bell deflection commands. A functional overview of the DAP processing is shown in figure 2.5-1. Additional information is provided in figures 2.5-2, 2.5-3, and 2.5-4. Detailed figures of the control loops are referenced.

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* ACCELERATION FDBK IS USED ONLY DURING LOAD RELIEF

① see figures 2.5-2
2.5-3
2.5-4

Figure 2.5-1.- Ascent DAP functional overview (first stage).

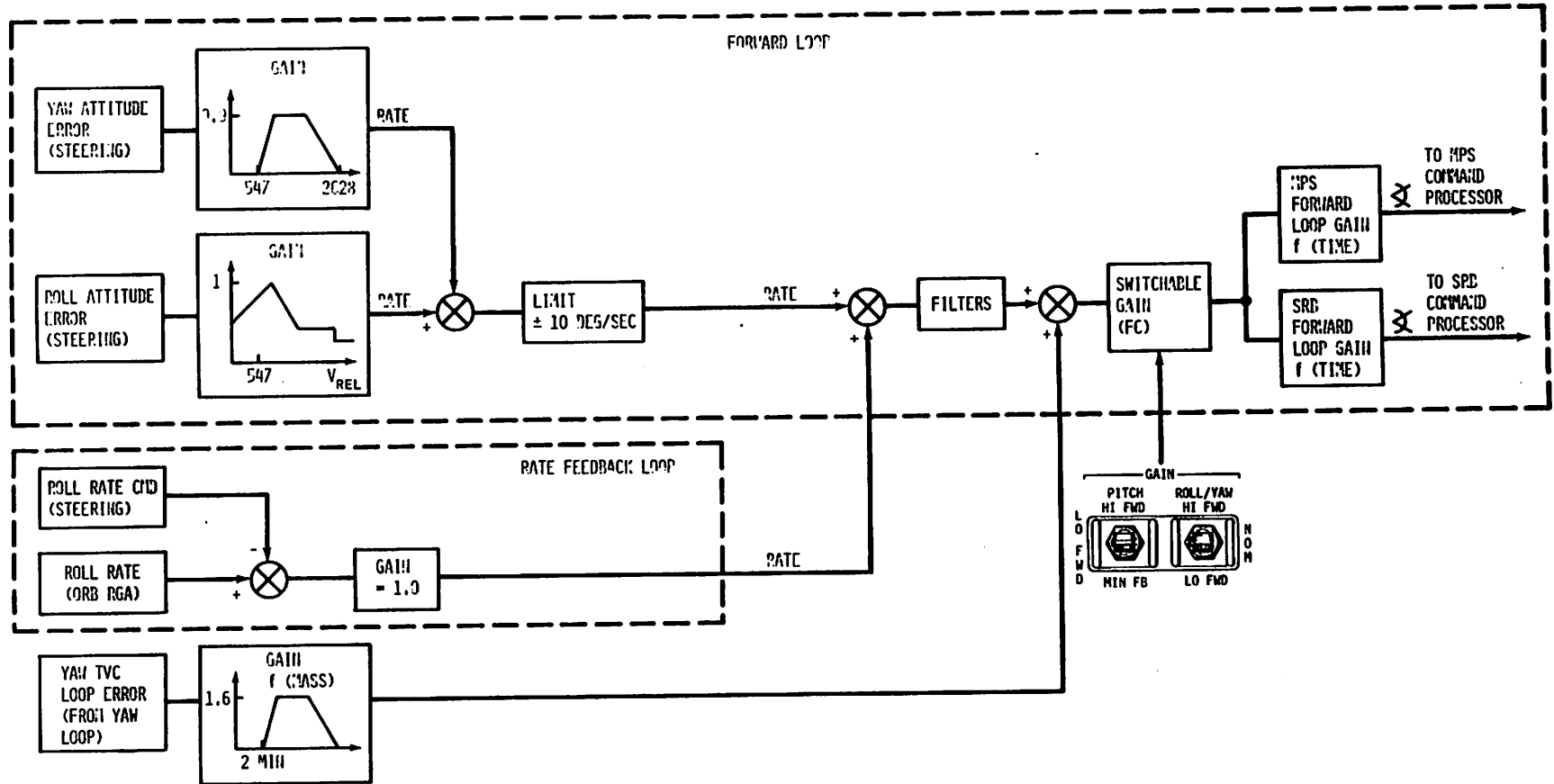


Figure 2.5-2.- First stage roll loop.

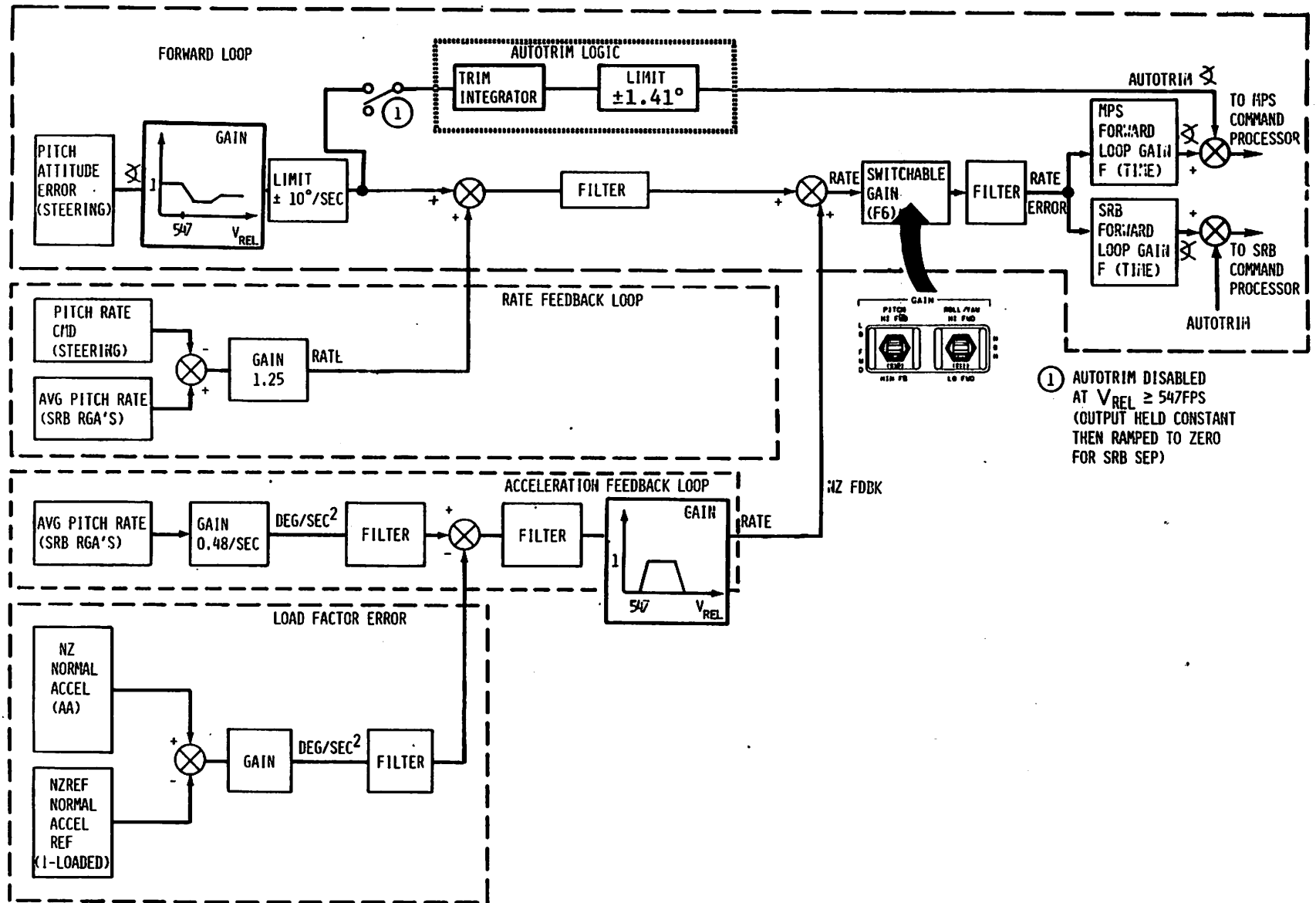


Figure 2.5-3.- First stage pitch loop.

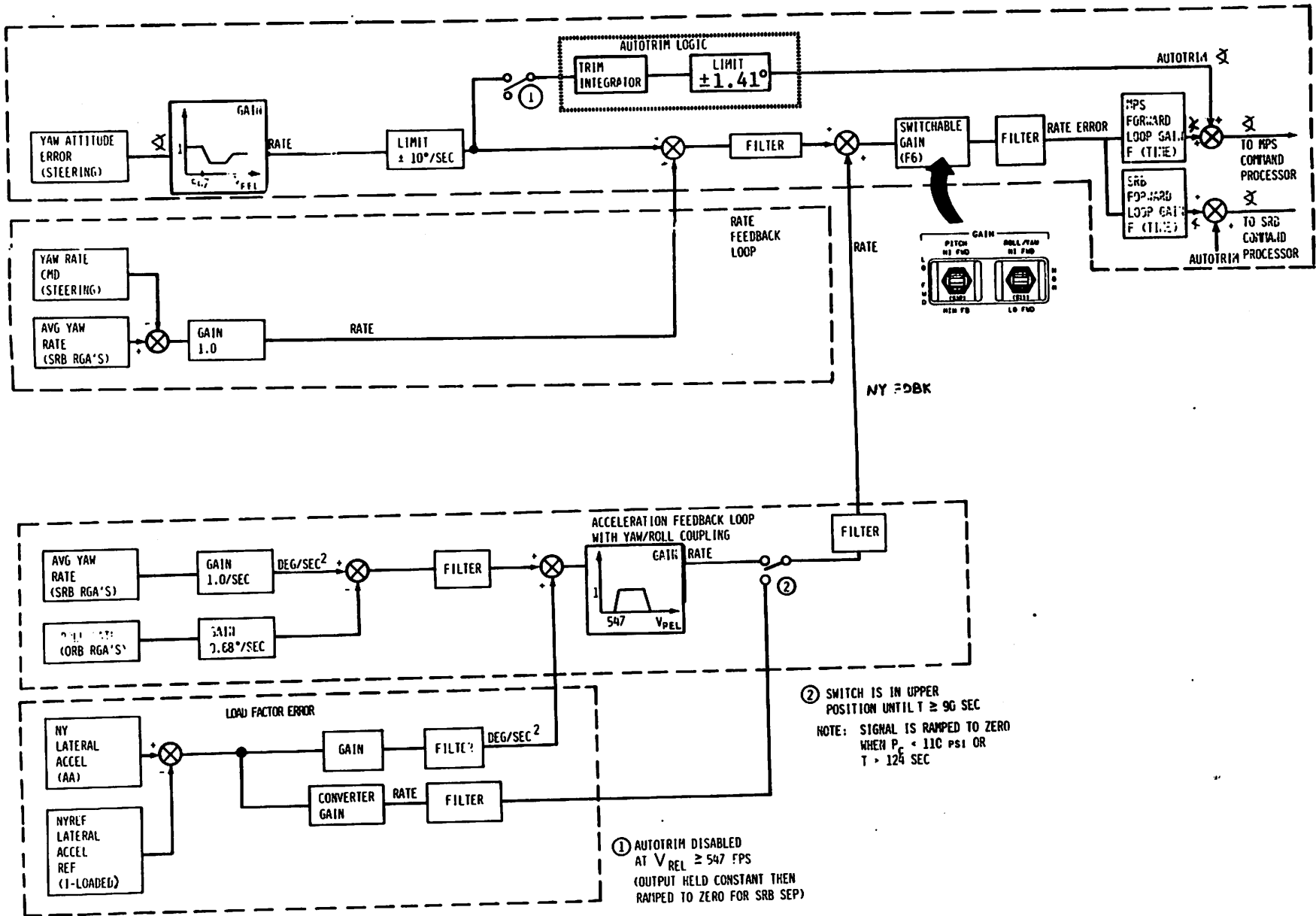


Figure 2.5-4.- First stage yaw loop.

The GAIN switches reduce the error signal (in the appropriate axis) by 50 percent when deflected down. When in the middle and upper positions the signal is not affected. (Note: the appropriate GAIN ENA PBI must be depressed to enable the gain switches).

Following the computation of body axis attitude commands, the DAP processes these commands (roll, pitch, and yaw) and converts them to engine bell (SSME and SRB) deflection commands. The SRB's receive the appropriate combination of tilt and rock commands and the SSME's receive the appropriate combination of (engine) pitch and (engine) yaw commands. The usage of the SSME's for vehicle control is summarized in table 2.5-2. An x means that that actuator (engine deflection) is used for control in the corresponding axis.

TABLE 2.5-2.- SUMMARY OF SSME USAGE FOR VEHICLE CONTROL

Engine conditions	Body axis control	Left SSME		Center SSME		Right SSME	
		Pitch	Yaw	Pitch	Yaw	Pitch	Yaw
All 3* ME'S operating	R	X			X	X	
	P	X		X		X	
	Y		X		X		X
Center** ME failed	R	X				X	
	P	X				X	
	Y		X				X
Left or right** ME failed	R	X	X	X	X	X	X
	P	X		X		X	
	Y		X		X		X
Any two *** ME's failed	P	X		X		X	
	Y		X		X		X

*bell deflection rate limited to 80/sec

(loss of one APU - 50/sec)

(loss of two APU's - 3.50/sec)

**failed engine deflection rate limited to 20/sec while driving the FAIL position (I-LOAD).

***Roll control is provided by RCS

Load relief: Load relief is performed by utilizing acceleration feedback from the AA's during the high \bar{q} region (beginning at $V_{REL} = 547$ ft/sec). During this time, the control loops become more sensitive to acceleration error than attitude error. This results in roll, pitch, and yaw commands that orient the vehicle so as to reduce normal and lateral accelerations (results in small dispersions from guidance commands).

Manual thrust vector control: To assume Manual Thrust Vector Control (MTVC), the CSS PBI's must be depressed. Once MTVC is selected, the rate commands from the RHC replace the rate commands from G/C STEER (guidance) (fig. 2.5-1). The attitude error is computed by integrating RGA rates. MTVC is implemented as discrete rate/attitude hold. The limits on the initiation of attitude hold are:

rate <10.0 deg/sec roll, yaw
< 1.0 deg/sec pitch

attitude <20.0 deg roll, pitch, yaw
error

Aerosurface Control: The elevons are driven to I-loaded positions during first stage to alleviate hinge moments loads in the load relief region. The I-loaded profile is dependent upon V_{REL} . The elevon profile is shown in figure 2.5-5 (note that the elevons are slaved).

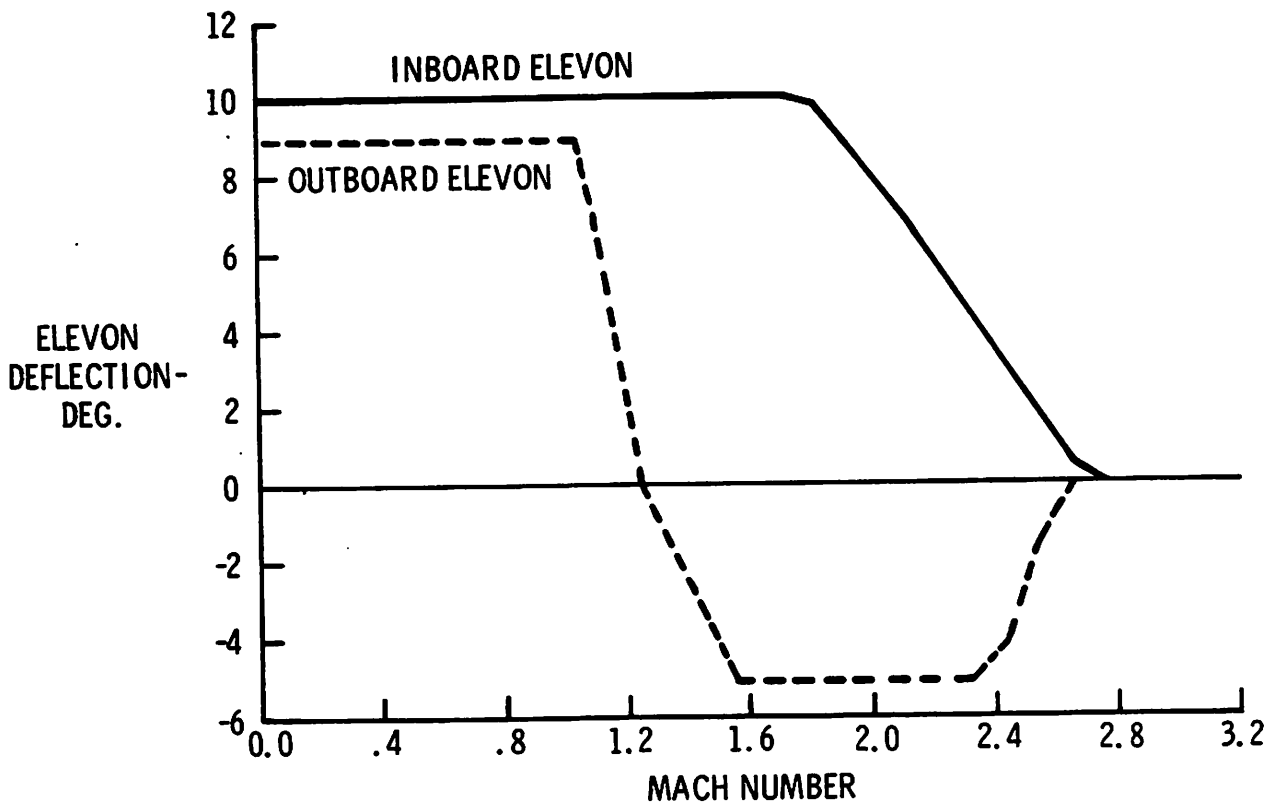


Figure 2.5-5.- Elevon profile.

The profile can be biased by software in the event that the primary ΔP on any of the actuators exceeds an I-loaded value. Once the ΔP is reduced, the bias is removed.

The rudder, speed brake, and body flap are held in place and not used during ascent.

RCS jet roll control: In the event of two SSME failures, the computed roll error is routed to the RCS jet logic for roll control if MET >90 sec. The remaining SSME is used only for pitch and yaw control.

2.5.2 Second Stage Flight Control

The flight control software modes from first stage control to second stage control at SRB SEP command. Attitude hold is commanded through separation and until second stage guidance (PEG 4) is converged. When PEG 4 has converged, flight control begins processing of the thrust vector commands in the same manner as first stage. A functional overview of the DAP processing is shown in figure 2.5-6. A flight control software summary of inputs and outputs during MM 103 is provided in table 2.5-3.

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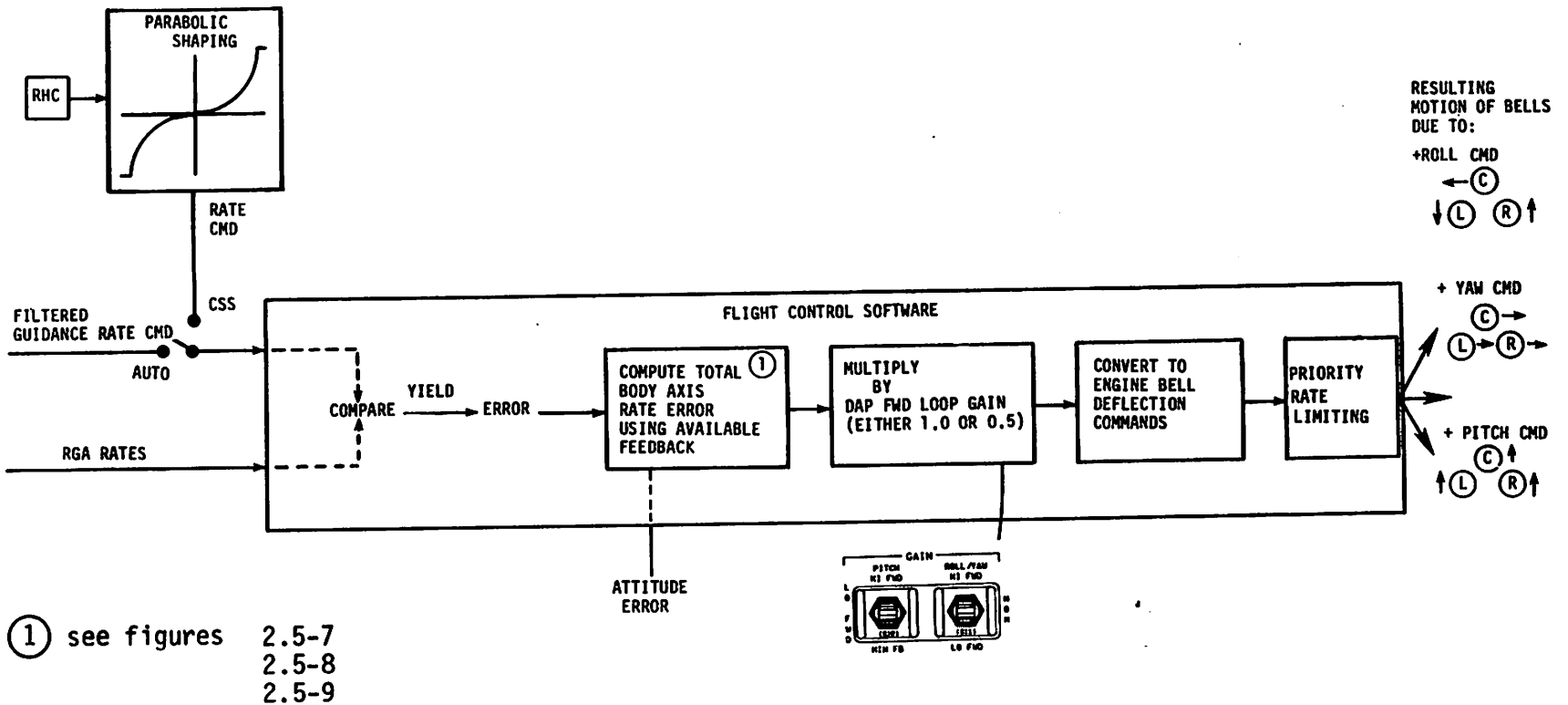


Figure 2.5-6.- Ascent DAP functional overview (second stage).

TABLE 2.5-3.- FLIGHT CONTROL SOFTWARE SUMMARY, MM 103

Subfunction	Inputs/source	Outputs/destination	Purpose
Guidance/Control Steering (G/C STEER)	<ul style="list-style-type: none"> ● ALT/ascent UPP ● VREL/ascent UPP ● Shuttle position/ascent UPP ● Wing angle criteria/guidance ● Thrust steering parameters/guidance 	<ul style="list-style-type: none"> ● Attitude error/digital autopilot ● Attitude error/ADI processor (for display) ● Rate command/digital autopilot 	<ul style="list-style-type: none"> ● Filter (smooth) guidance commands ● Command wings level ● Compute attitude error (error between current attitude and filtered steering command) ● Generate rate commands from commanded attitude ● Convert the above to body coordinates
Digital Autopilot (DAP) Thrust Vector Control (TVC)	<ul style="list-style-type: none"> ● Attitude errors/G/C STEER ● Rate commands/G/C STEER ● ORB RGA roll, pitch, yaw rate/ORB RGA RM 	<ul style="list-style-type: none"> ● SSME bell deflection commands/MPS TVC CMD SOP 	<ul style="list-style-type: none"> ● Generate engine bell deflection commands required to zero out attitude error
Aerosurface control	<ul style="list-style-type: none"> ● Body flap position/body flap position feedback SOP 	<ul style="list-style-type: none"> ● Body flap drive command/AERO ACT SOP 	<ul style="list-style-type: none"> ● Drive body flap to zero if it exceeds deadband about zero degrees ● Body flap logic is active until SSME stow complete
RCS JET roll control (used only in event of dual SSME failure)	<ul style="list-style-type: none"> ● Body axis roll error/TVC 	<ul style="list-style-type: none"> ● RCS JET ON/OFF command/ASC RCS CMD SOP 	<ul style="list-style-type: none"> ● Performs roll control of vehicle in event of dual SSME failure

G/C STEER: The guidance thrust vector commands are filtered (smoothed) in G/C STEER. Prior to filtering the guidance thrust vector commands, alpha (α) and beta (β) are limited to prevent heating problems. This limiting is done until MET > 150 sec or altitude > 200,000 ft. The α limit is +70/-30. Sideslip (β) is held to 00. After being filtered and limited, the commands are converted to body axis attitude and rate commands.

The body axis attitude error is computed by comparing command to actual (via IMU). The attitude error is displayed on the ADI and sent to the DAP for further processing. The rate command is sent to the DAP.

Digital autopilot: The roll, pitch, and yaw DAP control loops are shown in figures 2.5-7, 2.5-8, and 2.5-9.

The GAIN switches reduce the error signal (in the appropriate axis) by 50 percent when in the lower position. When in the middle and upper positions, the signal is not affected. (Note: the appropriate GAIN ENA PBI must be depressed to enable the GAIN switches).

SSME deflections for vehicle roll, pitch, and yaw control are summarized in table 2.5-4.

TABLE 2.5-4.- SUMMARY OF SSME USAGE FOR VEHICLE CONTROL

Engine conditions	Body axis control	Left SSME		Center SSME		Right SSME	
		Pitch	Yaw	Pitch	Yaw	Pitch	Yaw
All 3* ME'S operating	R	X			X	X	
	P	X		X		X	
	Y		X		X		X
Center** ME failed	R	X				X	
	P	X				X	
	Y		X				X
Left or right** ME failed	R	X	X	X	X	X	X
	P	X		X		X	
	Y		X		X		X
Any two *** ME's failed	P	X		X		X	
	Y		X		X		X

*bell deflection rate limited to 80/sec
(loss of one APU - 50/sec)
(loss of two APU's - 3.50/sec)

**failed engine deflection rate limited to 20/sec while driving the FAIL position (I-LOAD).

***Roll control is provided by RCS

2-265

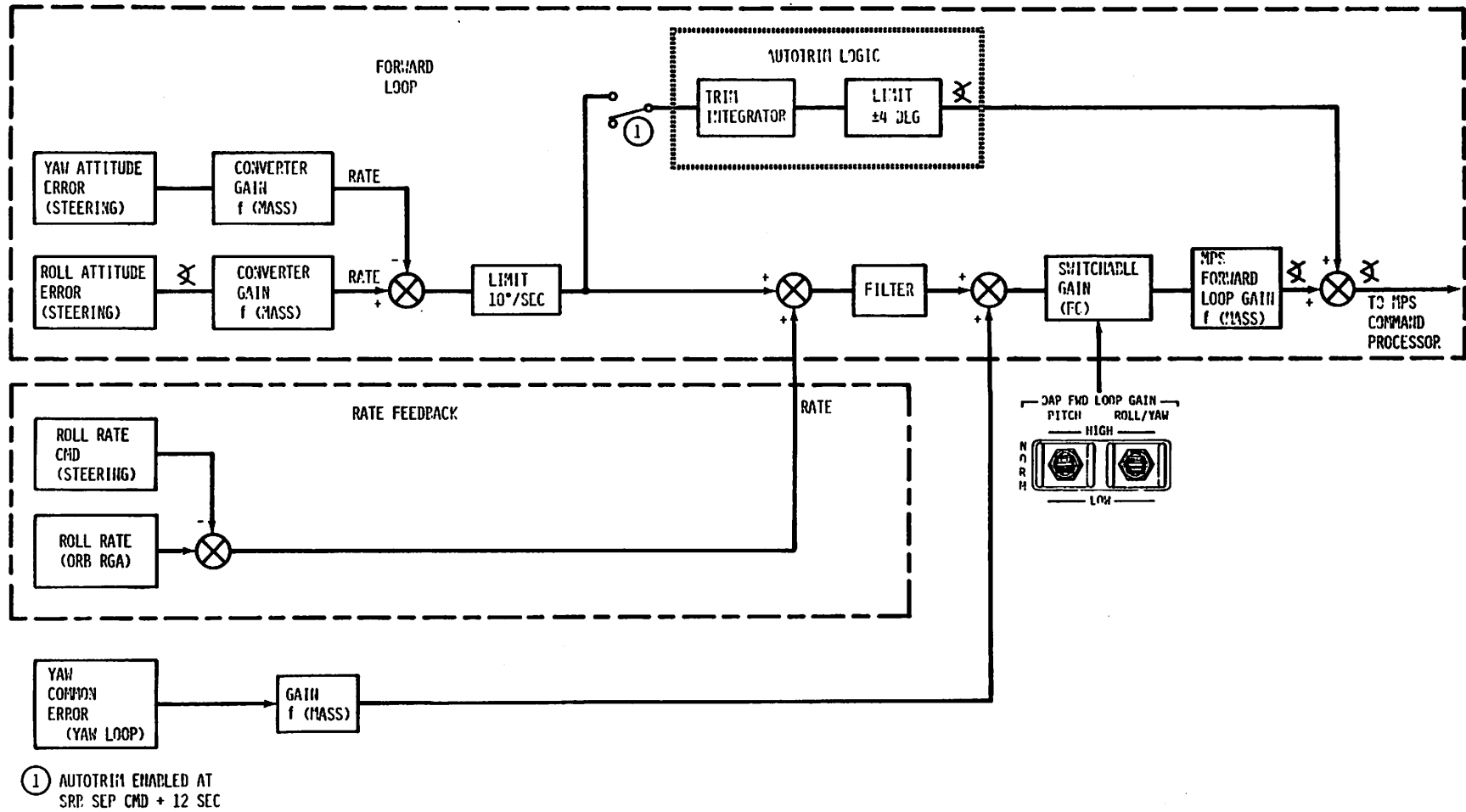


Figure 2.5-7.- Roll loop (second stage).

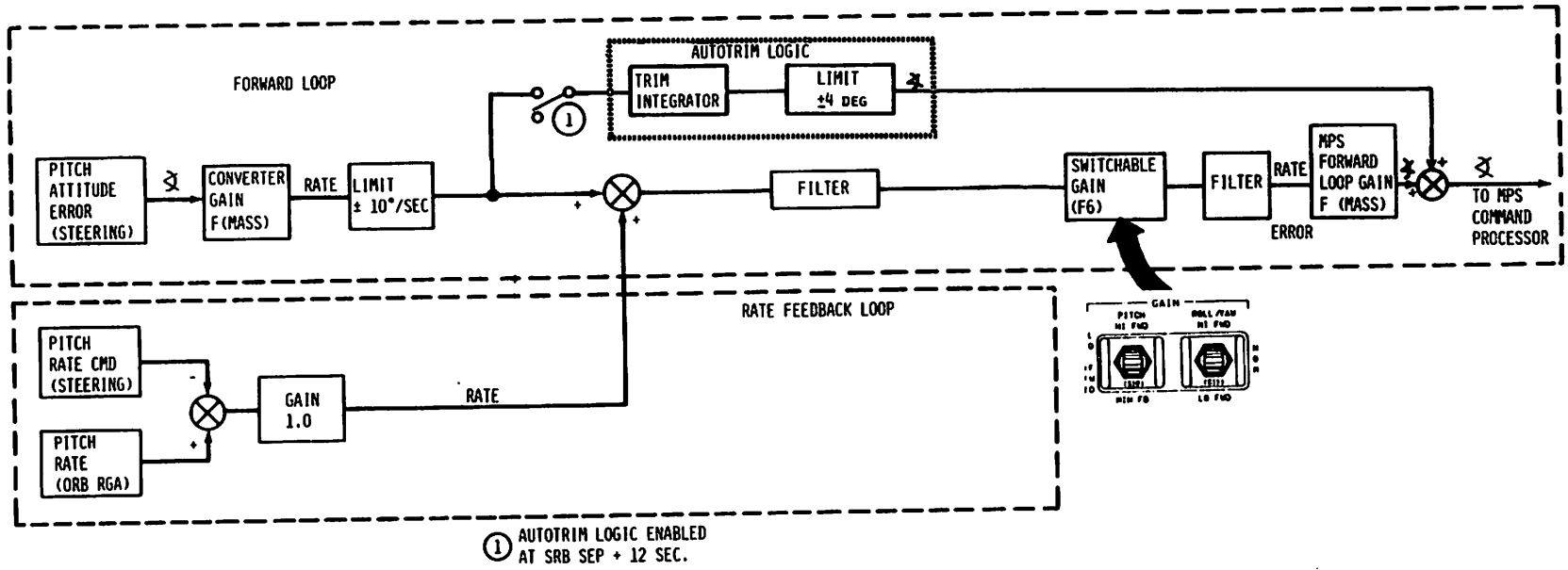
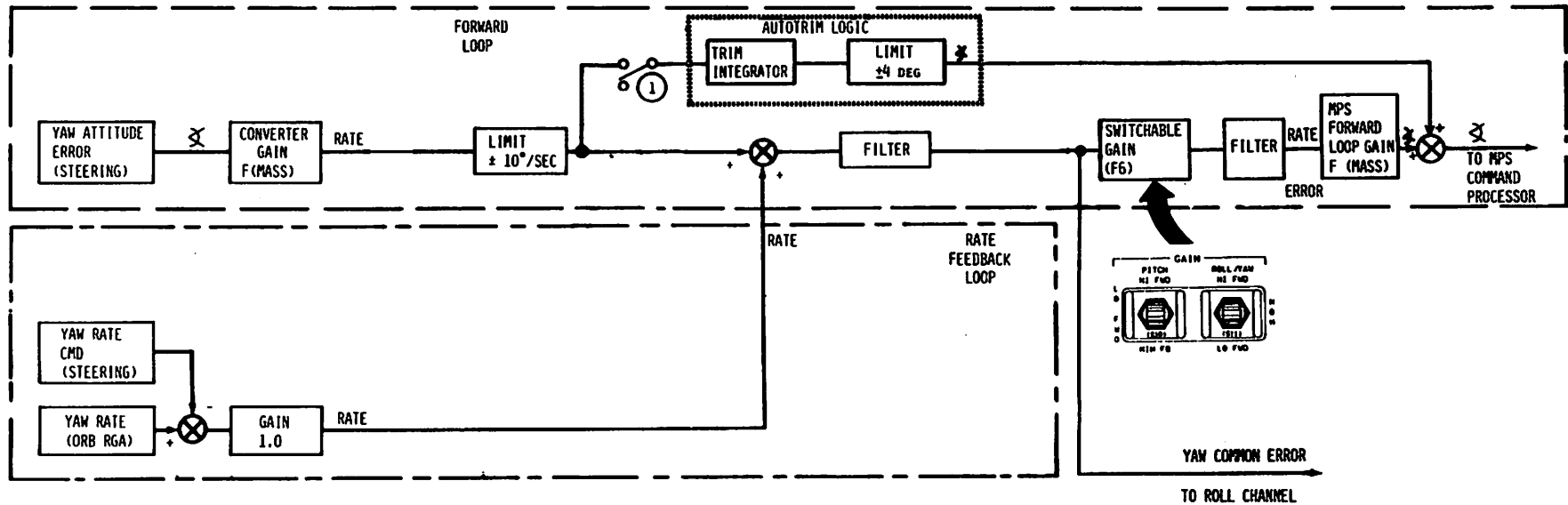


Figure 2.5-8.- Pitch loop (second stage).

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① AUTOTRIM ENABLED AT
SRB SEP CMD + 12 SEC

Figure 2.5-9.- Yaw loop (second stage).

Overview: The TRANS DAP is made up of two major software subsections, the reconfiguration logic, and the control laws. The control laws are further divided into three subsections: (1) the steering processor, (2) the OMS processor, and (3) the RCS processor. The basic relationship between these software modules and the inputs to and outputs from the TRANS DAP are shown in figure 2.5-10.

A detailed description of each of the TRANS DAP control modes follows:

- **ET Separation - ET SEP** is moded automatically 16 seconds after MECO is confirmed. The DAP may be in either AUTO or MANUAL, as the DAP control mode has no effect on the automatic ET SEP feature. The ET SEP processor is activated by the ET SEP sequencer and, when activated, causes the RCS jets to fire resulting in a -Z translation. When -4 ft/s ΔV is reached, the translation command is terminated and a separation complete flag is set. If the THC is moved out of detent during this process, the automatic separation feature is irrevocably terminated. However, the separation complete flag is still set when -4 ft/s ΔV_z is reached.
- **Auto TVC: Two Engines** - The two engine automatic mode is active when the AUTO control PBI has been depressed, the RHC is in the detent, and both OMS engines have ON commands. The DAP receives a commanded (desired) inertial velocity vector and its turning rate (set to zero in STS-1), plus a vector time tag from guidance. Incremental velocity, current time, and the body to inertial quaternion are sent to the DAP from the attitude processor. These parameters are essentially compared with the guidance commands and, using a process called cross product steering, a commanded body rate vector is generated. This body rate command is compared with sensed RGA body rates, and error commands are determined for each axis. These error commands are filtered, processed through proportional plus integral logic, and limited where appropriate. The body rate commands are converted to OMS engine TVC commands with engine yaw gimbal commands effecting changes in body yaw rate, engine pitch gimbal commands effecting changes in body pitch rate, and differential pitch gimbal commands effecting changes in body roll rate. The RCS processor is active in this mode as it acts in a wraparound mode to support the OMS processor if needed (see RCS wraparound section following).
- **Auto TVC: One Engine** - The one engine automatic mode is active when the AUTO control PBI is depressed, one OMS engine is ON, and the other has its OMS ENG switch on panel C3 OFF. Guidance commands are handled identically to the two-engine case; the major difference being that the RCS processor must handle the roll axis, as differential gimbaling is not available in the one-engine mode. The one thrusting engine controls both yaw and pitch commands through its gimbal system.

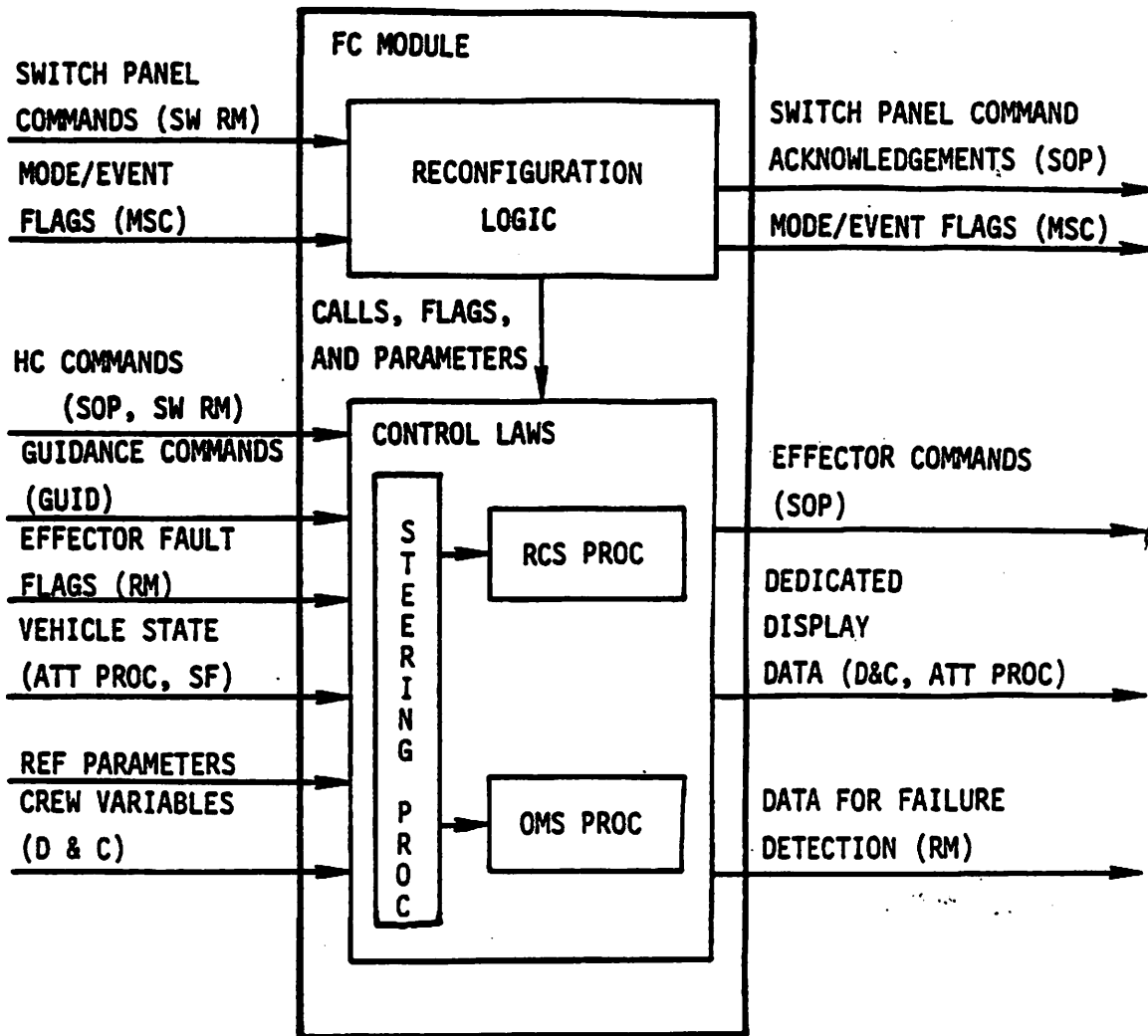


Figure 2.5-10.- TRANS DAP I/O.

- **Manual TVC: Two Engines -** The manual TVC two-engine mode is active when the manual control PBI has been depressed (or the RHC moved out of detent when the DAP is in AUTO) and both OMS engines are ON. Crew control is available through movement of the RHC, which supplies a limited (2 deg/s), proportional command to the DAP. The RCS processor is available in a wraparound mode unless the RHC is moved past the softstop(s), at which time the RCS processor goes into an acceleration mode. This process could result in the OMS and RCS processor being in conflict since the OMS processor would then be operating in its normal closed loop manner, whereas the RCS processor would be operating open loop for the axis in acceleration mode (see OMS/RCS conflict section following). The OMS TVC controls vehicle body rates about the three axes as in the AUTO mode.

- **Manual TVC: One Engine** - The manual TVC one-engine mode is active when one OMS engine is ON and the other is OFF (OMS ENGINE switch on panel C3 OFF), the RHC is out of detent, or the manual mode PBI has been depressed. The RHC SOP inputs deflections to the DAP and the OMS processor processes pitch and yaw commands, whereas the RCS processor handles roll commands in discrete rate, attitude hold, or acceleration (depending on RHC position). The RCS processor acts in a wraparound mode for pitch and yaw as in the two-engine case. If the roll softstop is passed and then the RHC returned between the softstop and the roll detent, the roll axis remains in free drift until reaching the detent, when attitude hold is commanded.
- **Discrete Rate/Attitude Hold** - The discrete rate/attitude hold mode is effected whenever AUTO control mode is active and the RHC is moved out of detent or remains in (or reaches) detent, respectively. The axis discrete rates are all I-loaded values, 0.5 deg/s, and the phase plane will maintain this rate within the I-loaded rate deadbands as long as the RHC remains out of detent. The rate commands are on an axis-by-axis basis. Attitude hold is commanded whenever the RHC is in the detent or returns to detent. The vehicle will remain within the I-loaded attitude deadbands as long as attitude hold is commanded. The reference attitude for attitude hold is that body attitude the vehicle is in when the RHC reaches detent.
- **Rotation Acceleration** - Rotation acceleration is accomplished when the RHC is deflected to the soft stop(s) for the axis or axes desired. RCS jets will fire in accordance with the jet select logic table hookup to give continuous rotation moment(s) about the axis or axes desired as long as the RHC is past the soft stop(s). This will occur whether the OMS engines are ON or OFF. When the RHC is brought back inside the soft stop(s), the vehicle will be in free drift about that axis or axes until the RHC detent(s) are reached. The mode then becomes attitude hold if AUTO control mode is active and free drift if MANUAL mode is active.
- **Translation Acceleration** - Translation acceleration is commanded on an axis-by-axis basis whenever the RHC is moved out of detent and results in translation commands to the RCS jets in accordance with jet select logic. The translation commands are available in conjunction with rotation commands as well as when the OMS engines are thrusting.

- Pulse - Pulse mode is accomplished when no OMS ENGINE ON command is present and the control mode PBI is in manual mode. Each time the RHC is moved out of detent(s) an 80 millisecond rotation command(s) is (are) effected. Free drift exists after the firings have concluded as well as when the RHC returns to detent. Pulse is not selectable on an axis-by-axis basis other than through RHC commands. Mixed modes of PULSE and DISCRETE RATE are not available.

The manual control modes with respect to engine configuration are summarized in table 2.5-6.

TABLE 2.5-6.- TRANSITION DAP MODES

RHC status (Axis by axis)	Engine configuration			
	Two OMS	One OMS	RCS (a)	RCS (b)
In detent	Attitude hold	Attitude hold	Attitude hold	Free drift
Out of detent, not past S/S	Limited rate command. Linear function of RHC deflection.	Pitch and yaw: same as two OMS. Rolls: discrete rate using RCS.	Discrete rate	80 ms pulse
Past S/S	Max (limited) rate command. RCS in accel. ^c	OMS: same as two engine. RCS: in accel mode. ^d	Accel mode	Accel mode
Between S/S and detent after having been past S/S	Same as out of detent.	Pitch and yaw: same as out of detent. Roll: free drift.	Free drift	Free drift

^aControl mode AUTO.

^bControl mode MANUAL.

^cCould result in OMS/RCS conflict.

^dCould result in OMS/RCS conflict in pitch or yaw.

NOTE: The phase plane is active when the RCS is in attitude hold, discrete rate, or wraparound modes and is inactive during acceleration or free drift modes. RCS wraparound capability exists for all OMS burns except for the roll axis during single engine thrusting.

Table 2.5-7 is a summary of TRANS DAP control deadbands and rates.

TABLE 2.5-7.- TRANS DAP DEADBANDS

TRANS DAP controllability	Control deadbands (R-P-Y)				Commandable rates °/sec			
	Attitude ° ±		Rates °/sec ±		Min		Max.	
2 OMS auto	* TVC		none TVC		none TVC		2°/sec	
	50-100-100 RCS		2.05-2.05-2.05 RCS		.08-.04-.04 RCS		total per axis RCS + TVC	
1 OMS auto	* TVC		none TVC		none TVC		2°/sec (P&Y)	
	50-100-100 RCS		2.05-2.05-2.05 RCS		.08-.04-.04 RCS		1°/sec (R) total per axis RCS + TVC	
2 OMS manual	N/A TVC		N/A TVC		N/A TVC		2°/sec	
	50-100-100 RCS		2.05-2.05-2.05 RCS		1°/sec per axis RCS		total per axis RCS + TVC	
1 OMS manual	N/A TVC		N/A TVC		N/A TVC		2°/sec (P&Y)	
	50-100-100 RCS		2.05-2.05-2.05 RCS		1°/sec roll RCS		1°/sec (R) total per axis RCS + TVC	
Discrete rate	N/A		N/A		1°/sec		1°/sec	
Attitude hold	3.50-3.50-3.50		0.30-0.30-0.30		.08-.04-.04		1°/sec	
Rotational acceleration	N/A		N/A		N/A		N/A	

*No deadbands as such/steady state error of TBS is acceptable

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OMS/RCS conflict in ACCEL mode: If the RCS processor is downmoded to the acceleration mode, it operates open loop with continuous thrusting. The OMS processor continues to operate closed loop with a $2^{\circ}/\text{sec}$ rate command, and it will attempt to maintain this rate as the vehicle rate increases above $2^{\circ}/\text{sec}$ due to continuous RCS thrusting. The OMS TVC will cause the gimbals to deflect in an effort to maintain $2^{\circ}/\text{sec}$ (could result in gimbals moving in opposite direction). However, the OMS control authority is limited due to gimbal travel limits, and eventually the RCS thrusting will cause the rotation rate about the axis (or axes) concerned to start increasing above $2^{\circ}/\text{sec}$.

RCS wraparound: The RCS wraparound mode supports the OMS processor if for any reason any body axis rate error exceeds the rate deadband during a burn ($2.05^{\circ}/\text{sec}$). In this event the RCS processor commands appropriate RCS jets to fire to assist the OMS engines in achieving the desired rate. This is likely to occur after an OMS engine or gimbal failure during a burn.

OPS 1 BFS flight control: This is an outline of the major differences between the PASS and BFS for flight control during OPS 1.

BFS flight control does not support prelaunch slew checks on the SSME's or aerosurface actuators. At T-60 seconds the BFS initializes and starts flight control functions. The BFS does not support SRB ignition but that is the point after which the BFS can take over control of the vehicle.

When the BFS is engaged, the actual SRB and SSME gimbal positions are read once to initialize flight control to the PASS state and after that it is assumed that the commanded state reflects the actual state. During first and second stage BFS flight control does not acknowledge manual steering commands but it does monitor the gain enable pushbuttons.

Only manual thrust vector control is available for OMS maneuvers when the BFS is engaged. The RHC is used for the manual inputs and flight control maintains attitude hold when the RHC is in detent. During the OPS 1 coast period, BFS flight control provides attitude hold and accepts manual rotational and translational commands.

2.5.4 Orbit Flight Control

The on-orbit DAP for STS-1 (the orbit 1 DAP) is an RCS-only DAP with automatic or manual rotation modes, manual translation, and primary or vernier (rotation only) RCS capabilities. The automatic or guided rotation commands are supplied by the universal pointing processor and manual rotation or translation commands by the RHC's or THC's, respectively. Crew commands from the forward or aft station are accepted. Three selectable manual rotation modes and two selectable translation modes (for X, Y, and minus Z translations) are provided. Three modes are provided for plus Z translations. The capability to select nose or tail only for pitch and/or yaw control is provided for the primary jets. Primary jet roll control is only provided by the aft RCS pods.

The vernier jets are used for tight attitude deadbands and fuel conservation. The loss of one vernier jet results in loss of the entire vernier mode.

DAP modes: The manual rotation modes of the on-orbit DAP are a function of the RHC state and PBI selected (table 2.5-8). Anytime the RHC is out of detent in any axis, no PBI change requests in rotation modes are recognized or remembered in any of the three axes. Translation moding is similarly inhibited if the THC is out of detent.

TABLE 2.5-8.- ORBITAL DAP MANUAL ROTATION MODES

PBI selected (a)			
RHC status (Axis by axis)	DISC RATE	ACCEL	PULSE
In detent	Attitude hold ^b	Free drift	Free drift
Out of detent, not past S/S	Discrete rate ^b	ACCEL mode	Rate cmd to establish rate, then free drift
Past S/S	ACCEL mode	ACCEL mode	ACCEL mode
Between S/S and detent, after having been past S/S	Free drift	ACCEL mode	Free drift

^aSelected on an axis by axis basis.

^bPhase plane active.

Figure 2.5-11 is a sketch of the on-orbit DAP control panel, one of which is located on panel C3 and the other on panel A6.

The DAP is initialized as follows:

- Rotation - discrete rate (all axes)
- Translation - pulse (all axes)
- RCS jets - normal (primary)
- Control - manual
- DAP select - DAP load A

In the event of an IMU attitude data bad flag being set, manual rotation and translation modes are set to the acceleration mode, and the only PBI's active are the RCS JETS. All others are ignored and only manual control is available.

The translation pulse mode is activated on an axis-by-axis basis by depressing one or more of the manual translation mode pulse PBI's on panel C3 or panel A6. Translation is affected whenever the THC is moved out of detent for the axis or axes in the pulse mode, and the resultant delta-V is the amount selected on the DAP CONFIGURATION display. This is a cumulative, open-loop system, as the delta-V is incremented each time the THC is moved out of detent and there is no feedback in the control loop. When the commanded increments reach the calculated delta-V, the jet translation command will be terminated (set to zero), whether the THC is in or out of detent.

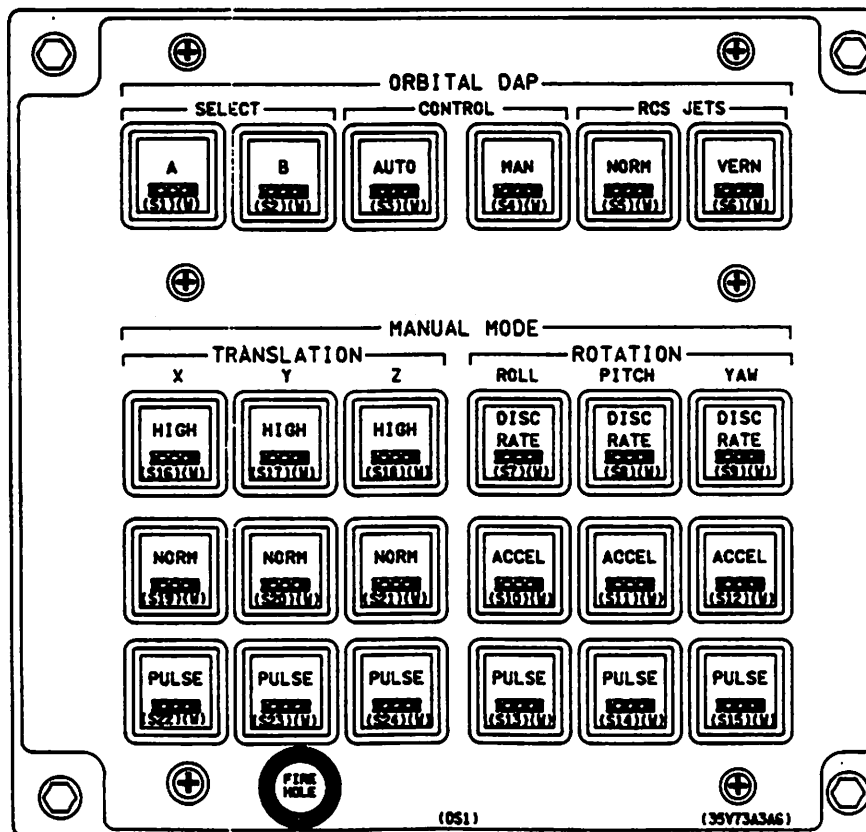


Figure 2.5-11.- DAP panel.

The discrete rate mode is affected whenever one or more of the DISC RATE PBI's on panel C3 or panel A6 are depressed and the RHC is moved out of detent. The rate commanded will be the rate selected on the DAP CONFIG display and the control loop is closed through the state estimator which estimates vehicle attitude and rate. These estimators are derived using attitude increments, supplied by the attitude processor, and estimated rotation rates due to the RCS jets firing, supplied by the jet select logic module. The phase plane will cause appropriate rotation commands to be sent to the jet select logic whenever the rate or attitude errors are outside the deadbands selected on the DAP CONFIG display as well as in accordance with the phase plane logic. Returning the RHC to detent will command attitude hold about the vehicle attitude existing at the time the RHC reached the detent. Force fire commands are issued whenever the RHC departs from detent to provide rate damping by imparting 80-ms rotational accelerations as required.

The rotation pulse mode is active for each axis whose manual rotation pulse mode PBI on panel C3 or A6 has been depressed. Each time the RHC is moved out of detent on the active axis (or axes), the vehicular rate entered on the DAP CONFIG display is commanded. The logic is open-loop and cumulative as multiple RHC deflections will result in a multiple of the vehicular rate. When the calculated rotational RCS firings have ceased, the DAP remains in a free drift mode whether the RHC is in or out of detent.

The automatic mode is entered when the RHC's are in detent and the AUTO control PBI is depressed on panel C3 or A6. Commands are issued to the DAP by the universal pointing processor which in turn interfaces with the ORBIT OPS display. Essentially, the DAP AUTO mode has the capability to:

- Perform a 3-axis automatic maneuver
- Track the local vertical about any body vector
- Rotate about any body vector at the DAP discrete rate
- Stop any of these options and command attitude hold

DAP configuration spec display: An example of the DAP CONFIG SPEC display with the initialized values currently being planned for STS-1 is shown in figure 2.5-12. The spec items and their units are identified in table 2.5-9. The A and B values determine which parameters the DAP will use in accordance with the DAPLOAD (A or B) selected on the ORBITAL DAP control panel. The maximum and minimum limits of the spec items are given in table 2.5-10.

2011/020/

DAP CONFIG

2 000/10:22:01
000/00:28:22

TRANSLATION
PULSE

	A		B
1	<u>0</u> <u>.</u> <u>1</u> <u>0</u>	16	<u>0</u> <u>.</u> <u>0</u> <u>2</u>

ROTATION

DISC RATE	NORM	2	<u>0</u> <u>.</u> <u>5</u>	17	<u>0</u> <u>.</u> <u>5</u>
	VERN	3	<u>0</u> <u>.</u> <u>2</u> <u>0</u>	18	<u>0</u> <u>.</u> <u>1</u> <u>0</u>
PULSE	NORM	4	<u>0</u> <u>.</u> <u>1</u> <u>0</u>	19	<u>0</u> <u>.</u> <u>0</u> <u>4</u>
	VERN	5	<u>0</u> <u>.</u> <u>0</u> <u>1</u> <u>0</u>	20	<u>0</u> <u>.</u> <u>0</u> <u>0</u> <u>2</u>
COMP	NORM	6	<u>.</u> <u>0</u> <u>0</u>	21	<u>.</u> <u>0</u> <u>0</u>
	VERN	7	<u>.</u> <u>0</u> <u>0</u> <u>0</u>	22	<u>.</u> <u>0</u> <u>0</u> <u>0</u>

INERTIAS

31	IX	<u>0</u> <u>.</u> <u>8</u> <u>7</u> <u>4</u>
32	IY	<u>6</u> <u>.</u> <u>8</u> <u>0</u> <u>2</u>
33	IZ	<u>7</u> <u>.</u> <u>0</u> <u>7</u> <u>0</u>

DEADBAND

ATT	R	8	<u>-</u> <u>6</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>	23	<u>-</u> <u>1</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>
	P	9	<u>-</u> <u>6</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>	24	<u>-</u> <u>1</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>
	Y	10	<u>-</u> <u>6</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>	25	<u>-</u> <u>1</u> <u>.</u> <u>0</u> <u>0</u> <u>0</u>
RATE	NORM	11	<u>0</u> <u>.</u> <u>2</u>	26	<u>0</u> <u>.</u> <u>2</u>
	VERN	12	<u>.</u> <u>0</u> <u>2</u>	27	<u>.</u> <u>0</u> <u>2</u>

JET OPT

P	13	<u>1</u>	28	<u>1</u>
Y	14	<u>1</u>	29	<u>1</u>

CNTL ACCEL

15	<u>0</u>	30	<u>0</u>
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Figure 2.5-12.- DAP CONFIG display.

TABLE 2.5-9.- DAP CONFIGURATION SPEC USAGE

<u>Item no.</u>	<u>Description</u>
1 and 16	Size of translation pulse in ft/sec (ΔV) for desired translation
2 and 17	Size of discrete rate command for primary jets in deg/sec rotation
3 and 18	Size of discrete rate command for vernier jets in deg/sec
4 and 19	The pulse size to achieve a desired rotational rate per pulse using the primary jets only (in deg/sec)
5 and 20	The pulse size to achieve a desired rotational rate in deg/sec using vernier jets only (rate is per pulse)
6 and 21	Allowable rotation rate (deg/sec) for axes resulting from a rate command about another axis when firing primary jets. Setting zero bypasses compensation.
7 and 22	Allowable rotation rate (deg/sec) for axes resulting from a rate command about another axis when firing vernier jets. Setting zero bypasses compensation.
8 and 23	Attitude deadband in deg for roll (X-axis)
9 and 24	Attitude deadband in deg for pitch (Y-axis)
10 and 25	Attitude deadband in deg for yaw (Z-axis)
11 and 26	Rate deadband (all axes) in deg/sec when using primary jets
12 and 27	Rate deadband (all axes) in deg/sec when using vernier jets
13 and 28	Jet options for pitch: 1: Forward and aft jets 2: Forward jets only 3: Aft jets only
14 and 29	Jet options for yaw (code as above)
15 and 30	Control acceleration selected for either primary or vernier jets 0: Nominal 1 thru 5: Prestored I-loaded values (used for vernier jets only)
31 thru 33	Principal axis moments of inertia (X, Y, and Z, respectively) in slug ft ² x 10 ⁻⁶ .

TABLE 2.5-10.- ORBIT DAP CONFIGURATION PARAMETER LIMITS

<u>Parameter</u>	<u>Range</u>	<u>Units</u>
Trans pulse level A & B	0.01 to 5.00	ft/s
Norm discrete rate A & B	0.1 to 2.0	deg/s
Vern discrete rate A & B	0.01 to 1.0	deg/s
Norm ROT pulse level A & B	0.04 to 1.0	deg/s
Vern pulse level A & B	.001 to .500	deg/s
Roll, pitch, & yaw att. DB A & B	0.01 to 40.0	deg
Norm rate limit A & B	0.2 to 5.0	deg/s
Vern rate limit A & B	.01 to .50	deg/s
Vehicle X inertia	0.75 to 0.90	slug ft ² x 10 ⁻⁶
Vehicle Y & Z inertia	6.000 to 7.000	slug ft ² x 10 ⁻⁶
Norm ROT COMP DB A & B	.00 to .99	deg/s
Vern ROT COMP DB A & B	.000 to .999	deg/s

Control Laws: The orbital DAP flight control schematic is given in figure 2.5-13. The following paragraphs discuss the principal blocks in the diagram.

Auto maneuver track: The auto maneuver track module generates control signals to achieve a commanded vehicle attitude and angular rate or to hold a vehicle attitude. The commanded attitude and rotation initiation are entered on the ORBIT OPS display. The module outputs the following commands:

- Body rate
- Attitude
- Attitude increment

All of the commands are inputs to the RCS errors module.

Rotation pulse: This module operates open loop and provides a command to generate a predetermined body rate increment about a selected axis or axes each time the RHC is out of detent(s). The increments are additive. The body rate increments are set on the DAP CONFIG spec display, and they are set for both vernier and primary jets. The control acceleration available for each body axis for both primary and vernier jets is also provided the module by the DAP CONFIG spec display. It is used in the calculation of the time required for the vehicle to reach the commanded rate increment (or rate increment sum), hence the time the command is active to the jet select logic.

Translation pulse: The translation pulse module operates open loop to command an incremental velocity (pulse size), entered on the DAP CONFIG spec display, each time the THC is moved out of its detent(s). The module receives vehicle mass as an I-loaded value and uses it in calculating how long the translation command should be active in order to achieve the desired velocity increment.

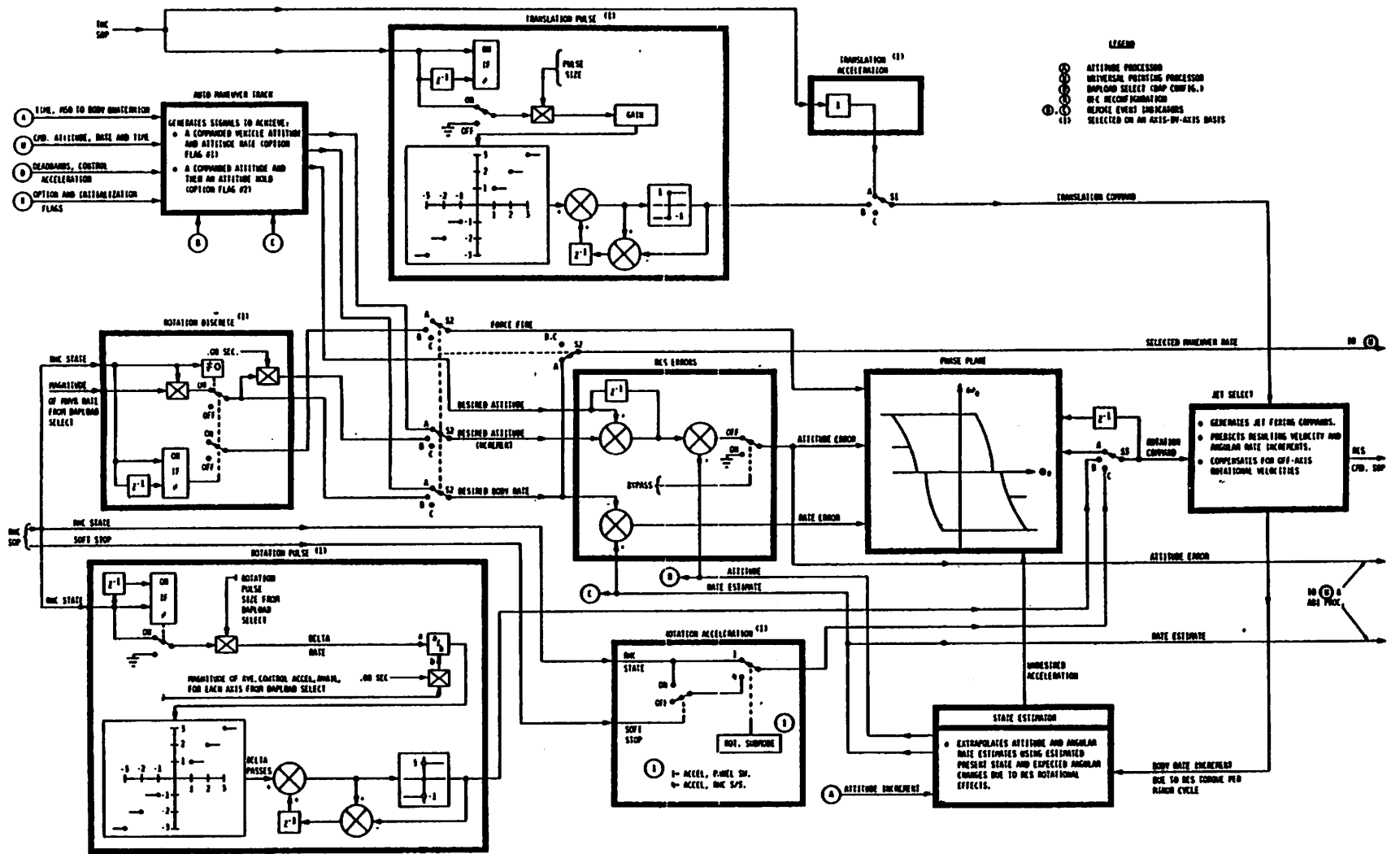


Figure 2.5-13.- Orbit DAP schematic.

Notes for figure 2.5-13:

<u>Switch</u>	<u>Contact</u>	<u>Moding Logic</u>
S1	A	Primary jets <u>and</u> trans accel selected
	B	Primary jets <u>and</u> pulse mode selected
	C	Vernier jets selected
S2	A	Auto
	B	Discrete rate/attitude hold selected and manual
	C	Manual <u>and</u> (pulse or accel mode selected)
S3	A	Attitude data good <u>and</u> (auto <u>or</u> DRC/AH)
	B	Manual <u>and</u> pulse mode selected
	C	Manual <u>and</u> accel mode active <u>or</u> attitude data bad

State estimator: The state estimator determines vehicle state by estimating body rate and attitude. It first integrates attitude increment inputs from the attitude processor to obtain measured attitude. This data is filtered to obtain estimated attitude and attitude rate data.

The data is further refined using jet-select, modeled, angular rate increments due to RCS torques in order to extrapolate the vehicle state. The state estimator also determines undesired acceleration (or disturbance acceleration) about each body axis for phase plane use. This parameter is used in reshaping the switching lines of the phase plane as well as in the basic phase plane logic.

Phase plane: When the DAP is in the discrete rate mode (AUTO or MANUAL) there is a phase plane module for each body axis which determines whether an angular rate command is required and the desired sign and magnitude of the command. It causes a force fire flag to be set when the RHC is moved out of detent in one or more axes which results in a firing command(s) to initiate rotation. The phase plane considers undesired (or disturbance) accelerations in its command logic when vernier jets are selected. These accelerations can be the result of gravity gradient, venting drag forces, etc. The primary DAP phase plane also limit cycles on attitude error just as the transition DAP phase plane does. The deadbands for the phase plane logic are set using the DAP CONFIG SPEC.

Jet select logic: The on-orbit jet select logic uses a table look-up technique for the primary jets similar to that used by the transition DAP. Unlike the transition DAP, the on-orbit jet select logic must check for high or low rotation mode for both pitch and yaw for the nose and tail jets. There are special tables for the high +Z translation mode, also not used by the transition DAP. Angular rate increment predictions are made for

each group of primary jets and for each specific vernier jet for state estimator use and rotation compensation. The primary jet predictions are based on one nominal inertia matrix and c.g., and the vernier predictions are based on nominal or one of the five different payload configurations (with the manipulator arm extended) inertia matrices. The rotation compensation logic uses a combination threshold (set on the DAP CONFIG spec display) and counter (or accumulator) in determining when to command jets on for rotation compensation. Compensation is active for all rotation modes when selected.

2.5.5 Deorbit Flight Control

The trans DAP is the digital autopilot used for deorbit. Refer to section 2.5.3.

2.5.6 Entry Flight Control

The aerojet 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 OPS 3, but is used only during MM 304 (ENTRY) and MM 305 (TEAM and A/L). This is accomplished by using either manual (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 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-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, 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 PRL logic and the Jet Select Logic (JSL). The control laws are further broken down into the major entry and TAEM/A/L software divisions.

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, speed brake, and body flap. For exoatmospheric flight or flight during the trajectory where certain surfaces are rendered ineffective by adverse aerodynamics, a combination of aerosurface and reaction jet commands are 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.

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 discretizes 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 PBI's on panels F2 and F4.

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.

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.

- Longitudinal Control Laws - For this document the pitch channel, the speed brake channel, and the body flap 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.
- Pitch Channel (fig. 2.5-14) - The pitch channel software issues two separate effector commands to control vehicle pitch motion. These are (1) the number of pitch jets to be fired to effect desired response, and (2) an elevator deflection command when in atmospheric flight. The effectiveness of the jets and elevator is a function of the dynamic pressure 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 20 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 $\bar{q} \geq 0.5$ lb/ft² thereby relieving the pitch jets of this function and saving fuel. Prior to achieving $\bar{q} = 20$ lb/ft² elevator trim is calculated by numerically integrating the dynamic elevator and pitch jet commands. After $\bar{q} = 20$ lb/ft² elevator position feedback 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 body flap activation and deployment of the speed brake. The speed brake is nominally deployed at $M = 10$ to induce a more positive elevator trim deflection. This is accomplished by biasing the elevator position feedback during speed brake activation. The elevator trim deflection is sent to the body flap channel to facilitate the body flap 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 ADI processing for display on the ADI pitch error needle. This occurs at all times while in MM 304 whether in AUTO or CSS.

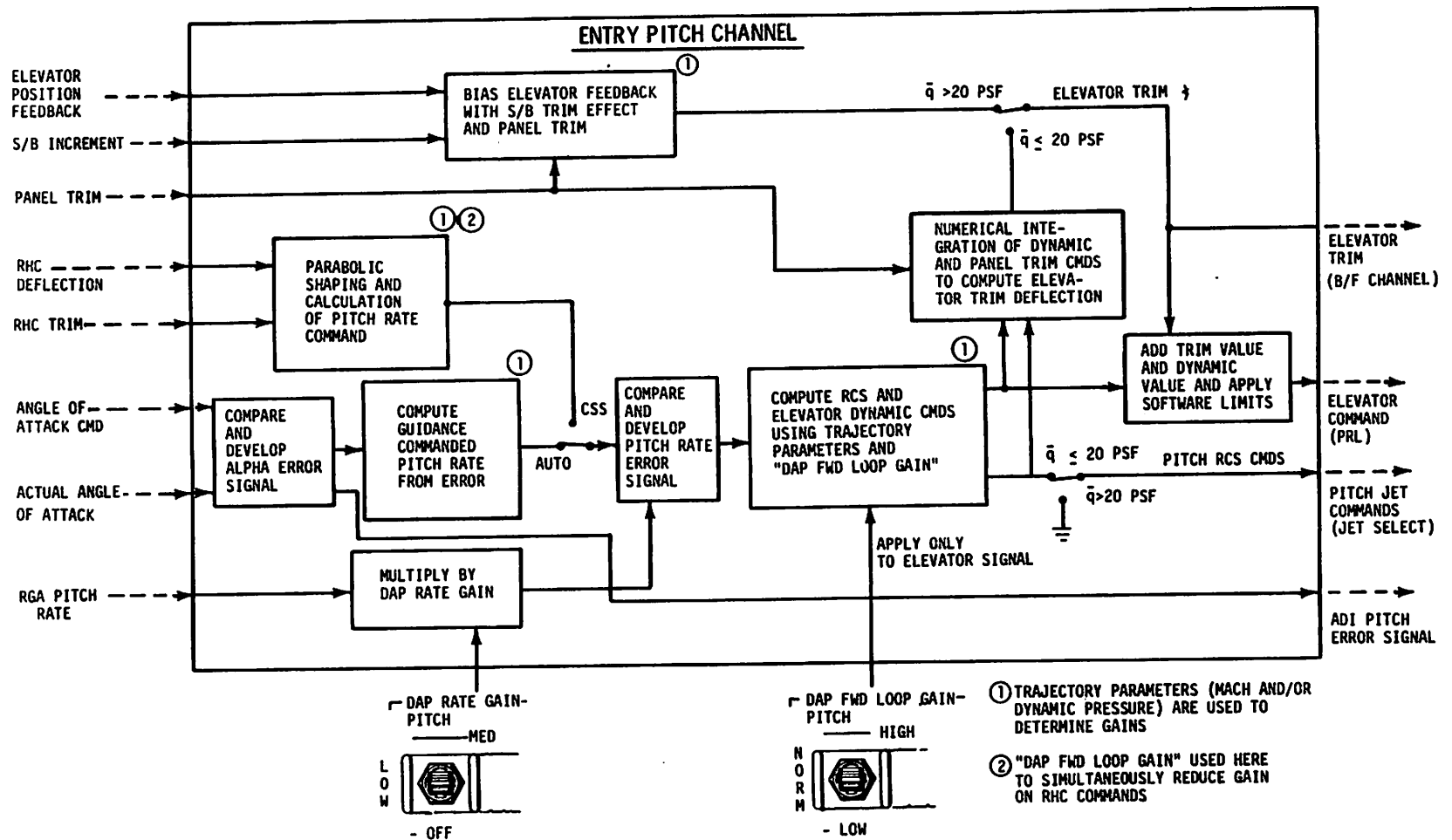


Figure 2.5-14.- Entry pitch overview.

- Speed Brake Channel (fig. 2.5-15) - Two outputs of the speed brake software are provided during MM 304. These are: (1) the commanded speed brake deflection in hingeline degrees going to PRL and (2) 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 speed brake mode for crew reference and tracking.

When the speed brake is in AUTO, a command is developed by using Mach numbers to select a deflection from a deflection vs. Mach number scale. A negative bias is added to this command at high Mach numbers to keep the speed brake 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 SBTC deflection is input from the selection filter and a constant gain is applied to convert SBTC degrees to speed brake hingeline degrees. This MANUAL speed brake deflection command is then processed exactly as the AUTO command by applying the same bias and software limits.

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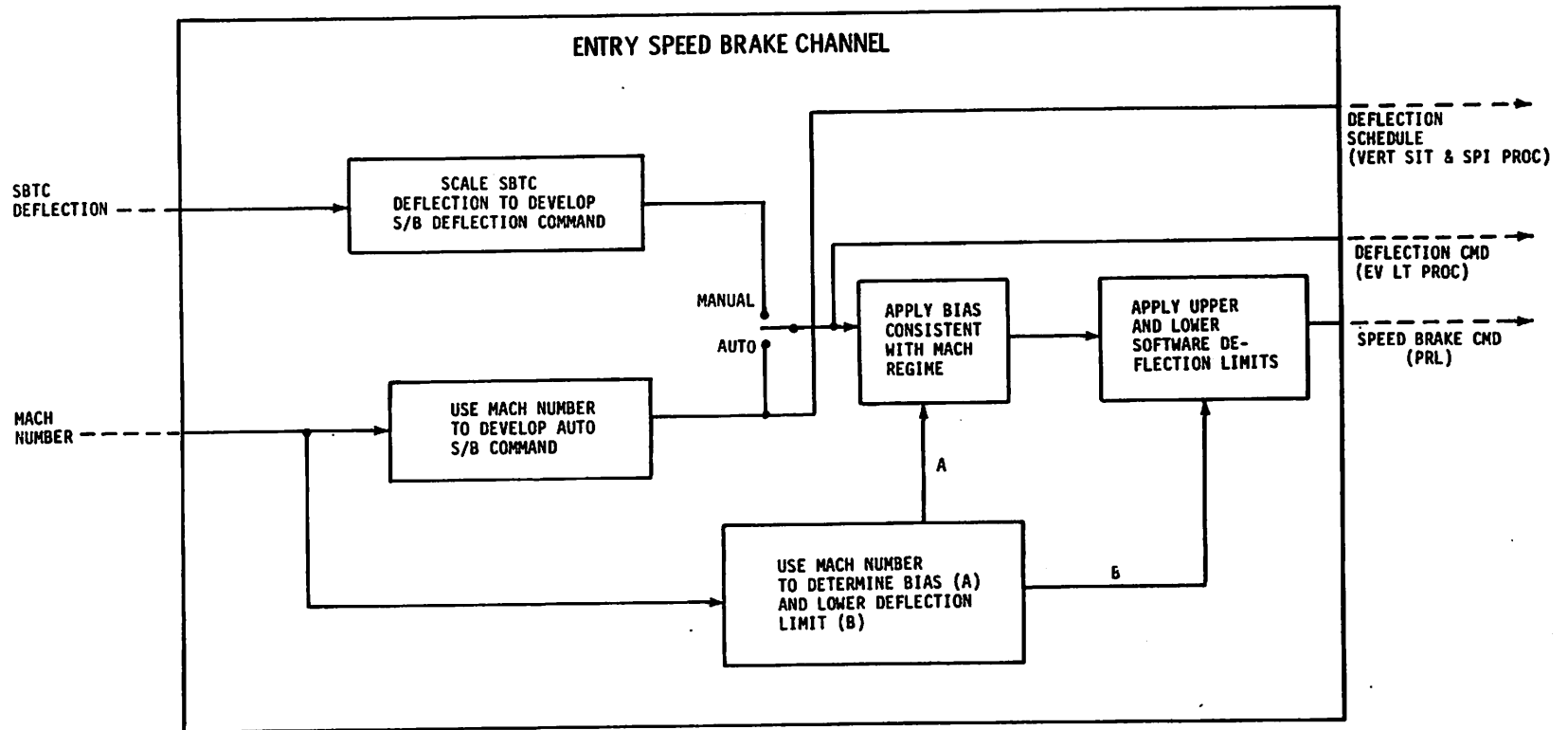


Figure 2.5-15.- Entry speed brake overview.

- Body Flap Channel (fig. 2.5-16) - The elevon/body flap interconnect provides the capability for the body flap to automatically deflect to maintain a preselected elevator trim deflection schedule. The MANUAL mode provides the capability to position the body flap independent of the trim deflection schedule.

When the body flap 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 body flap to deflect at a rate that changes with Mach number. This rate command is integrated to produce a body flap position command. A constant upward (-) deflection limit and a Mach variable downward (+) deflection limit is applied to the integrator output. This limited body flap position command is then compared to the actual body flap 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 body flap 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 body flap rate command that is a constant rather than Mach dependent. This means that the body flap 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 due to slow body flap movement and the size of the hysteresis deadbands. This allows the body flap to respond instantly to a crew command regardless of where the position error is inside the deadband.

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 position feedback value each time the output of the slew switch changes value. This automatically nulls the input to the final hysteresis.

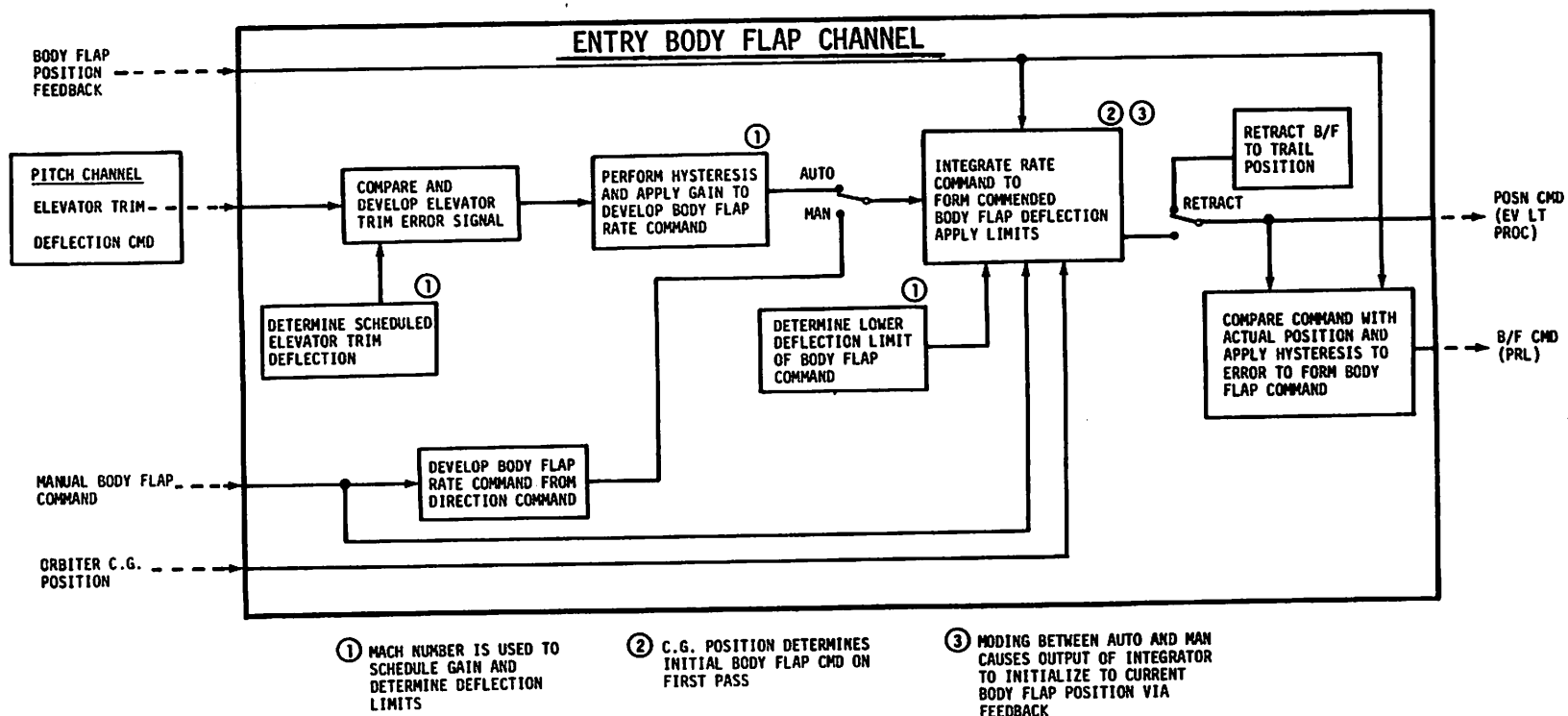


Figure 2.5-16.- Body flap overview.

Roll/yaw control laws: Refer to figures 2.5-17 and 2.5-18. 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 a great deal of 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 $> 2.0 \text{ lb/ft}^2$ and the roll jets are turned off upon reaching a dynamic pressure of 10 lb/ft^2 . In the yaw channel, the rudder is not commanded to deflect for dynamic control or trim until reaching $M = 4.5$ and the yaw jets are used until $M = 1.0$.

For normal roll/yaw configuration (ENTRY ROLL 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 damping 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 component of limited roll rate command. Below $Mach = 0.9$, the roll channel provides all the effector power for initializing sustaining roll maneuvers with the yaw channel providing the conventional coordination.

Manual transitioning between the early and late systems can be accomplished by exercising the ENTRY ROLL MODE switch. The YAW JET/RUD position corresponds to the early configuration and the AILERON position corresponds to the late configuration. In the AUTO position, the transitioning occurs automatically and independently of TAEM transition criteria.

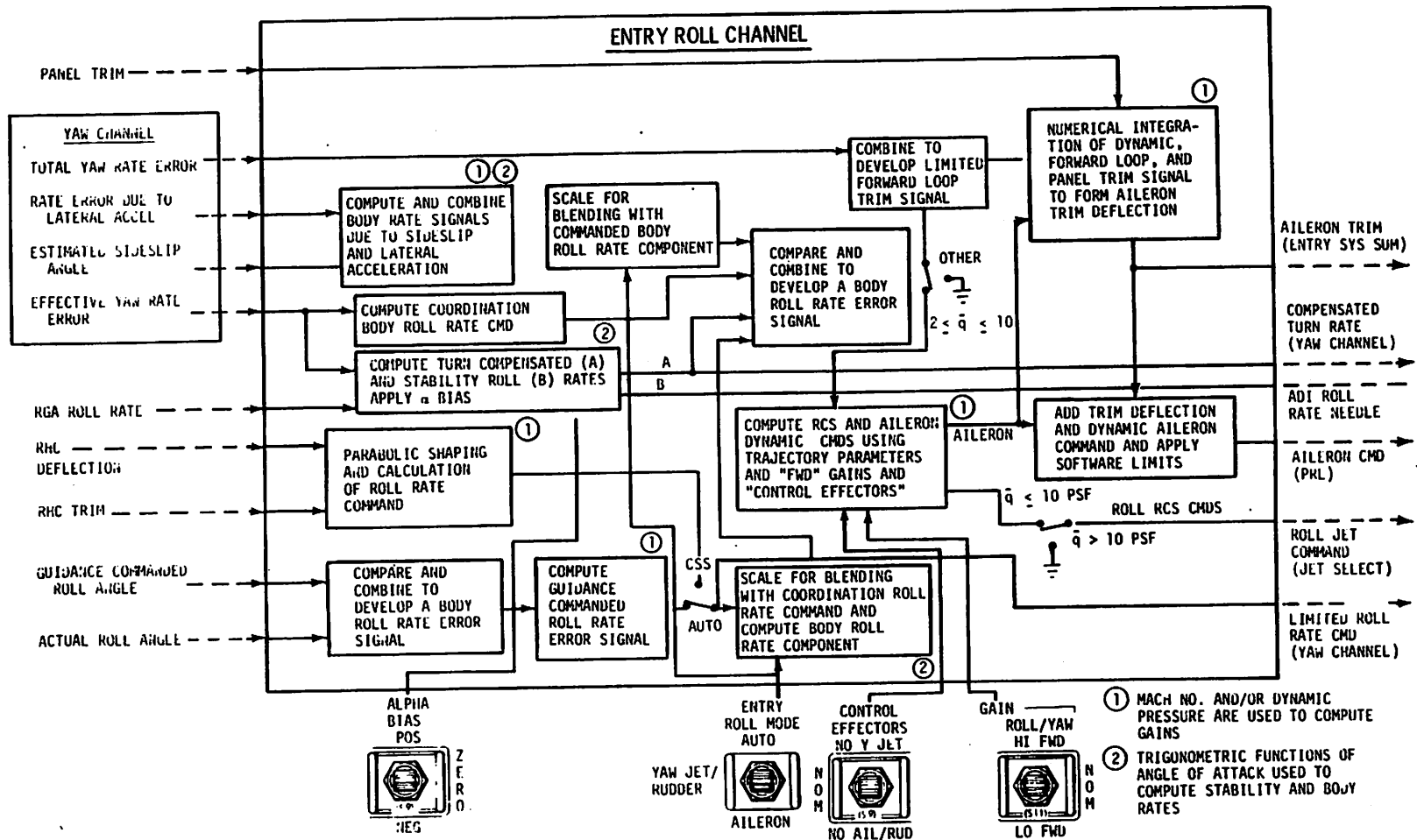


Figure 2.5-17.- Entry roll channel.

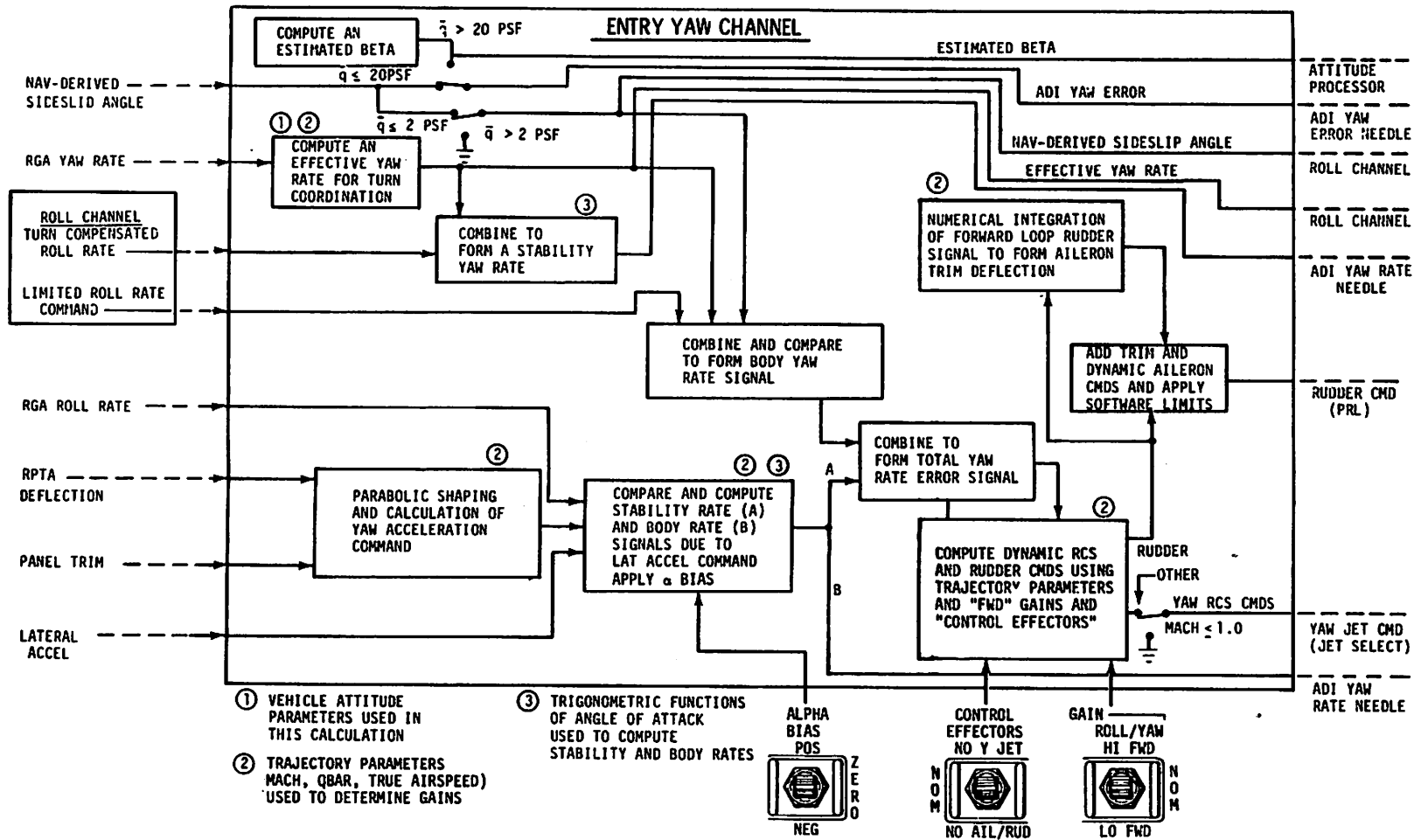


Figure 2.5-18.- Entry yaw channel.

TAEM/AL (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 aerog-jet DAP. Although there is some commonality between entry and TAEM/AL control laws, there are also several significant differences. These configurations for TAEM/AL are shown in the overviews in figures 2.5-19, 2.5-20, and 2.5-21

The changes occurring in the roll/yaw channel are more subtle than in pitch, but none the less significant. These changes primarily involve deleting certain terms at various times during the TAEM/AL trajectory.

Longitudinal control laws: During TAEM/AL, the speed brake is used to modulate aircraft energy. This dynamic use of the speed brake means that the speed brake and body flap will play a more active role in determining and maintaining an elevator trim position.

Pitch channel: The basic structure of TAEM/AL pitch channel corresponds to the entry configuration at the TAEM interface. Steady state and dynamic elevator trim deflection are provided by elevator position feedback, speed brake induced pitch effects are trimmed out with speed brake cross-feed, 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 body flap 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.

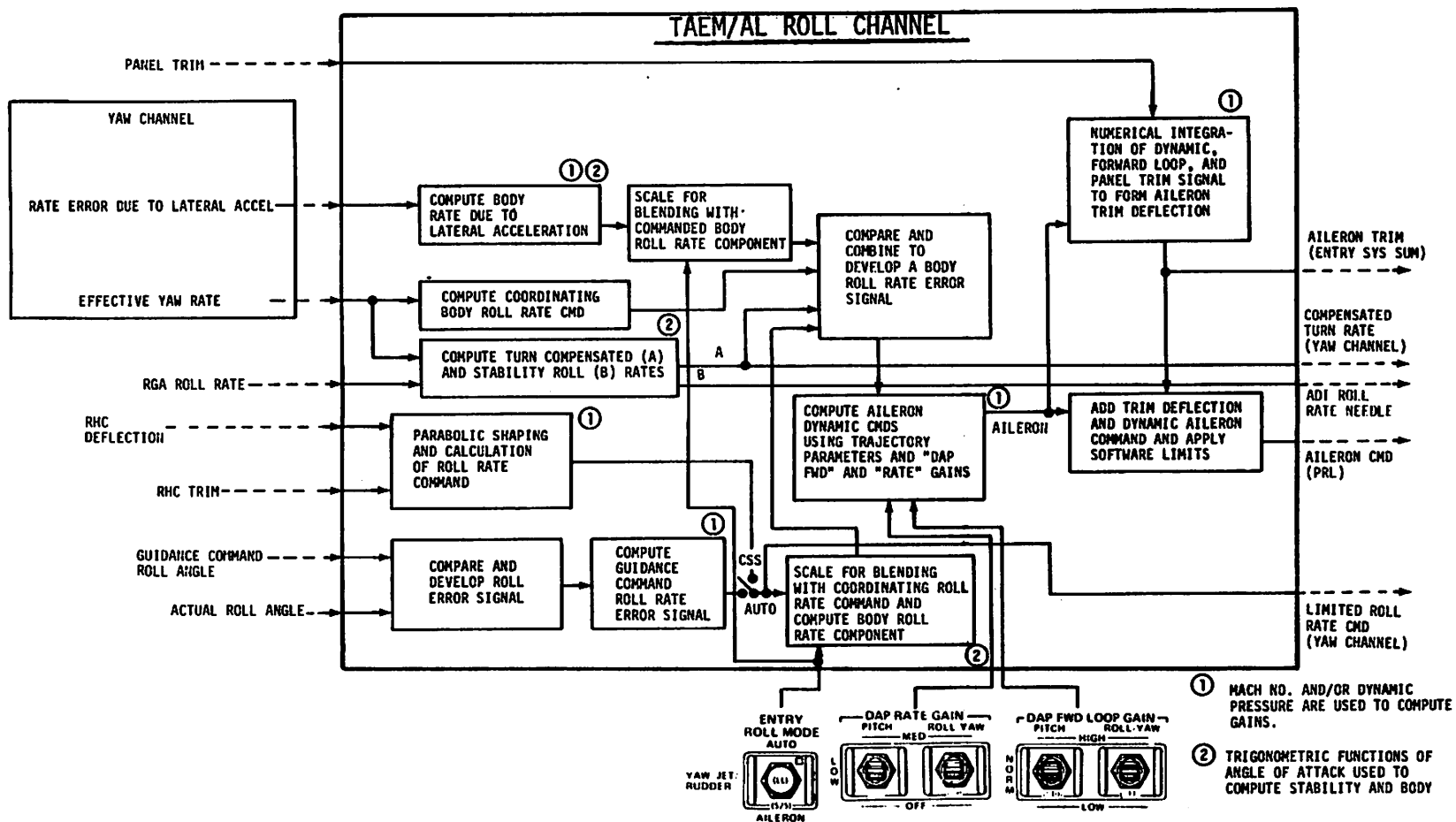


Figure 2.5-19.- Roll TAEM overview.

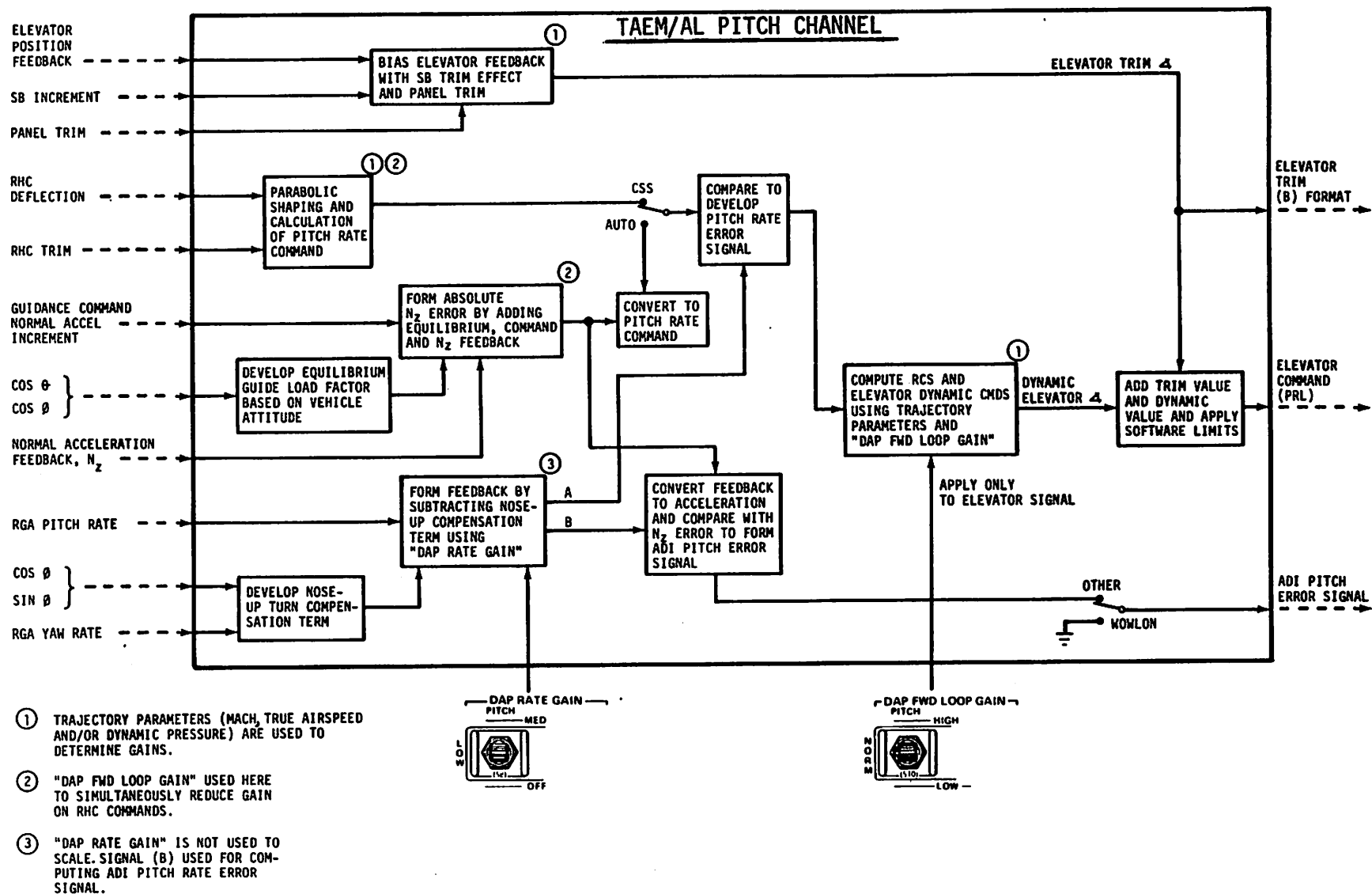


Figure 2.5-20.- Pitch TAEM overview.

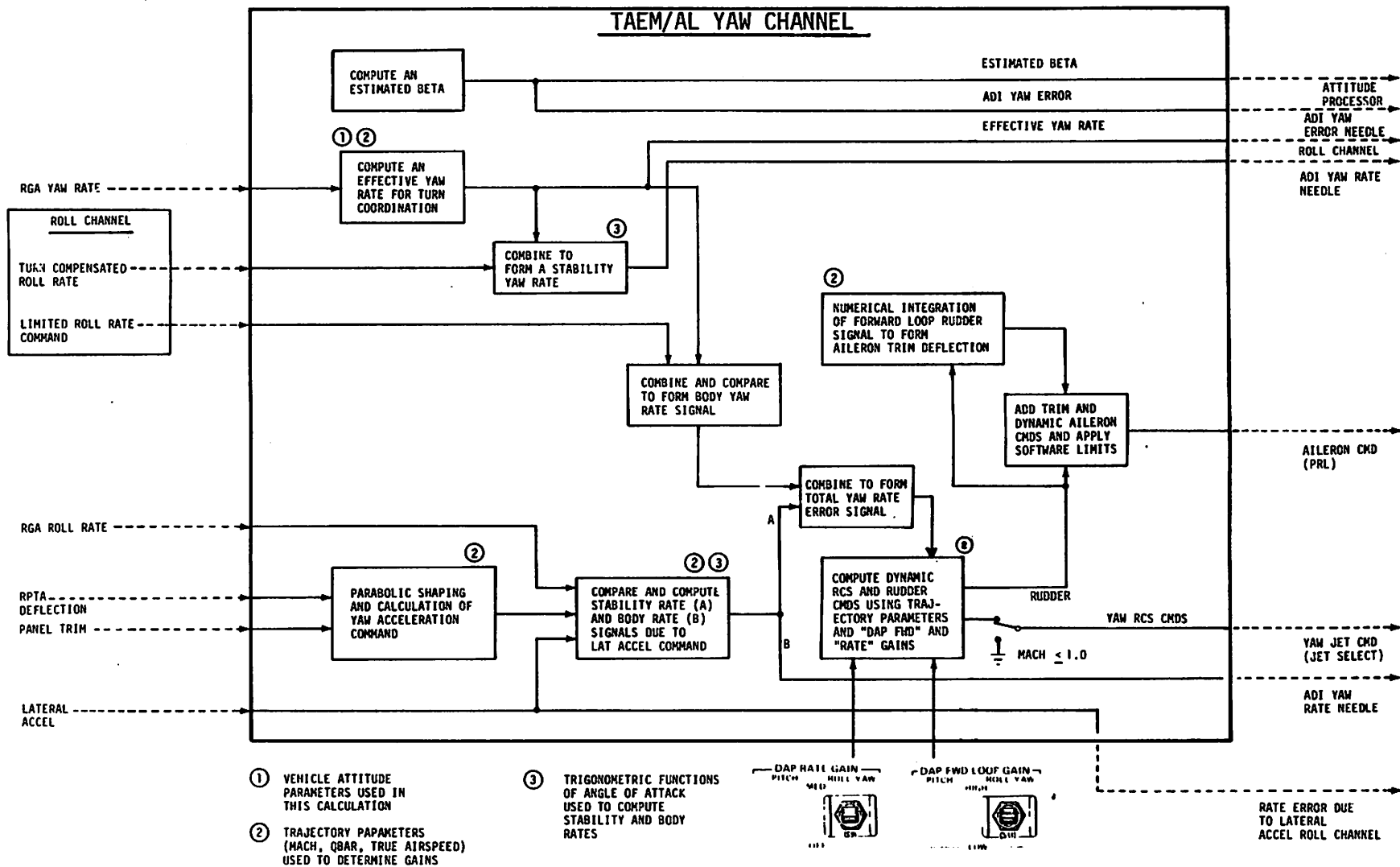


Figure 2.5-21.- Yaw TAEM overview.

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.

Speed brake channel: During TAEM/AL the speed brake channel software follows closely that of entry except that the auto speed brake command is formed in guidance and is used to modulate equivalent airspeed. The speed brake is now used in a dynamic sense to modulate the TAEM trajectory. The speed brake crossfeed into pitch channel will be critical in damping speed brake-induced pitching moments.

Guidance logic determines when the speed brake should be retracted (commanded CLOSED). This command is received by the speed brake channel as a deflection command so that no special moding or logic must take place in speed brake channel software to retract the speed brake.

Body flap channel: The only body flap logic peculiar to the TAEM/AL major mode is the body flap retract logic. Upon receiving the body flap retract discrete from A/L guidance, the body flap is commanded to deflect to 0.0° or the trail position. For the retraction to take place, the body flap mode must be auto and the speed brake deflection must be less than 50° . If the body flap mode is manual or if the speed brake is deflected 50° or more, the body flap 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 $-2^{\circ}/\text{sec}$. 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 (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.

Roll/yaw channel: The most obvious differences may be noted by comparing the entry and TAEM/AL diagrams. The TAEM/AL roll and yaw channel signal flows as shown in figures 2.5-19 and 2.5-21. As the Mach no. decreases below 3 the yawjet/rudder aileron blend gain decreases such that the ratio of aileron and yawjet/rudder control approaches that of a conventional aircraft.

When in AUTO the aileron trim angle is held constant upon receiving the flatturn discrete, and upon commencing rollout the aileron trim angle is set to zero. Neither of these events occur 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 being known as Mach Require 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 $\leq .9$) 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 $\leq .9$, 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.

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.

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 Groundspeed Enable (GSENB). This occurs as ground speed decreases below 115 kn.

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 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 kn. Unlike all other effector commands generated by the aeroflex DAP the nosewheel deflection command does not go the PRL. It is sent directly to the aero-actuator SOP.

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 speed brake with a special speed brake slow closing rate for small speed brake deflection angles. This slow closing rate is for protection against slamming the two rudder/speed brake panels together. Another feature of the PRL module is the lower rudder rate limiting and maximum rudder/speed brake closing rate when either the rudder or the speed brake panels exceed a prescribed deflection limit. This is commonly referred to as the rudder panel margin.

PRL overview: The inputs to PRL are the commanded positions for the elevator, aileron, rudder, and speed brake from the flight control channels (fig. 2.5-22). 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 body flap and leakage is subtracted from the total amount of hydraulic flow available for the speed brake, which is used for determining the speed brake opening and closing rate limits. The speed brake command is rate-limited for determining the speed brake position command and the speed brake increment. The rudder position and speed brake position commands are combined and then compared to the rudder/speed brake summed position limit to determine if the rudder/speed brake 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 speed brake closing rate limit is changed.

In PRL all rate-limited surface deflection commands are dependent upon the number of good hydraulic systems available. For the rudder/speed brake section, the rudder panel margin is computed by the rudder/speed brake position limit (52.9°) minus the absolute value of the rate-limited rudder deflection command minus one-half the rate-limited speed brake 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 speed brake rate limits as well as the amount of hydraulic flow available for the speed brake.

PRIORITY RATE LIMITING OVERVIEW

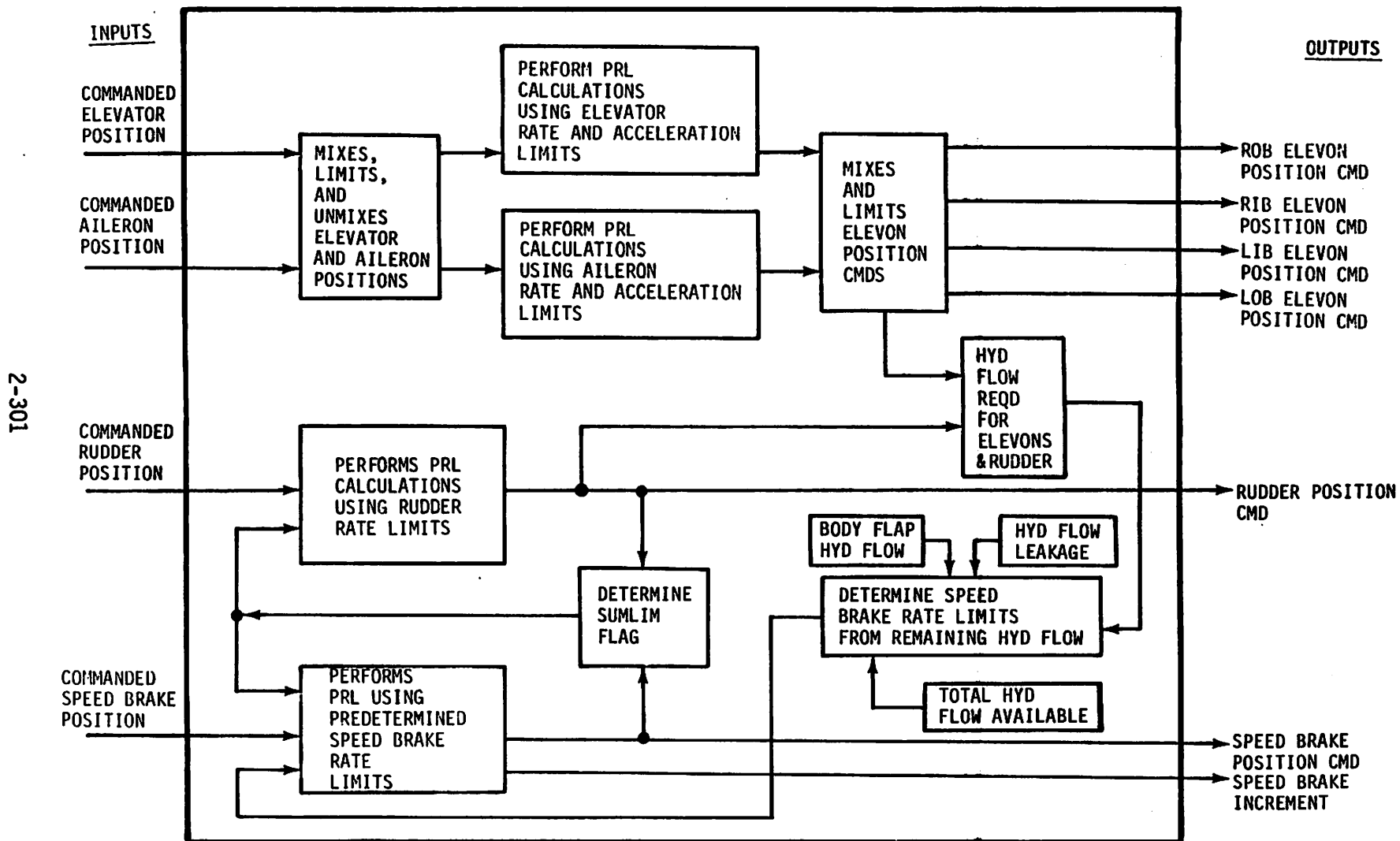


Figure 2.5-22.- Priority rate limiting overview.

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 body flap and the hydraulic flow allowed for leakage are subtracted from the total hydraulic flow available to give the remaining flow available. If the SUMLIN flag is set, the hydraulic flow required for closing the speed brake 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 speed brake.

The hydraulic flow available to the speed brake is converted to a speed brake rate limit. The lessor of this rate limit and the speed brake negative or closing rate limit is used for speed brake negative or closing limit. The lessor of the rate limit and the speed brake position or opening rate limit is used for the speed brake positive or opening limit.

The speed brake command is back-differenced with the post speed brake 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 speed brake 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 speed brake command are combined to form the rudder panel margin which was discussed previously.

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 preengage tracking of the primary flight control system. It also provides for takeover-engage initialization with acceptable transients. The D DAP combines the Glide Return to Landing Site (GRTLS) and AERO JET functions into a single software unit. Unified AERO_PRL and JET SELECT software is provided in backup.

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:

1. Only the four Y-axis and six Z-axis RCS jets on each aft pod are utilized.
2. No X-axis nor any vernier jets are utilized.
3. RM sends the discrettes which designate which jets are available for firing.
4. A maximum of four jets per pod may be fired simultaneously.
5. 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:

1. RM checks to see which RCS jets are good.
2. The logic sets all roll and pitch jet commands to off.
3. 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).
4. 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.
5. The logic checks for reconfiguration or for the crossfeed valve being open. If either event is in process, the number of jets total in both aft RCS pods is four.
6. 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.
7. The logic issues JET-ON commands to the appropriate jets.

During entry, the flight control system used the RCS jets in the following regimes:

TABLE 2.5-11.- RCS USAGE

RCS jets	Jets activated	Jets deactivated
Pitch	Entry interface	$\bar{q} = 20 \text{ lb/ft}^2$
Roll	Entry interface	$\bar{q} = 10 \text{ lb/ft}^2$
Yaw	Entry interface	$M = 1.0$

BFS variations from the primary jet select logic: The BFS combines the jet select logic for transition, GRTLs, and descent. Another major difference is that BFS is single fault tolerant to jet failures whereas 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 cross-coupling.

Entry flight control system downmoding: The STS entry/GRTLs flight control system 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 flight control system may not handle very well or that a variation of the flight control system might be required 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 are known as the downmoding switches. The concept of downmoding is defined as the crew use of cockpit panel-mounted Pushbutton Indicators (PBI's) and switches selectively to apply multipliers to the forward loop and/or rate feedback loop gains in the FCS pitch and roll/yaw axes, to bias the angle-of-attack input, and to effect reconfigurations in the roll/yaw channel control logic.

The basic design rules regarding the selection of the gain enable mode and the use of the gain switches during entry are as follows:

1. GAIN ENA PBI's must be depressed to activate the switches in each respective axis.
2. All FCS mode PBI's are independent; i.e., mixed modes are allowed.

3. Pressing the GAIN ENA PBI's automatically downmodes the FCS to CSS. The downmoding switches cannot be activated when the FCS is in AUTO. When the GAIN ENA PBI's are depressed, the FCS is configured to the state defined by the positions of the respective switches. The switches can be positioned as desired after being enabled to determine gain magnitude, alpha bias, or an alternate roll control logic.
4. Depressing a lighted GAIN ENA PBI restores the nominal gain structure in that axis and extinguishes the GAIN ENA lamp (mode remains in CSS).
5. The downmoding functions are applicable in the BFS as well as PASS.

The PITCH GAIN ENABLE PBI activates the following switch:

- PITCH GAIN - This switch has three positions, HI FWD, LO FWD, and MIN FB. The HI FWD position increases the forward loop gain on the pitch rate error signal by a factor of 1.5. The gain increase commands a larger elevon deflection for a given rate error. HIGH FWD may be used if pitch response is sluggish and damping is oscillatory, though of medium frequency and small amplitude. Feedback gains are unaffected. Similarly, the LOW FWD position decreases the forward loop gain on the pitch 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 over gained (high-frequency oscillatory) in pitch, selection of LOW FWD should allow improved damping. If large amplitude diverging rates are induced as might be encountered in a large signal instability, the LOW FWD gain may retard the vehicle accelerations, allowing recovery.

The use of the Minimum Feedback (MIN FB) position is intended to counter the effects of control surface buzz, ringing, or a bending instability, caused by rates fed back to the FCS from the RGA's. A multiplier of 0.0 is placed on the pitch feedback gain, automatic elevon trim is eliminated, and the RHC pitch commands are reduced by half. Because the vehicle is generally unstable without rate feedback damping, pilot workload is high while maneuvering. Therefore, this switch position is intended to be used only for short periods until the cessation of surface oscillations.

The ROLL/YAW GAIN ENABLE PBI activates the following switches:

- ROLL/YAW GAIN - This switch is analogous to the PITCH GAIN switch, having the positions HI FWD and LO FWD. The HI FWD position increases, by 1.5, the forward loop gains for the aileron and rudder commands, based on roll and yaw rate errors. With this gain selected the roll response and damping should be slightly quickened over the nominal response.

The LOW FWD position reduces by 0.5 the aileron and rudder forward loop gains for surface command. If only one APU/hydraulic system remains for control, low pitch and roll/yaw gains may potentially prevent RHC commands from causing surface rate saturation in high \bar{q} conditions, because surface rates may be reduced. The low gain position may also be applied if overgained system response roll rate ringing is encountered.

The Nominal (NOM) position allows the roll and yaw channel gains to remain as scheduled, while the other two ROLL/YAW functions are involved.

- ALPHA BIAS - The utility of the alpha bias capability has been diminished, because the incorporation of NY feedback in the roll and yaw channels reduces the FCS sensitivity to α errors.

Previous STS flight control systems were sensitive to errors in nav-derived angle of attack; e.g., α errors due to wind effects. Errors in the α used in the internal FCS control laws led to loss of control especially in cases of aerovariations and yaw jet failures. Yaw jet firings, β , roll rate, and aileron and rudder trim deflections in response to roll commands provided cues that indicated the nav-derived α input to flight control was higher or lower than the actual environmental α . With these cues, a pilot could bias the angle of attack being used by flight control to correct roll response and lessen chances of commands, loss of control. If β , δa , trim, and yaw jets aided in the roll commands, the bias was added. If β , δa , trim and yaw jets opposed the roll commands, the bias was subtracted. The ALPHA BIAS switch allows the pilot to apply a bias of $\pm 3^\circ$ to the input angle of attack. The biased α is then used in flight control trigonometric functions calculating roll and yaw rate errors. This bias on angle of attack affects only flight control, not guidance or navigation. The stability roll and yaw rates displayed on the ADI will reflect the application of an α bias.

The NOM position applies no α bias, allowing other roll/yaw switch functions to be selected independently.

- CONTROL EFFECTORS - 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 3.5, 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.

If elevon or rudder surface ringing or bending instability is encountered, the NO AIL/RUD position is used to isolate the roll and yaw channels somewhat from RGA and AA feedback; i.e., body yaw rate, roll rate, β , and N_y feedback. This switch position would be used in conjunction with the pitch gains MIN FB position to isolate the control surfaces from the sensors.

In the NO AIL/RUD mode, stability roll and yaw rates are computed with body roll and yaw rate components, but the error terms are passed through a fourth-order body bending filter. This alternate rate feedback loop assists in rate damping in the yaw channel. A large dead band on the aileron forward loop prevents the aileron from responding to error commands. Aileron deflections still provide lateral trim in response to the automatic trim loop or the roll panel trim switches. The yaw panel trim switch inputs are integrated for rudder trim commands, although the rudder no longer responds to yaw rate commands.

The center, nominal switch position, NOM implies the baseline FCS roll/yaw control logic configuration.

2.5.7 Powered RTLS Flight Control

The ascent DAP remains active during PRTLS (section 2.5.2). A few modifications are made to handle events peculiar to RTLS. They are listed below:

- G/C STEER remains active until MM 602 to support last commanded attitude at MECO. (It can be as long as 28 sec from MECO to MM 602.)
- GRTLS DAP is activated at MECO to hold last commanded attitude using RCS.
- Ascent DAP is terminated at ZERO THRUST.

2.5.8 Gliding Flight Control

The ascent DAP is terminated at zero thrust and GRTLS DAP is initiated at MECO commanded so that an approx 2.5 second overlap of control is provided during transition from PRTLS to mated coast. GRTLS (aerojet DAP) holds the last commanded guidance attitude using RCS until completion of the -Z translation (ET SEP) maneuver.

Summary of FCS deltas (GRTLS vs. AERO JET DAP):

- GRTLS DAP uses both RCS thrusters and aerodynamic surfaces to control attitude after the ET SEP burn. (The rudder becomes active at MECO to aid in yaw control.)
- Moding parameters are different since guidance phases are different.
- GRTLS DAP commands RCS jets by developing rate errors proportional to rate and attitude commands from GC/STEER to perform mated coast and ET SEP.

The GRTLS flight control system automatically schedules the elevons and controls the body flap to null the elevon trim command. Pitch and roll commands will be provided by the elevons while yaw control will be provided by the rudder when $V_{rel} < 5000$ ft/s and $\alpha \leq 18^\circ$. RCS jets remain active for the following times:

- pitch jets remain active until $\bar{q} = 10$ lb/ft²
- roll jets remain active until $\bar{q} = 20$ lb/ft²
- yaw jets remain active until $M = 1.0$

Elevon trim is established throughout GRTLS by elevon position feedback.

BFS flight control: Three different DAP's control the vehicle through an RTLS. Ascent DAP has command until MECO. RTLS DAP assumes control between MECO and ET SEP. Descent DAP takes over at ET SEP until roll out on the runway. There are some significant differences in all the DAP's compared to PASS DAP's. The most significant difference in the ASC DAP is that there is no manual control capability. ASC DAP does not provide for reconfiguration in the case of multiple main engine failures.

The RTLS DAP also contains several important differences. Only automatic control of the RCS jets and the aerosurfaces shall be provided in the backup system, the jet select logic is a separate principal function and combines all three DAP's unlike the PASS jet logic which changes with each DAP. Also, the BFS processes at 12.5 Hz instead of 6.25 Hz in PASS. RTLS DAP uses 28 primary RCS jets during the mated coast. They are the forward Y and -Z and the aft Y and Z jets. During the ET separation, only the forward -Z jets and the aft Y and -Z jets are used. The BFS chooses individual jets within a group in different priorities than PASS. BFS order is based on maximizing a commanded moment or minimizing cross-coupling. The BFS protects against violations of tank constraints at all times, (i.e., exceeding the single aft tank constraint of five jets with crossfeed closed during mated coast).

Differences between descent DAP and nominal entry DAP (PASS) are as follows:

- CSS mode only is available
- The ADI error needle displays are terminated at TAEM
- No automatic nosewheel steering. Descent DAP (GRTLS) is restricted to 20 jets; the aft Y and Z. All other functions are the same as in PASS.

2.5.9 Attitude Processor

Mainstage (ascent): During ascent and entry the same attitude processor, called ascent/entry Attitude Processor (ATT PROC), is used. Only finite differences occur depending upon the major mode.

The function of the attitude processor is to derive attitude-related data for several user principal functions. These data include attitude, display variables, derived air data, and coordinated velocity in the body frame. ATT PROC is intended to meet these requirements during ascent and transition modes (transition being orbit insertion and deorbit). ATT PROC is performed at 12.5 Hz for both ascent and transition modes.

A summary of the inputs and outputs to ATT PROC for ascent and transition are shown in figure 2.5-23.

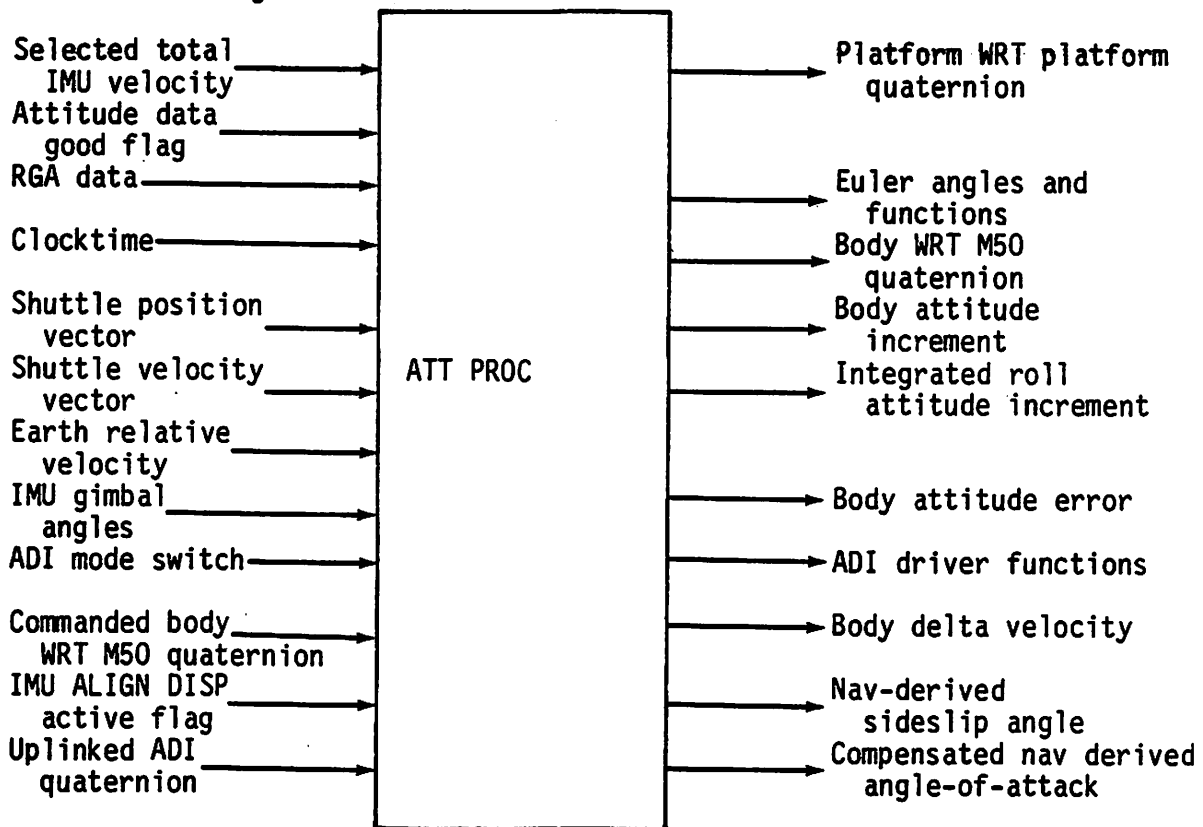


Figure 2.5-23.- ATT PROC inputs and outputs for ascent and transition.

Attitude processor modes:

- **Ascent Mode** - Ascent mode is initialized at prelaunch in OPS 1. During the actual launch, ATT PROC actually processes as if there are two sub-modes: (1) ascent before SRB separation and (2) ascent after SRB separation. Upon receipt of the OPS 1 MECO_CMD flag, the ATT PROC terminates ascent mode processing and commences transition mode processing.

- Transition Mode - Transition mode processing commences when the ATT PROC receives the MECO_CMD flag. Upon receipt of the OPS 2 command, transition mode processing is terminated and on-orbit processing commences.
- RTLS - The ATT PROC is moded directly from ascent to entry during RTLS at the transition to MM 602.

BFS attitude processor: The attitude processor of the BFS provides attitude information whenever the BFS is active, whether it is engaged or not.

An IMU is selected for attitude using an algorithm based on platform-to-platform misalignment.

The function of ATT PROC is to take the gimbal angles of the selected IMU and the body-mounted rate gyro outputs and compute the vehicle attitude. This attitude is then made available as quaternions, Euler angles, derived air data, and ADI ball-driving functions for other programs.

The processing actually takes place as a function of the major mode which is supplied by the Backup Events Controller (BEC).

MM 101 to MM 103* and MM 601	ASCENT
MM 103* to MM 106 } MM 301 to MM 303 }	TRANSITION
MM 304 to MM 305 and MM 602 to MM 603	ENTRY

*MECO zero thrust for nominal ascent, AOA, and ATO

Inputs and outputs: A brief summary of the ATT PROC inputs and outputs is shown on figure 2.5-24.

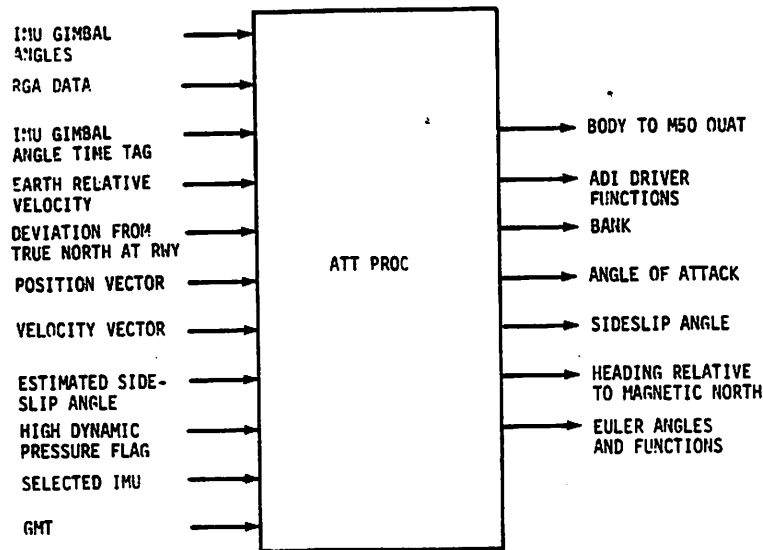


Figure 2.5-24.- BFS ATT PROC.

Depending upon which major mode, the inputs and outputs can vary slightly according to user demand requirements.

On-orbit attitude processor: The On-Orbit Attitude Processor (ATT PROC ORBIT) is used while the Orbiter is on-orbit during OPS 2 and OPS 8. The ATT PROC ORBIT is less complicated than the ascent/entry ATT PROC because the RGA data are not used on-orbit.

The function of ATT PROC ORBIT is to provide common coding to derive attitude and attitude rate information for several user functions. These users include the FCS and displays. ATT PROC ORBIT uses IMU SOP and NAV outputs to calculate these data.

ATT PROC ORBIT is performed at 6.25 Hz. ATT PROC initialization is accomplished by carryover from OPS 1 and OPS 3 and the internal initialization module. On-orbit processing commences upon receipt of the MM 201 command and terminates upon receipt of the MM 301 command.

Inputs and outputs: A summary of the inputs and outputs of ATT PROC during on-orbit are shown in figure 2.5-25.

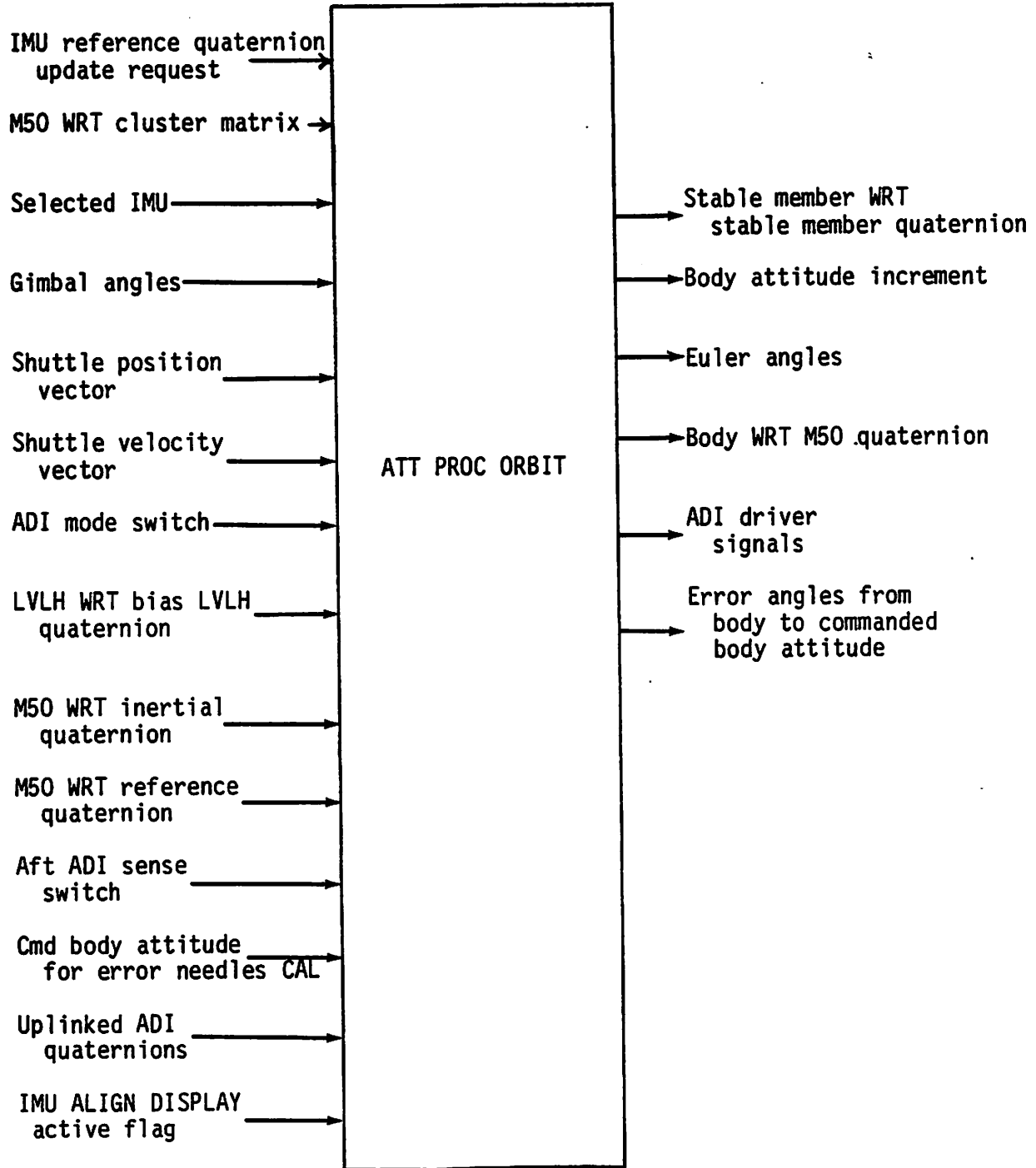


Figure 2.5-25.- On-orbit ATT PROC.

Brief description of internal functions:

- Initializes the quaternion stable member with reference to (WRT) M50, Q-SM-M50_k, for each IMU following on-orbit alignment. All other initialization is accomplished through constants and initial values or through carryover of OPS 1/3 variables.
- Comprises largely common coding for outer and inner loop passes at 1.04 Hz. Outer-loop processing establishes exact values for sines and cosines of half-gimbal angles' sums and differences. At 6.25 Hz, these functions are propagated using small angle approximations via first-order Taylor series. Common coding is used for forming platform WRT platform quaternions for use in IMU selection and the body WRT M50 quaternion. In outer-loop passes, low rate CRT display data, payload attitude data, and the LVLH frame are derived when required. An internal counter is used for determining whether inner-or outer-loop processing is done.
- Computes body attitude error angles and prepares ADI ball driver functions.

Entry ATT PROC description: The entry ATT PROC is basically the same as the ascent ATT PROC. Only finite differences occur depending upon the particular vehicle's major mode.

The function of the attitude processor is to derive attitude-related data for several user principal functions: These data include attitude, display variables, derived air data, and velocity in the body frame. ATT PROC is intended to meet these requirements during transition and entry modes.

ATT PROC is structured to derive the vehicle attitude quaternion using a selected IMU at low rate (1.04 Hz) and to propagate the attitude by integrating a quaternion differential equation, driven by selected pre-filtered RGA outputs, at high rate (12.5 Hz in transition and 6.25 Hz in entry). All other quantities are derived using the attitude quaternion at the rate and during those mission phases in which users require them.

The ATT PROC for the gliding portion of RTLS (GRTLS) is the same as the entry ATT PROC.

Inputs and outputs: A summary of the inputs and outputs to ATT PROC for Transition (T) and Entry (E) are shown in figure 2.5-26.

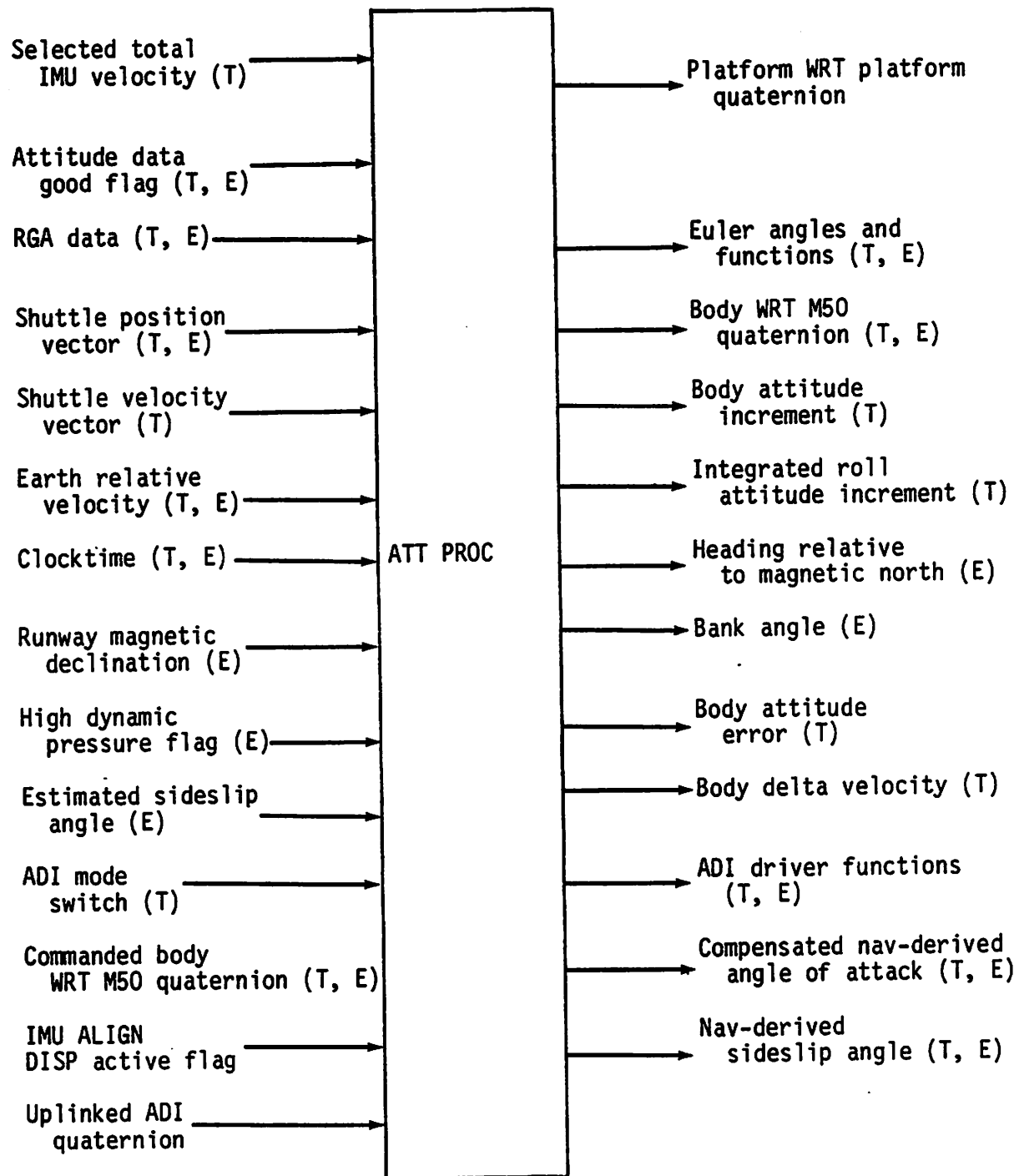


Figure 2.5-26.- Entry ATT PROC.

T = Transition
E = Entry

2.6 AOA/ATO ABORTS

This section covers two intact abort modes for ascent:

- Abort to Orbit (ATO)
- Abort Once Around (AOA)

2.6.1 AOA/ATO Overview

ATO and AOA abort modes are used when an engine failure or some other system failure makes a nominal orbit unattainable or undesirable. An ATO would be selected when the inertial velocity at MECO is less than the target velocity (MECO underspeed) or when called for by mission rules. The resulting orbit for an ATO would normally be a circular orbit at an altitude of 105 n. mi. as opposed to 150 n. mi. for a nominal orbit. However, an ATO could result in an orbit with a perigee altitude as low as 80 n. mi. The ATO option allows for time in orbit to analyze vehicle problems. Depending upon the situation, an ATO mission might end with a deorbit as early as the fourth revolution or it might be continued for the full duration of the planned mission.

For cases when the MECO underspeed is more severe, an AOA would be required. The AOA option could also be selected when some other system problem necessitated a return to the ground before the end of the first orbit. An AOA involves performing an OMS 1 maneuver to achieve an intermediate orbit and then a deorbit maneuver to return for a landing at the AOA designated landing site. There are two kinds of AOA trajectories. The OMS 1 target used for a normal AOA is the same as the ATO target; only the OMS 2 target is different. A normal AOA begins as an ATO with a downmode to AOA prior to the OMS 2 burn.

On the other hand, a shallow AOA, which is done when the MECO underspeed is greater than a predetermined amount, has a different target for both OMS 1 and OMS 2 burns and is selected as an AOA prior to OMS 1. The shallow AOA gets its name from its shallow approach to EI. The shallow AOA requires less engine performance than the normal AOA but it is less desirable because it involves a closer approach to thermal limits and to the skipout boundary. How all of the abort modes are related in terms of preference and mission duration is graphically shown in figure 2.6-1.

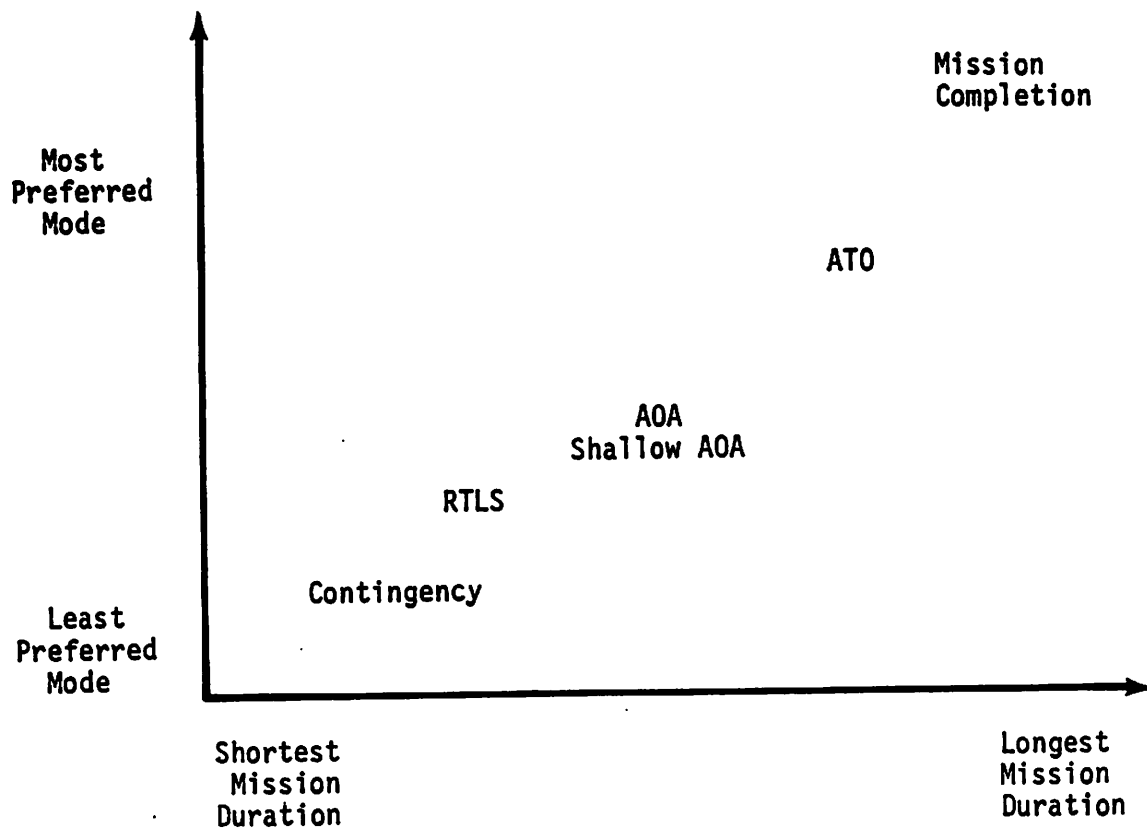


Figure 2.6-1.- Abort mode preference.

2.6.2 AOA/ATO Targets

Targets for the OMS 1 and OMS 2 maneuvers consist of the following parameters.

HT: Desired target radius in n. mi.

θ_T : OPS 1 - In-plane downrange angle from the launch site to the target radius vector in deg

OPS 3 - In-plane angle from EI to the point of OMS 2 ignition in deg

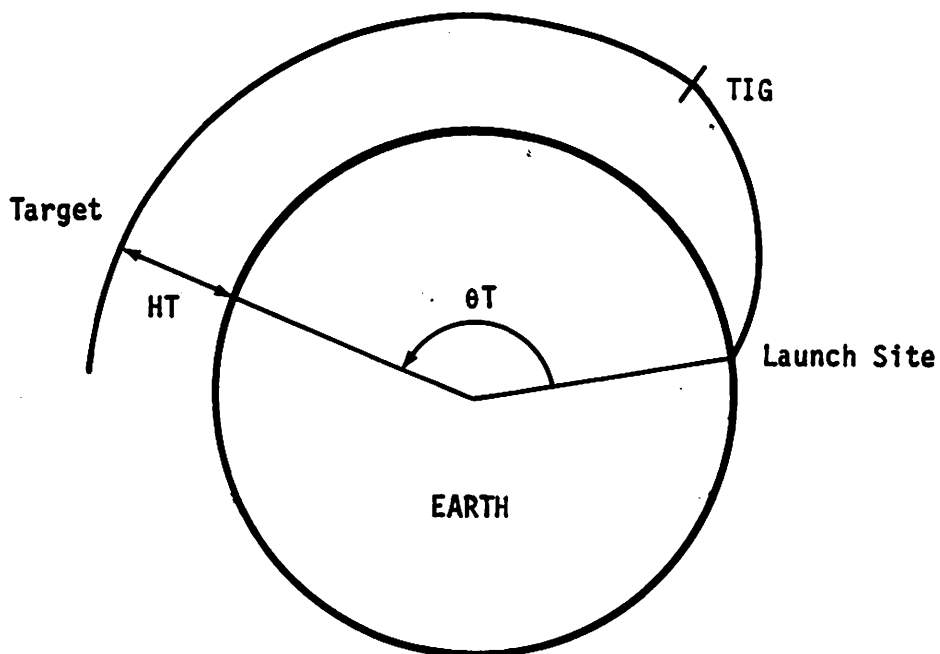
C1 and C2: Velocity target parameters for deorbit only, defined by the following equation:

$$V_{\text{VERTICAL}} = C1 + C2 \times V_{\text{HORIZONTAL}}$$

TIG: Time of ignition in sec (MET)

These target parameters are described graphically in figure 2.6-2. OMS 1 and OMS 2 target sets for STS-1 are listed in table 2.6-1.

OPS 1



OPS 3

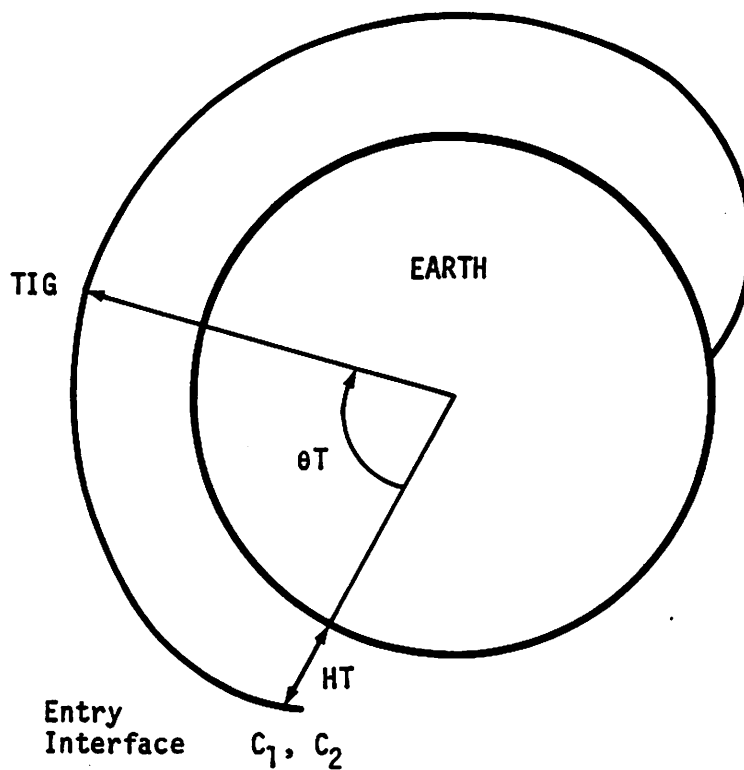


Figure 2.6-2.- OMS target parameters.

TABLE 2.6-1.- OMS 1 AND OMS 2 TARGETS FOR STS-1

Option no.	MECO state	OMS 1 (C1 = 0, C2 = 0)				OMS 2					
		Targets	TIG (MET) hr:min:sec	HT (n.mi.)	θT (degrees)	Targets	TIG (MET) hr:min:sec	C1	C2	HT (n.mi.)	θT (degrees)
1	Nominal	Nominal I-load	10:43	149	157	Nominal I-load	43:09	0	0	157	337
2	Nominal	Nominal Manual	20:43*	149*	237*	Nominal Manual	1:03:45*	0	0	157	337
3	USPD <185 ft/sec	ATO I-load	MECO + 2	106	231	ATO I-load	1:02:37	0	0	105	51
4	USPD <286 ft/sec	ATO I-load	MECO + 2	106	231	AOA Manual	54:23*	14589*	-0.5852*	66*	32*
5	USPD <340 ft/sec	ATO I-load	MECO + 2	106	231	AOA (Shallow) Manual	33:48*	20035*	-0.78716*	66*	161*
6	USPD <630 ft/sec	AOA (Shallow) I-load	MECO + 2	80	193	AOA (Shallow) Manual	51:31*	20924*	-0.8211*	66*	74*
7	Nominal	ATO I-load	20:43*	106	231	ATO I-load	1:00:22*	0	0	105	51
8	Nominal	ATO I-load	20:43*	106	231	AOA Manual	52:10*	14589*	-0.5852*	66*	82*
9	Nominal	AOA (Shallow) Manual	20:43*	90*	146*	AOA (Shallow) Manual	52:31*	20266*	-0.7966*	66*	67*
10	Nominal	Nominal I-load	10:43	149	157	AOA I-load	53:14	14738	-0.5912	66	87
11	Nominal	Nominal I-load	10:43	149	157	AOA (Shallow) Manual	23:42*	17216*	-0.6795*	66	195*

*Requires crew input
 NOTE: Targets are subject to change and are presented here as examples only.

2.6.3 Abort Selection

There is a boundary in the ascent trajectory called the press-to-MECO boundary. When the vehicle passes this point, a call is made to the crew indicating that even if one SSME fails, MECO can be achieved with the capability to perform at least an AOA. Later in the trajectory there is another call when the vehicle has passed the single engine press-to-MECO boundary, which means that the capability exists to perform at least an AOA even if two SSME's fail. At MECO, the magnitude of the underspeed (if one exists) will determine whether the required mode is nominal, ATO, AOA, or shallow AOA. The abort regions are shown in figure 2.6-3, the abort selection logic for STS-1 is illustrated in figure 2.6-4.

2-320

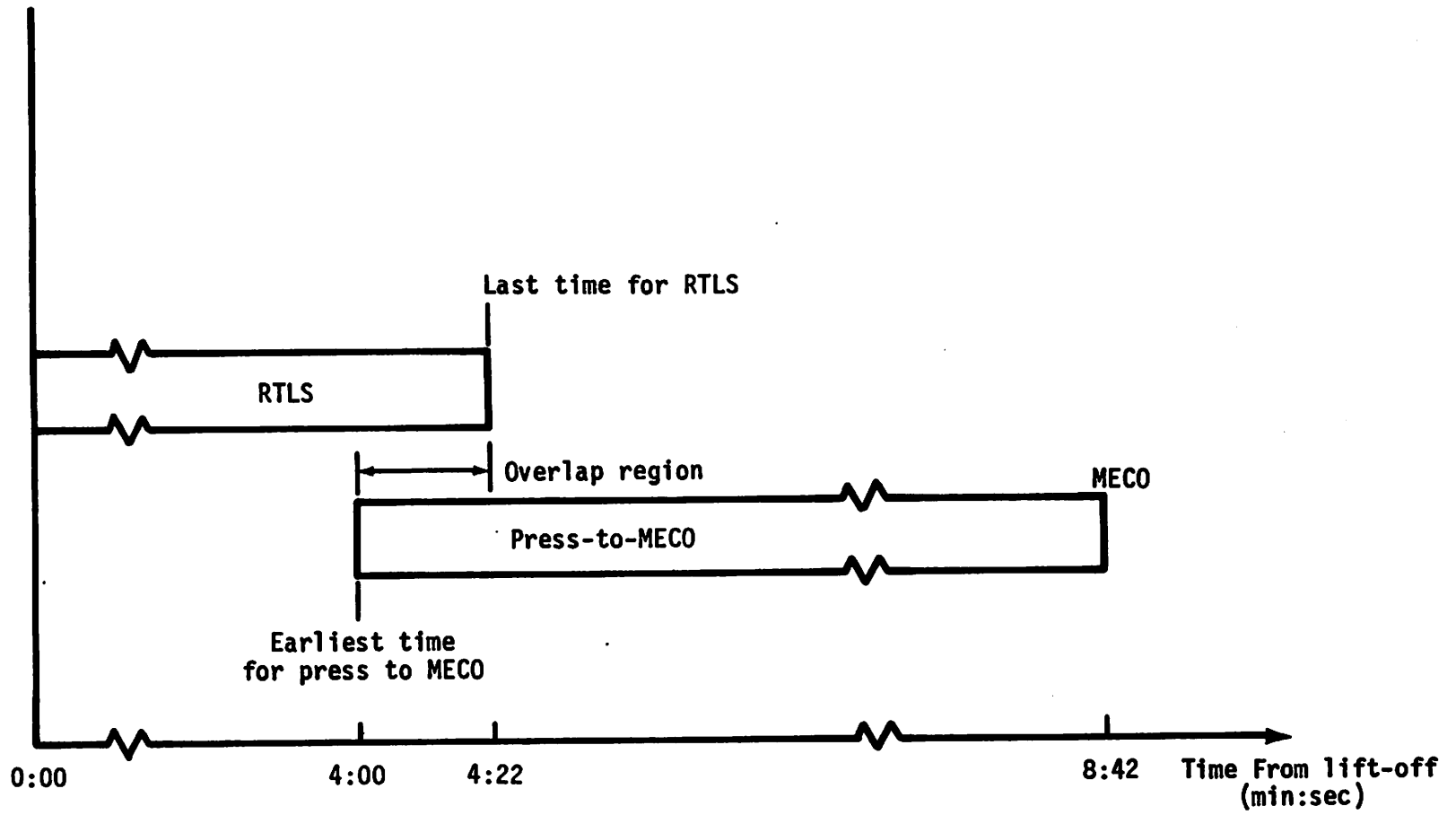


Figure 2.6-3.- The RTLS and press-to-MECO regions.

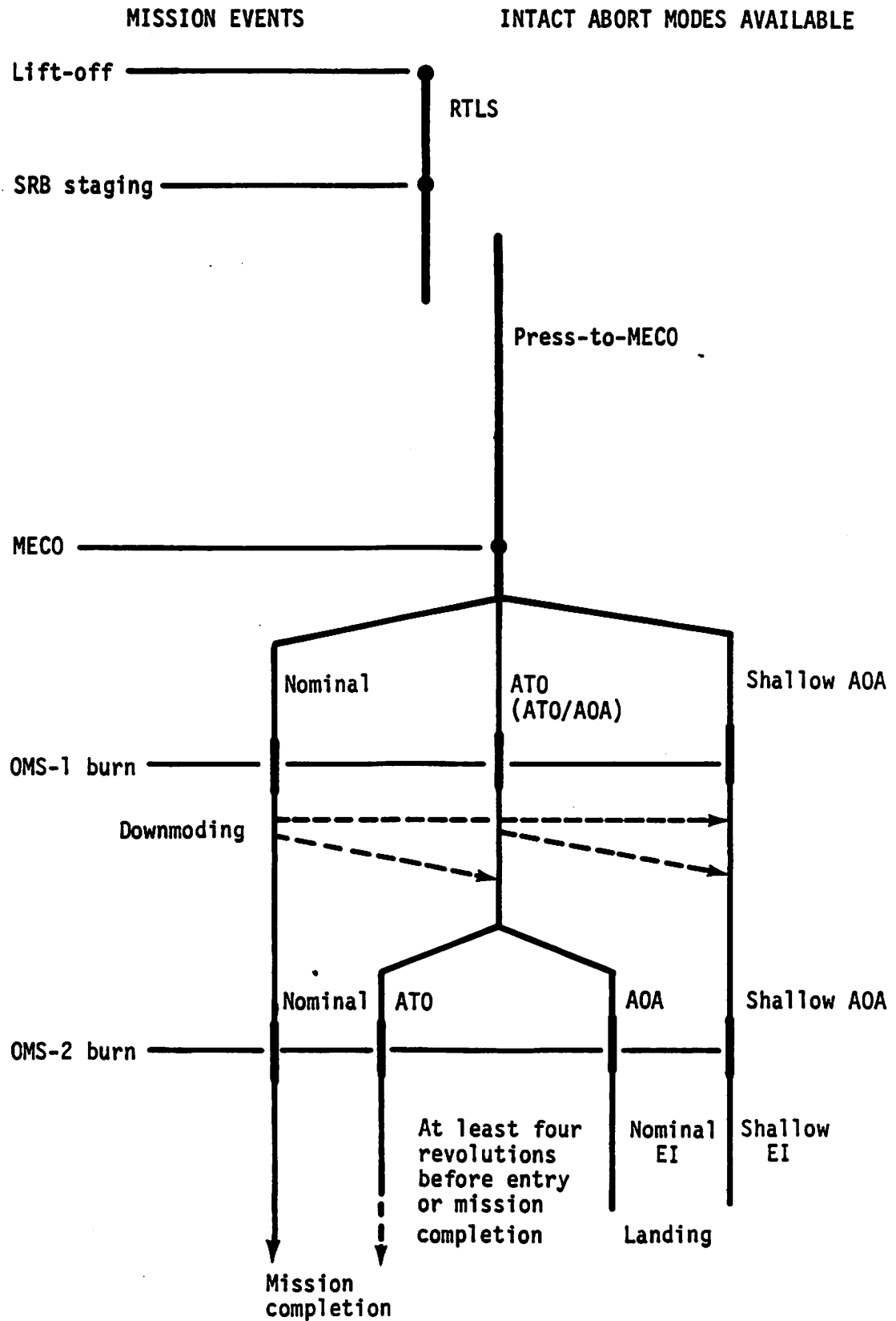


Figure 2.6-4.- Abort mode decision logic.

The following is a summary, outlined by mission phase, of target selection and other aspects of ATO and AOA aborts.

First and Second Stage Ascent

If an SSME fails:

- The remaining SSME's are throttled to 100 percent.
- Auto shutdown of remaining engines is inhibited.
- In first stage, guidance switches to an engine-out I-load.
- The velocity at which the engine failed is stored by guidance.

Prior to press-to-MECO:

- Trajectory is shaped to reach press-to-MECO boundary as soon as possible.
- As a precaution, guidance assumes that an SSME will fail at the predicted press to MECO time.
- The ABORT PBI is illuminated for RTLS and contingency aborts only.

After press-to-MECO:

- At least the AOA option is available as long as two SSME's continue to operate.
- No selection of AOA or ATO is made prior to MECO, and abort PBI is not illuminated even if AOA/ATO abort conditions exist.

After single engine press-to-MECO:

- At least the AOA option is available as long as one SSME continues to operate.

After MECO

- There is only one MECO target for STS-1. The target inertial velocity is 25,668 ft/s.
- The OMS TARGETS cue card and the OPS 104 MNVR display are consulted to determine proper OMS 1 target.
- Depending on the underspeed, the target will be nominal, ATO, or shallow AOA.

1. If nominal - the I-loaded target can be used.
2. If ATO - rotary switch (set to ATO) can be used to load ATO target or it can be loaded manually. (This target is also used for an AOA, when possible).
3. If shallow AOA is required - rotary switch (set to AOA) can be used to load shallow AOA target or it can be loaded manually.
4. For systems failures, the cue card provides the proper targets.

- If MECO is nominal, the OMS 1 burn can be delayed using manual targets provided on the cue card.

After OMS 1

- OMS TARGETS cue card and MNVR display are again consulted to determine proper OMS 2 or deorbit target.
- Downmoding can be done at this time. The cue card provides OMS 2 targets corresponding to whichever mode was selected for OMS 1.
- If ATO is the selected mode, the OMS 2 burn is used to raise the perigee as high as possible, up to 105 n. mi.
- If AOA is selected, the OMS 2 burn becomes a deorbit burn and is performed in OPS 3. There are different targets for normal AOA and shallow AOA.
- If the OMS 1 burn was delayed there are separate sets of targets which must be used for the OMS-2 burn.

After the OMS 2 Maneuver (ATO)

- The vehicle will be in an orbit with perigee altitude of at least 80 n. mi.
- The vehicle can complete four revolutions and make an entry and land at Edwards AFB.
- If conditions warrant, the mission can be continued for a longer time, up to full mission duration.

After the Deorbit Maneuver (AOA)

- The vehicle completes entry and lands at the Northrup Strip in New Mexico.

As was noted above, if the MECO state is nominal or nearly nominal (under-speed <80 ft/sec), the OMS 1 burn can be delayed by 10 min, at which time the Madrid tracking station is acquired. This might be done if, for example, problems were encountered with ET separation. A full range of options is available with the delayed burn including nominal, ATO, or AOA. In the case of a delayed OMS 1 ATO, it is possible to achieve a 105 by 80 n. mi. orbit with only one burn. This orbit would be acceptable for at least four revolutions.

The software is capable of more complex abort targeting logic than will be exercised on STS-1. This includes:

- Two MECO targets, one for nominal and one for aborts
- Selection of AOA and ATO prior to MECO
- Use of OMS and RCS burns prior to MECO

2.6.4 BFS for ATO and AOA

The BFS has the capability to perform ATO and AOA aborts. The only differences between the primary and backup systems as related specifically to aborts are the following:

- The BFS provides no automatic guidance during OMS maneuvers. The crew must maintain proper attitude using the ADI and manual controls.
- The MPS propellant dump is done automatically when the BFS is engaged and there is no option to perform the dump manually.

Prior to OMS maneuvers, the BFS MNVR display should be monitored to insure that the targets and burn parameters are consistent with the primary system.

2.7 CONTINGENCY ABORTS

There is very little software specific to contingency aborts. There is virtually no guidance and the control mode is CSS. A plot of H_A vs. V_A is shown in figure 2.7-1. From this plot a spec was developed (SPEC 52) and the trajectory is flown using this spec. The various regions of a contingency abort are shown in the figure.

There are two sequences specifically associated with contingency aborts. They are:

- Contingency dump sequence
- Fast ET SEP sequence

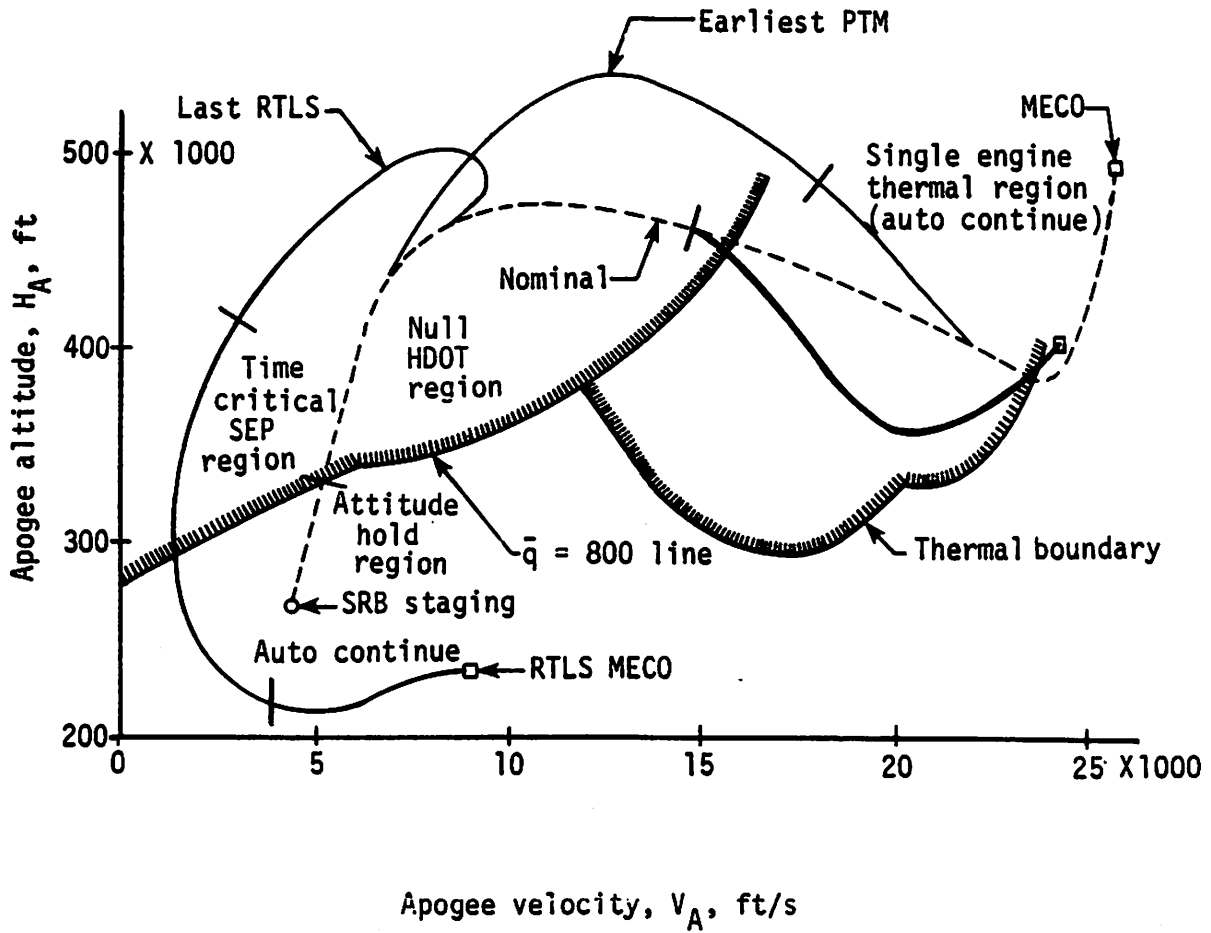


Figure 2.7-1.- Contingency abort H_A - V_A plot.

2.7.1 Contingency Dump Sequence

The contingency dump is initiated via SPEC 52. The dump is needed to achieve a proper c.g. for the contingency entry.

The abort sequencer turns on the two OMS engines and interconnects OMS propellant to 24 RCS jets. The two OMS engines and the 24 RCS jets burn ≈ 100 lb/s of OMS propellant. The dump terminates on an OMS quantity level (currently 8.6 percent) which allows enough OMS fuel for single engine roll control during powered flight.

2.7.2 Fast ET Separation

Fast SEP is available in MM 102 and 601. The fast SEP sequence is initialized by the ET SEP pushbutton. The sequence immediately shuts down the SSME's and 3 sec later fires the SEP bolts. At SEP, the -Z translation jets (4 forward, 6 aft) are turned on. MM 602 will come up automatically when $\alpha > 10^\circ$ and 10 sec have passed since sep. A timeline of fast SEP is shown in figure 2.7-2.

- ∇ SSME - 1 Shutdown command
- ∇ SSME - 2 Shutdown command
- ∇ SSME - 3 Shutdown command

- ∇ Start fast SEP

$P_c \leq 30$ percent for all SSME'S

- ∇ SSME - 1 LOX prevalve close command
- ∇ SSME - 2 LOX prevalve close command
- ∇ SSME - 3 LOX prevalve close command

- ∇ SSME - 1 LH2 prevalve close command
- ∇ SSME - 2 LH2 prevalve close command
- ∇ SSME - 3 LH2 prevalve close command

- ∇ Arm and fire umb. plate PIC's
- ∇ Arm and fire PIC's (struct. sep.)

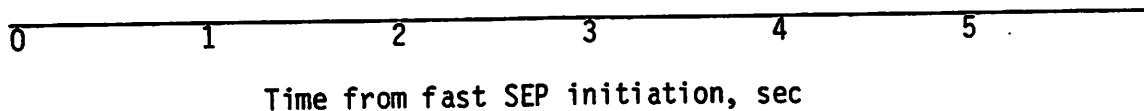


Figure 2.7-2.- Fast SEP sequence.

SECTION 3 SYSTEMS MANAGEMENT

Systems Management (SM) software provides the user a means to monitor vehicle subsystems. Information related to the vehicle subsystem health and status is provided to the user through automatic fault detection, performance data readouts, and configuration status displays. Vehicle subsystem management is accomplished by user assessment of information provided by SM and subsequent user actions to affect vehicle configuration. SM also performs special computations and sequences to provide additional information and to provide assistance in the maintenance of several vehicle subsystems.

Data monitored include Operational Instrumentation (OI) data from the Pulse-Code Modulation (PCM) master unit, data from the payload MDM's (includes midbody MDM's), and software-generated data.

In addition to the monitoring of the vehicle subsystem, SM software provides the user the capability to update parameters which control SM processing.

The SM structure consists of the three major functions of basic SM processes, special processes, and display and uplink controlled processes. The functions and the major processes included in each are shown in the hierarchy diagram in figure 3.0-1.

Basic SM processes acquire data from the various vehicle subsystems and monitor this data to provide fault detection and warning in the event of a vehicle subsystem failure or performance degradation. Data acquisition acquires the data and makes it available for further processing. Performance monitoring provides the control for FDA and precondition processing (dynamic limit selection for FDA processing).

Special processes, under the control of a special processes executive, provides specialized computations and processing sequences to determine and control the status and performance of various vehicle subsystems. Vehicle subsystems data necessary for special processes are acquired by special processes data acquisition and made available to basic SM processes. Error conditions determined as a part of computation or sequences are annunciated from the special processes.

Display and uplink controlled processes provide functions which can be used to initiate, alter, or terminate certain processing within basic SM processes and special processes. Some of these functions provide the on-board capability to modify certain tables used by SM processes. Display and uplink controlled processes consist of table maintenance, subsystem configuration management, communications, instrumentation, antenna management, mass memory patch, payload bay doors, payload control, remote manipulator system, and the uplink processor.

An overview of the SM basic process flow for PASS is shown in figure 3.0-2.

3-2

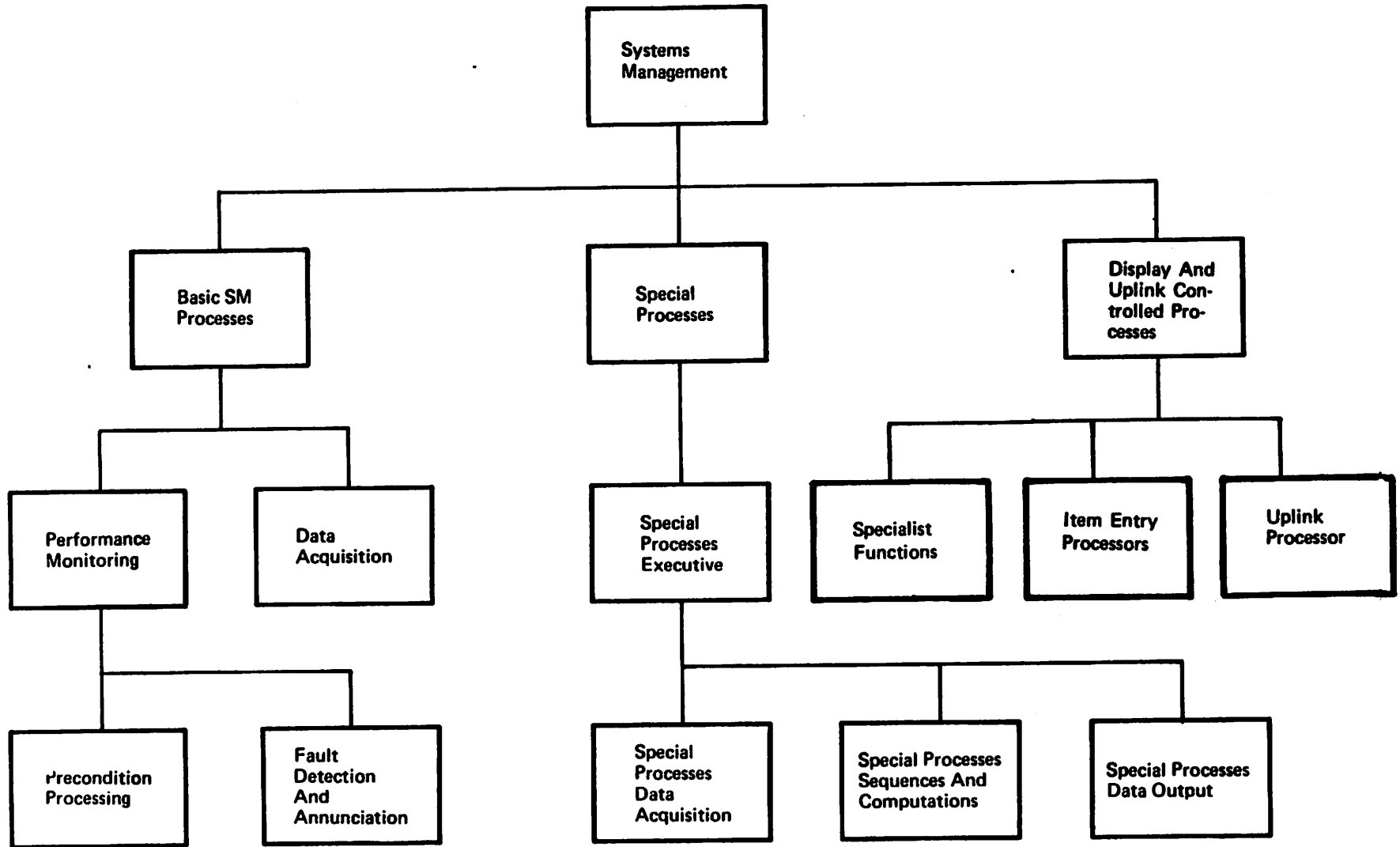


Figure 3.0-1.- Systems management diagram.

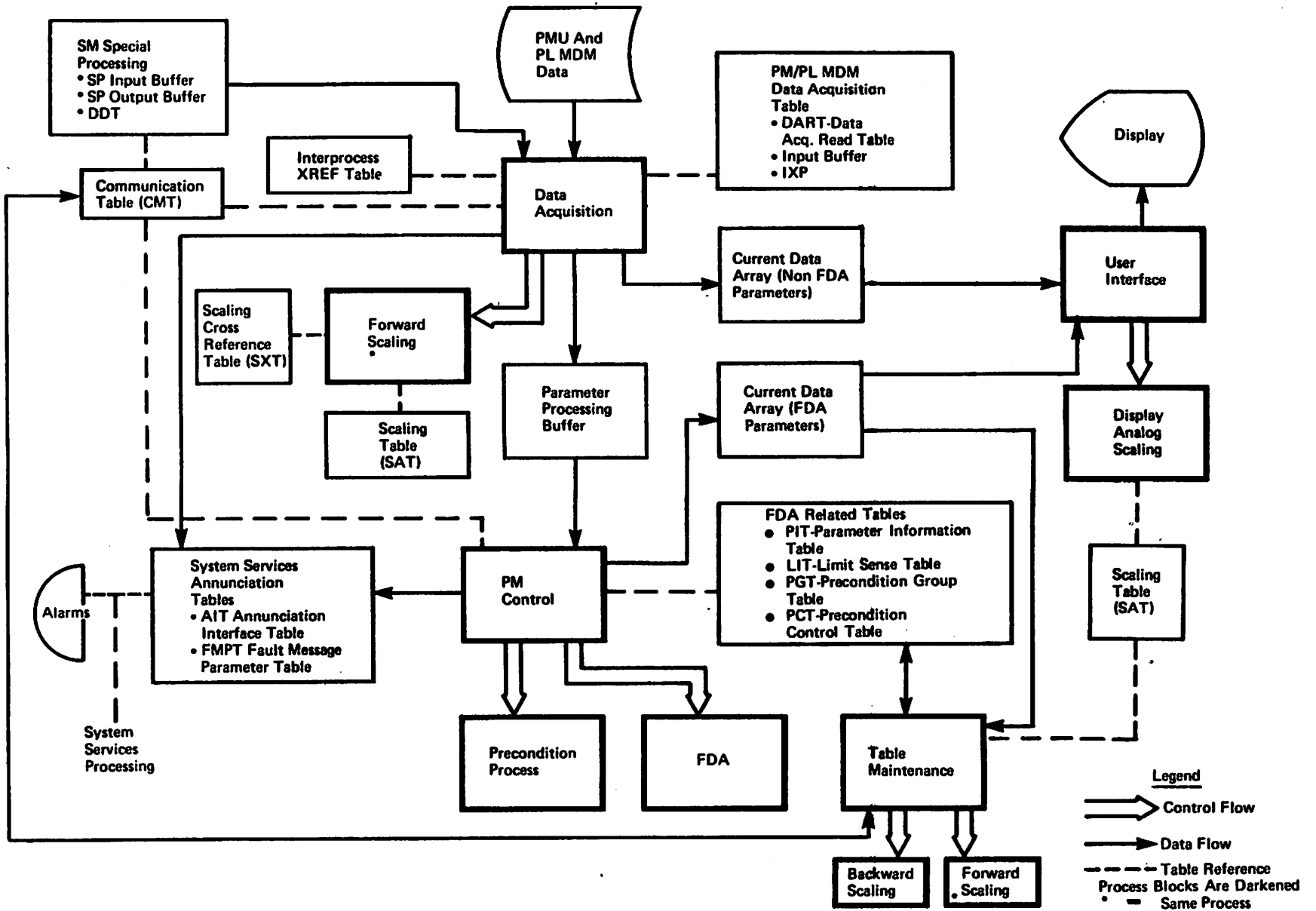


Figure 3.0-2.- Basic SM processes.

3.1 BASIC SM PROCESSES

The SM Fault Detection and Annunciation (FDA) system is designed to provide backup C&W and an alert system for the crew on most systems. The FDA receives its inputs from sensors through the OI Pulse Code Modulation Master Unit (PCMMU). The parameters are identified and processed in the FDA by a modified Master Measurements List (MML) number. Block diagrams of the FDA major task are found in figures 3.1-1, 3.1-2, and 3.1-3. Some of the inputs have to go through precondition steering prior to being compared against limits. Precondition steering is the decision-making portion of FDA whose function is to determine which fault detection testing is required and the conditions which must be met for each individual parameter. Precondition steering is executed once per second for each parameter. It also determines if additional parameters are required in the process. The same precondition steering is used for several parameters where it is applicable. A specific limit set is selected when certain preconditions are met. If no preconditions are met, then the parameter faults back to the prime or first limit set. Precondition steering will also allow parameters or groups of parameters to be bypassed for up to 32,767 seconds to allow the subsystem to stabilize. An example of this is the APU gearbox lube oil output pressure which is disabled for 10 seconds when it changes limit sets.

The FDA has the capability to limit sense analog parameters, engineering unit parameters, and discretetes. The limits are classified as simple or multiple. A simple limit set for an analog or engineering unit parameter is one high and one low constant value. A simple limit set for discrete parameter is a single discrete value. A multiple limit set contains either two or three simple limit sets from which one set is selected by preconditioning. Class 2 alarms (backup C&W) only have simple limit sets. Class 0, 3, or 4 alarms can have either simple or multiple limit sets.

BFS basic SM processes is a similar version of the PASS system primarily intended for use only during ascent and entry flight phases. An overview of the SM basic process flow for BFS is shown in figure 3.1-2.

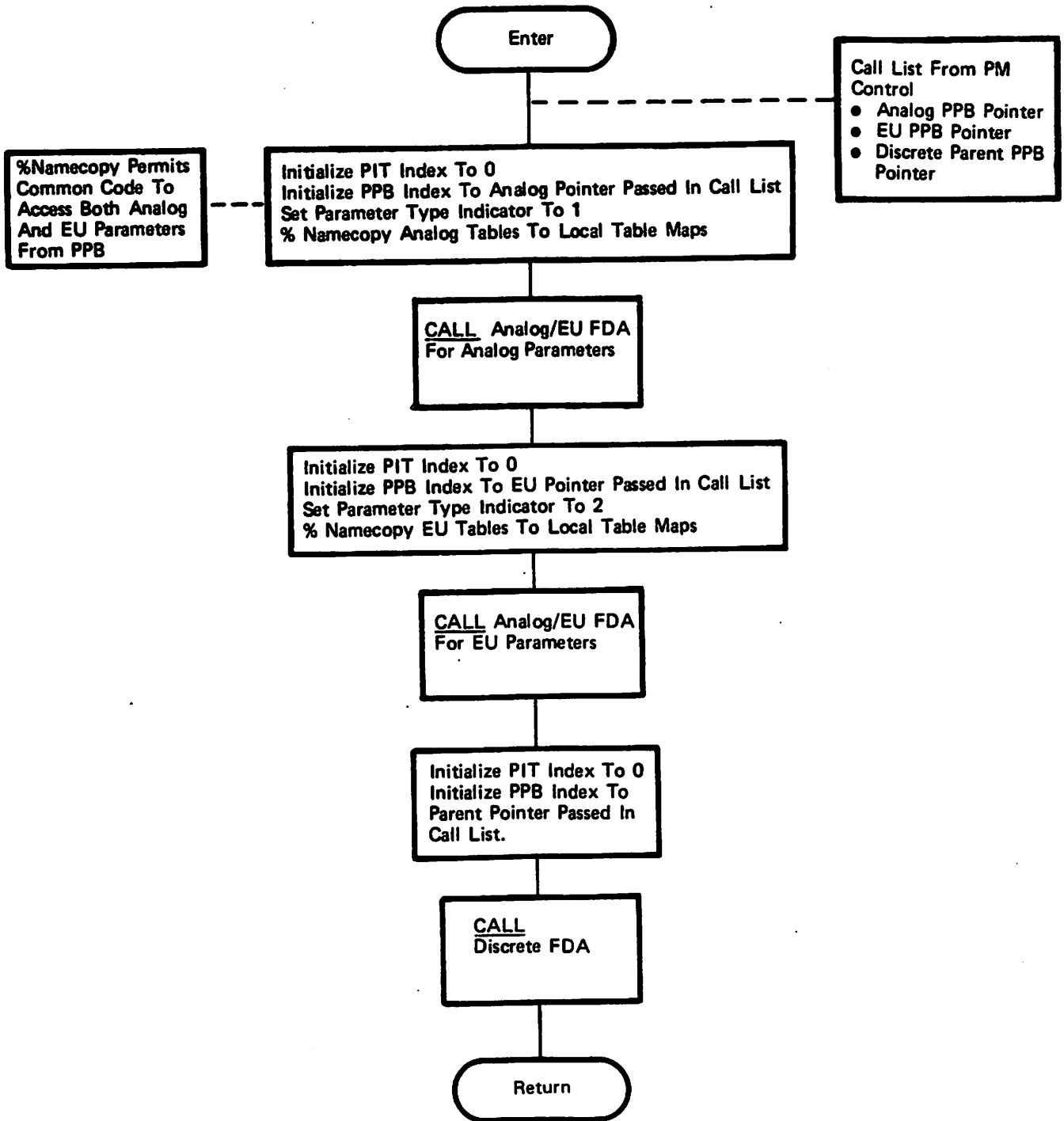


Figure 3.1-1.- Fault detection and annunciation.

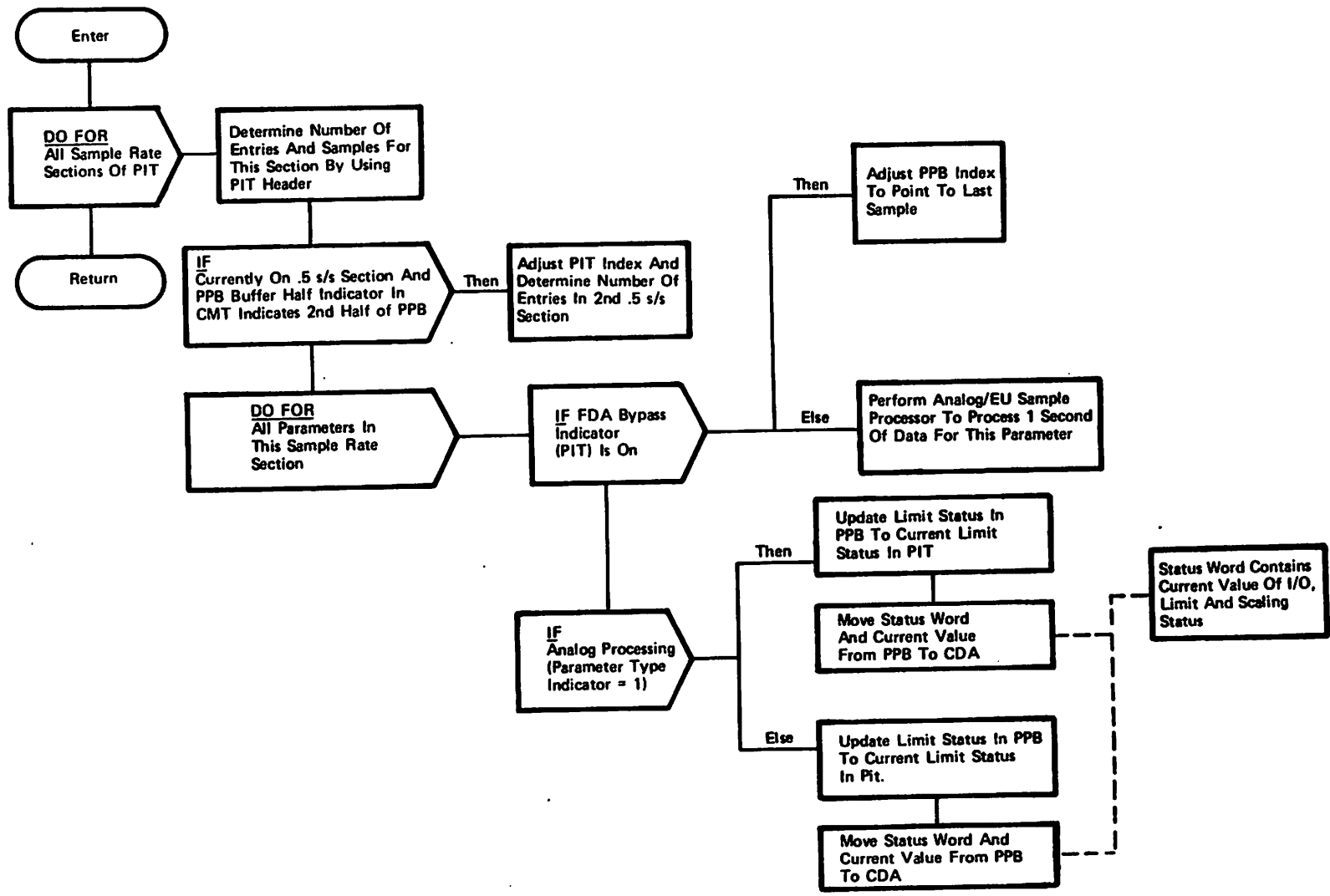


Figure 3.1-2.- Analog/EU FDA.

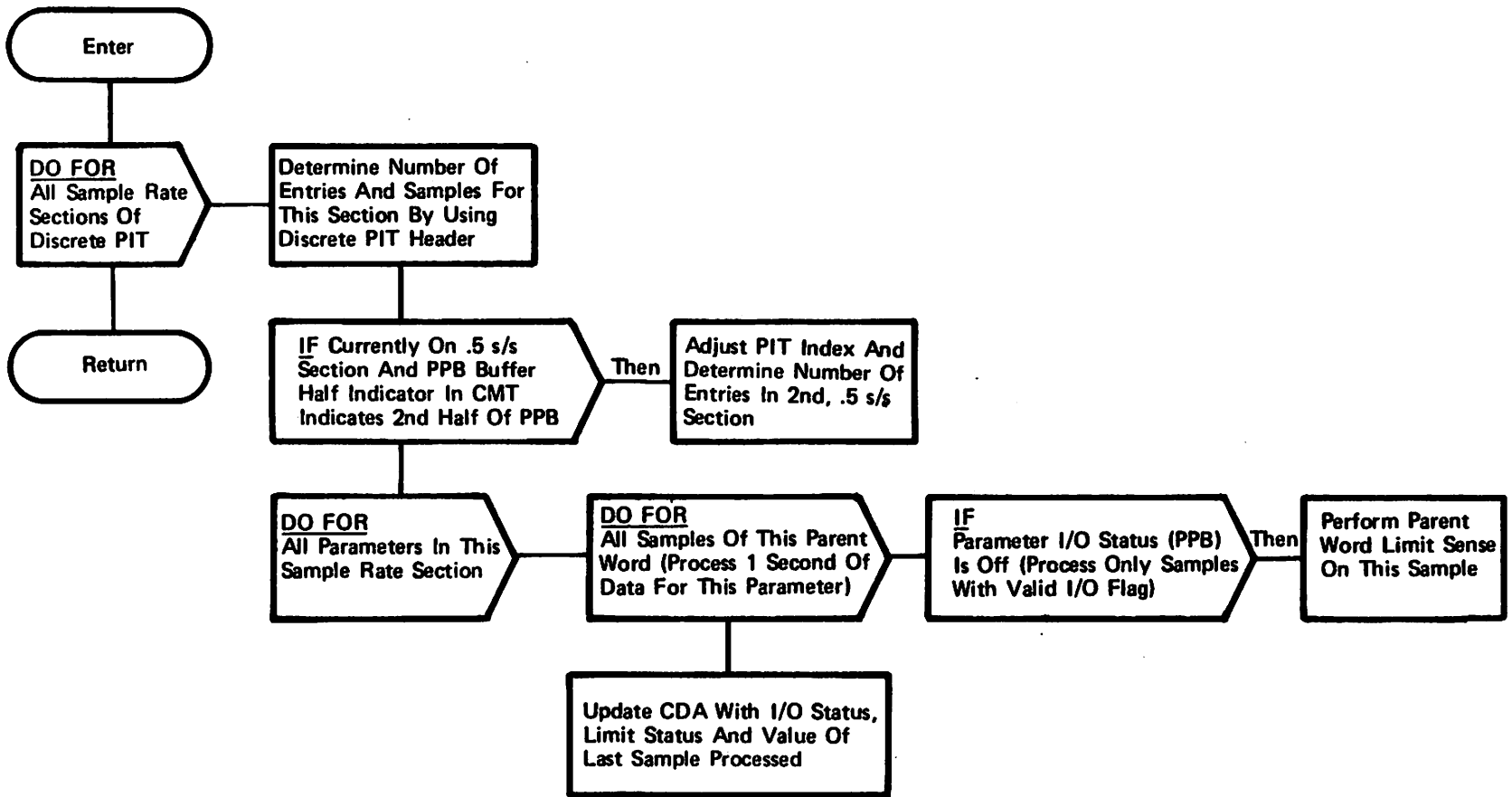


Figure 3.1-3.- Discrete FDA.

SM PROGRAM FUNCTIONS

FAULT DETECTION AND ANNUNCIATION

- Precondition Steering
- Limit Sensing
- False Alarm Avoidance
- Fault Annunciation Output
- Scaling

SYSTEM CONFIGURATION MANAGEMENT

- Switch monitoring
- Analog value monitoring

CHECKPOINT

- Retrieves and stores constants

PCMMU TELEMETRY FORMAT LOAD (TFL)

- Retrieves the selected TFL and loads it in the operating PCMMU

INITIALIZATION

- Allows changes of mission dependent data
- Starts current time counting

PDI DECOM LOAD AND FETCH

- Acquires, verifies, and stores external data

MASS MEMORY PATCH

- Modifies mass memory

PSP COMM

- Communicate with the payload signal processor

DISPLAY FUNCTIONS

- Handles interfaces with the following displays

OPS

- Antenna
- Payload bay doors

SPEC

- SM table maintenance
- Systems check
- PCMMU/payload COMM
- PDRS control

DISP

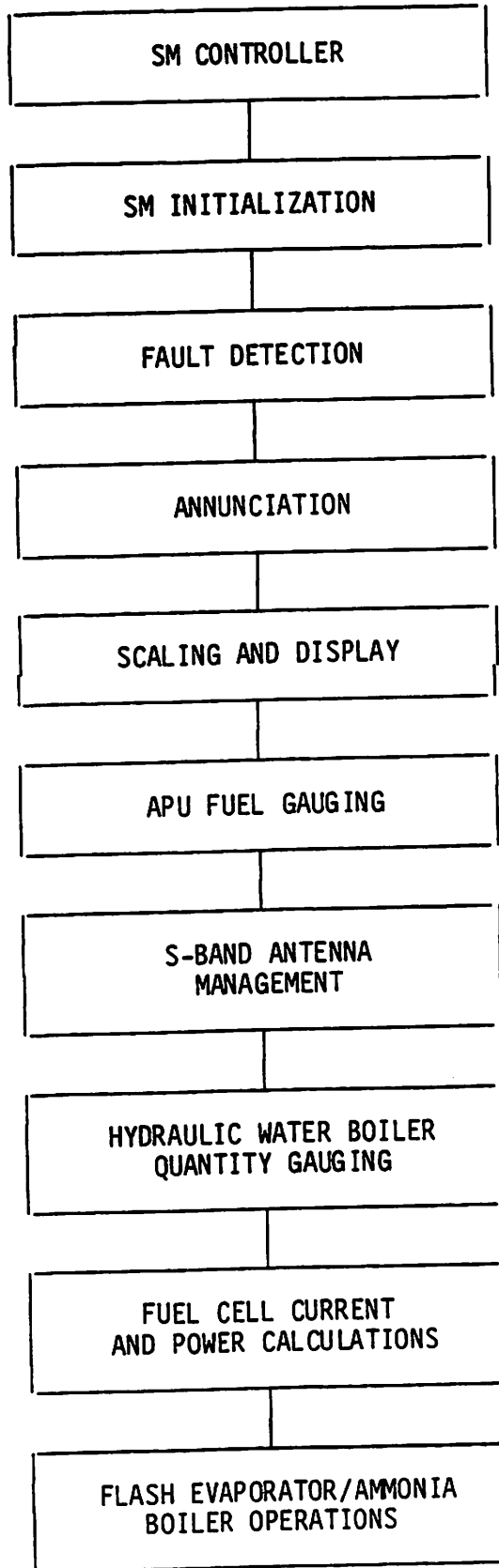
- SM systems summary 1 & 2
- Electric
- Environment
- APU hydraulic
- Cryogenic/fuel cells
- COMM/recorder
- APU/environment thermal
- Hydraulic thermal
- Propellant thermal
- PDRS status

SPECIAL PROCESSING

- APU fuel QTY
- Fuel cell computation
- Antenna management
- Hydraulic water boiler QTY
- Fuel cell purge
- Hydraulic fluid temp control

- Payload bay door sequences
- O₂ + N₂ QTY computation
- RMS
- Standby H₂O coolant loop control
- H₂O pump delta press
- Recorder percent tape comps

BFS SM/SP PROGRAMS



3.1.1 Data Acquisition

The basic SM data acquisition function acquires all data to be provided by the other basic SM functions. The SP data acquisition function acquires all data required by SP from the PCMMU and the payload MDM's. General data acquisition is a function called by both basic SM and SP to acquire and screen the data.

The basic data acquisition module controls cyclic acquisition of data for Subsystem Configuration Management (SCM), precondition, and FDA. It gets this data from the PCMMU, payload MDM's, and Special Processes (SP) com-pools. Data not acquired due to I/O failure is flagged invalid. The basic data acquisition also gets data from internal SM tables.

SM data acquisition overview for PASS is shown in figure 3.1-4.

Data acquisition in BFS is done by the Backup Operating System (BOS). The basic functions of BOS is to perform inputs, run jobs, and perform outputs over a 960 ms period.

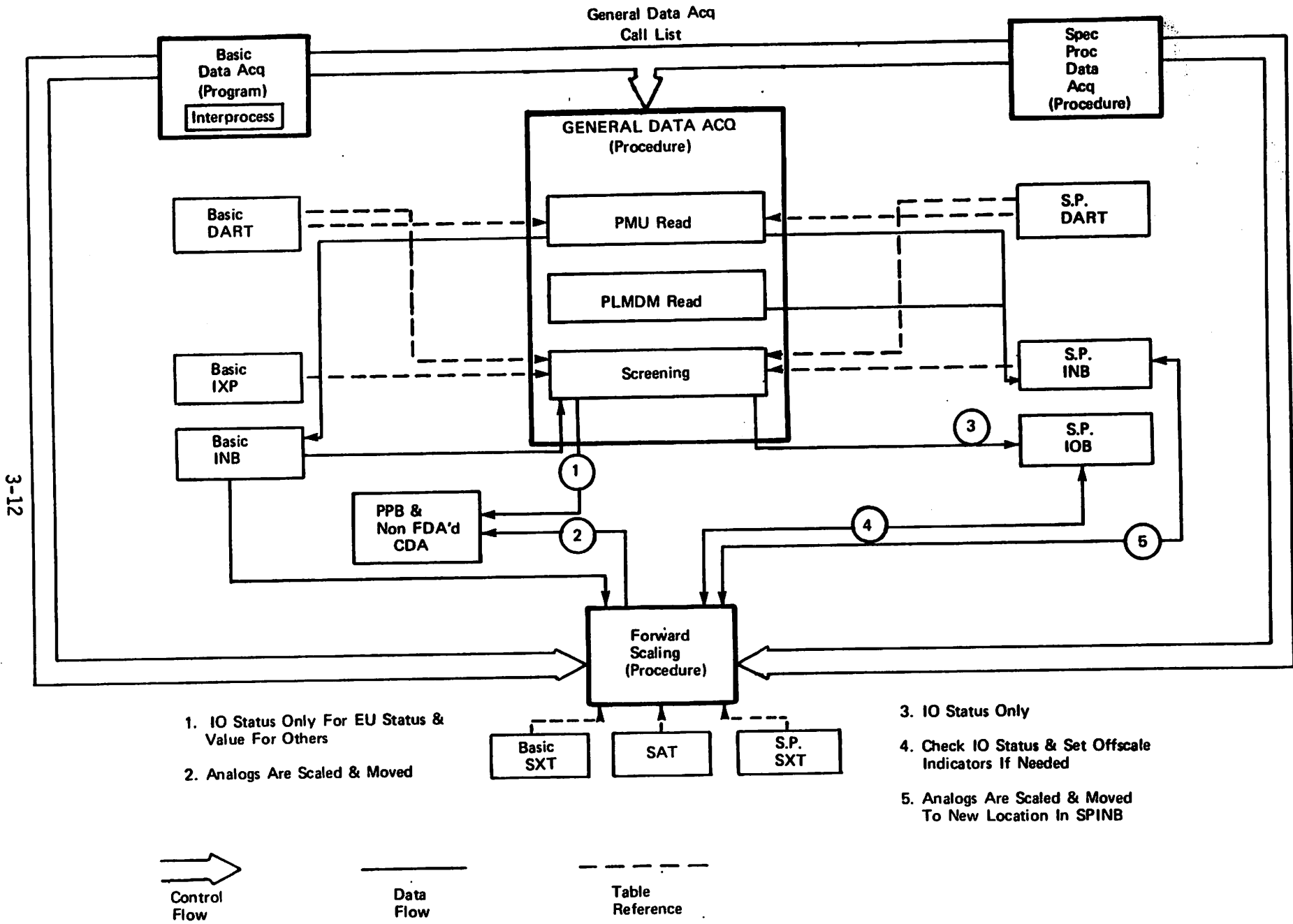


Figure 3.1-4.- Systems management data acquisition overview.

3-12

Analog/EU processing: FDA interrogates the FDA bypass indicator in the Parameter Information Table (PIT) entry for each parameter to determine if FDA processing is bypassed for that parameter. The FDA bypass indicator is set on by precondition processing for each parameter of a group during group warmup and reset by precondition processing when warmup is completed. In any case, parameters with the FDA bypass flag enabled do not undergo FDA processing; however, the last sample and associated I/O status are moved from the Parameter Processing Buffer (PPB) to the CRT Display Annunciation (CDA), and the limit status indicators are cleared. FDA processes successive PIT entries similarly until a parameter that is not bypassed is found. All samples from one second of data are processed for that parameter as follows:

- Status flags in the PPB are checked to determine if the parameter sample is valid. If a sample is invalid, further FDA processing of that sample is skipped. Limit sense processing is thus performed on all valid samples of that parameter. FDA obtains the current sample from the PPB and the current limit values from the Limit Table (LIT). An indicator of the current limit set is found in the PIT entry for this parameter. The current value is compared to the high limit, then the low limit. If either of these comparisons results in an indication that the current value is beyond one of the limits, the parameter is determined to be out of limits. Noise filtering is then performed.
- If the limit status for this parameter has changed from in limits to out of limits (or vice versa), the current noise (N) count is re-initialized and the limit status for this parameter is updated in the analog/Engineering Units (EU) portion of the PIT. Checks are then made on the parameter's N count and the limit status and annunciation status to determine if annunciation should be enabled. A limit failure is annunciated only if three conditions have been met:
 - This parameter has failed limit sense processing N (maximum noise count) consecutive times.
 - Annunciation for this parameter is not inhibited (annunciation inhibit indicator in PIT is off).
 - This parameter is not currently in a hard fail state; i.e., it has not previously been out of limits or it was out of limits and subsequently came into limits N consecutive times.
- Annunciation is enabled by issuing a user interface macro. Inputs to the macro are an index to a system Fault Message Parameter Table (FMPT) and a limit status indicator. The FMPT index resides in the PIT entry of each parameter. Annunciation for alarm class 0 parameters causes no alarms or fault messages to be generated. (Alarm classes are defined in SFOM, Volume 1.)
- When the last sample of a parameter has been processed, FDA moves that sample and its associated I/O status to the FDA CDA. Processing continues on all analog/EU parameters (PIT entries) until all have been processed.

3.1.2 Fault Detection and Annunciation

The FDA function executes cyclically, when enabled, to perform the limit sense and false alarm avoidance functions on parameters that have limit defined and to initiate the annunciation process for those which have failed.

Inputs to FDA are PPB pointers passed in the call list as discussed above and various table entries of information needed to perform fault detection and annunciation on each parameter. Parameter values and corresponding I/O statuses are located in the PPB. Other information such as noise counts, limit status flags, limit values for discrete parameters, pointers to various tables, and annunciation information is located in the PIT. Limit values for analog and EU parameters reside in limit table. All such table information resides in compool.

The control flow is presented in figures 3.1-1 through 3.1-3. The FDA module is a table driven processor. When initiated, FDA accesses the PIT and initiates processing on the first entry in the PIT. Every sample of a parameter is processed. Parameter samples are buffered by the data acquisition module in the PPB.

Access into the PPB is gained through the PIT. This accessing is not done by means of an index in the PIT for each parameter, but rather a mapping scheme which relates the location of a parameter in the PIT to its location in the PPB. The PIT is organized into nine sections - three blocks (a block each for analogs, EU's, discretets) of three groups each (5 s/s, 1 s/s, .5 s/s). The .5 s/s group is further subdivided into two sections - one for those parameter samples acquired in data acquisition cycles 1-5 and one for those acquired in data acquisition cycles 6-10. The PPB is also organized into nine blocks with the same arrangement; however, the PPB is further subdivided such that parameter samples acquired in data acquisition cycles 1-5 are located in the first half of the PPB and samples acquired in data acquisition cycles 6-10 are located in the second half of the PPB or the second half as determined by PPB pointers passed to FDA in the call list.

Parameters that have both alarm class 0, 3, or 4 and alarm class 2 limits specified have two PIT entries and respective PPB entries such that the alarm class 2 limit entry follows the alarm class 0, 3, or 4 limit entry. Thus, all FDA processing is performed on both entries, as if they were different parameters.

FDA processing is performed serially on all analog parameters, all EU parameters, then all discrete parent words. The logic and control of analog and EU processing is identical. In some instances, unique identifiers must be used since the table entries are different data types. For example, analog parameter values in the PPB are integer whereas EU values are scalar.

Discrete parent processing: Discrete FDA processing of parent words is similar to analog/EU processing except there is no FDA bypass logic with discrete parents. An FDA bypass indicator is associated with each parameter within a parent and is discussed later. All samples from one second of data are processed for each discrete parent as follows:

- I/O status flags in the PPB are checked to determine if the parent word sample is valid. If the sample is invalid, further FDA processing of that sample is skipped. Limit sense processing is thus performed only on all valid samples of that parameter.
- The FDA parameters within a parent word are isolated by using mask 1 in the parent word PIT entry and the parent word is limit sensed. If any parameter within the parent word is out of limits, or the parameter count (number of parameters within the parent that have reached max N count) does not equal the number of FDA parameters within the parent, limit sensing is performed on each discrete within the parent. The Discrete Measurement Subentry Table (DMST) is used to control limit sensing and noise filtering for the discrete parameters. The parent word's PIT entry contains a pointer to the sequential DMST entries for the parent word.
- The FDA bypass indicator in the DMST is interrogated to determine if this discrete parameter is to be FDA processed. If the parameter is to be FDA processed, discrete measurement limit sensing and noise filtering is performed. The current value of this parameter sample is compared to its expected limit in the mask 2 entry in the PIT to determine the current limit state. The expected state of mask 2 can either be set to the should-be or should-not-be state according to the state of the discrete described below. The SM offline preprocessor initializes discrete limits to the should-be state. If the current limit status differs from the last sample of this parameter, the N count is reinitialized and the limit status for this parameter is updated in the limit status field of the PIT. If this parameter changed from a maximum good state (has been within limits n consecutive times) to a fail state or from a hard-fail state to an in-limits state, the parameter count of this parent word is decremented. Once the parameter's current noise count equals the maximum noise count, the parent word parameter count is incremented. If a fail state has occurred, the user interface annunciation macro is issued provided the annunciation inhibit indicator and the hard fail indicator are off. The hard fail indicator is then turned on and the should-be state in mask 2 is changed to the should-not-be state. This is done to avoid limit sense processing on individual discretely once they reach a hard fail state.
- If the parameter has reached a maximum good state, a check is made to see if the hard fail indicator is on. If so, the should-not-be state in mask 2 is changed back to the should-be state and the hard fail indicator is reset.

When the last sample of a parent word has been processed, that sample and its associated I/O status are moved from the PPB to the CDA. The limit status is moved from the PIT to the CDA. Processing continues on all parent words (PIT entries) until all have been processed. When discrete parent processing is complete, FDA returns control to the calling function (PM control).

Fault detection and annunciation BFS: During PASS operations, the limits can be changed from the keyboards through the SM TABLE MAINT display (SPEC 60) (fig. 3.1-5). Items 2 through 7 change the alert parameter low and high limits, and items 11 and 12 change the backup C/W low and high limits. If the parameter being displayed does not have a backup C/W limit, items 11 and 12 will be blank. Discrete parameters are limit-sensed by comparing the parameter state to a predefined state. If the two are the same, then the discrete is out of limits. FDA also has the capability to count the parameter processing cycles for which a parameter is out of a specific limit set or within that limit set. If the parameter has been out of the limit set for a specified, consecutive number of times, fault annunciation for that parameter is enabled. The number of times a parameter has to be out of limits to annunciate an alarm or in limits to reenale the alarm is variable from 1 to 99. This value is controlled through items 8 and 13 of SPEC 60. If a different set of limits is selected by preconditioning, then alarm annunciation will be reset. Item 8 covers the class 3 alarms and item 13 covers the class 2 alarms.

A detailed list of the parameters that are processed by FDA during PASS operations can be found in the OFT FSSR for SM.

During the period of time when the SM function is provided by the BFS, the FDA continues the same basic type operation as PASS. The number of parameters acted upon is greatly reduced. Only three groups of parameters are multi-limit set sensed during BFS. They are fuel cell stack coolant out temperatures, APU speed, and main bus voltages. No provision has been made in BFS to support the SPEC 60 functions; however, limits can be changed via GPC read/write or uplink. The filter value for all parameters is set at 2 for all BFS parameters. A list of the parameters that are included in the BFS FDA function are found in PRD MGO3-8103. FDA is active in both the pre-engaged and postengaged mode. When SM PASS is initiated, control of the payload buses is transferred from BFS to PASS, thereby inhibiting BFS annunciation of SM parameters.

3.2 DISPLAY AND UPLINK CONTROLLED PROCESSES

This section describes the specialist function control segments and the display controlled processes. The display controlled processes are on-demand processes that are called by either an OPS or SPEC control entry or an item entry to a display. Two unique SPEC's may be active within each OPS. There is a generic capability to cyclically display analog, EU, and discrete parameter values and statuses on the displays pertaining to this section.

2011/060/

SM TABLE MAINT

2 237/14:35:02

000/01:30:00

PARAM

UPLINK

1 ID 0470215

VALUE = [+]58000.00 ↑

LIMITS

ALERT

BACKUP C&W

LOW

HI

LOW

HI

2 *[+]14000.00

3 [+]50000.00

11 *[+]12000.00

12 [+]57500.00

4 [+]13900.00

5 [+]52500.00

6 []

7 []

8 FILTER 03

13 FILTER 03

ENA 9*

ENA 14*

INH 10

INH 15

CONSTANT

16 ID 0914123

17 VALUE = [+]75541.0000

CHECKPT 236/00:05:00

FDA

INITIATE 18

ENA 19*

STATUS GOOD

INH 20

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Figure 3.1-5.- SM TABLE MAINT display.

3.2.1 Display Operations

Display Controlled Processes (DCP) covers scaling and display requirements. DCP scaling is used to scale computation parameters for processing in EU, as required both for display and calculations, and to perform scaling of digital parameters. Scaling is used both for converting the EU values entered via the keyboard and analog values entering from transducers. All analog measurements which require scaling are identified with an 'A' as the ninth character of its MML.

SMM provides control display processing for the following displays.

1 - SPEC 60	SM TABLE MAINTENANCE
2 - SPEC 61	SYS CHECK
3 - OPS 201	ANTENNA
4 - SPEC 62	PCMMU/PL COMM
5 - SPEC 67	ELECTRIC
6 - SPEC 66	ENVIRONMENT
7 - SPEC 86	APU/HYD
8 - SPEC 68	CRYO/FUEL CELLS
9 - SPEC 76	COMM/RCDR
10 - SPEC 88	APU/ENVIRON THERMAL
11 - SPEC 94	PDRS CONTROL
12 - SPEC 87	HYD THERMAL
13 - OPS 202	PL BAY DOORS
14 - SPEC 89	PRPLT THERMAL
15 - SPEC 97	PDRS STATUS
16 - SPEC 78	SM SYS SUMM 1
17 - SPEC 79	SM SYS SUMM 2

A detailed description of what is on each of these displays is found in SFOM, Volume 14. In this section, only the background processing that goes on behind the display will be discussed.

The SM Table Maintenance (TM) display is used to display the current value and the applicable limit values used by FDA of the parameter specified by item 1. These items are input on demand from the TM display and are output on demand with the exception of the parameter value which is output cyclically. One section of this display is used to control FDA. A detailed description of this portion of the display is covered in section 3.1.2.

The second section of the TM display is used to display or change the current value of constants used in the SM PASS program by special processing. However, not all constants in processing can be changed. For example, in the APU fuel quantity calculations, there are 15 constants used. Five of these cannot be changed or accessed through the TM display. The third portion of the TM display is used to initiate and status an SM checkpoint. This operation stores into mass memory and then checks to see that the transfer was executed correctly or incorrectly. The success or failure of the operation is displayed on the TM display and a time log is displayed if the operation was successful. Processing for this display includes the handling of up to 10 uplink parameters at a time.

The system check display processing consists of comparing switch configuration against a list stored in memory and outputting the deltas to the CRT. The ability to change the comparison list is included in the display processing. The program also checks analog parameter values against variable limits to see if a system is operating within those limits.

The PCMMU/PL COMM display support consists of that portion that controls loading and verification of format data into the 128 kbps and 64 kbps programmable PCMMU output downlink format rams, and that portion that supports payload communications. Measurement management for this display consists of status monitoring and display output. As in other displays, error protection is provided by Systems Measurement Management (SMM) for inputs to this display.

The antenna display is not completely supported for STS-1 because the Ku-band will not be active. All functions for this display are covered under SCM and special processes. The electric display simply takes measurements from the electrical power distribution and control and on fuel cell subsystems that have already been processed and displays them on the CRT. No special processing is required for this display.

The remainder of the displays are the same as the electrical display in that SMM only provides the capability to monitor system configuration and operations. Detailed support requirements for the Payload Deployment and Retrieval System (PDRS) control and status displays are not available at this time and will be added at a later date.

The capability exists to control the payload bay door special processing computations via the PL BAY DOORS display. SMM control of this display consists of item interface with the auto sequence special processing. The capability exists to select the payload bay door sequence to be performed via item entry or deselect the sequence via item entry. These sequences include auto sequence, center line latch sequences (4), bulkhead latch sequences (4), left door open/close sequence, and right door open/close sequence. Processes are provided to bypass the control switch.

Class 5 error message generation provides for out-of-sequence manual operations or illegal operations. The current select/deselect status of each item is displayed. When the display is initialized or if SM OPS mode 202 is terminated, the power OFF item is selected and the auto and manual mode is deselected by the SM program.

Display operations - BFS: During BFS operations, SMM requirements are handled by FDA on a reduced scale. There are only four BFS SM displays. These are THERMAL/PL, SM SYS SUMM 1, SM SYS SUMM 2, and PL BAY DOORS. The SMM requirements for these displays are the same as in PASS except for the THERMAL/PL display which is not fully defined at this time.

3.3 SPECIAL PROCESSES

SP covers the acquisition, processing, and output of data associated with special computations and sequences. An overview of special processes is shown in figure 3.3-1. A sample 2 second cycle is shown in figure 3.3-2. One complete executive cycle consists of 10 executive executions (2 seconds).

The special processes data acquisition module causes the acquisition of all data required by SP from the PCM master unit and/or the payload MDM's. Data not acquired due to I/O failure is flagged invalid.

3.3.1 APU Fuel Quantity PASS

The capability exists to compute the fuel quantity remaining for each of the three APU's via a single pressure-volume-temperature equation. The SP acquires data from the MDM at 2-second intervals and from the TM display upon request. If any of the input parameters are missing, calculations will continue and an M will be displayed with the output. Output from this SP is formatted in PCM analog form to drive the meter.

APU fuel quantity BFS: The APU fuel quantity function is the same as PASS.

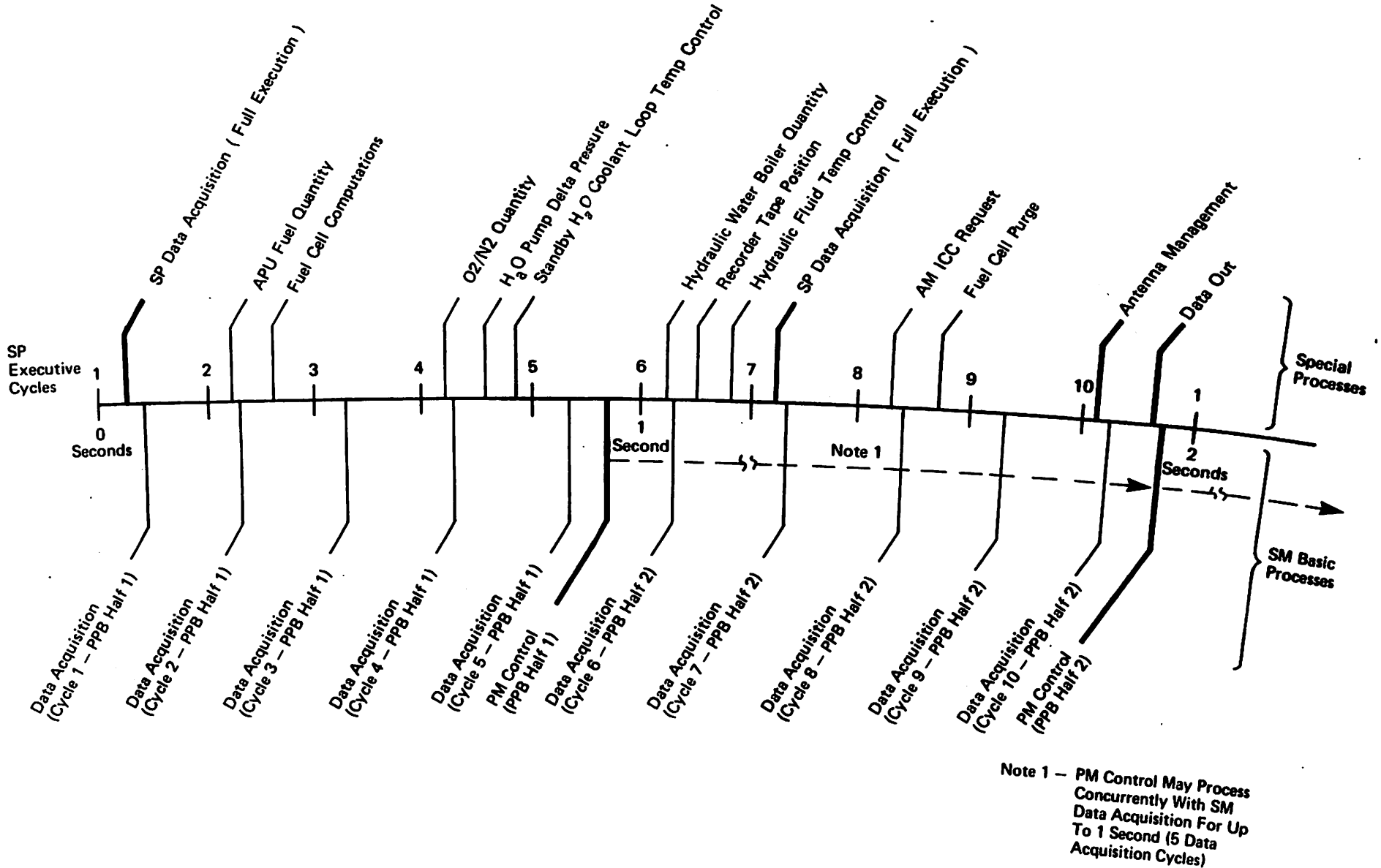


Figure 3.3-2.- Sample load balancing

3.3.2 Fuel Cell Computations

The SP computes the total current produced by the three fuel cells by summing the current output from each of the three fuel cells. SP then computes the total power produced by the three fuel cells by summing the power produced by each of the fuel cells in kilowatts. The power calculations are made for FDA. The capability exists to compare the power produced by each fuel cell to two power levels which define the limit change points for FDA. The limit change points are constants which can be changed by the SM TABLE MAINT SPEC. If any parameters are missing for these calculations, the calculation will be bypassed.

Fuel cell computations BFS: The fuel cell computations in BFS are the same as PASS.

3.3.3 Antenna Management

The antenna management sequence consists of Ku-band steerable antenna management, S-band quad and hemi antenna management, and a common function used to determine the position of the Tracking and Data Relay Satellites (TDRS's) and when the TDRS's are in view. The antenna management SP outputs about 90 parameters. The parameters acquired from the GNC application via Intercomputer Channel (ICC) are processed as groups, so that parameters from the two ICC-updates are not used in one computation.

For early OFT missions, there will not be a Ku-band antenna on-board; therefore, these computations will not be active for these flights.

Included in the antenna management calculations are the TDRS position calculations. The TDRS east and west positions are computed using an angle and delta time process with the current time tag of the NAV state vector, the inclination of the satellite orbit, the longitude of the ascending mode for a specific equatorial crossing, the magnitude of the TDRS position vector, and the GMT time tag of the ascending mode crossing. Position vectors for the satellites are downlisted. Orbiter attitude data are used to compute a line-of-sight vector from the Orbiter to the satellite, and Ku-band antenna pointing is computed on these vectors.

The Ku-band antenna control process consists of two subroutines. The subroutine to be used is determined by the position of the Ku-band ANTENNA STEERING switch. If the switch is in the GPC ACQUISITION position, the Amplitude Modulation (AM) sequence controls the acquisition sequence of the Ku-band COMM A antenna. If the switch is in any other position, a search-initiate discrete is disabled. More detailed explanations of Ku-band antenna operations will be covered in SFOM, Volume 4.

S-band antenna management is done similar to the Ku-band. The selected TDRS computation for the S-band quad antenna is controlled via one of three items - auto, west, and east on the AM mode display. The program selects

the TDRS that is in view or the next one to come into view when the auto mode is selected. In the east or west mode, the software selects the appropriate TDRS if it is in view. Other computations are provided to select quad and hemi antennas to cover communications with the Space Tracking and Data Network/Space Ground Link System (STDN/SGLS) sites.

Site locations and range coverage are stored in a table in the SM program. As in other programs, the S-band routine checks switch discrettes and SPEC item selections in its processing. Additionally, if S-band AM is enabled on the AM mode display and the S-band Phase Modulation (PM) ANTENNA SELECT switch is in the GPC position, the quad antenna commanded during the last previous computation cycle for either the TDRS or STDN/SGLS is compared with the actual quad antenna selected at the Right Forward (RF) switch during this computation cycle. If this comparison fails, a failure discrete is enabled.

Antenna management BFS: The S-band antenna management function is the only BFS-supported function. For the S-band PM subsystem, the optimum quad antenna is selected to establish and maintain communications via satellite or directly to the ground. The communications mode for the PM link is selectable via ground uplink to the Ground Control Interface Logic (GCIL) during the ascent and entry flight phase. The automatic antenna select function may be bypassed. Initially, this can be done by setting a discrete in the compool via I-load. Subsequent decisions to bypass or reset the automatic antenna select function are accomplished via uplink.

For the S-Band Frequency Modulation (FM) subsystem, the hemi-antenna is selected for a one-way transfer of wide-band data, including TV, from the Orbiter to the ground station. Since the selection of the hemi-antenna is determined from the selection of the quads, the management function for the FM link is provided only when the transmission mode is direct, not via the satellite.

As in PASS, the function computes the current position of the Orbiter, the current attitude of the Orbiter, and the current position of the target communications site. Using this data, it selects the appropriate antenna.

3.3.4 Hydraulic Water Boiler Quantity

A routine exists to compute the hydraulic water boiler quantity remaining for each of the three water tanks using a single Pressure, Volume, Temperature (PVT) equation. Output of these computations are provided to the payload MDM for display. This computation includes a system pressure initial value that is input through the TABLE MAINT display (constant).

Hydraulic water boiler quantity BFS: The hydraulic water boiler quantity computation sequence in BFS is the same as in PASS.

3.3.5 Fuel Cell Purge Sequence

The SM program includes an auto purge sequence in OPS 2. The auto purge sequence is performed for those fuel cells whose PURGE VALVE CONTROL switches have been configured to the GPC position. The routine is initiated when it receives a discrete from the GPC PURGE SEQUENCE switch. The program is designed to turn on and monitor heaters in the purge lines. It also includes timing sequences that set limits on sequences and actions like time of purge and time for temperatures to reach set values. Details of the fuel cell purge operation are covered in SFOM, Vol 2. Heater turn-off time after purging is completed can be set through the TABLE MAINT display.

Fuel cell purge sequence BFS: There is no fuel cell purge sequence in BFS.

3.3.6 Hydraulic Fluid Temperature Control Sequence PASS

The capability exists to control the temperature of the hydraulic fluid in each of the three hydraulic systems within predefined limits by controlling the on/off duty cycle of the three hydraulic system circulation pumps. The operating order of the hydraulic system's circulation pumps is based upon the order in which the individual systems satisfy the following test. The first system to satisfy this test is assigned the highest priority. If the highest priority is already assigned, succeeding systems which satisfy the test will be assigned the next highest unassigned priority number. The test consists of limit checking a predefined set of fluid line and actuator temperature parameters to see if the temperature falls below certain limits. These limits are unique to this system and are not necessarily the same as those used in FDA. However, these limits can be changed via the constant function of the TABLE MAINT display. If the predefined temperature of the fluid line is out-of-limits low, that particular system is a candidate for having its circulation pump commanded ON. Only one pump is commanded ON at a time. Time delays are also set into the system to limit the ON time of the pumps and the time between turning one pump OFF and another ON. This routine is active when OPS 2 is up and the pump switches are in the GPC position. When there is an OPS transition, the pumps are commanded OFF.

Hydraulic fluid temperature control sequence BFS: There is no hydraulic fluid temperature control sequence included in BFS.

3.3.7 Payload Bay Door Sequence PASS

The purpose of the payload door sequence is to open and close the payload bay doors and associated latches and to provide latch, door, and switch position status. The program computes and displays the position of each latch actuator, both doors, and the PL BAY DOORS switch via combinational logic. If a parameter value required for this processing computation has an M parameter status indicator enabled, the computation continues to be performed. The sequence to be performed is determined by the PL BAY DOORS display. If the Payload Bay Door (PBD) switch bypass item is deselected, the the selection sequence is controlled by the PBD hardware switch on

panel R13. Otherwise, item selection will determine which portion of the program is being executed. The system contains protection logic to prevent the operator from executing a manual door operation in improper sequence. Also, the program contains protection which keeps the operator from deselecting OPS 202 while the sequence is in the process of opening or closing the doors. A detailed opening and closing sequence is covered in SFOM, Vol 10.

Payload bay doors BFS: The payload bay door sequence in BFS is the same as in PASS.

3.3.8 O2 and N2 Quantity Computations

O2 and N2 computations done by the SM program are simple and straight forward. Using the input parameters, the software does a PVT computation and outputs these quantities to displays. Cryo O2 quantities are provided by quantity sensors in the tanks and do not require special processing. In this routine, if a parameter has an M status indicator, the computation is bypassed.

O2 and N2 quantity computations BFS: There is no O2 or N2 quantity computation sequences in BFS.

3.3.9 Remote Manipulator System

The Remote Manipulator System (RMS) function has the capability to monitor and control the RMS for payload deployment and retrieval operations. The software contains the following major functions:

- RMS executive function
- Hand controller interface function
- Coordinate transformation function
- Resolved rate processor
- Automatic control function
- Rate transformation function
- Position hold function
- Kinematic data generator
- Hardware configuration control function
- System health monitor function
- Dedicated display processor
- Single control function
- Inverse transformation function
- Temperature processing function
- Command output processor

Several of these functions require I-load parameters. The number of sets of these I-load parameters required for a mission are dependent upon the payloads, number of arms, end effectors, and preprogrammed sequences. Some of the functions will vary from flight to flight.

RMS executive function: This function is initiated when the RMS SPEC function is first initiated in SM OPS 2 and remains active until SM OPS 2 is deselected. This function is programmed to control the configuration of the RMS software. The function converts RHC and THC inputs, filters, monitors and set mode discrettes, and determines the appropriate reference system for display.

Hand controller interface function: The Hand Controller Interface Function (HCIF) conditions the signals from the manual translation and rotation hand controllers for input to the Coordinate Transformation Function (CTF). HCIF monitors the vernier/coarse select discrete and the payload-captured discrete and determines the maximum end effector rate. The end effector rates can be changed via the TM display. Payload maximum translation rate coarse and rotation rate coarse parameter ID's are 0927600 and 0927601. The vernier rate ID's are 0927605 and 0927606. These parameters are changed via the constant function and the translation rates are in ft/s and the rotation rates in deg/s.

Other functions handled by the HCIF are:

- Scales the THC inputs so that the end effector translation rates output to the CTF are proportional to the THC deflection in each axis and do not exceed the maximum limits.
- Scales the RHC inputs so that the end effectors rotation rates output to the CTF are proportional to the RHC deflection in each axis and do not exceed the maximum limits. The relationship between RHC and end effector rotation rates is non-linear.
- Monitors the rate-hold-select discrete and, while set, freezes all rate outputs to the CTF at their current levels.
- When the rate hold discrete has been reset, it returns processing of actual RHC and THC commands.
- While rate hold is in effect, waits for the hand controller null flag to be set; it then processes subsequent signals from the hand controllers as trims to the held rates.

Coordinate transformation function: The CTF accomplishes coordinate transformations as required to support RMS rate control inputs to the Resolved Rate Processor (RRP). There are four coordinate systems selectable by the operator in manual mode. All CTF outputs go to the RRP and are in the arm referenced coordinate system. In the automatic mode, all translation inputs to the CTF are in the Orbiter structural system and all rotational inputs are in the end effector-referenced system.

Resolved rate processor: This function resolves commanded end effector translation and rotation rates (in arm-referenced system) into arm joint angle rates. The algorithm used also monitors for and handles singularities and command angle rate limits. The point of control resolution for the resolved rate algorithm is on the wrist roll joint roll axis. Joint angle rates are commanded via the terms of a Jacobian matrix transformation.

Automatic control function: The Automatic Control Function (ACF) controls the processing of automatic sequences. The following two types of automatic sequences are processed. The operator-commanded sequence consists of the current end effector position and attitude plus a single point desired position entered by the operator via RMS control display. The prestored sequence consists of a set of prestored points for desired positions. Storage is provided for 200 such points. The ACF function goes through these prestored points in groups called sequences. The sequences are executed automatically with ACF computing directions and rates.

Rate transformation function: The Rate Transformation Function (RTF) derives end effector command rates from the command joint angles. All computations performed by the RTF are for the active arm. RTF computes the commanded end effector instantaneous translation and rotation rate components in the arm reference system based on commanded arm joint angle rates, end effector length, arm constants, and the present arm configuration. This function also zero's all outputs when the software is in temperature monitor configuration.

Position hold function: The Position Hold Function (PHF) processes joint angle commands to produce joint angle rate commands to hold the commanded position. It determines where to get the commands by checking the configuration flags. Also, when the software is in the auto configuration, PHF computes the difference between the commanded and actual joint angles and converts to a joint angle rate by multiplying this by a specific constant. PHF then determines that none of the rates exceed the limits.

Kinematic data generator: The Kinematic Data Generator (KDG) function converts encoder data to arm joint angles and joint angle rates; derives actual end effector rates, position, and attitude from the joint angle data; and computes the jaw singularity distance. To accomplish this for the active arm, the KDG performs the following:

- Determines which RMS arm is active and applies the corresponding set of biases to the encoder data.
- Converts the biased encoder data into actual joint angles.
- Differentiates and filters the actual joint angles to obtain actual joint angle rates.
- Computes the sine and cosine of each arm joint angle.
- Computes the transformation matrices for the reference systems.
- Computes the components and resultant of the actual end effector instantaneous translation and rotation rate in the arm reference system.
- Provides referenced components for display in the Orbiter structural, Orbiter referenced, End Effector (EE) referenced, and payload referenced systems.
- Computes for the selected arm the end effector position and attitude in the Orbiter structural referenced system.

Hardware configuration control function: The Hardware Configuration Control Function (HCCF) controls the RMS hardware configuration to make it compatible with the RMS software configuration and operational status of the RMS. It also commands zero rates when the idle or temperature monitor configurations are in effect or when the software stop discrete is set. When the software is in the idle or temperature monitor configuration or the software stop discrete is set HCCF, the HCCF sets all joint motor current limits to produce maximum joint motor torque levels. When the software is in neither the idle nor the temperature monitor configuration and the software stop discrete is not set, HCCF computes joint motor current limits.

System health monitor function: The System Health Monitor Function (SHMF) supplements the RMS hardware BITE by monitoring certain kinematic, critical RMS operational status parameters to detect anomalies in the operational health of the RMS. It does this by applying the appropriate biases to the tachometer data from the Manipulator Control Interface Unit (MCIU), setting the motor stall flag and encoder check flag, setting derigidized warning discrete, setting the release warning discrete, setting the joint angle caution flag, setting the software stop discrete, and setting the reach limit caution discrete when any of the conditions warranting these flags are determined to be met. SHMF also determines conditions that warrant setting of the caution flag, setting the software stop discrete, and setting the reach limit caution discrettes associated with CRT caution, singularity caution, motor stall caution, and RMS master alarm and then sets these discrettes.

Dedicated display processor: The Dedicated Display Processor (DDP) function accomplishes processing required to condition output parameters to drive meters and digital displays on the RMS D&C panel. DPP scales rates, displays them, and sends them to the MCIU. It also monitors discrettes from the parameter select switch. When one of the parameter select discrettes is set, DDP converts the appropriate data into the proper format to drive the three digital monitors.

Single control function: The Single Control Function (SCF) provides for control of individual arm joints (one at a time) in accordance with operator inputs via the JOINT SELECT switch and SINGLE/DIRECT DRIVE switch on the RMS dedicated D&C panel. When first enabled, SCF checks each joint's actual angles and commands the joints that are deselected to their current actual position. SCF monitors such discrettes as vernier/coarse, payload captured, and single/direct to determine the maximum rates.

Inverse transformation function: The Inverse Transformation Function (ITF) simply computes a set of joint angles corresponding to an end effector position and attitude for the selected arm. This function is used to validate operator commanded sequence data entries and to support the automatic control function.

Temperature processing function: The Temperature Processing Function (TPF) supplements the RMS hardware BITE by monitoring certain non-kinematic, critical RMS operational status parameters to detect anomalies in the operational health of the RMS. Specifically, the TPF converts thermistor

data from the MCIU to degrees Fahrenheit, identifies and annunciates out-of-limit conditions, determines the critical temperature, and supports the D&C hardware BITE function. This function resembles a miniature FDA program with the filter values set at 3.

Command output processor: The Command Output Processor (COP) function primarily converts rate position commands to the appropriate form for output to the MCIU. While in the idle or temperature monitor configuration or the software stop discrete is set, COP uses the HCCF angle rate output commands. While in the manual augmented configuration, COP uses outputs from RRP. While in the auto configuration, it uses either RRP or PHF joint rate commands. While in the single configuration, COP monitors for the joint select discretely excluding joint select EE and joint select critical temperature. For a selected joint, it converts the angle rate command from the SCF to provide an appropriately scaled rate demand output to the MCIU. For the unselected joints, it converts the rate command from the PHF to approximate rate demand outputs for the MCIU.

RMS BFS functions: There are no provisions for RMS to run in the BFS systems.

3.3.10 Standby Water Coolant Loop Control

The capability exists to control the standby water coolant loop via a special processing computation. This computation is performed in OPS 2 at a once per 2 second rate. Each time an SM OPS transition occurs or 240 minutes after the pump was last commanded OFF, the sequence will turn the pump on for 6 minutes if GPC is selected on the pump switch. The off time in the above sequence can be changed via SPEC 60 ID number 0920640 using items 16 and 17 of the display. The 6-minute cycle-on time can also be changed using ID number 0920641.

BFS standby water coolant loop control: There are no provisions in BFS for standby water coolant loop control.

3.3.11 H2O Pump Delta Pressure Computations

The SM program computes the pump delta pressures of loops 1 and 2 by subtracting the inlet pump pressure from the pump outlet pressure. This is done only in OPS 2 and at a rate of once every 2 seconds. The delta pressure is also output at the same rate. If any of the input parameters for this calculation has an M parameter status indicator enabled, the computation will be bypassed and the last calculated value and the M will be displayed.

BFS H2O pump delta pressure computation: There are no provisions for H2O pump delta pressure computations in BFS.

3.3.12 Recorder Tape Position Computations

The capability exists to compute the tape position for each of the three operational recorders via a summation. The inputs for this summation are the OPS 1, OPS 2, and payload recorders percent tape bits 1 through 5. If any of these inputs have an M parameter status enabled, the computation is bypassed and M along with the last value is passed on for display. The computation is done once every two seconds in OPS 2 only.

BFS recorder tape position computations: There are no provisions for tape recorder position computations in BFS.

APPENDIX A
ACRONYM LIST

AA	Accelerometer Assembly
ABS	Absolute
ACF	Automatic Control Function
ADI	Attitude Direction Indicator
ADS	Air Data System
ADTA	Air Data Transducer Assembly
AIF	Auto/Inhibit/Force
A/L	Approach and Landing
α	Angle-of-Attack
ALTM	Altimeter
AMI	Alpha/Mach Indicator
AOA	Abort Once Around
APU	Auxiliary Power Unit
ASA	Aerosurface Actuator
ATO	Abort to Orbit
ATT PROC	Attitude Processor
ATT PROC ORBIT	On-Orbit Attitude Processor
ATVC	Ascent Thrust Vector Control
AVVI	Altitude/Vertical Velocity Indicator
β	Sideslip
BEC	Backup Events Controller
BFS	Backup Flight System
BITE	Built-In Test Equipment
BOS	Backup Operating System
CAPRI	Capacitance Reset Integrator
CDA	CRT Display Annunciator
CDR	Commander
CMD	Command
COAS	Crew Optical Alignment Sight
COP	Command Output Processor
CSS	Control Stick Steering
CTF	Coordinate Transformation Function
CTPD	Crew Training and Procedures Division
CUR	Current
C&W	Caution and Warning
DAP	Digital Autopilot
DCP	Display Controlled Processes
DDP	Dedicated Display Processor
DG	Data Good
DIP	Display Interface Processor
DK	Display/Keyboard
DMST	Discrete Measurement Subentry Table
DPS	Data Processing System
E	Entry
EAS	Equivalent Airspeed
EE	End Effector

EI	Entry Interface
eij	relative attitude vector differences
EIU	Engine Interface Unit
EOM	Equations of Motion
ERR	Error
ET	External Tank
ET SEP	External Tank Separation
EU	Engineering Units
FASCOS	Flight Acceleration Safety Cutoff System
FCOS	Flight Control Operational Software
FCS	Flight Control System
FDA	Fault Detection and Annunciation
FDI	Fault Detection and Identification
FDIR	Failure Detection, Identification, and Reconfiguration
FM	Frequency Modulation
FMCA	Forward Motor Control Assembly
FMPT	Fault Message Parameter Table
FOD	Flight Operations Directorate
FOV	Field-of-View
GC	Guidance and Control
GCIL	Ground Control Interface Logic
G/C STEER	Guidance/Control Steering Interface
GMT	Greenwich Mean Time
GN&C	Guidance, Navigation, and Control
GPC	General Purpose Computer
GRTLS	Glide Return to Landing Site
GSENB	Groundspeed Enable
h	Pressure altitude rate
HAC	Heading Alignment Circle
HCCF	Hardware Configuration Control Function
HCIF	Hand Controller Interface Function
HIP	Hardware Interface Program
hpc	corrected pressure altitude
hpu	uncorrected pressure altitude
HSD	Horizontal Situation Display
HSI	Horizontal Situation Indicator
ICC	Intercomputer Channel
IMU	Inertial Measurement Unit
INH ALL	Inhibit All
I/O	Input/Output
ITF	Inverse Transformation Function
JSL	Jet Select Logic
k	threshold tolerance
KDG	Kinematic Data Generator
KEAS	Knots Equivalent Airspeed
KSC	Kennedy Space Center

L/D	Lift-to-Drag (ratio)
LIT	Limit Table
LOS	Line-of-Sight
LPS	Launch Processing System
LRU	Line Replaceable Unit
LTVC	Linear Terminal Velocity Constraint
LVLH	Local Vertical Local Horizontal
M	Mach number
MCC	Mission Control Center
MCIU	Manipulator Controller Interface Unit
MDM	Multiplexer/Demultiplexer
MDS	Malfunction Detection System
ME	Main Engine
MEC	Master Event Controllers
MECO	Main Engine Cutoff
MEP	Minimum Entry Point
MET	Mission Elapsed Time
MLS	Microwave Landing System (MSBLS)
MM	Major Mode
MML	Master Measurement List
MPS	Main Propulsion System
MRB/RRXF	Mach Require Bank/Roll Rate Crossfeed
MSBLS	Microwave Scan Beam Landing System
MTVC	Manual Thrust Vector Control
MVS	Middle Value Select
N	Noise
NAVAID	Navigation Aid
NAVDAD	Navigationally-Derived Air Data
NEP	Normal Entry Point
NWU	North/West/Up
OI	Operational Instrumentation
OMS	Orbital Maneuvering System
OPS	Operational Sequence
PASS	Primary Avionics Software System
PBD	Payload Bay Door
PBI	Pushbutton Indicator
Pc	Chamber Pressure
PCM	Pulse-Code Modulation
PCMMU	Pulse-Code Modulation Master Unit
PDRS	Payload Deployment and Retrieval System
PEG	Powered Explicit Guidance
PHF	Position Hold Function
PIC	Pyro Initiator Controller
PIT	Parameter Information Table
PLT	Pilot
PM	Phase Modulation
PPA	Powered Pitch Around
PPB	Parameter Processing Buffer
PPD	Powered Pitch Down
PRL	Priority Rate Limiting

PRTL	Powered Return to Launch Site
PVT	Pressure, Volume, Temperature
q	dynamic pressure
RCS	Reaction Control System
RESID	Residual
RF	Right Forward
RGA	Rate Gyro Assembly
RHC	Rotational Hand Controller
RJD	Reaction Jet Driver
RM	Redundancy Management
RMS	Remote Manipulator System
ROT	Rotation
RPTA	Rudder Pedal Transducer Assembly
RRP	Resolved Rate Processor
RTF	Rate Transformation Function
RTL	Return-to-Launch Site
S&A	Safe and Arm
SBTC	Speed Brake Thrust Controller
SCF	Single Control Function
SCM	Subsystem Configuration Management
SF	Selection Filter
SGLS	Space/Ground Link System
SHMF	System Health Monitor Function
SM	Systems Management
SMM	Systems Measurement Management
SMRD	Spin Motor Run Detector
SOP	Subsystem Operating Program
SP	Special Processes
SPI	Surface Position Indicator
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
S TRK	Star Tracker
STDN	Space Tracking and Data Network
STS	Space Transportation System
SUMLIM	Sum Limiting
T	Transition
tacan	Tactical Air Command and Navigation System
TAEM	Terminal Area Energy Management
TAS	True Air Speed
TDRS	Tracking and Data Relay Satellite
TERM	Terminate
THC	Translational Hand Controller
TIG	Time of Ignition
TM	Table Maintenance
TPF	Temperature Processing Function
TPS	Thermal Protection Subsystem
TRANS DAP	Transition Digital Autopilot
TVC	Thrust Vector Control
UPP	User Parameter Processing

V
VU

Velocity
Vehicle Utility

WOW
WRT

Weight on Wheels
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