

WORLD SPACEFLIGHT NEWS SPECIAL REPORT:

**GNC HARDWARE/SOFTWARE
OVERVIEW**

GNC HS OV 2102

Advanced Training Series

**Training Division
Flight Training Branch
Mission Operations Directorate**

May 17, 1984



**SECTION 1
GNC FUNCTIONS**

1.1 INTRODUCTION

The function of GNC hardware (H/W) is to respond to guidance, navigation, and control (GNC) software (S/W) commands to effect vehicle control and to provide sensor and controller data to GNC software, enabling it to compute those commands.

The GNC software has five functions:

- Flight control
- Guidance
- Navigation
- Hardware data processing
- Crew display

The specific tasks of each function, as well as the GNC hardware used to support them, vary with mission phase. A brief description of the first four of these functions is given in this section. The other sections of this workbook give an overview of each type of GNC hardware and the data processing associated with the hardware. The first four sections consider only the primary avionics software system (PASS). Section 5 describes the differences between the PASS and the backup flight system (BFS) for the GNC hardware data processing.

1.2 FLIGHT CONTROL

Figure 1-1 is a block diagram of the flight control interfaces discussed below.

Vehicle control is maintained and trajectory changes are made during powered flight by firing and gimbaling engines and during atmospheric flight by deflecting aerosurfaces. The part of the GNC software that computes and issues the engine fire and gimbal commands (CMDs) and the aerosurface deflection commands is called "flight control." Flight control includes attitude processing, steering, thrust vector control, and digital autopilots. Flight control receives body commands from guidance software or from the crew controllers, which are in terms of vehicle dynamics (attitudes, rates, and accelerations), and does processing to convert them to effector (engine, gimbal, or aerosurface) commands. Flight control output commands are based on errors for stability augmentation. The errors are the difference between the commanded attitude, aerosurface position, body rate, or body acceleration and the actual attitude, position, rate, or acceleration. Actual attitude is derived from inertial measurement unit (IMU) angles; aerosurface position is provided by actuator feedback transducers in the aerosurface servo amplifier (ASA); body rates are sensed by rate gyro assemblies (RGAs); and accelerations are sensed by accelerometer assemblies (AAs). In addition, during atmospheric flight, flight control adjusts control sensitivity based on air data parameters derived from local pressures sensed by air data probes and performs turn coordination using body attitude angles derived from IMU angles. The GNC hardware required to support flight control is a function of the mission phase and will be defined later.

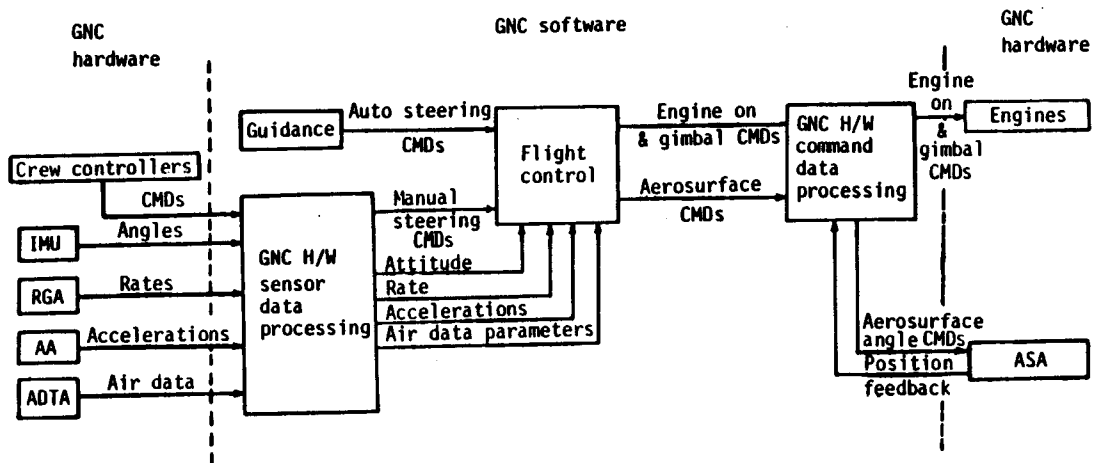


Figure 1-1.- Flight control interfaces.

1.3 GUIDANCE

Figure 1-2 is a block diagram of the software interfaces discussed below.

The steering commands used by flight control software may be computed by guidance software or may be the result of crew hand-controller deflections. When flight control software uses the steering commands computed by guidance software, that is sometimes called automatic, or auto, guidance. It follows that when the crew is controlling the vehicle, the guidance is called manual. The commands computed by guidance are those required to get from the current state (position and velocity) to a desired state (which may be specified by target conditions -- e.g., altitude, airspeed, runway centerline). The steering commands take the form of translational and rotational angles, rates, and accelerations. Guidance receives the current state from navigation software. The desired state or targets are part of the initial software load, and some may be manually changed in flight.

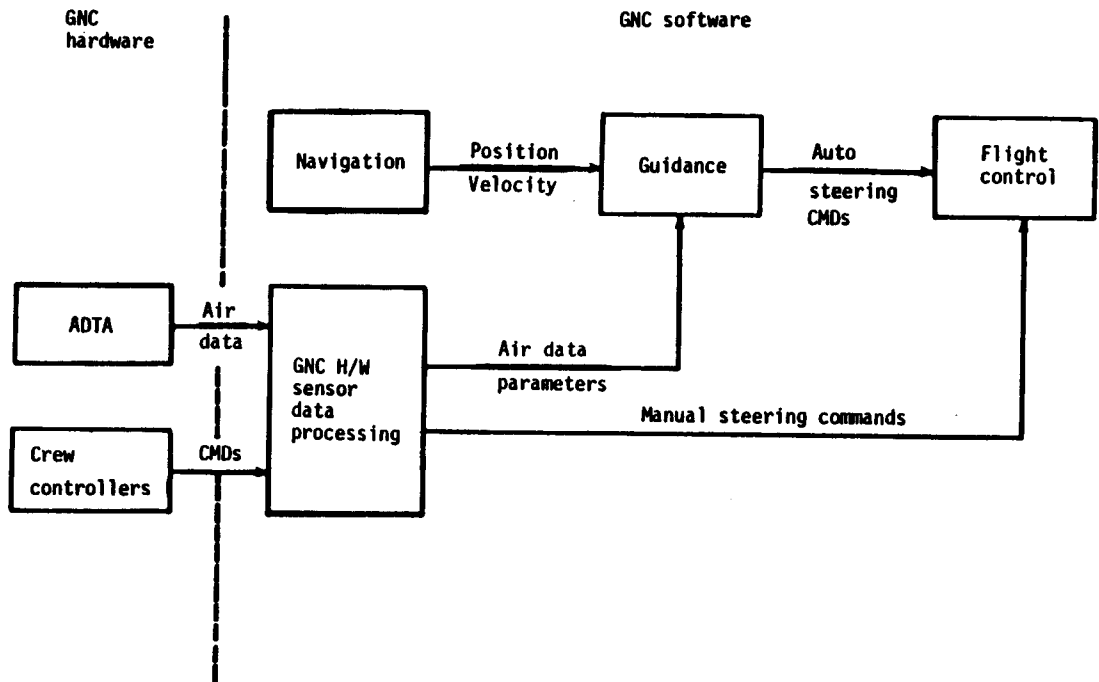


Figure 1-2.- Guidance interfaces.

1.4 NAVIGATION

Figure 1-3 is a block diagram of the navigation data interfaces discussed below.

The function of navigation is to maintain an accurate estimate of the vehicle position and velocity. This estimate is called a state vector. From position, velocity, and attitude, other parameters (e.g., acceleration, angle of attack) are calculated for use by guidance and for display to the crew. The current state vector is mathematically determined from the previous state vector by integrating the equations of motion using accelerations of the vehicle. These accelerations are sensed by the IMUs and/or computed from gravity and drag models. The alignment of the IMU, and hence the accuracy of the resulting state vector, deteriorates as a function of time. Celestial navigation instruments, star trackers and crew optical alignment sight (COAS), are used to keep the inertial sensors aligned while in orbit. The accuracy of the IMU-derived state vector is, however, insufficient for either guidance or the crew to bring the Orbiter to a pinpoint landing. Data from other navigation sensors (air data probes, TACAN units, and microwave landing systems) are blended into the state vector during the entry phase to provide the necessary accuracy.

During rendezvous and proximity operations the onboard navigation system maintains the Orbiter state vector and the state vector of the target vehicle. During close operations (less than 15 miles separation) very good accuracy of these two state vectors is required so that the relative state vector is accurate. Rendezvous radar (RR) measurements (range and range rate) are used from about 15 miles to 100 feet separation to provide the necessary relative state vector accuracy. When the separation of the two vehicles is less than 100 feet, the crew relies mostly on visual monitoring (windows and CCTV).

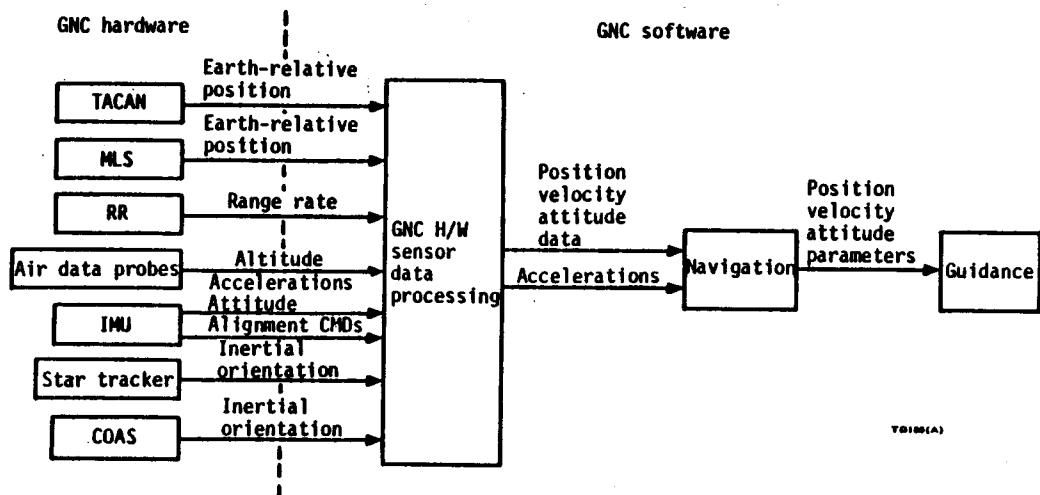


Figure 1-3.- Navigation interfaces.

1.5 GNC HARDWARE DATA PROCESSING

The GNC hardware data functions have been divided into two categories of processing:

1. Subsystem operating programs (SOPs)
2. Redundancy management (RM)

Before reading about SOPs and RM, it is important to understand the paths and processes by which the hardware data reach software.

Each piece of GNC hardware is hardwired to one of eight interface devices called flight-critical multiplexer/demultiplexers (MDMs). The flight-critical MDMs are connected to each of the five general purpose computers (GPCs) by data buses. Each GPC is assigned to command on one or more data buses, and the assignments can be changed. Simplified diagrams of the data and command flows are shown in figure 1-4.

A sensor, controller, or flight control effector cannot be assigned or rerouted to another MDM during flight, but the MDM it is wired to can be assigned to a different GPC. A piece of GNC hardware is sometimes called a line replaceable unit (LRU). Although the units can be individually replaced, replacement must be done between flights, not during flight.

A string includes a flight-critical forward (FF) MDM and a flight-critical aft (FA) MDM, each of which is connected to the five GPCs via two data buses, as depicted in figure 1-5. A string also includes three display driver units (DDUs), three engine interface units (EIUs), and two master events controllers (MECs). There are four strings currently defined.

There are multiple units of each type of GNC hardware, and each unit is hardwired to a different MDM. Take the four accelerometer assemblies as examples. AA1 is wired to MDM FF1 and is consequently part of string 1, AA2 is wired to FF2 on string 2, AA3 is wired to FF3 on string 3, and AA4 is wired to FF4 on string 4.

Diagrams specific to each type of hardware will be shown later in this workbook.

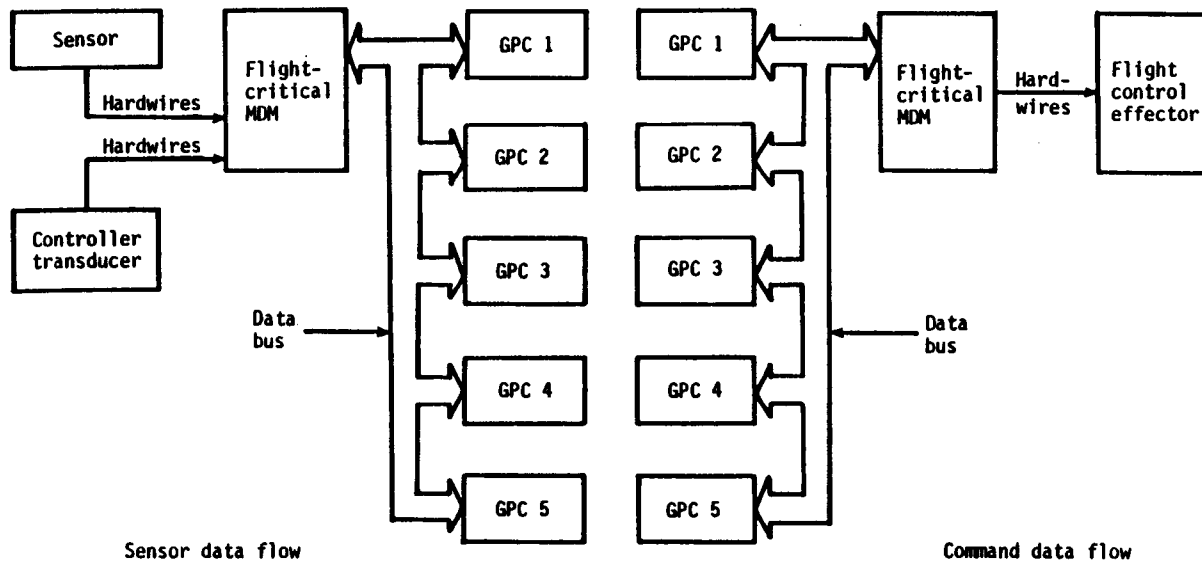


Figure 1-4.- Data and command interfaces.

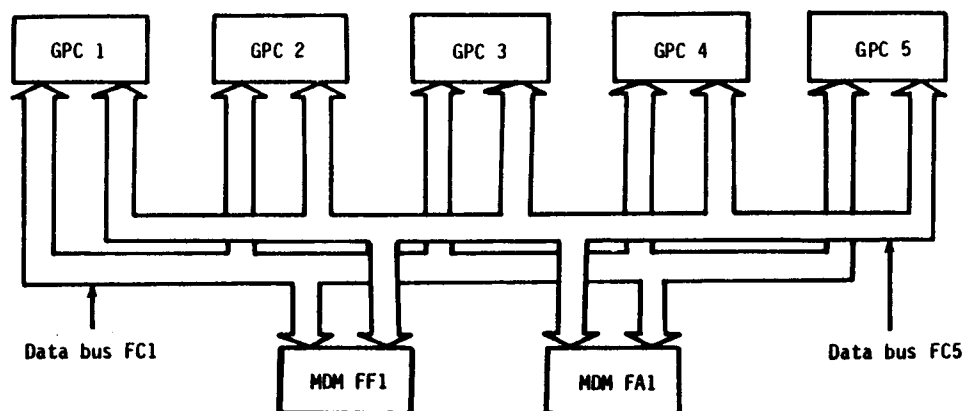


Figure 1-5.- GPC string.

GNC software acquires the data it requires by the following process. At a cyclic rate, the flight computer operating system (FCOS) in systems software in each of the redundant set GPCs activates a GNC executive program and issues commands to the flight-critical MDMs to provide a set of input data. Each MDM receives the command from the GPC assigned to it, acquires the requested data from the GNC hardware wired to it, and sends the data to all the GPCs. When the input data arrives at each of the GPCs, systems software (FCOS) checks to see that the data arrived, that no problems or errors were experienced (quantity check rather than quality), and that all the redundant set GPCs received the data. The data are stored in each GPC in locations accessible by GNC software. If, for two consecutive GPC cycles, the data are not received or contain errors, a flag called "commfault" is set for that piece of data to advise GNC software that a communications fault has occurred and the expected data is "missing". In documentation, commfault may appear as one word, two words, with one M, or with two M's.

The following are examples of failures or conditions resulting in commfault:

1. GPC transmitter failure - request for data did not get out
2. No response from MDM due to power being off or failure of the MDM
3. Data bus failure - command did not reach MDM or data did not reach GPC
4. Noise on data bus or internal MDM failure caused data to be in wrong format or incomplete
5. GNC hardware failure so that MDM detected problem in GNC H/W data
6. Digital GNC hardware being unpowered

When systems software sets the commfault flag, it also annunciates the problem with

1. SM alert light and tone
2. Fault message on active CRT displays
3. M in the status column on any CRT displays where the data or status of the GNC hardware with the commfault is displayed

The fault message depends on whether the data loss from an MDM is partial or total:

1. BCE STRING YYYYY, where X identifies the string (1, 2, 3, or 4) having the GNC hardware unit(s) that is (are) commfaulted and YYYY identifies the type of hardware
2. I/O ERROR FFX or I/O ERROR FAX, where X identifies which flight-critical MDM has had a commfault and has caused the loss of all the GNC hardware units wired to it

The GNC hardware data processing done in each GPC is not done on data from one hardware unit but on as many sets of data as there are redundant units. The case of the two radar altimeters (RAs) which sense altitude is shown in figure 1-6.

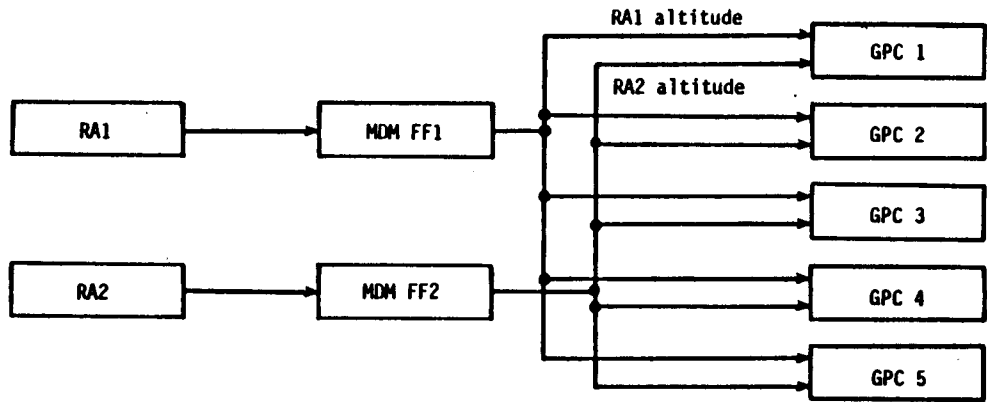


Figure 1-6.- RA data flow.

Each GPC has received radar altitude from both RAs.

It was stated earlier that the hardware use varied with mission phase. This is not a hardware constraint but is due to electrical power, computer memory, and operational limitations. The microwave landing system (MLS), as an example, could be turned on during the ascent and orbit phases, but it would consume electrical energy, the software required to process the data would take up computer memory, and it would serve no useful purpose. The MLS could not provide valid data during those phases nor would the data be needed then.

1.5.1 Subsystem Operating Program. When the sets of GNC hardware data arrive at the GPCs via MDMs and data buses, the data are generally not in the proper format, units, or form for use by flight control, guidance, or navigation. A subsystem operating program (SOP) for each type of hardware processes the data to make them usable by GNC software. An SOP also contains the software necessary for hardware operation, e.g., activation, self-test, and moding. Because each SOP is different, it is difficult to describe them in general terms. Each will be discussed with its associated hardware type. Figure 1-7 shows the general flow for the microwave landing system (MLS). Examples of the type of processing done are

1. Applying a deadband to hand controller commands
2. Converting local air pressures to angle of attack, equivalent airspeed, etc
3. Converting RGA voltages which represent body rates into engineering units of degrees per second

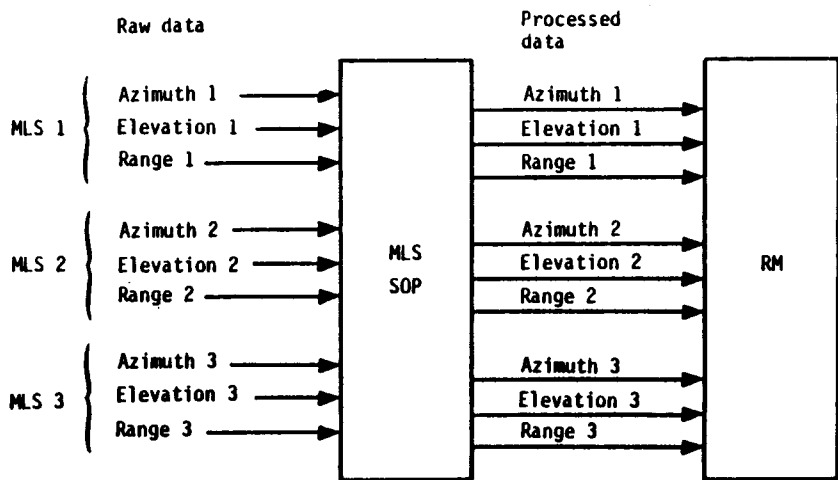


Figure 1-7.- MLS data flow.

1.5.2 Redundancy Management. The Orbiter design philosophy is that in the event of a flight-critical component failure, the vehicle must be operationally capable of completing the mission and, in the event of a second failure of the same type of component, the vehicle must be able to return the flight crew safely. The philosophy is called fail ops/fail safe (FO/FS) and includes two implications:

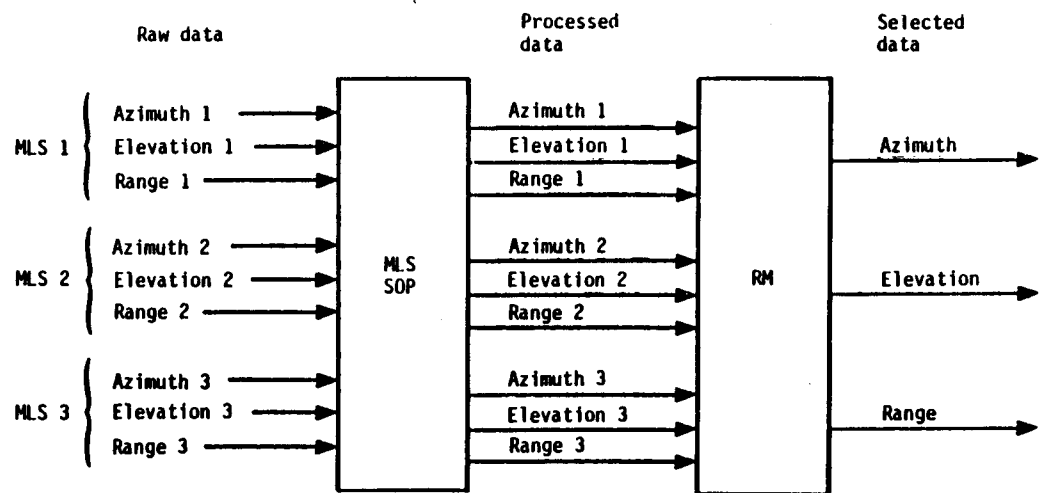
1. The first failure can be detected and isolated to a unit
2. After the first failure there is a complete set of GNC H/W data available to GNC so that there is not a loss of GNC capability

As a consequence, there are redundant units of the GNC hardware which are considered flight-critical. The level of redundancy varies from two to four depending on the particular unit. The FO/FS philosophy led to the design stated before: each of the redundant GNC units is hardwired to a different MDM, there are two data paths for the GNC H/W data to take from each MDM to the GPCs, and each GPC performing GNC functions receives data from all the redundant GNC hardware and executes the same processing on it.

The software that processes data from the redundant GNC hardware is called redundancy management (RM). RM performs two functions, which have been named

1. Selection filter. Selects, from redundant sets of hardware data, one set of data for use by guidance, navigation, and flight control.
2. Fault detection, identification, and reconfiguration (FDIR). Detects data that are out of tolerance, identifies the faulty unit, annunciates the failure to the crew, and may make the data from the faulty unit unavailable to the selection filter.

The selection filter and FDIR processes are common to many types of GNC hardware and will be explained in section 3. Figure 1-8 shows the flow of data from the hardware through RM for the MLS.



TD100(A)

Figure 1-8.- MLS data flow.

SECTION 2
GNC HARDWARE/SOFTWARE

2.1 INTRODUCTION

This section gives a brief description of all GNC hardware plus overview descriptions and diagrams of the power and software interfaces. The redundancy management software function will be discussed in section 3.

The sensors used by navigation are

- Inertial measurement units
- Star trackers
- Crew optical alignment sight
- Tactical air navigation
- Air data system
- Microwave landing system
- Radar altimeters
- Rendezvous radar

The second set of hardware to be discussed is that hardware used by the flight control system:

- Accelerometer assemblies
- Orbiter rate gyro assemblies
- Solid rocket booster rate gyro assemblies
- Controllers
- Aerosurface servo amplifiers/actuators

At the end of the discussion of each type of hardware, a power and data interface diagram and a diagram showing the flow of data through the software are presented. It is suggested that the reader not dwell on the details of these diagrams since these details are discussed further in the later workbooks.

2.2 NAVIGATION SENSORS

2.2.1 Inertial Measurement Unit. There are three inertial measurement units (IMUs) on the Orbiter. They are mounted on a navigation (nav) base inside the pressure hull forward of the forward flight deck (see figure 2-1). Each IMU contains two accelerometers and two gyros mounted on an inertially stabilized four-gimbal platform called the cluster.

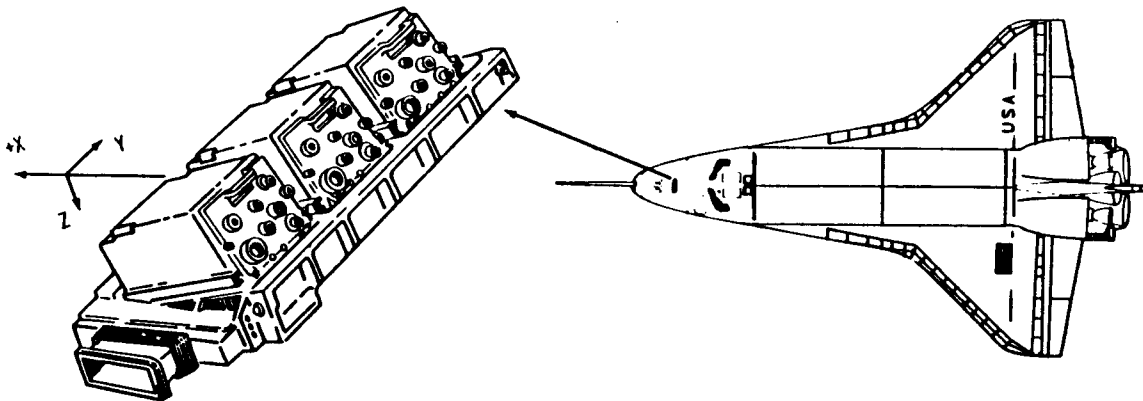


Figure 2-1.- IMU Location.

The accelerometers are pendulous masses anchored to a case and mounted on the cluster platform. Acceleration produces relative motion between the mass and the case. The motion is sensed and voltage is applied to counteract the motion. The voltage required is proportional to the acceleration. The voltage is electronically converted to pulses which are accumulated. The accumulated acceleration pulses represent velocity and are sent to the data processing system. One accelerometer senses acceleration along two axes, X and Y, and the other senses acceleration along a Z axis. X, Y, and Z are the inertial coordinate axes of the IMU cluster and not the vehicle axes. The velocities can be transformed to vehicle axes or any other coordinate system desired.

These accelerometer-accumulated velocities are used by navigation to determine the three position and three velocity components of the state vector. This information is then available for other users who need to know the position or velocity of the Shuttle.

The gyros are two degree-of-freedom gyros mounted on the cluster platform. As for any gyro, angular motion about any axis perpendicular to the spin axis causes rotation about a third orthogonal axis which is electrically sensed. In response to this sensed rotation the appropriate gimbals are torqued to null the gyro rotation. This gimbal torquing in response to vehicle motion results in an inertially-stabilized platform. In addition, the gyros may be pulse torqued (small angles) or slewed (large angles) by software, which results in gimbal torquing to position or reposition the platform. Each gimbal has a coarse (one-speed or 1X) resolver and a fine (eight-speed or 8X) resolver on it. The resolver readouts are sent to the data processing system, which determines the gimbal angles.

These gimbal angles determine the attitude of the Orbiter, which is used by

1. Flight control for turn coordination
2. Guidance for computing steering commands
3. Attitude director indicator (ADI) to display attitude
4. Navigation to determine drag and height of center of gravity (CG) above wheels

The attitude accuracy of the IMUs deteriorates with time. If the errors are known, they can be physically or mathematically corrected. Software based on preflight calibrations is used to compensate for most of the inaccuracy. The star trackers and/or COAS are used to determine additional inaccuracy.

The three IMUs have skewed orientations. That is, their axes are not co-aligned and are not aligned with the vehicle axes. This is done for two reasons. Gimballed platforms have problems at certain orientations. This skewing ensures that not more than one IMU will have an orientation problem for a given attitude. Also, this relative skewing is used by redundancy management to determine which IMUs have failures (see section 3).

Each IMU contains heaters, temperature sensors, a pressure sensor, and built-in test equipment (BITE). If the temperatures are out of limits or if BITE detects a failure, a fault message and SM alert are annunciated. The message is IMU BITE/T X, where X identifies the faulty IMU.

The IMU has two modes:

- Warmup/standby. Heaters are turned on to warm up the IMU.
- Operate. When temperatures are within limits the IMU may be commanded to the operate mode. In this mode the IMU is fully powered and the digital interface with the data processing system is activated. The operate mode has submodes which are under software control.

The IMU subsystem operating program (SOP) processes the data from the IMUs and converts it to a usable form for other users. The following computations are performed in the IMU SOP:

- Converts velocities to Mean of 1950 coordinates
- Converts resolver outputs to gimbal angles
- Computes accelerations for display
- Performs additional software BITE checks
- Supports selection, control, and monitoring of IMU submodes. The submodes are submodes of the operate mode. Preflight submodes include calibration, attitude determination, alignment, gyrocompassing, and inertial processing. Flight submodes are:
 1. Inertial. Gimbals are torqued to null gyro rotation due to vehicle motion and uncompensated gyro errors. IMUs are normally in this mode.
 2. Alignment. Gimbals are torqued or slewed to realign platform in response to SOP commands or the software matrix defining the IMU orientation is recomputed to reflect the actual IMU alignment.
- Computes torquing commands based on the misalignment determined by the star trackers, crew optical alignment sight (COAS), or another IMU. Misalignments are due to gyro drift.

Figure 2-2 shows the power and data interfaces of the IMUs. Notice that each IMU is redundantly powered so that a single electrical failure will not cause the loss of an IMU.

Figure 2-3 shows the flow of IMU data through the flight software. Note that IMU RM and the star tracker/COAS SOP are discussed later in this workbook.

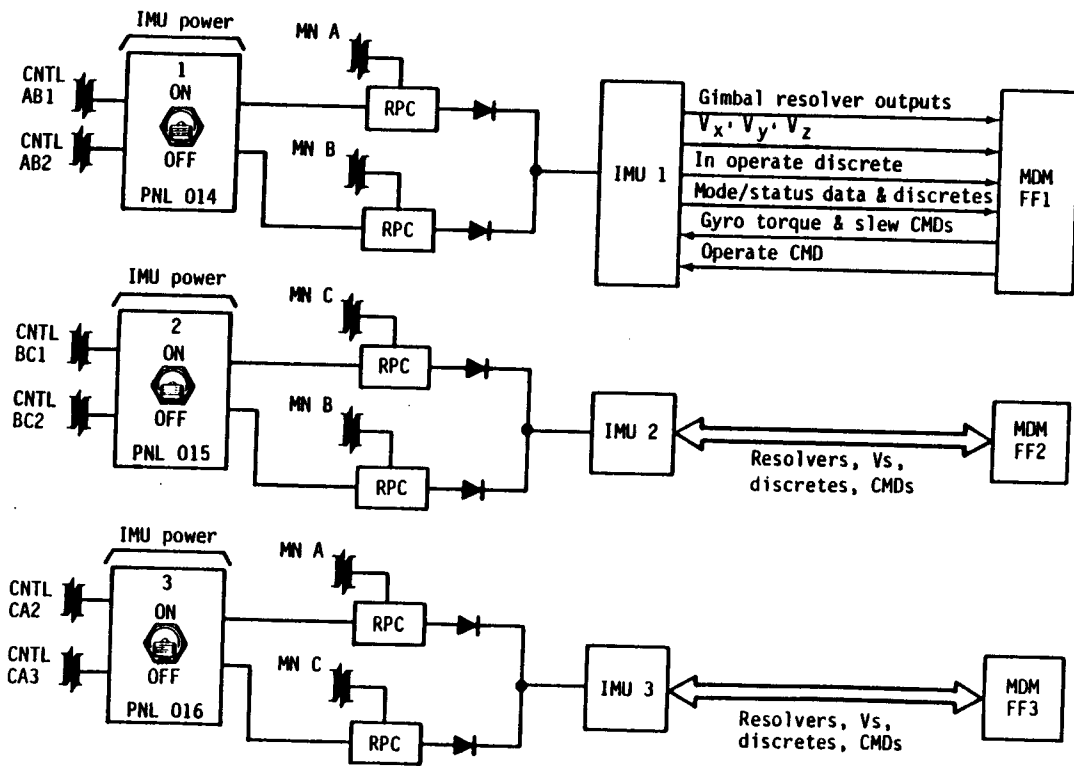


Figure 2-2.- IMU power and data interfaces.

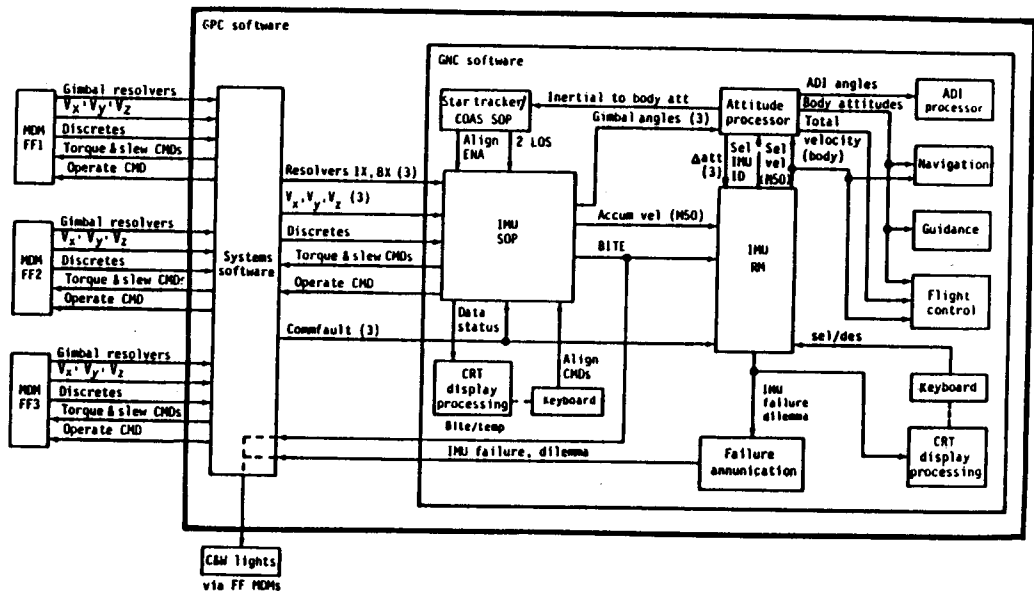


Figure 2-3.- IMU data flow through software.

2.2.2 Star Trackers. There are two star trackers on the Orbiter: -Y and -Z. They are located on an extension of the nav base on which the IMUs are mounted (figure 2-4). The trackers are slightly inclined off the vehicle axes for which they are named.

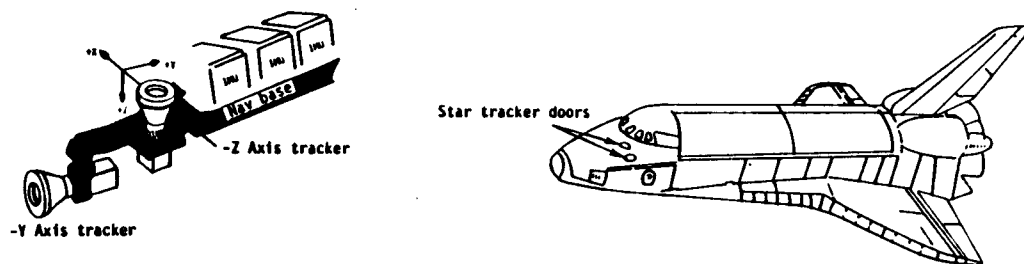


Figure 2-4.- Star tracker location.

Each star tracker is covered by a door. The doors are closed for ascent and entry and opened during orbit. Each door is driven by two motors. Limit switches stop the motors and drive a talkback when the door has reached limits of full travel, open or closed.

The star tracker includes a light shade assembly and an electronics assembly. The field of view of the star tracker is a 10° square. It contains a shutter mechanism which may be opened or closed manually by the crew or automatically by a bright object sensor or target suppress software. The bright object sensor reacts before a bright object such as the sun or moon could cause damage to the star tracker (The sensor has a larger field of view than the star tracker shutter). The target suppress software reacts to broad light sources (such as the sunlit earth) which may not trip the

bright object sensor but which produce overall illumination large enough to cause photo currents larger than desired.

The electronics assembly contains an image dissector tube. The star tracker itself does not move. The field of view is scanned electronically. It may be commanded to scan the entire field of view or a smaller offset field of view (1° square) about a point defined by horizontal and vertical offsets. An object is tracked when the proper intensity and the correct location is sensed. Star tracker outputs are the horizontal and vertical position within the field of view of the object being tracked and its intensity.

The star trackers determine Orbiter attitude in inertial space and are used to correctly align the IMU inertial platforms. A star tracker searches for, acquires, and tracks stars whose approximate position is determined by the IMUs. By knowing the relationship of the star tracker to the Orbiter and the location of the star in space, a line-of-sight vector from the Orbiter to the star is defined. Two such vectors define the Orbiter inertial attitude.

The difference between the inertial attitudes defined by the star tracker and the IMU is processed by software and results in IMU torquing angles. If the IMU gimbals are physically torqued or the matrix defining its orientation is recomputed, the effects of IMU gyro drift are removed and the IMU is restored to its inertial attitude. If the IMU alignment is in error by more than 1.4° , the star tracker is unable to acquire and track stars. In this case, the COAS must be used to realign the IMUs to within 1.4° , and then the star trackers can be used to realign the IMUs more precisely. The reason the star tracker cannot be used if the IMU alignment error is greater than 1.4° is that the angles the star tracker is given for searching are based on the current knowledge of the Orbiter attitude, which is based on IMU gimbal angles. If that attitude is greatly in error, the star tracker may acquire and track the wrong star.

In addition to aligning the IMUs, the star trackers can be used to provide angular data from the Orbiter to a target. This capability can be used during rendezvous or proximity operations with a target satellite.

The star tracker SOP performs the following software functions:

1. Supports the modes commanded manually

- Self-test. Software and hardware test.
- Star track. Star tracker will do an offset scan search for the star and acquire and track it. The star may be selected by the crew or by the computer. In either case, field of view and occultation checks are made. As of this writing, there are 50 stars in the star catalog and there are plans to have a total of 100 stars in the catalog.
- Target track. Same as star track but the crew must specify the target and its threshold.
- Break track. Forces the star tracker to stop tracking a star and to do a search scan from the current location to track the next star it acquires.
- Term/idle. The star tracker continues its operation, but all star tracker software processing ceases.

2. Maintains the star table

- When a star tracker has acquired and tracked a star and the data has passed software checks, the star ID, time tag, and line-of-sight (LOS) vector are stored. The ID and time elapsed since time tag are displayed in the star table on SM SPEC 22. When two or three stars are in the table, the angular difference between their LOS vectors is displayed. The difference between the star tracker angular differences and the star catalog angular differences is displayed as an error. The star tracker SOP selects LOS vectors of two of the stars in the star table for IMU alignment and outputs the ALIGN ENA discrete. The software selects the star pair which has an angular difference closest to 90° or the pair which has an elapsed time of entry into the table of less than 60 minutes. The crew may manually override the SOP selection or clear the table if desired.

3. Determines and displays star tracker status

The power and data interfaces of the star tracker are shown in figure 2-5, and the flow of star tracker data through the software is shown in figure 2-6.

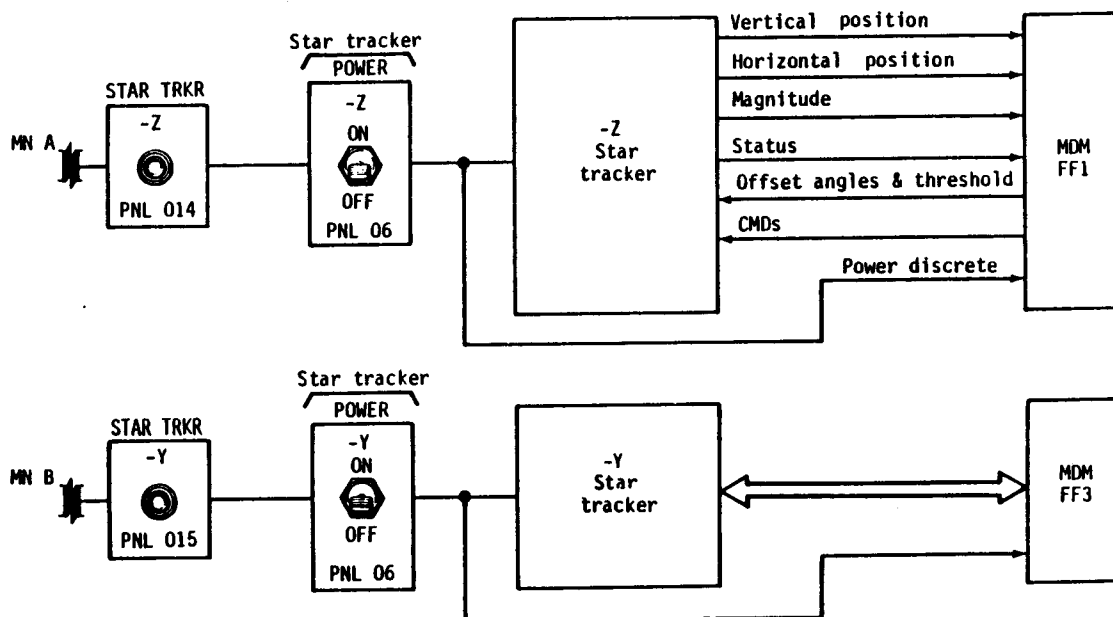


Figure 2-5.- Power and data interfaces of the star trackers.

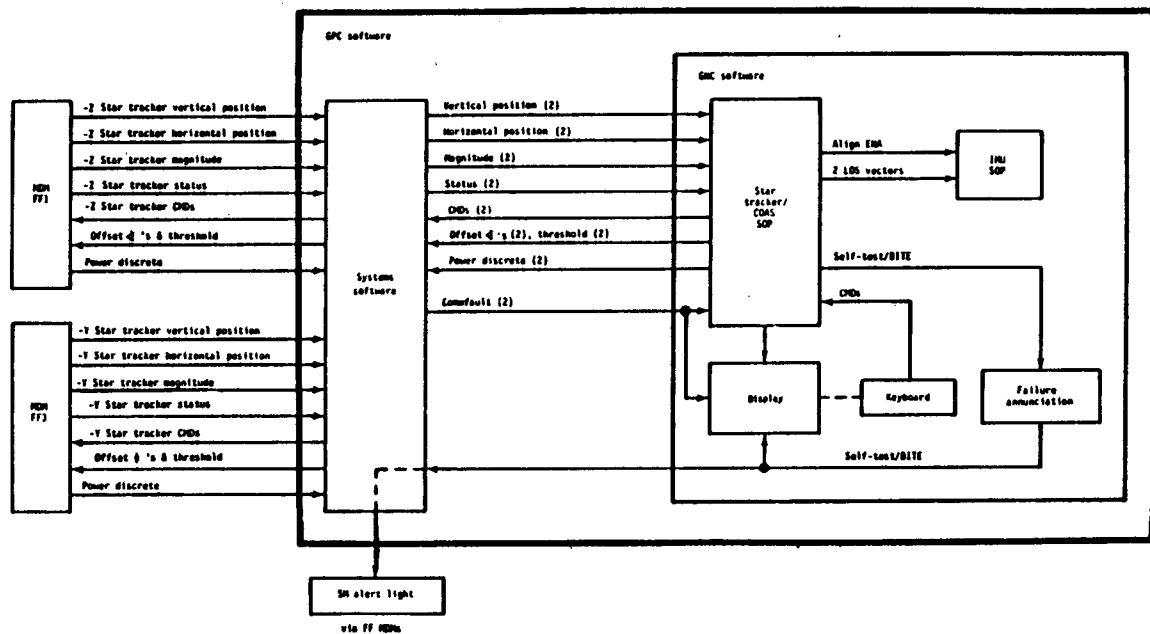
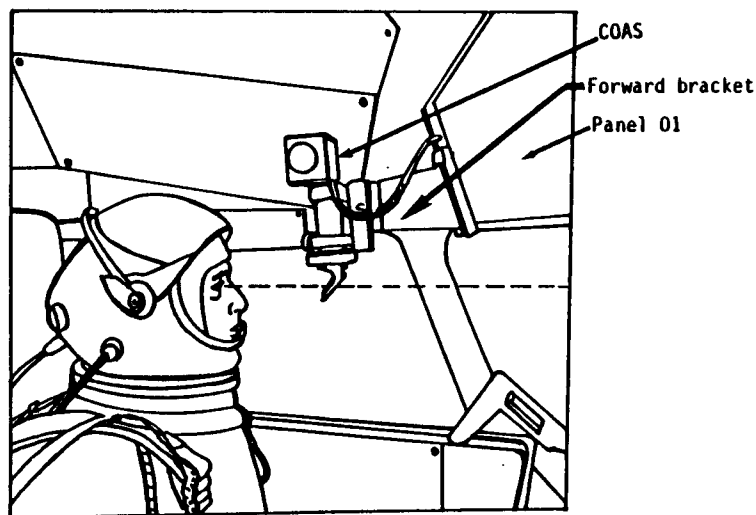


Figure 2-6.- Flow of star tracker data through applicable software.

2.2.3 Crew Optical Alignment Sight. The crew optical alignment sight (COAS) is an optical device with a reticle projected on a combining glass focused on infinity. A light bulb with variable brightness illuminates the reticle. The COAS may be mounted in two locations, defined by the vehicle axis along which it sights (figure 2-7):

- o +X above the commander's window
- o -Z next to the starboard (right) overhead window



+X Location



-Z Location

Figure 2-7.- COAS locations

The COAS is used for

1. Aligning the IMUs when their misalignment is greater than the star tracker can handle or when the star trackers have failed
2. Visual tracking of targets during proximity operations or visually verifying tracking of correct star by -Z star tracker

Recall that the star tracker cannot be used for tracking stars when the IMU misalignment is greater than 1.4° . In that case, the COAS would be used to align the IMUs to within that accuracy. Then the star trackers could be used to further refine the accuracy.

After mounting the COAS in the desired location, the crewmember must manually maneuver the Orbiter until the selected star is in the field of view. The crewmember then maneuvers so that the star will cross the center of the reticle. At the instant of crossing, the crewmember makes a mark by depressing the most convenient ATT REF (attitude reference) pushbutton. There are three of these on panels F6, F8, and A6. At the time of the mark, software stores the gimbal angles of the three IMUs. The mark can be taken again if it is felt the star was not centered as well as possible. When a good mark has been taken, the crewmember notifies software to accept it. Good marks for two stars are required for an IMU alignment. The separation between the two stars should be between 60° and 120° .

By knowing the star being sighted and the COAS location and mounting relationship in the Orbiter, software can determine a line-of-sight vector from the COAS to the star in an inertial coordinate system. Line-of-sight vectors to two stars define the attitude of the Orbiter in inertial space. This attitude can be compared to the attitude defined by the IMUs and, if the IMUs are in error, they can be realigned to the more correct orientation described by the COAS sightings.

The COAS mounting relative to the nav base on which the IMUs are mounted is calibrated before launch. The constants are stored in software, and COAS line-of-sight vectors are based on known relationships between the COAS line of sight and the nav base. Zero gravity and pressure differential effects have been found to have negligible effects on this prelaunch calibration.

The COAS data processing is done in the star tracker/COAS SOP. This SOP performs the following COAS functions:

- Accepts and stores crew inputs on COAS location (+X or -Z), star ID, sighting or calibration mode
- Accepts marks, computes and stores the line of sight vectors, and enables the IMU alignment when two marks have been accepted
- Computes, updates, and provides data for display

The power and data interfaces of the COAS and ATT REF pushbuttons are shown in figure 2-8. The power supplied through the mounting connector and power cable is used only by the light bulb and dimmer. An intensity control on the COAS itself controls brightness of the light bulb.

The flow of COAS data through the software is shown in figure 2-9. The COAS and star tracker processing is done in one SOP.

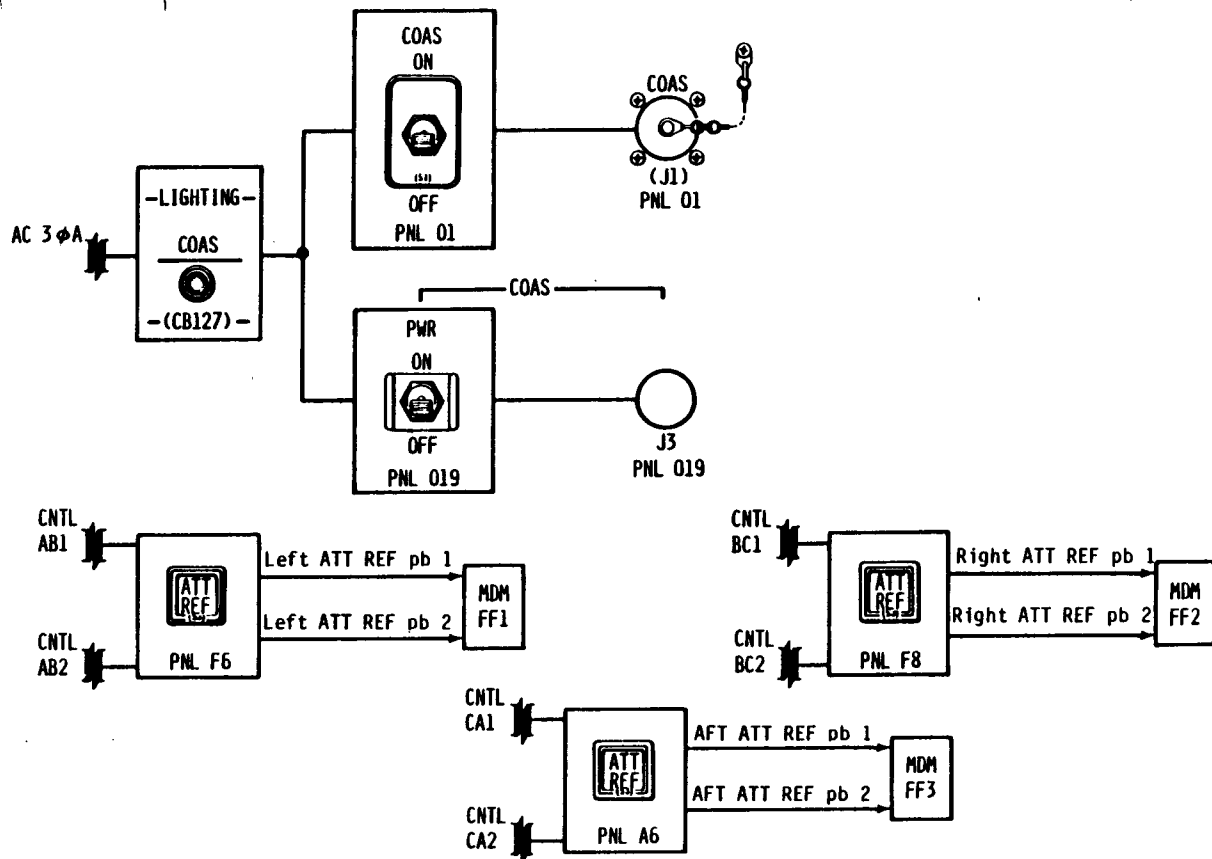


Figure 2-8.- Power and data interfaces of COAS and ATT REF pushbuttons.

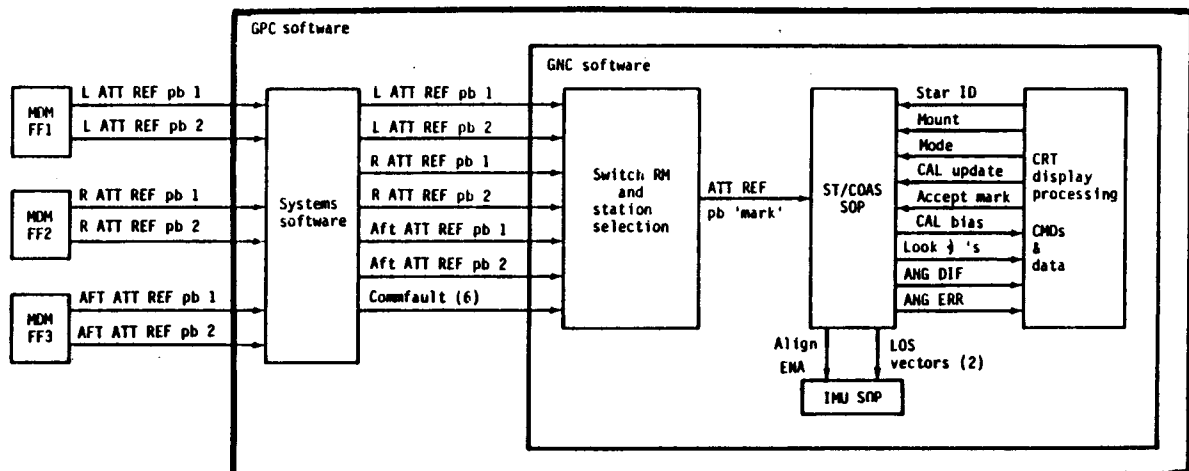


Figure 2-9.- Flow of COAS data through software.

2.2.4 Tactical Air Navigation. The onboard tactical air navigation (TACAN) units determine slant range and magnetic bearing of the Orbiter to a TACAN or VHF omnirange tactical air navigation (VORTAC) ground station (see figure 2-10).

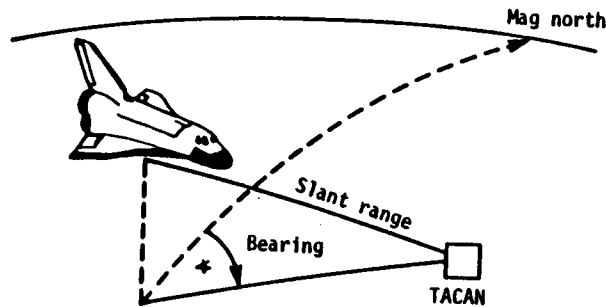


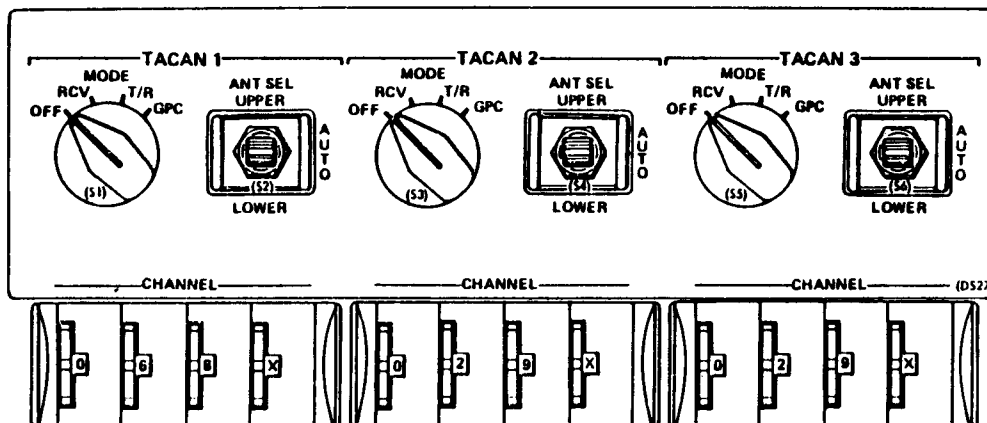
Figure 2-10.- Range and bearing.

The ground-based TACAN and VORTAC stations constitute a global navigation system for military and civil aircraft. Location and station data pertinent to a subset of all the stations are stored in the onboard software. The subset of selected stations represent those that will normally be used for landing at the planned site and at backup or contingency sites. Each ground station has an assigned frequency (L-band) and a three-letter ID. Channel selection is made with three thumbwheels on panel 07. The channel number represents a frequency. Selection of the proper station may be verified by listening to the ID transmitted by the station in Morse code using the TACAN ID audio controls on panels 05 (CDR) and 09 (PLT). The fourth thumbwheel selects X or Y mode. The ground beacon is omni-directional. When the Orbiter is over the ground station, or nearly so, it is in a cone of confusion. Within the cone, bearing is unusable.

The TACAN unit includes a transmitter, a receiver, and a data processor. The data processor decodes the selected channel and sends the frequency to the receiver.

- **Range.** The data processor triggers the transmitter to initiate interrogation pulses. The ground beacon returns a signal in response to the interrogation pulse and the range is determined by the data processor as a function of the elapsed time between signal transmission and return.
- **Bearing.** The ground beacon continuously transmits pulse pairs on its assigned frequency, the onboard receiver receives them, and the data processor decodes the pulse pair to compute bearing.
- **ID.** Approximately every 37 seconds the beacon transmits the station ID in Morse code.

There are three TACAN units on the Orbiter. They are located in the forward avionics bays. Each TACAN has two antennas: one upper and one lower. Each TACAN also has a set of controls, called the control head, which selects TACAN mode, antenna, and ground station (channel); see figure 2-11.



Panel 07

Figure 2-11.- TACAN controls.

The TACAN may operate in three modes, selected by the MODE rotary switch on panel 07:

- RCV. Only bearing is received and processed by the hardware unit. The TACAN SOP does not process this mode.
- T/R. Both bearing and range are processed by hardware and software. Channel selection is manual using the thumbwheels on panel 07 but must be from among a list of stations stored in software.
- GPC. Both bearing and range are processed by hardware and software. Channel selections are made by GNC software. Self test may be commanded in this mode only.

Antenna selection may be made

1. By the crew by placing the antenna select switches on panel 07 in the upper or lower positions
2. By TACAN SOP when the antenna select switch is in AUTO and the mode rotary switch is in the GPC or T/R position

TACAN range and bearing are used in the entry phase by

1. Navigation to update the state vector position components and increase their accuracy
2. Area navigation for display on the horizontal situation indicator, after being transformed to other parameters in the runway coordinate system

TACAN range and bearing are used from approximately 160 000 feet altitude until MLS (microwave landing system) acquisition at approximately 18 000 feet. If the MLS is not acquired, the TACAN data will be used until 1500 feet altitude. The navigation software has an initial two-unit lock-on requirement in bearing and range, unless two of the three parameters are deselected or commfaulted. This range lock-on requirement is dropped when the navigation state converges within certain specifications.

TACAN range and bearing cannot be used to produce a good estimate of the altitude position component, so navigation uses barometric altitude derived from air data in conjunction with TACAN range and bearing to update the onboard state vector.

Processing of the TACAN data is done in the TACAN SOP, which converts range and bearing to units of feet and radians, respectively. It responds to a request from RM for self test and determines whether the TACANs passed or failed self test. It controls automatic channel and antenna

selection. Automatic channel selection is made from a list of TACANs available near the selected landing site. The selection is based on lock-on, range to the landing site, and range to the TACAN station. Automatic antenna selection is made by selecting the lower antenna when a channel change is made. If the lower antenna fails to lock on to the range within 10 seconds, the upper antenna is tried for 5 seconds, etc. If all three TACANs lose lock on both antennas, a different channel is selected.

Figure 2-12 shows the TACAN power and data interfaces and figure 2-13 shows the flow of TACAN data through the software.

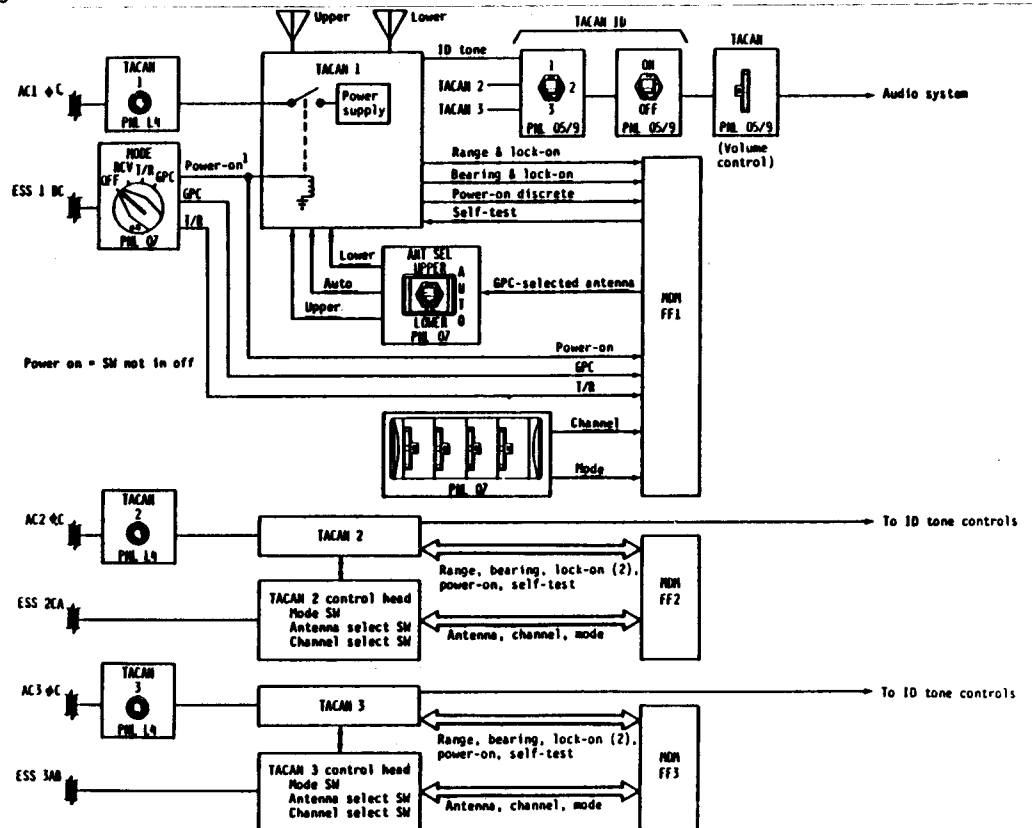


Figure 2-12.- TACAN power and data interfaces.

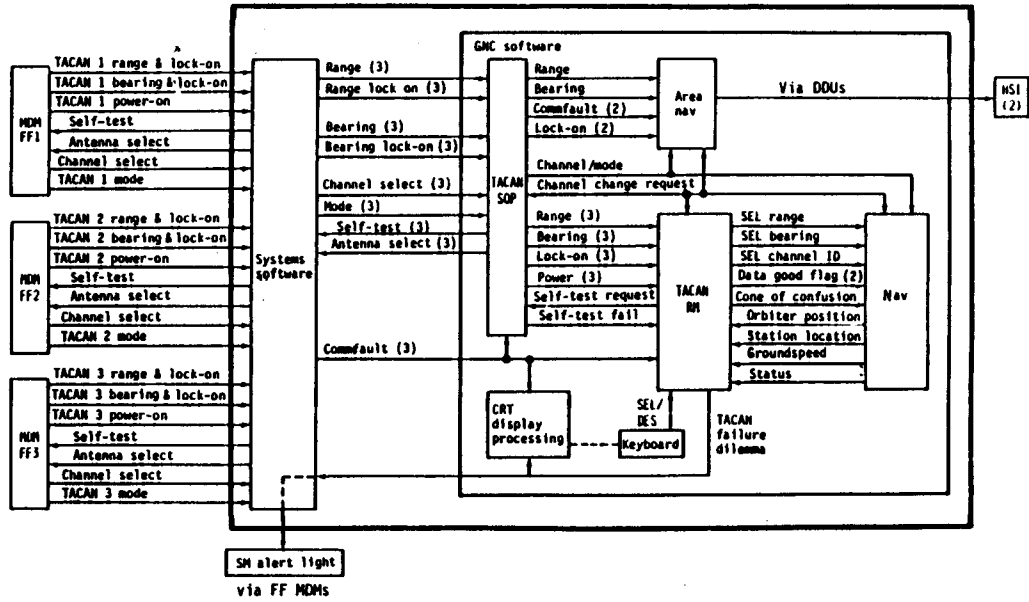
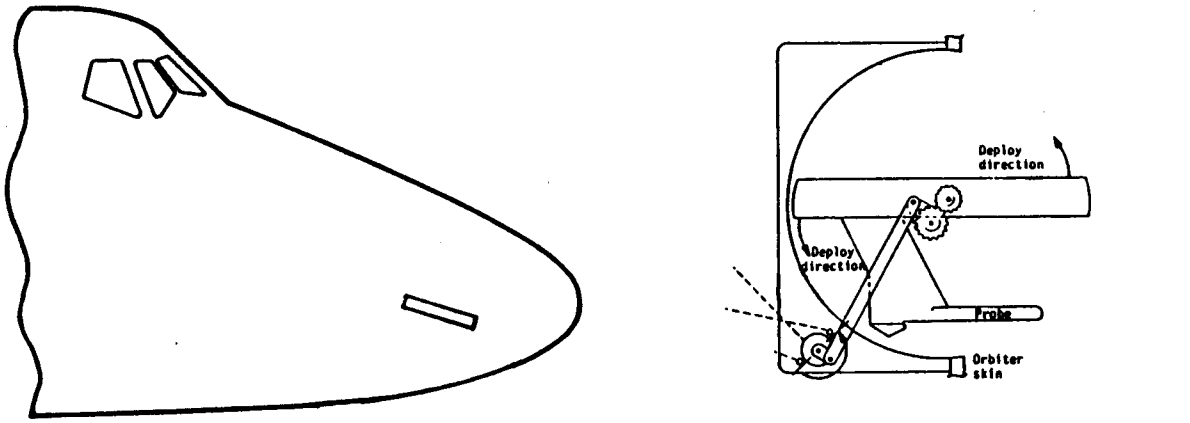


Figure 2-13.- Flow of TACAN data through software.

2.2.5 Air Data System. There are two air data probes located on the left and right sides of the Orbiter nose. During the ascent, orbit, and deorbit phases, the probes are stowed underneath the Orbiter skin. At Mach numbers depending on mission requirements (usually around Mach 4 or 5), the probes are deployed by rotating them outward so that they face forward. Each probe is deployed by an actuator driven by two motors (see figure 2-14).



Right air data probe location

Right air data probe top view, half deployed

Figure 2-14.- Air data probe location and deployment.

Each probe has four pressure ports (holes) and two temperature sensors (figure 2-15). The pressures sensed are

1. P_S - static pressure (ambient)
2. $P_{\alpha U}$ - upper alpha pressure
3. $P_{\alpha C}$ - center alpha pressure (pitot or impact)
4. $P_{\alpha L}$ - lower alpha pressure

The ports are connected via a set of pneumatic lines to two air data transducer assemblies (ADTAs). The temperature head contains two sensors which are each wired to an ADTA. De-icing, if needed, is provided by heaters on the probes.

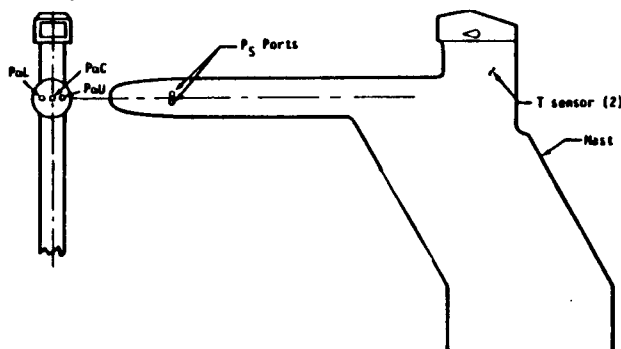


Figure 2-15.- Air data probe ports and temperature sensors.

An ADTA contains four pressure transducers and a temperature bridge. An algorithm using sensed pressure and temperature and calibration constants is applied to the four pressures. The pressures and temperature are converted to digital format for output to the data processing system. Software does not receive or process temperature data so no further mention will be made of it. Each ADTA contains built-in test equipment, the results of which are continuously output in the mode/status word. Additional tests are performed in response to a self test command from software.

The ADTA SOP uses the data from the ADTA's to compute the following seven parameters:

1. Angle of attack (α)
2. Mach number (M)
3. Equivalent airspeed (EAS)
4. True airspeed (TAS)
5. Dynamic pressure (\bar{q})
6. Barometric altitude (h)
7. Altitude rate (\dot{h})

The air data system is used during entry, approach, and landing phases of atmospheric flight. The computed air data parameters are used

1. For display to the crew on the AMI, AVVI, and CRT displays
2. For updating the navigation state vector in the case of altitude
3. By guidance in steering and speedbrake command calculations
4. By flight control for control law computations and for reconfiguration and moding

Figure 2-16 shows the ADTA power and data interfaces. The air data probe deploy and stow power and logic interfaces can be found in the ADS workbook. Figure 2-17 shows the flow of ADTA data through the software.

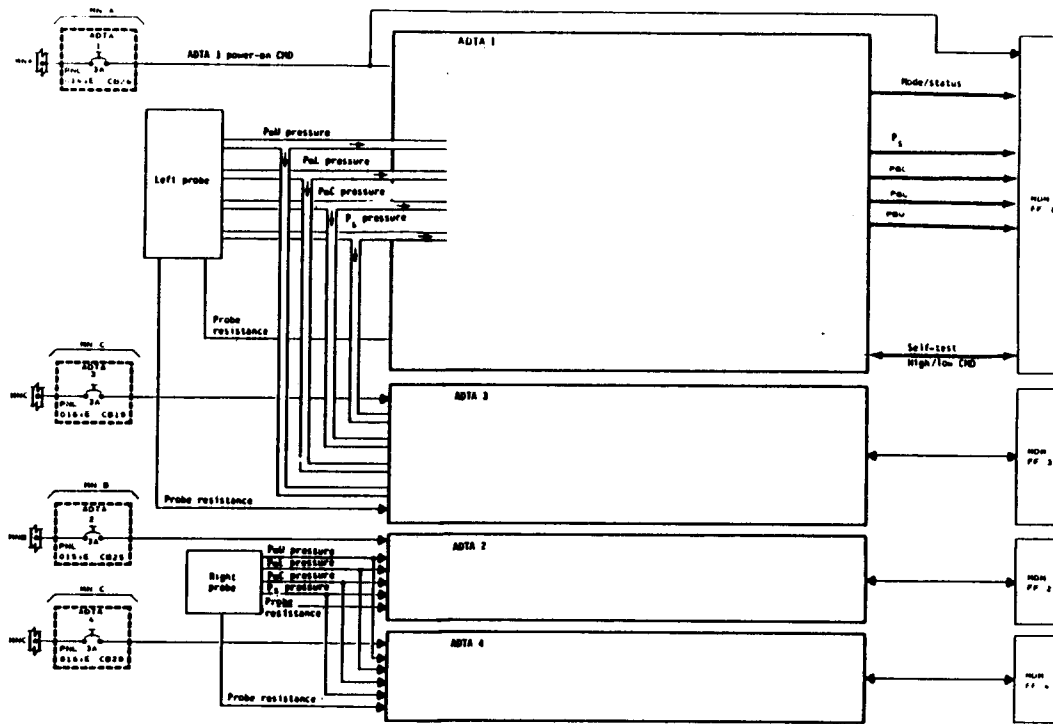


Figure 2-16.- ADTA power and data interfaces.

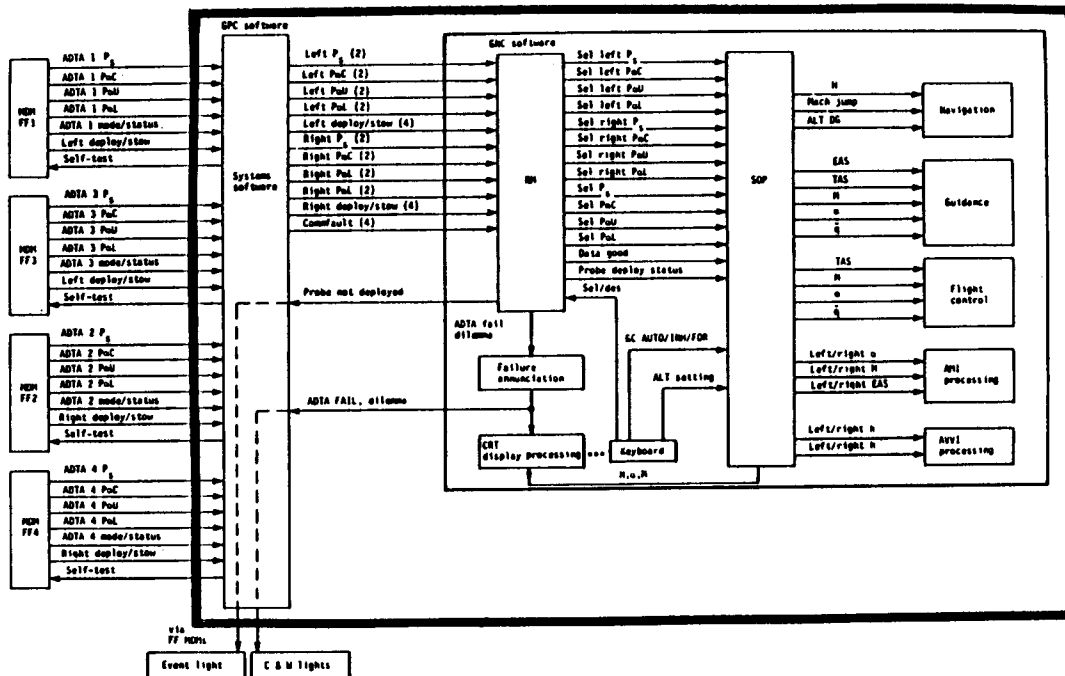


Figure 2-17.- Flow of air data through software.

2.2.6 **Microwave Landing System.** The onboard microwave landing system (MLS), or microwave scanning beam landing system (MSBLS), units determine slant range, azimuth, and elevation to the ground stations located along the side of the landing runway. The glideslope ground station provides elevation; the azimuth/DME (distance measuring equipment) ground station provides azimuth and slant range. Figure 2-18 defines these quantities and shows the angular limits of the beams.

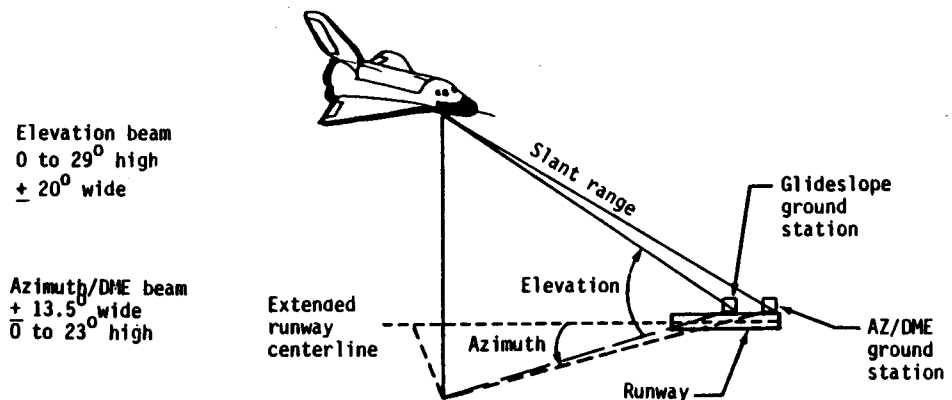


Figure 2-18.- MLS quantities.

MLS azimuth, elevation, and range are used in the approach and landing phase by

1. Navigation to update the state vector components and increase their accuracy
2. Area navigation for display on the horizontal situation indicator, after being transformed to other parameters in the runway coordinate system

The MLS system locks on at approximately 18 000 feet altitude and remains through touchdown. The MLS data replaces the TACAN and air data parameters in the navigation software.

There are three MLS units on the Orbiter. They are located in the forward avionics bays. Each MLS has one antenna on the upper surface of the nose. The ground stations operate on a frequency (Ku-band) corresponding to a channel number selected by the crew using thumbwheels on panel 08.

The ground station transmits a distance measuring equipment (DME) solicit pulse. The onboard MLS receiver responds with a DME interrogation pulse. The ground equipment then responds by transmitting return pulses. A decoder in the MLS decodes the pulses to determine range, azimuth, and elevation.

- **Range.** Range is a function of the elapsed time between interrogation pulse transmission and signal return.
- **Azimuth.** The pulses are returned in pairs. The spacing between the two pulses in a pair identifies the pair as azimuth and indicates which side of the runway the Orbiter is on. The spacing between pairs of pulses defines the angular position from runway centerline.
- **Elevation.** The spacing between the two pulses in a pair identifies the pair as elevation. The spacing between pairs of pulses defines the angular position of the Orbiter above the runway.

The MLS data processing is done in the MLS SOP. This SOP scales and converts data to appropriate units for use by other software modules.

The power and data interfaces of the MLS are shown in figure 2-19, and the flow of MLS data through the software is shown in figure 2-20.

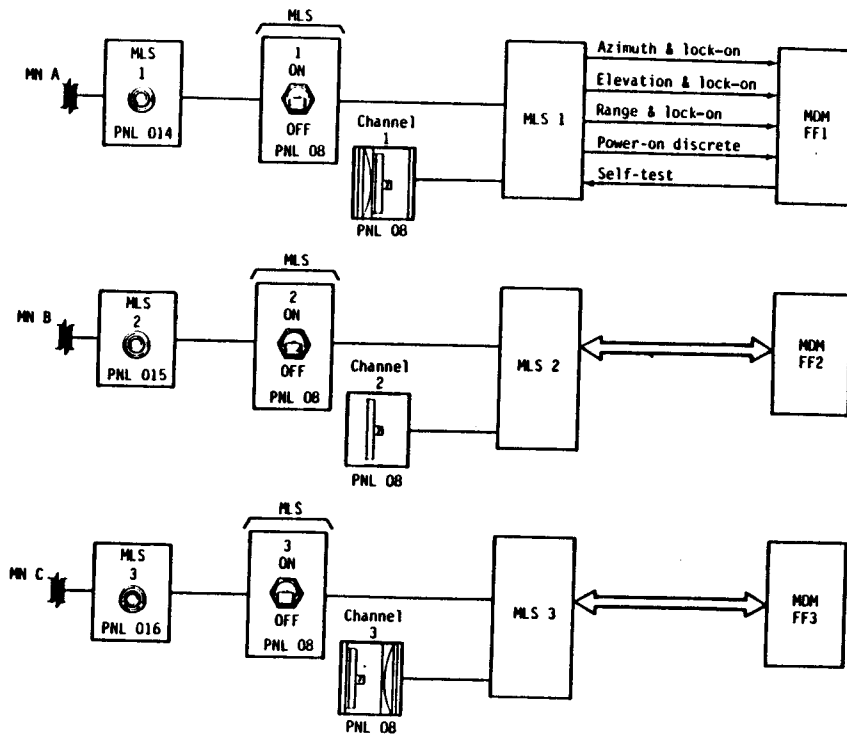


Figure 2-19.- MLS power and data interfaces.

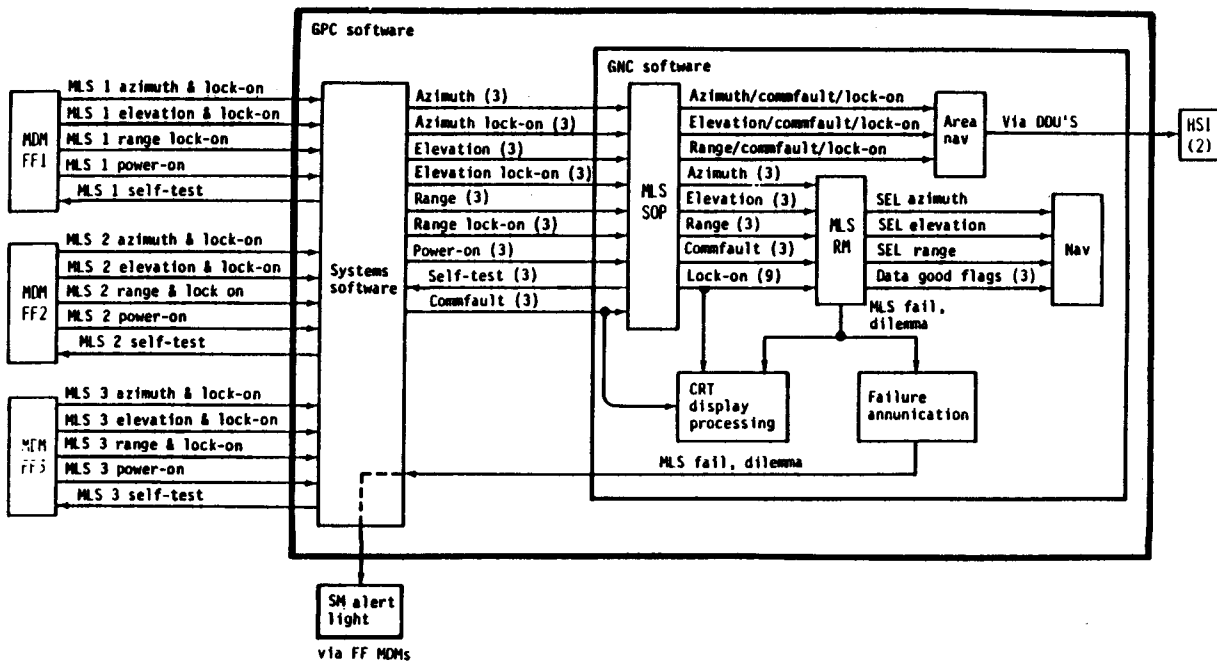


Figure 2-20.- Flow of MLS data through software..

2.2.7 Radar Altimeter. The radar altimeters (RAs) measure absolute altitude from the Orbiter to the nearest terrain within the beam width of their antennas. The altitude is used for display to the crew on the altitude/vertical velocity indicators (AVVIs).

There are two RAs on the Orbiter. They are located in avionics bays 1 and 2. Each RA has a transmitting antenna and a receiving antenna. The four antennas are located on the underside of the Orbiter nose (figure 2-21). The beam width is a 40° cone centered around the Z axis.

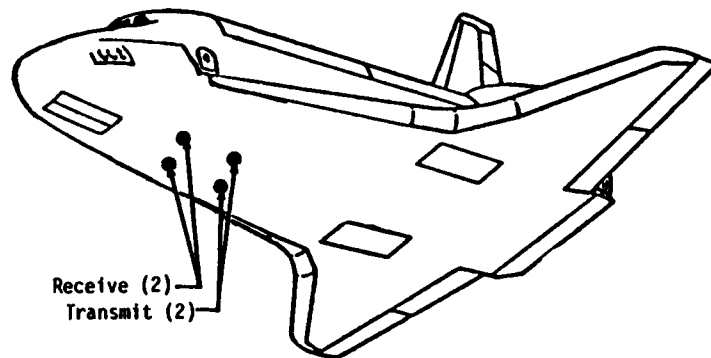


Figure 2-21.- RA antenna locations.

Each RA transmits a C-band (4300-MHz carrier modulated at 8.5 KHz) pulse through its transmitting antenna. The signal is reflected by the nearest terrain, and the leading edge of the return radar echo is locked on by the RA through its receiving antenna. The altitude output by the RA is proportional to the time delay between the transmitted and received pulses. The range of the RA output is from 0 to approximately 5000 ft. Specification tracking rate is 2000 ft/s maximum. A lock-on flag is output to indicate that the RA is tracking the terrain. The RA will not lock on if the vehicle has large pitch or roll angles.

Figure 2-22 shows the RA power and data interfaces and figure 2-23 shows the flow of RA data through the software. It should be noted that, at the time of the writing of this workbook, the RA RM processing is turned off by I-loads so that the data is not used by navigation; however, the software processing is still present for possible future use.

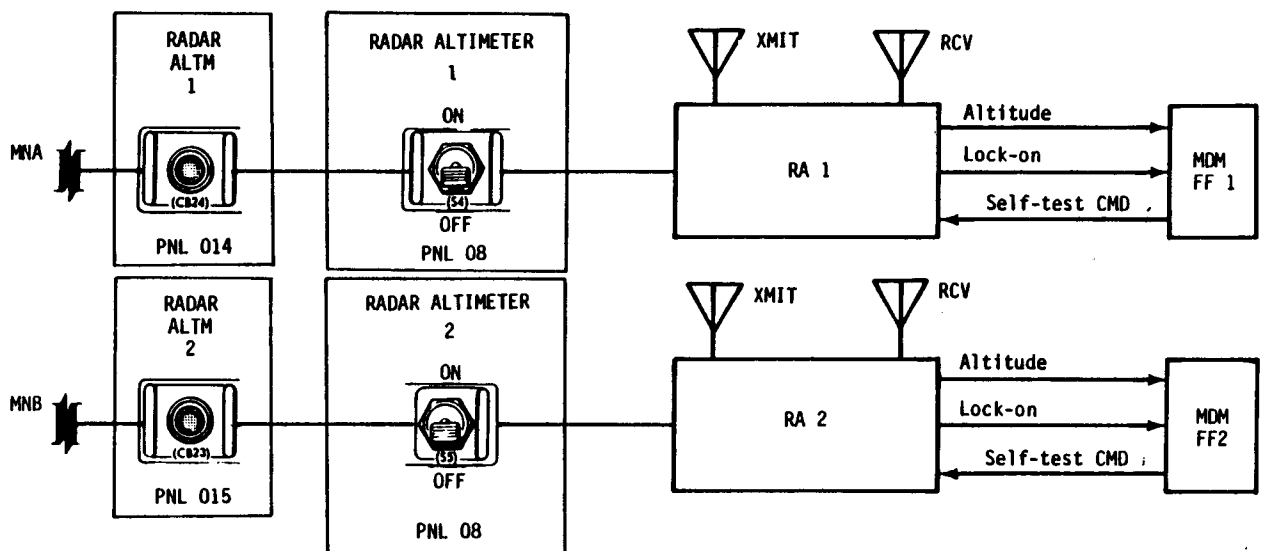


Figure 2-22.- RA power and data interfaces.

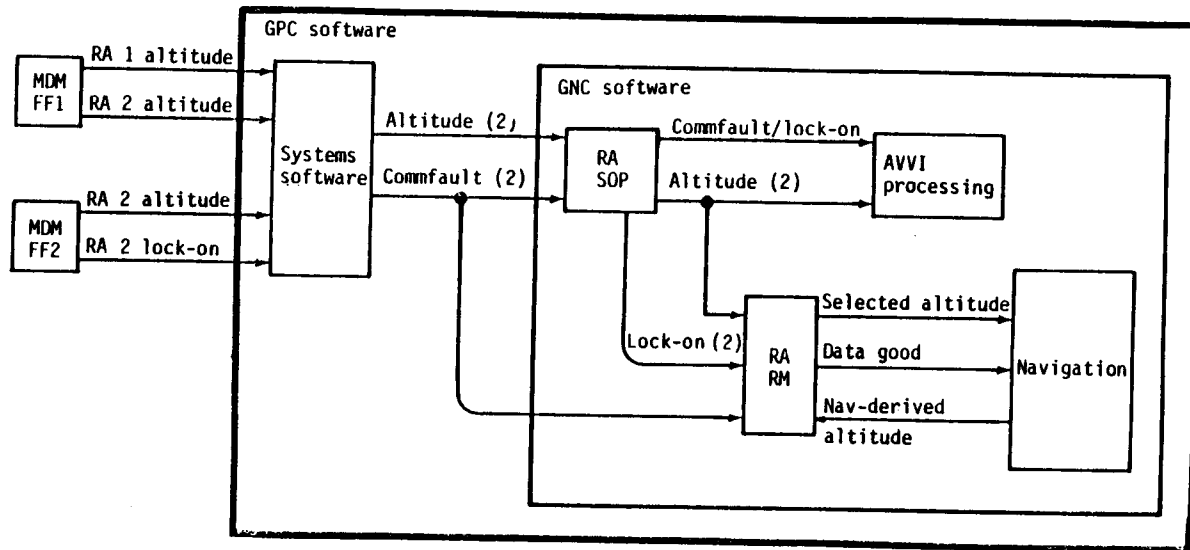


Figure 2-23.- Flow of RA data through software.

2.2.8 Rendezvous Radar. The rendezvous radar is one operating mode of a Ku-band radar and communications dual-purpose system used during orbital operations. The system performs as either a rendezvous radar to detect and track detached payloads or other spacecraft during rendezvous maneuvers, or as a wide-band, two-way communications system for data exchange with the ground via the tracking and data relay satellite system (TDRSS). The functions of radar and communications are not available simultaneously since they share a common high-gain antenna mounted on a deployable assembly on the starboard side of the Orbiter payload bay. During launch and entry, the deployable assembly is stowed internal to the payload bay. Figure 2-24 depicts the dual functions of this system.

Radar rendezvous range

- 12 n. mi., max - passive payload
- 300 n. mi., max - active payload

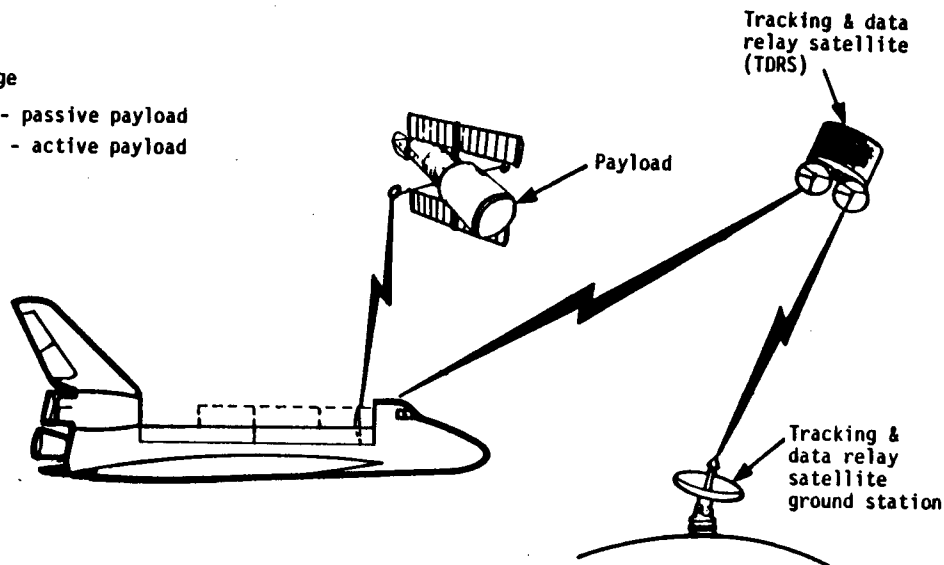


Figure 2-24.- Ku-band radar and communications system.

The Ku-band system consists of four LRUs as shown on the power block diagram (figure 2-25). The radar subsystem can skin track a target within a range of 10 feet to 12 nautical miles or track an active target equipped with a transponder within a range of 10 feet to 300 nautical miles. The radar automatically tracks targets to provide range, range rate, pointing angle and inertial angle rate data to the Orbiter data interfaces (MDMs).

The RR requires two GPCs: one processing GNC software (GNC GPC) and one processing Orbiter systems software (SM GPC); see figure 2-26. In proximity operations or rendezvous operations, the GNC GPC maintains the state vector of the Orbiter and the target vehicle. The data from the RR is used by the GNC GPC to better define the relative state vector between the Orbiter and the target. The SM GPC uses the state vector information from the GNC GPC to compute and command the antenna angles and to give it a range estimate in which to search for the target. The data flow diagram shows the primary data interfaces of the two GPCs.

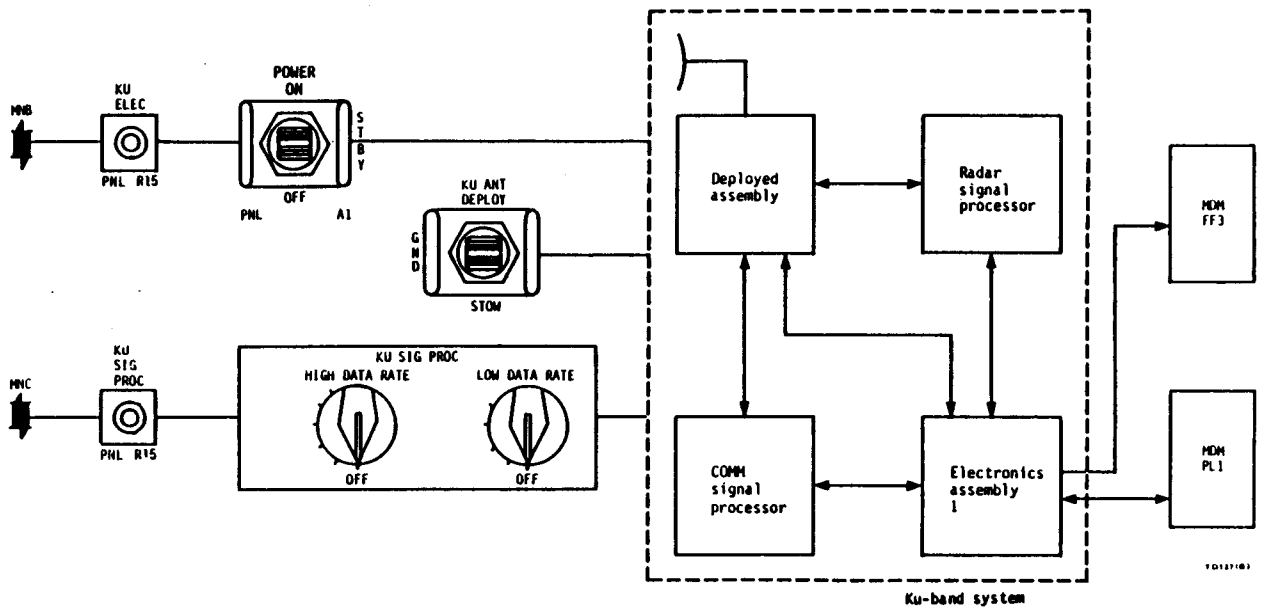


Figure 2-25.- Power and data interfaces of Ku-band system.

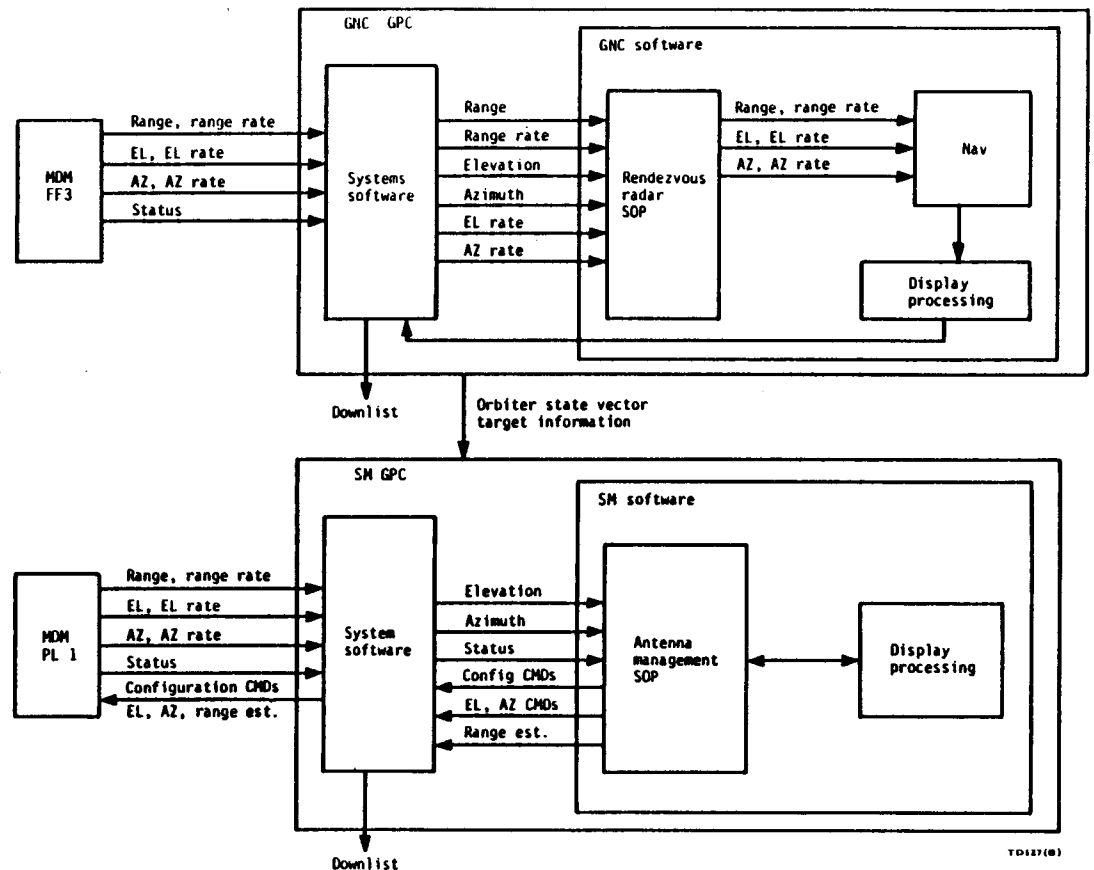


Figure 2-26.- Flow of RR data through software.

2.3 FLIGHT CONTROL SYSTEM HARDWARE

2.3.1 **Accelerometer Assemblies.** There are four accelerometer assemblies AAs on the Orbiter mounted in the forward avionics bays. Each AA contains two identical single axis accelerometers, one of which senses vehicle acceleration along the lateral vehicle axis while the other senses vehicle acceleration along the vertical (normal) vehicle axis (figure 2-27).

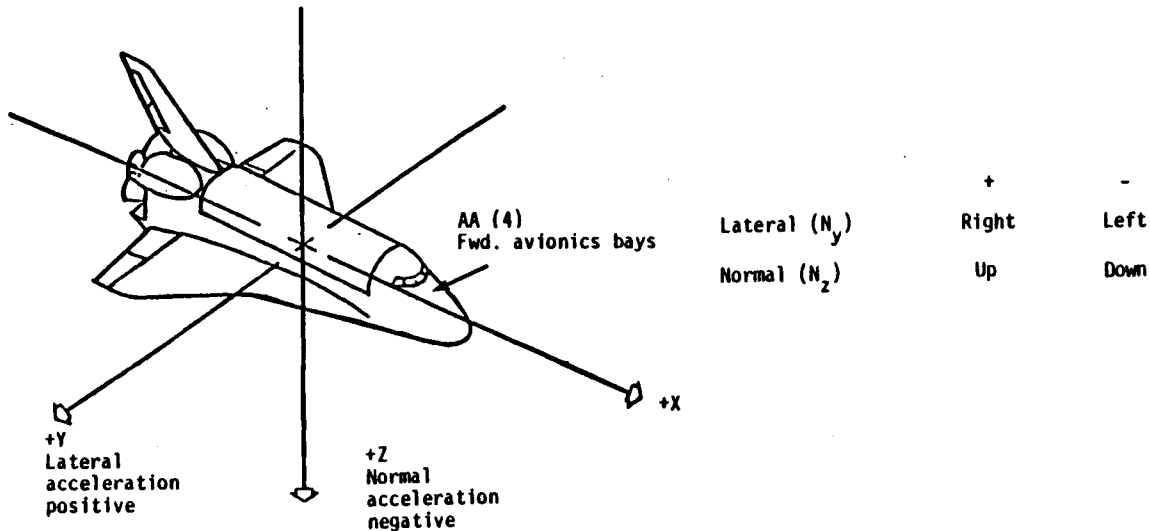


Figure 2-27.- AA location and acceleration axes.

Each accelerometer consists of a pendulum suspended so that its base is in a permanent magnetic field between two torquer magnets. A lamp is beamed through an opening in one of the torquer magnets. Photodiodes are on both sides of the other torquer magnet. When acceleration deflects the pendulum toward one photodiode, the resulting light imbalance on the two photodiodes causes a differential voltage. The differential voltage is used to increase the magnetic field on one of the torquer magnets to return the pendulum to an offset position; the magnitude of the current required to do that is proportional to the acceleration. The polarity of the differential voltage depends on the direction the pendulum moved, which is opposite to the direction of acceleration. The only difference between the lateral and normal accelerometers is the position in which they are mounted within the assembly. When the acceleration is removed, the pendulum returns to the null position.

The AA SOP converts the eight accelerometer output voltages to gravitational units. These accelerations are used by the flight control system for

1. Feedback to find acceleration errors which are used for stability augmentation during first stage ascent, aborts, and entry
2. Elevon load relief during first-stage ascent
3. Computing steering errors for display on the attitude director indicator (ADI) during TAEM and approach and landing phases

The power and data interfaces of the AAs are shown in figure 2-28. Note that AAs 3 and 4 are redundantly powered while AAs 1 and 2 are not.

The flow of data through the software is shown in figure 2-29.

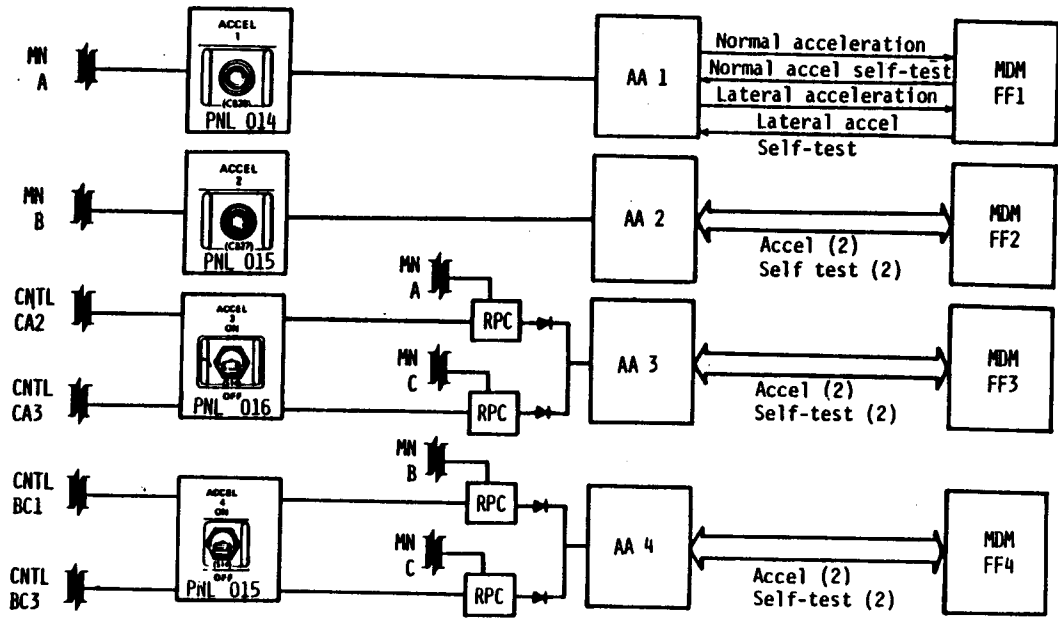


Figure 2-28.- AA power and data interfaces.

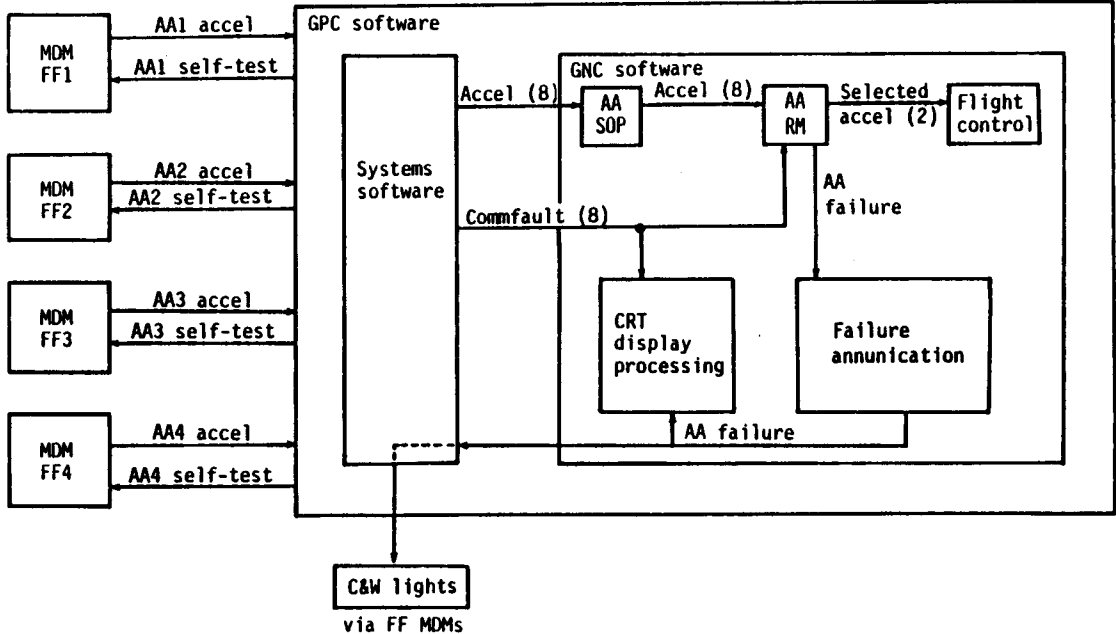


Figure 2-29.- Flow of AA data through software.

2.3.2 Orbiter Rate Gyro Assemblies. There are four rate gyro assemblies (RGAs) located on the aft bulkhead of the Orbiter. Each assembly contains three identical single degree-of-freedom rate gyros positioned so that each gyro senses rotation about one of the vehicle axes. Thus, each RGA contains one gyro sensing roll rate, one gyro sensing pitch rate, and one gyro sensing yaw rate. (See figure 2-30).

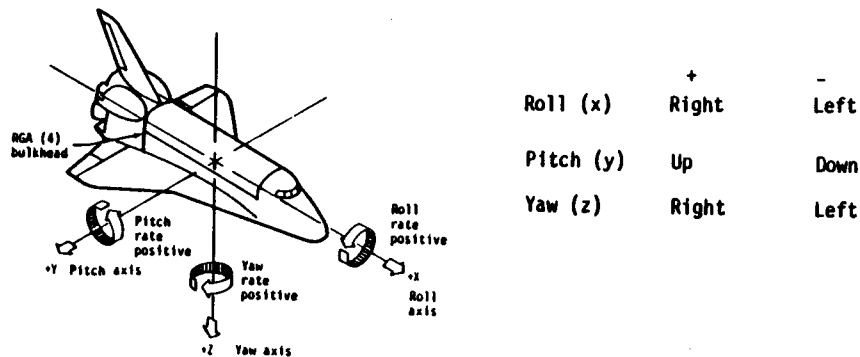


Figure 2-30.- RGA location and rotation axes.

A gyro has three axes. A motor forces the gyro to rotate about its spin axis. When the vehicle rotates about the gyro input axis, a torque results in a rotation about the output axis (figure 2-31). An electrical voltage, proportional to the angular deflection about the output axis and representing vehicle rate about the input axis, is generated and transmitted to the data processing system. This same voltage is used internal to the RGA to generate a counteracting torque to prevent excessive gimbal movement about the output axis.

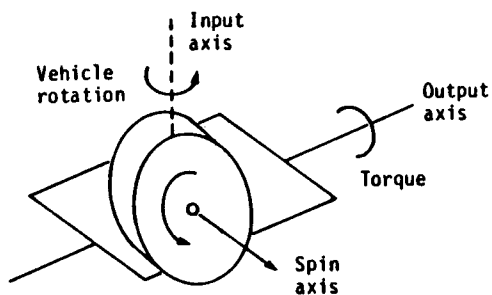


Figure 2-31.- Gyro axes.

The RGA SOP converts the 12 voltages representing a rate into units of degrees per second. These rates are used

1. By the flight control system during ascent, entry, and aborts as feedback to find rate errors which are used for stability augmentation
2. For display to the crew on the attitude director indicator (ADI)

The RGA power and data interfaces are shown in figure 2-32. Note that RGAs 1 and 4 are redundantly powered while RGAs 2 and 3 are not.

Figure 2-33 shows the flow of data through the software.

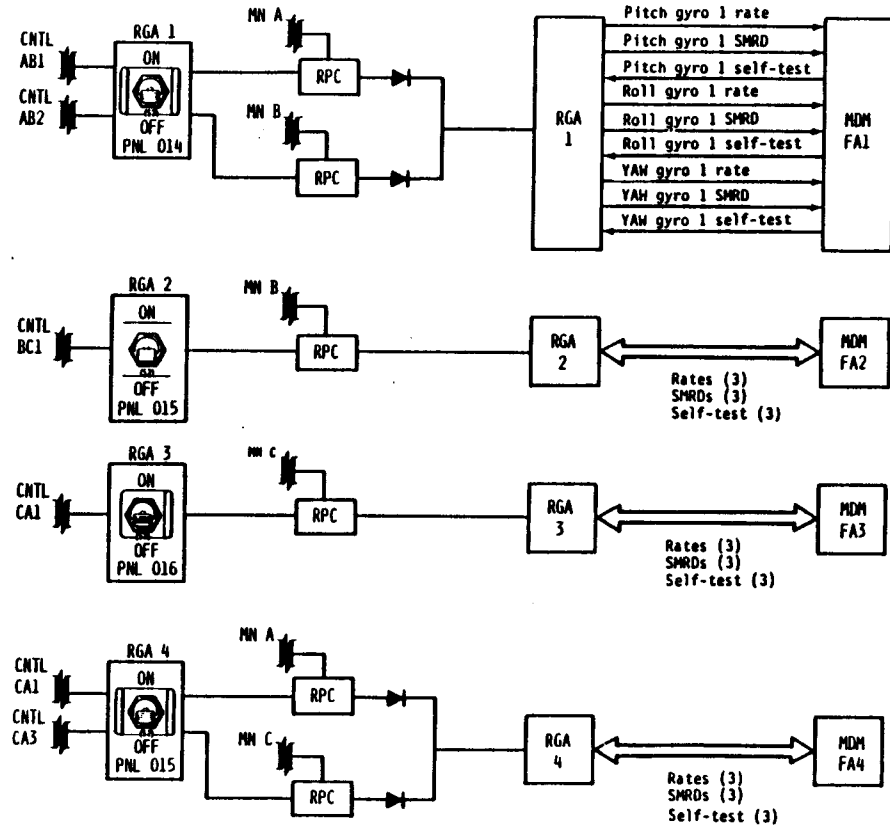


Figure 2-32.- RGA power and data interfaces.

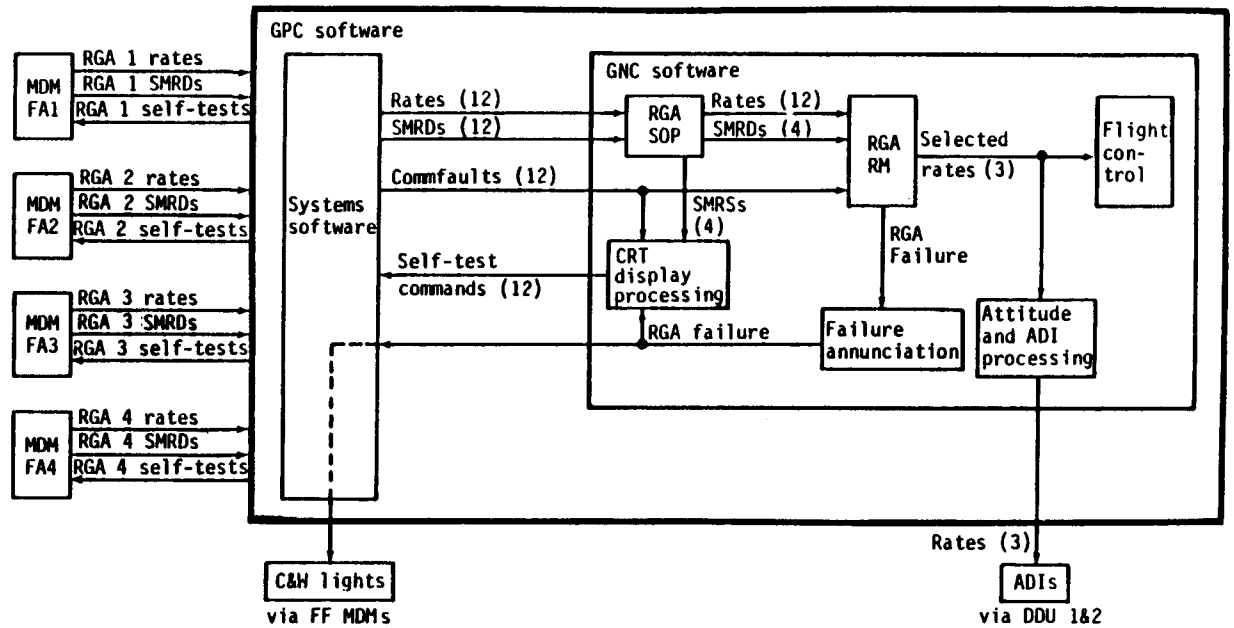


Figure 2-33.- Flow of RGA data through software.

2.3.3 Solid Rocket Booster Rate Gyro Assemblies. There are three RGAs on each SRB, left and right. The RGAs are mounted on the forward ring within the forward skirt of the solid rocket booster/external tank (SRB/ET) attach point (figure 2-34). Each RGA contains two identical single degree-of-freedom rate gyros. The rate gyros themselves are identical to the Orbiter rate gyros described previously. The only difference between the RGAs is that the SRB RGAs have only two gyros sensing pitch and yaw rates whereas the Orbiter RGAs have a third gyro sensing roll rate.

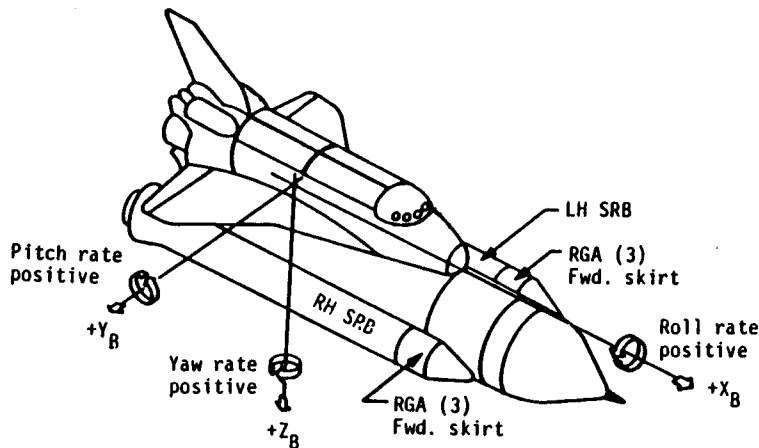


Figure 2-34.- SRB RGA location and rotation axes.

The SRB RGA SOP converts the 12 voltages representing a rate into units of degrees per second. These rates are used by the flight control system during first-stage ascent (from lift-off until seconds before SRB separation) as feedback to find rate errors which are used for stability augmentation. After first-stage ascent, flight control drops the SRB RGA rates and begins using rates from the Orbiter RGAs.

Figure 2-35 shows the SRB RGA power and data interfaces.

Figure 2-36 shows the flow of SRB RGA data through the software.

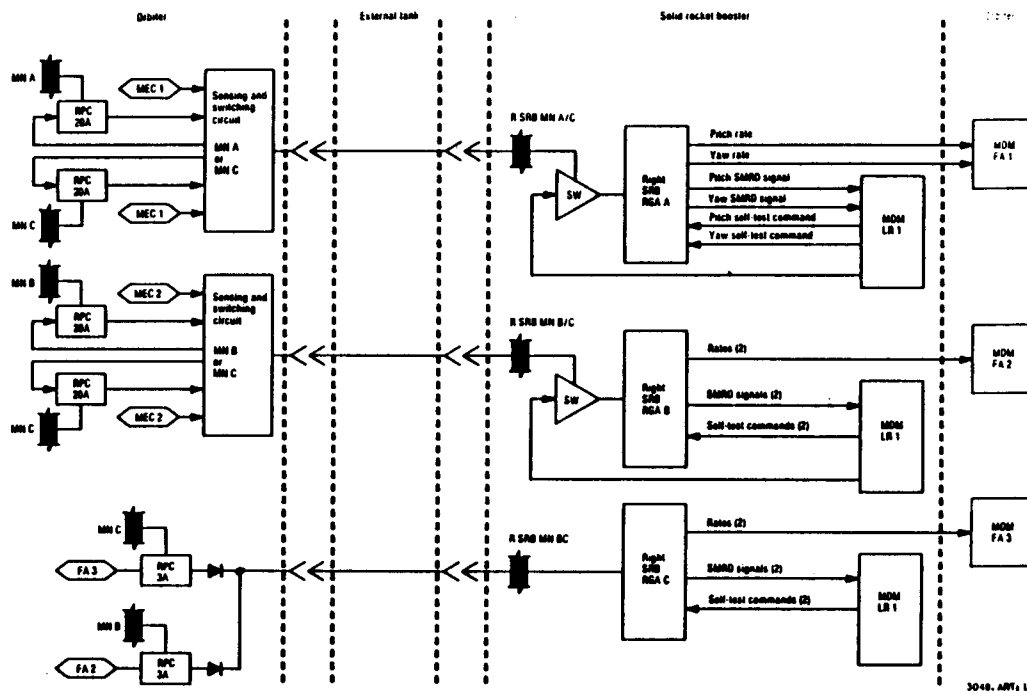


Figure 2-35.- SRB RGA power and data interfaces.

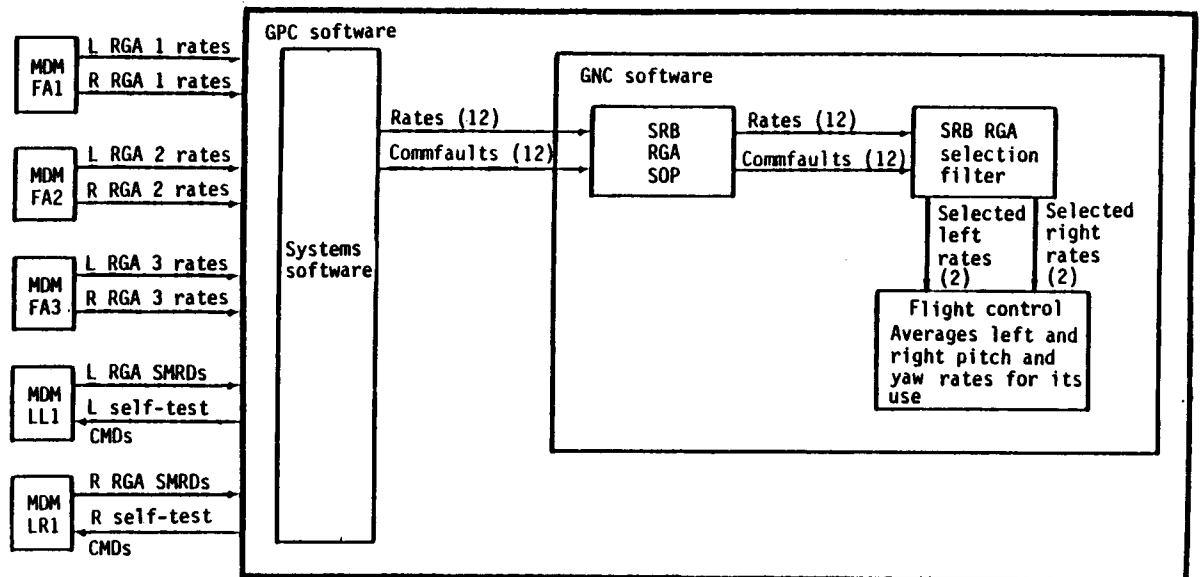


Figure 2-36.- Flow of SRB RGA data through software.

2.3.4 **Controllers.** The controllers are used to manually control the Orbiter attitude and trajectory by positioning engines and aerosurfaces and by firing RCS thrusters. The controllers are

- Rotational hand controllers
- Translational hand controllers
- Rudder pedals
- Speedbrake/thrust controllers
- Body flap switch
- Trim switches

Each is covered separately with its associated switches and indicators. The aerosurfaces they control are covered in the next section. The intricacies of selecting manual control are not discussed here.

2.3.4.1 **Rotational Hand Controller.** The rotational hand controllers (RHCs) control vehicle rotation about three axes: roll, pitch, and yaw.

The function and usage of the RHCs varies by flight phase:

1. **Ascent.** Left and right RHCs may be used to gimbal the main engines and solid rocket motors.
2. **Insertion and deorbit.** Left and right RHCs may be used to gimbal the OMS engines and fire RCS thrusters.
3. **Orbit.** Left, right, and aft RHCs may be used to fire RCS thrusters.
4. **Entry.** Left and right RHCs may be used to fire RCS thrusters during early entry and may be used to position elevons. RHCs provide aerosurface control in roll and pitch axes from entry through landing.

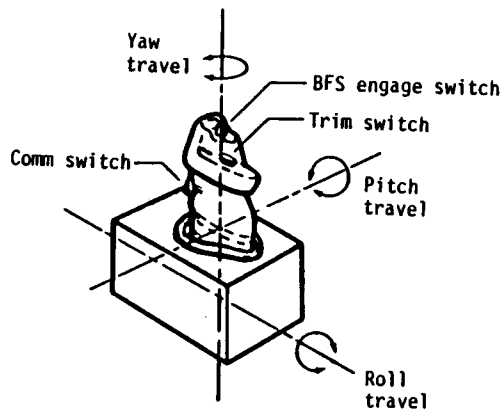


Figure 2-37.- RHC axes.

There are three RHCs:

- Left RHC at the commander's position
- Right RHC at the pilot's position
- Aft RHC at the mission specialist's position

Each moves about three axes as shown in figure 2-37.

The aft RHC is used on orbit only when the crewmember is looking out the rear (-X) or overhead (-Z) windows. Human factors dictate that an RHC deflection produces a rotation in the same direction relative to the crewmember's line of sight, regardless of direction, with respect to vehicle axes. Software performs the necessary transformations on aft RHC commands to make them common to the left and right RHCs. A sense switch at the aft station is used to specify which window (and transformation) is being used: -X or -Z.

Several switches are located on the RHC as shown in figure 2-37.

1. BFS mode button. The BFS mode button engages the backup flight system when depressed at the left or right RHC.
2. Trim switch. The trim switch is a two-contact switch that may be pushed forward or backward to add a trim rate to the RHC pitch command; pushing it left or right adds a roll trim rate. The trim switch on the aft RHC is inactive.
3. Comm switch. The comm switch is a push-to-talk switch that enables voice transmission while the switch is depressed.

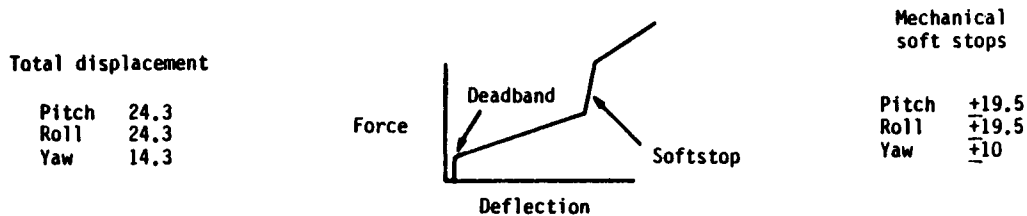


Figure 2-38.- RHC total and softstop displacements.

The RHC has an initial deadband of 0.25⁰ in all three axes. To move the RHC beyond the deadband, an additional force is required. At an amount of deflection called the "soft stop," a step increase in the force required for further deflection occurs. Total and soft stop displacements are shown in figure 2-38. The forces required at each deflection vary by axis. Software has defined deadbands and soft stops whose values differ from the mechanical RHC deadbands and soft stops. When a software detent position is exceeded, that RHC assumes manual control.

Each RHC contains nine transducers:

- Three redundant transducers sensing roll deflection
- Three redundant transducers sensing pitch deflection
- Three redundant transducers sensing yaw deflection

The transducers produce an electrical signal proportional to the deflection.

The three transducers are called channel 1, channel 2, and channel 3. Each channel is powered by a separate power supply in its associated DDU.

The RHC SOP performs the following functions:

1. If RM (section 3) determines that an RHC is bad, all commands from that RHC are set equal to zero.
2. RHC commands are compensated and biased. The aft RHC commands are transformed according to the sense switch position.
3. Discretes are set to indicate if an RHC deflection is outside the software detent or is beyond the software softstop.
4. The compensated and transformed commands are summed, by axis, to produce rotation commands for flight control software.

Figure 2-39 shows the power and data interfaces of the RHCs. The DDU is the display driver unit which provides the necessary power for the controllers.

Figure 2-40 shows the flow of data through the software.

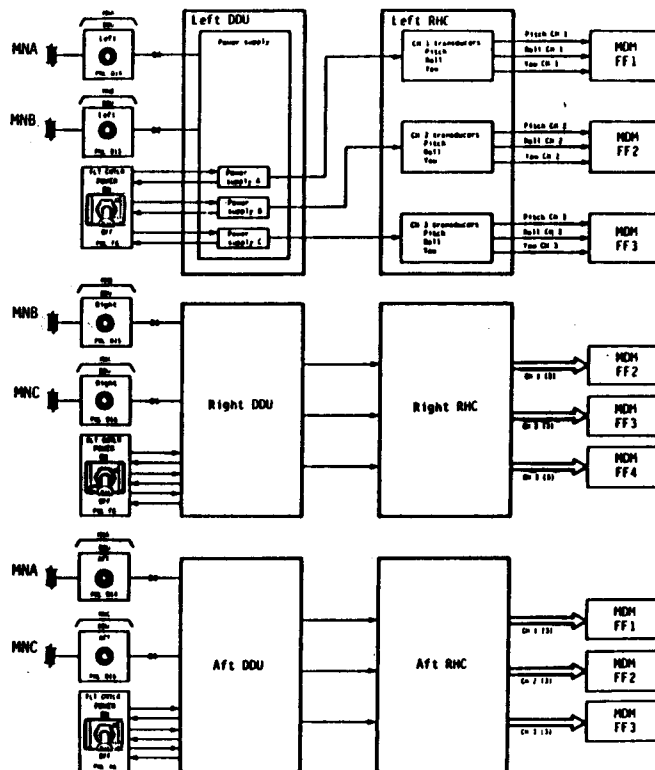


Figure 2-39.- RHC power and data interfaces.

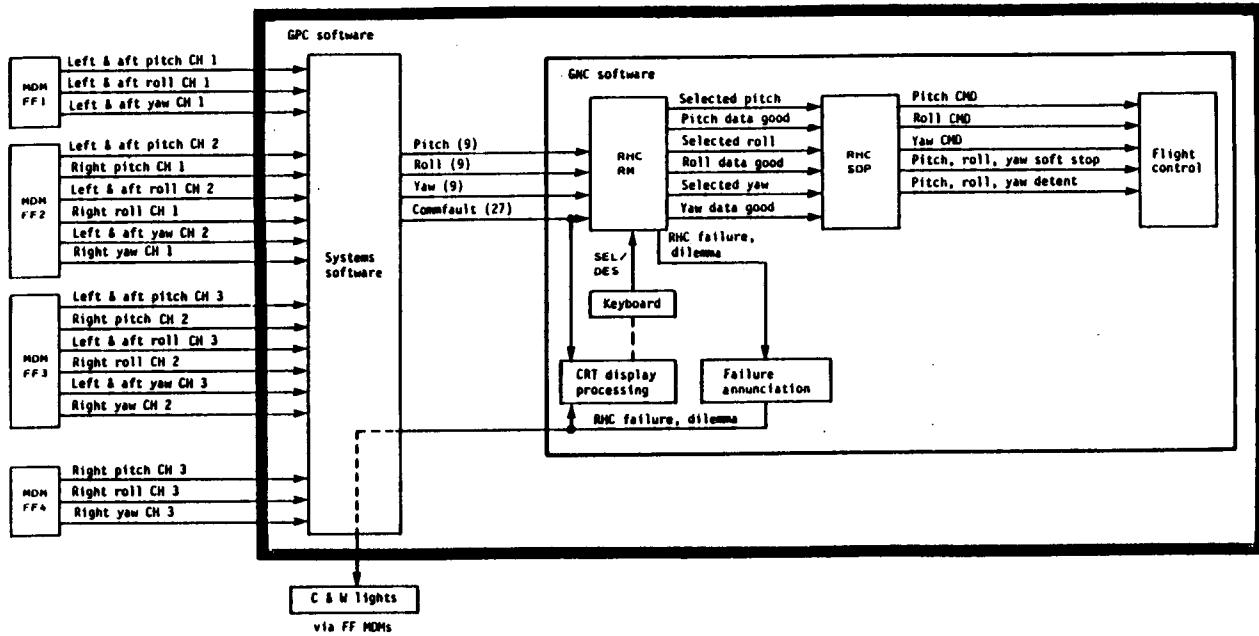


Figure 2-40.- Flow of RHC data through software.

2-57

2.3.4.2 Translational Hand Controller. The translational hand controllers (THCs) are used to manually control translation along the longitudinal (X), lateral (Y), and vertical (Z) vehicle axes (figure 2-41). The translation commands result in flight control software commands which fire RCS thrusters.

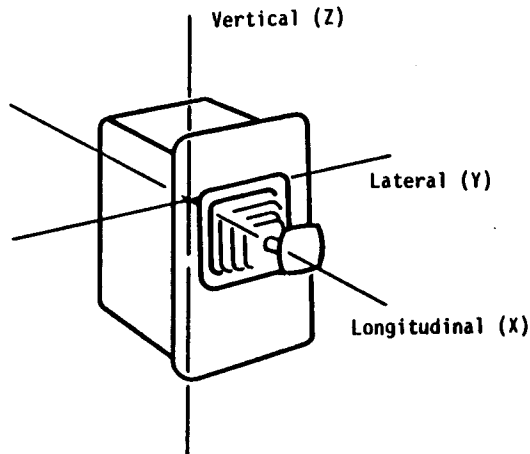


Figure 2-41.- THC translational axes.

There are two THCs: one at the commander's station (left) and one at the mission specialist's station (aft). The left THC is active during the orbital insertion, orbit, and deorbit flight phases; the aft THC is active during the orbit phase only.

The aft THC is used when the crewmember is looking out the rear or overhead windows. The commands are transformed for the same reason and in a similar manner as they are for the RHC. The same sense switch specifies which transformation to use.

Each THC contains six three-contact switches: one each in the + and - directions for each of the three axes. Moving the THC to the right closes three switch contacts called channels 1, 2, and 3. This information goes to the flight control function which would command translation along the +Y axis.

Figure 2-42 shows the power and data interfaces of the THCs.

Figure 2-43 shows the flow of data through the software.

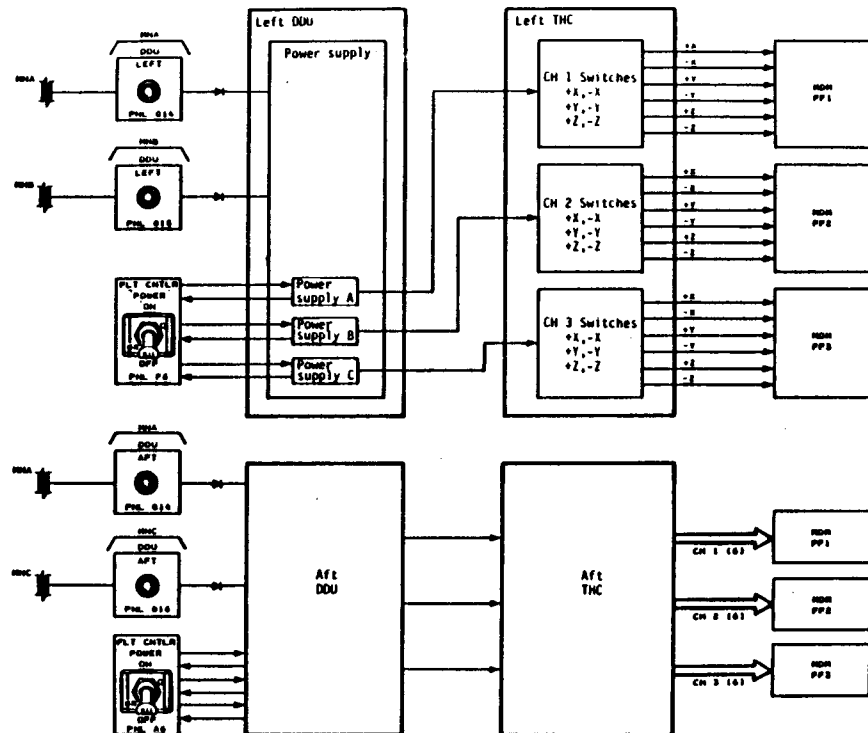


Figure 2-42.- THC power and data interfaces.

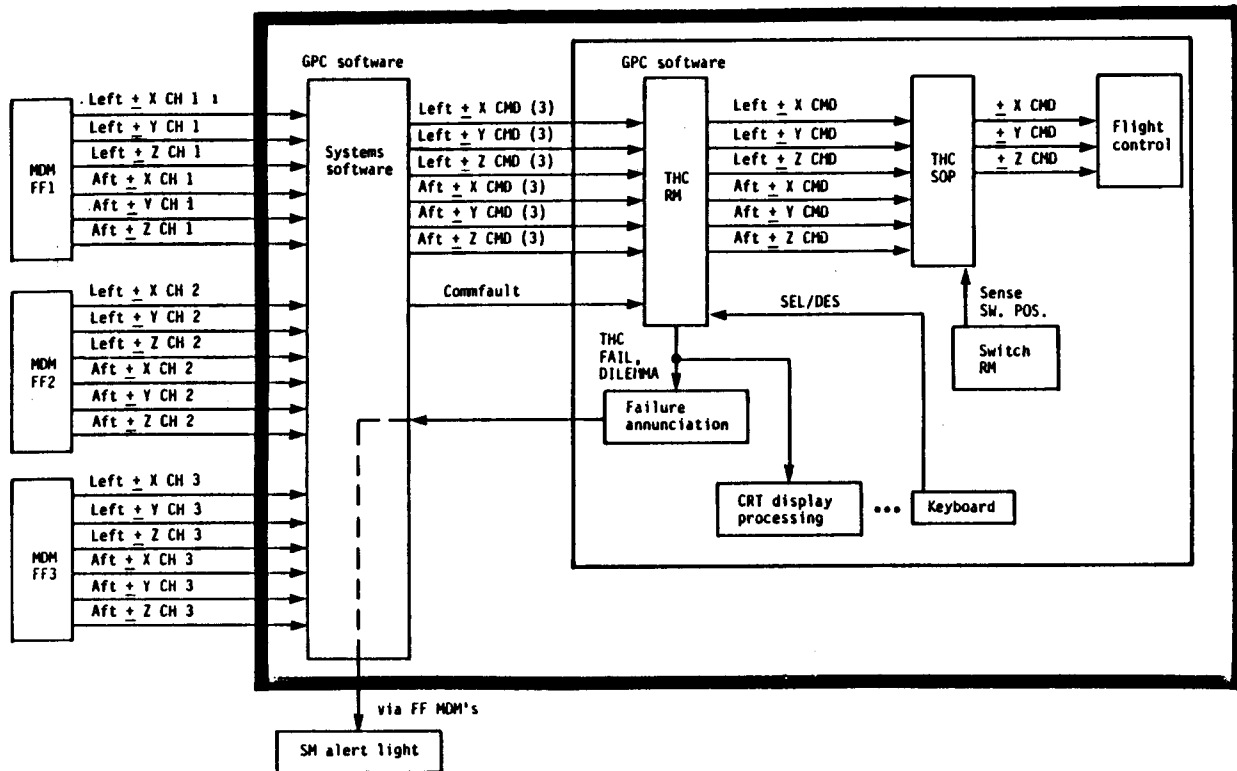


Figure 2-43.- Flow of THC commands through software.

2.3.4.3 Rudder Pedals. The rudder pedals command vehicle rotation about the yaw axis by positioning the rudder during atmospheric flight (see figure 2-44). Because the flight control software performs automatic turn coordination, the rudder pedals are not used until the wings are leveled prior to touchdown in actual practice.

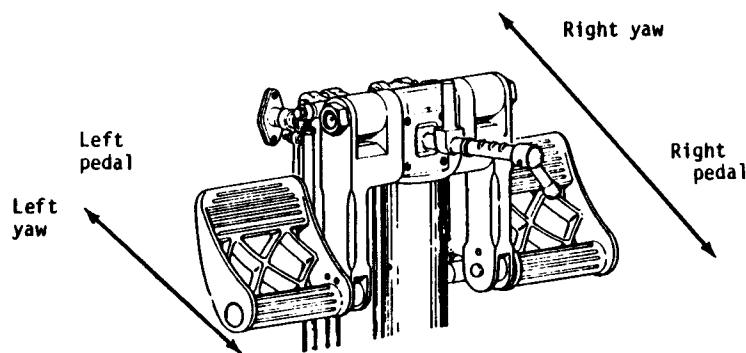


Figure 2-44.- Rudder pedals.

There are two pairs of rudder pedals: one at the commander's station and one at the pilot's station. When a pedal is depressed, it moves a mechanical input arm in a rudder pedal transducer assembly (RPTA). The RPTA contains three transducers (channel 1, channel 2, and channel 3) and each generates an electrical signal proportional to the rudder pedal deflection. The signals are output to the data processing system. The left and right rudder pedals are mechanically linked together so that movement on one side moves the other side.

The rudder pedals provide two additional functions after touchdown which are unrelated to software:

1. Rudder pedal deflections turn the nosewheel to provide steering.
2. Depressing the upper portion of the pedals provides braking. Differential braking may be used for steering.

The RPTA SOP does the following processing on the RPTA data:

1. Converts the selected left and right commands from volts to degrees.
2. If RM declares an RPTA bad, the command for that RPTA is set to zero.
3. Selects the largest of the left and right commands for output to flight control software after applying a deadband.

The power and data interfaces of the RPTAs is shown in figure 2-45 and the flow of RPTA data through the software is shown in figure 2-46.

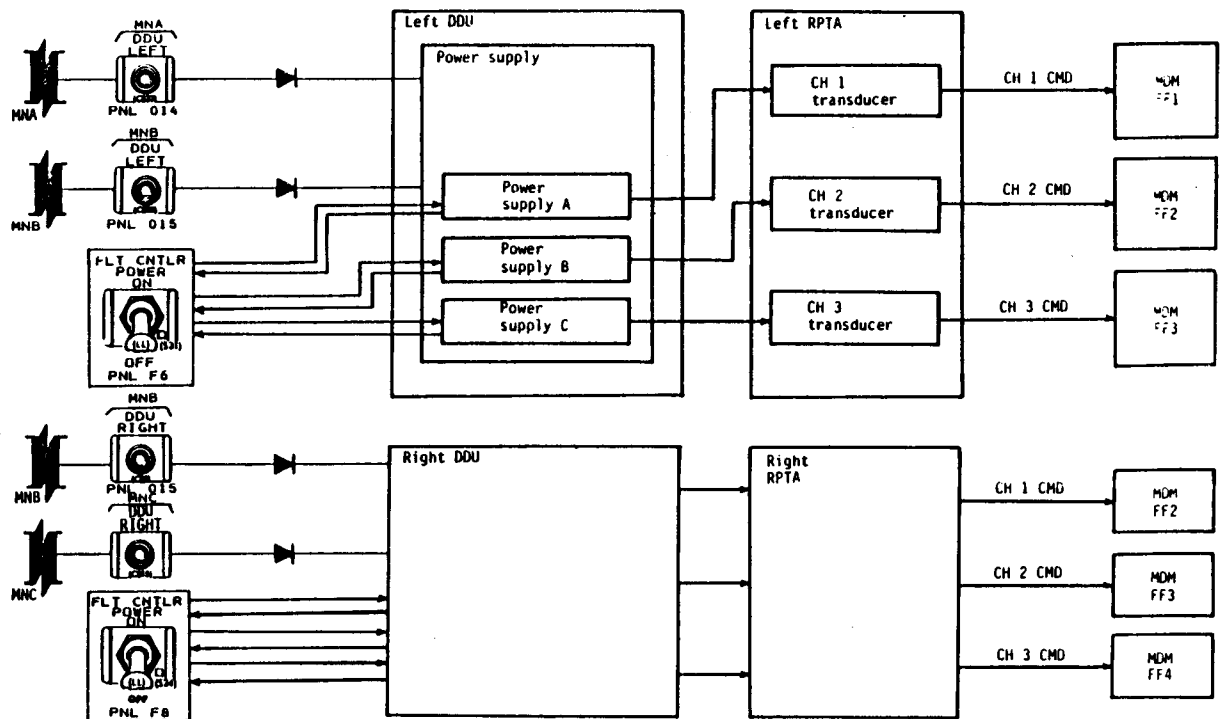


Figure 2-45.- RPTA power and data interfaces.

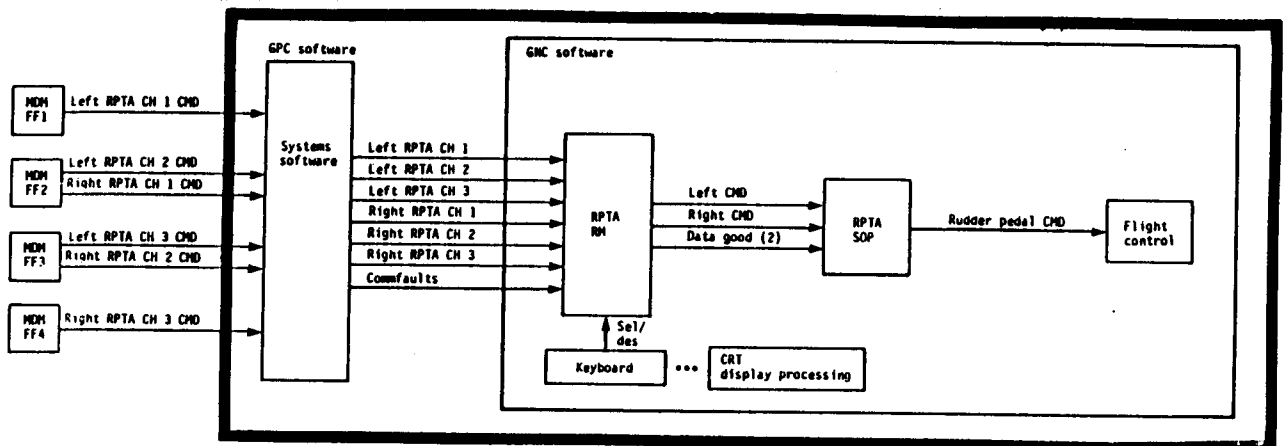


Figure 2-46.- Flow of rudder pedal commands through software.

2.3.4.4 Speedbrake/Thrust Controller. There are two SBTCs on the Orbiter (figure 2-47). One is at the commander's station and one at the pilot's station. At the forward setting, the thrust level is greatest or the speedbrake is closed. Rotating the controller back decreases the thrust level or opens the speedbrake. Each SBTC contains three transducers (channel 1, channel 2, and channel 3) which produce a voltage proportional to the deflection. A takeover switch (three contacts) on each SBTC gains manual control of the thrust setting or speedbrake position for that SBTC when it is depressed.

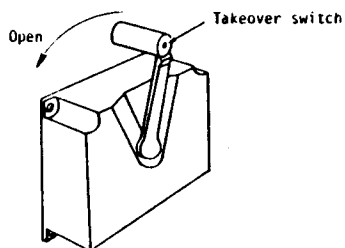


Figure 2-47.- Speedbrake/thrust controller.

The speedbrake/thrust controllers serve two distinct purposes:

1. During ascent, the pilot's SBTC may be used to vary the thrust level of the main engines.
2. During entry, both SBTCs may be used to control aerodynamic drag (and hence airspeed) by opening or closing the speedbrake.

The SBTC SOP performs the following functions:

1. Converts the selected speedbrake command from volts to degrees and converts the throttle command to a setting in percent.
2. Selects the speedbrake command from the SBTC whose takeover button was last depressed. The command is frozen if RM declared the SBTC bad. If both takeover buttons are simultaneously depressed, the commander's SBTC is given control.

Figure 2-48 shows the SBTC power and data interfaces, and figure 2-49 shows the flow of SBTC data through the software.

2-66

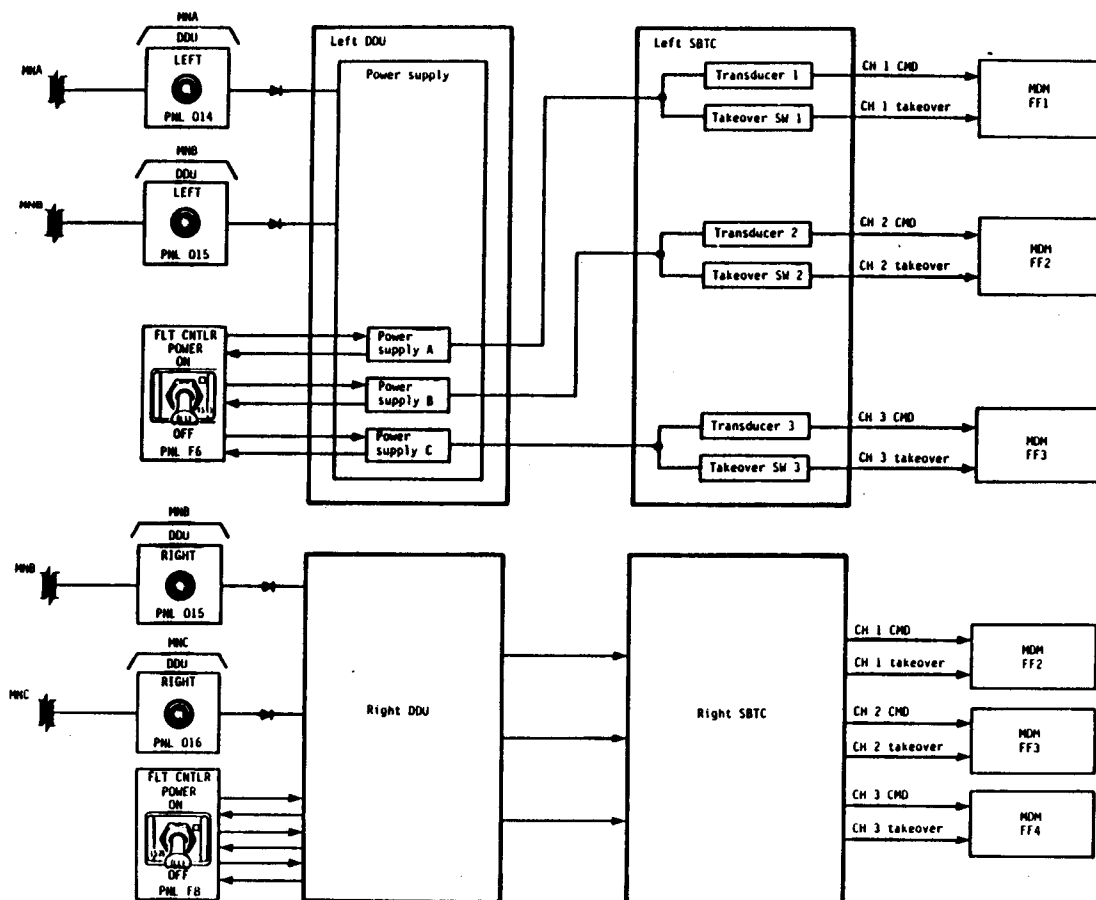


Figure 2-48.- SBTC power and data interfaces.

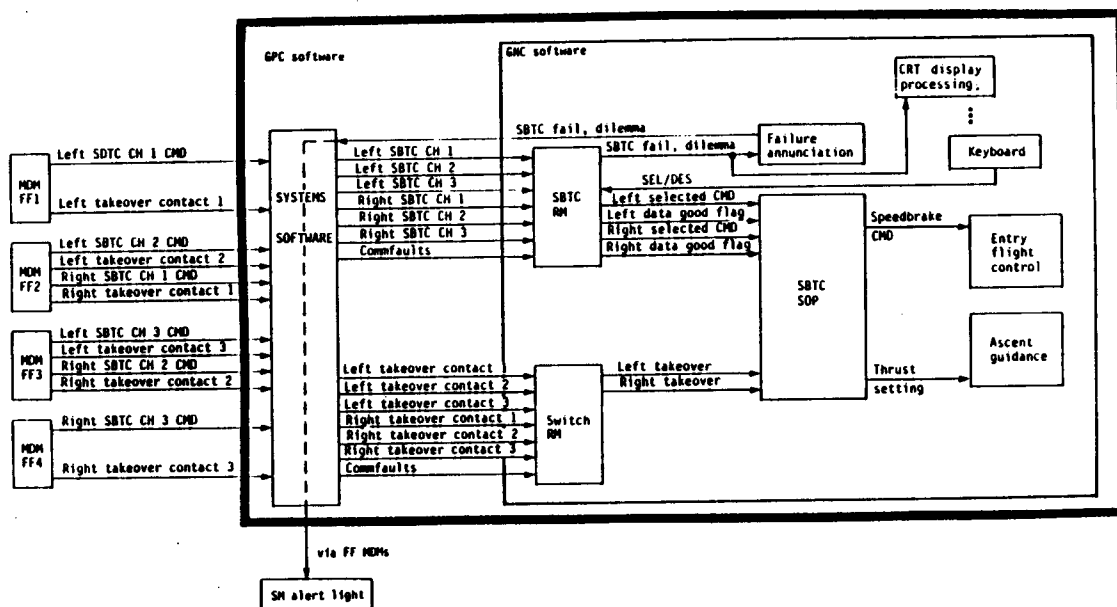


Figure 2-49.- Flow of SBTC data through software.

2.3.4.5 Body Flap Switch. The body flap switch provides manual control for positioning the body flap for main engine thermal protection and for reducing elevon deflections during the entry phase.

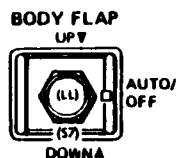


Figure 2-50.- Body flap switch.

There are two body flap switches on the Orbiter (see figure 2-50). One is at the commander's station, and one is at the pilot's station. They are lever-locked switches, spring loaded to the center position. The UP and DOWN positions of the switch each have two contacts. Software processes the contacts and selects a position for flight control to generate as a command. If the two switches generate opposing commands, UP has priority.

Each contact of the switches receives power from a control bus and each contact status is sent to the data processing system via an FF MDM.

2.3.4.6 Trim Switches. Panel trim switches allow the crew to make small changes to RHC commands by applying a trim to them. The trim switches actually move the aerosurfaces, whereas the RHC trim switches change the rate command.

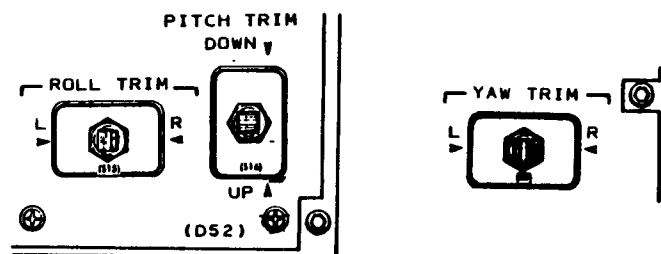


Figure 2-51.- Trim switches.

There are six panel trim switches controlling roll, pitch, and yaw at the commander's and pilot's crew stations (figure 2-51). Each switch has two two-contact positions and is spring loaded to the center (off) position. If two switches generate opposing commands, no trim is output.

A switch called PANEL TRIM on panel F3 enables control bus power to a set of three panel trim switches. Each contact receives power from a control bus and each contact status is sent to the data processing system via an FF MDM. A switch called RHC/PANEL TRIM on panel F3 must be in the enable position for panel trim signals to be recognized by the software.

2.3.5 Aerosurface Servo Amplifiers/Actuators. Vehicle control during atmospheric flight is accomplished in the entry phase by deflecting the seven aerosurfaces on the Orbiter. The aerosurfaces and vehicle axes they control are shown in figure 2-52.

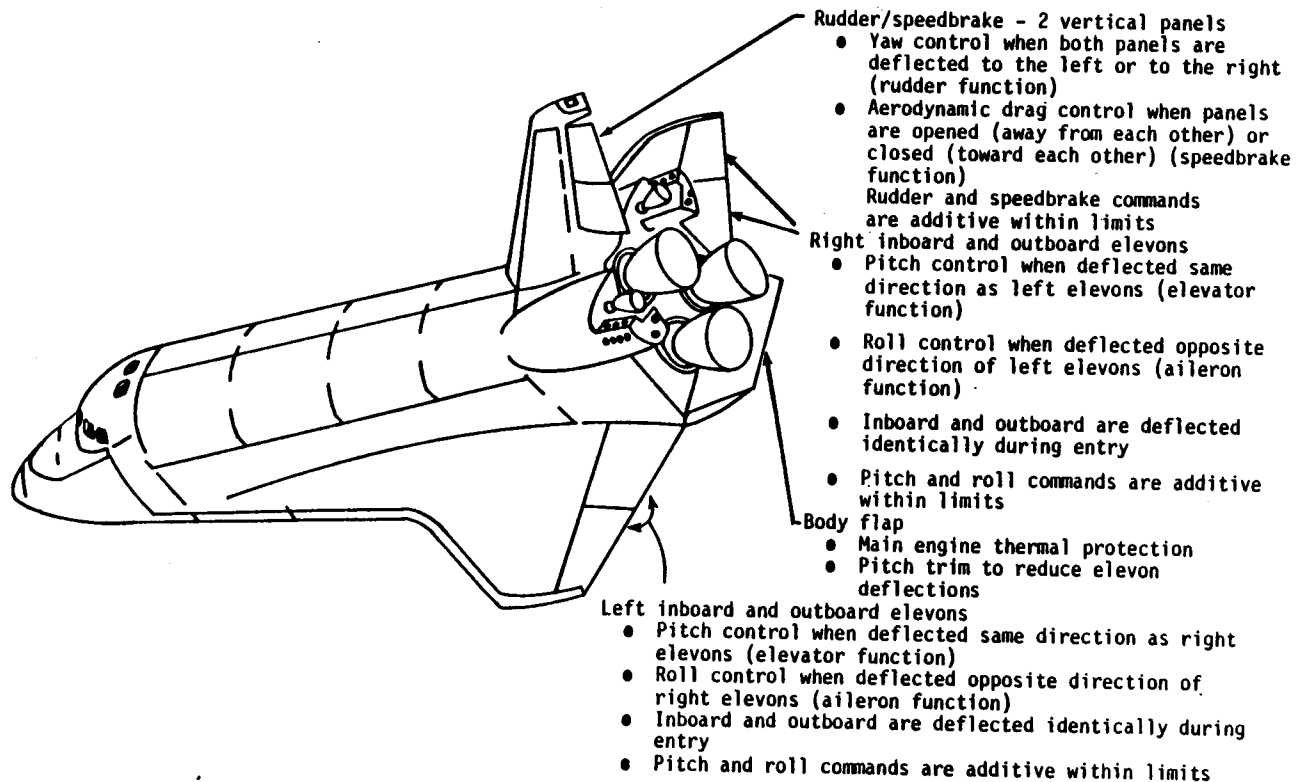


Figure 2-52.- Aerosurfaces and associated control axes.

Each aerosurface is driven by a hydraulic device (actuator) which is controlled by a redundant set of electrically driven valves (ports). There are four of these valves for each aerosurface actuator, except the body flap which has only three. These valves are controlled by electronic devices called aerosurface servo amplifiers (ASA). There are four Orbiter ASAs located in the aft avionics bay. Each ASA commands one valve for each aerosurface. ASA 4 does not command the body flap since it has only three valves. This gives command redundancy to each aerosurface.

In addition to the command channels from the ASAs to the control valves, there are data feedback channels to the ASAs from the aerosurface actuators. These feedback channels include:

1. Position feedback signals. Each aerosurface has four associated position feedback transducers and each transducer sends a position feedback signal to an ASA.
2. Pressure feedback signals. Each of the rudder, speedbrake, and elevons has four associated secondary pressure transducers and the four elevons also have four associated primary pressure transducers. Each pressure transducer sends a feedback signal to an ASA which is used to determine the performance of each valve. If a valve gets out of limits an isolation (bypass) command is issued which removes that valve from any actuator control.

The path from an ASA to its associated control valves and from the aerosurface feedback transducers to the ASA is called a flight control channel. Thus, there are four flight control channels with the ASAs being the control hardware component of each channel. Although the ASAs have automatic control over the isolation of a valve in the case of a failure, the crew may manually force all six valves in a flight control channel out of isolation by taking the FCS channel switch on panel C3 to OVERRIDE.

Each ASA is hardwired to a flight aft MDM. Flight control commands originate from manual inputs to the controllers or from guidance software. These inputs go to the flight control software where they are augmented and then routed to the ASAs.

There are several SOPs associated with the ASA commands and data. For purposes of this workbook they are not delineated by the various software modules, but are described in terms of their functions, which include the following:

1. Converts elevon, rudder, and speedbrake commands from flight control software from degrees to millivolts
2. Sets commands to body flap valves based on an enable command from body flap RM and up/down commands from flight control
3. Converts position feedbacks to degrees for the elevons, rudder, speedbrake, and body flap
4. Computes elevator position from elevon position feedbacks
5. Calculates body flap and speedbrake deflections as percentages and calculates elevon and rudder positions for display on the surface position indicator (SPI)
6. Monitors the flight control system (FCS) channel switches and, if any is positioned in override, sets the override command for that ASA
7. Monitors hydraulic system pressures for failures and rate limits aileron and elevator commands according to the number of failures

Figure 2-53 shows the power and data interfaces of the ASAs. Notice that each ASA contains two power sources. In the event that an ASA loses power, there remains power to bypass its channel.

Figure 2-54 shows the general flow of aerosurface commands and data through the software.

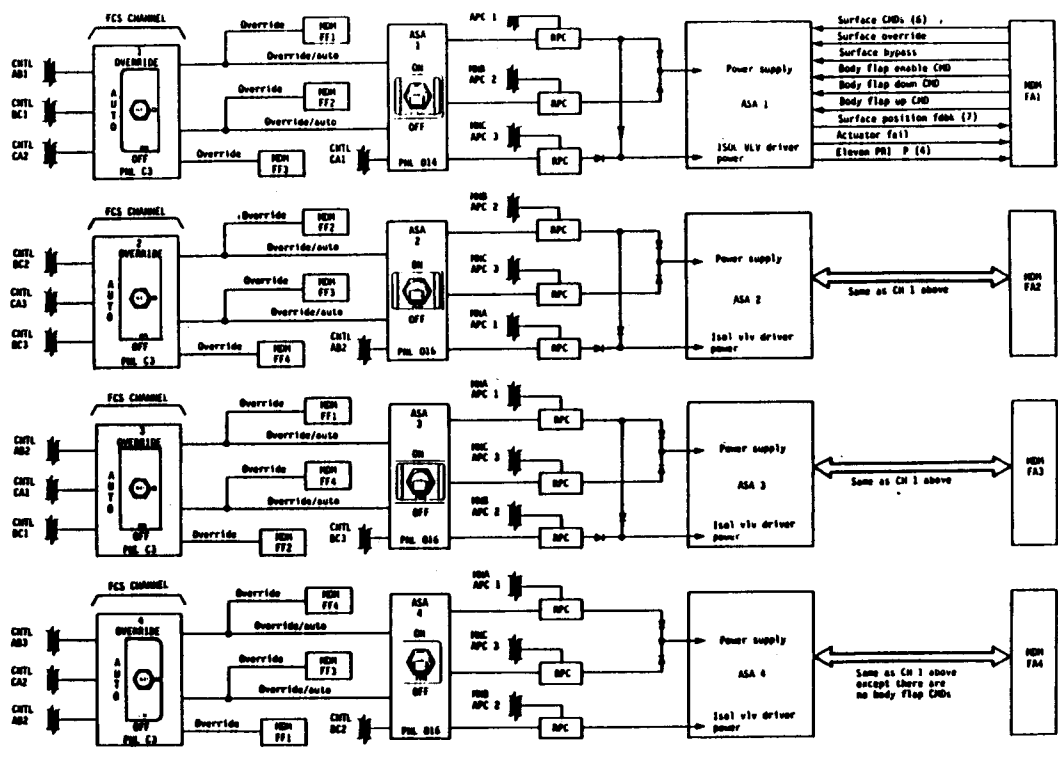


Figure 2-53.- Power and data interfaces of ASAs.

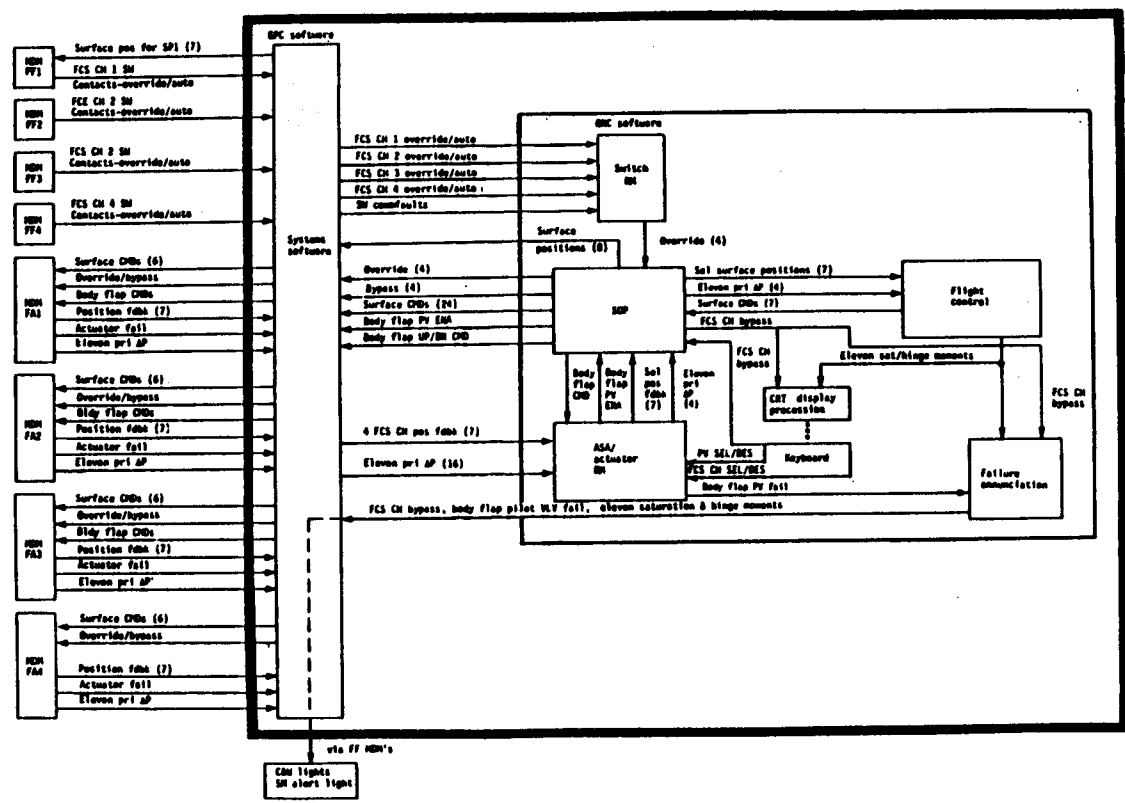


Figure 2-54.- Flow of aerosurface commands and data through software.

SECTION 3
REDUNDANCY MANAGEMENT

3.1 FUNCTIONS OF REDUNDANCY MANAGEMENT

The two functions of RM are repeated here. The figure on the following page depicts their relationship.

1. Selection filter. The selection filter selects, from redundant sets of hardware data, one set of data for use by flight control, guidance, and navigation.
2. Fault detection, identification, and reconfiguration (FDIR). The FDIR detects data which is out of tolerance, identifies the faulty unit, and may make the data from the faulty unit unavailable to the selection filter.

The selection filter and FDIR processes are of two types depending on the type of data they operate on:

1. Analog. In among processes the parameter may take on a range of values. Radar altitude which may be 0 feet to 5000 feet or anything in between is an example of this type.
2. Discrete. "Discrete" refers to switch position contacts. They are either open, represented by 0, or closed, represented by 1.

The first part of this section will discuss redundancy management for analog systems using hypothetical LRUs. This is followed by a discussion of the RM for the Orbiter GNC analog LRUs, which have many individual peculiarities. Finally, redundancy management for discrete signals is discussed, with a brief discussion of some of the switches for which this type of processing is done.

Figure 3-1 depicts RM inputs and outputs for a general case of three identical LRUs supplying two parameters (P_1 and P_2). Even this general diagram is not valid for all cases, since some systems have the LRU SOP processing after the RM processing.

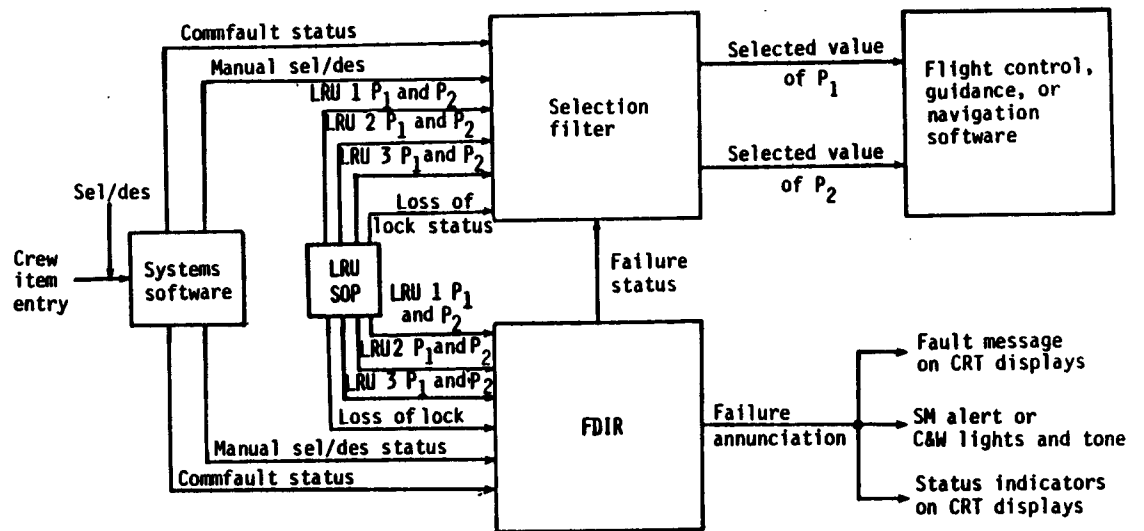


Figure 3-1.- General case RM inputs and outputs.

3.2 ANALOG DATA SELECTION FILTER

The purpose of selection filter processing is to choose, from redundant sets of hardware data, one set for use by guidance, navigation, and flight control. The selection process depends on the number of LRUs. For the sake of this section, suppose there is a navigation line replaceable unit (LRU) which measures Orbiter position with three parameters: latitude (LAT), longitude (LON), and altitude (ALT). A subscript indicates the number of the LRU. The number in this example, as well as in the real orbiter, signifies the string it is on. The letter S is used as subscript to indicate the selected parameter.

3.2.1 One-LRU Case. The selection process is simple. LAT₁, LON₁, and ALT₁ are selected. This is called the "pass" mode as the selection filter passes through the only data it receives (figure 3-2).

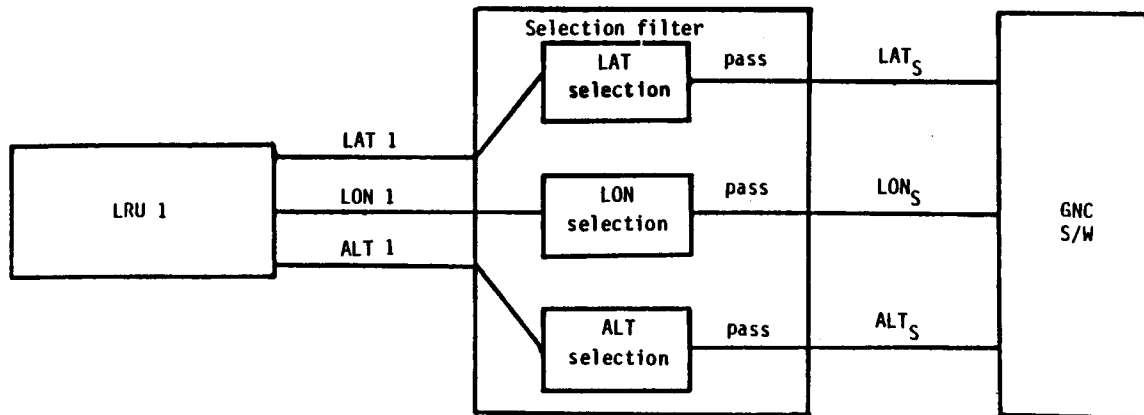


Figure 3-2.- Pass selection mode.

3.2.2 Two-LRU Case. The selected values are the average of the two parameter values.

$$LAT_S = \frac{LAT_1 + LAT_2}{2}$$

$$LON_S = \frac{LON_1 + LON_2}{2}$$

$$ALT_S = \frac{ALT_1 + ALT_2}{2}$$

This is called the "average" mode. Note that unless both of the parameters in a pair are identical, the data used by GNC is not what was sensed by the LRUs, but something in between the two.

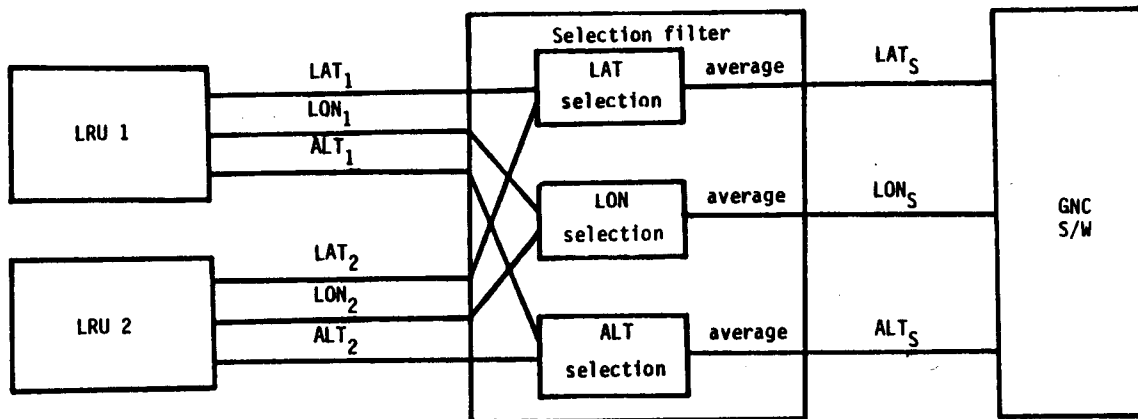


Figure 3-3.- Average selection mode.

3.2.3 Three-LRU Case. The selected values are the middle values of the three parameter values.

$$\begin{aligned} \text{LAT}_S &= \text{MIDVAL}(\text{LAT}_1, \text{LAT}_2, \text{LAT}_3) & \text{LON}_S &= \text{MIDVAL}(\text{LON}_1, \text{LON}_2, \text{LON}_3) \\ \text{ALT}_S &= \text{MIDVAL}(\text{ALT}_1, \text{ALT}_2, \text{ALT}_3) \end{aligned}$$

This is called the middle value select mode or MVS (figure 3-4). Suppose the three LRUs sense these values:

LRU 1: LAT₁ = 29°50'N LON₁ = 95°10'W ALT₁ = 25 ft.
 LRU 2: LAT₂ = 29°49'N LON₂ = 95°12'W ALT₂ = 24 ft.
 LRU 3: LAT₃ = 29°48'N LON₃ = 95°11'W ALT₃ = 26 ft.

For this example, LAT₂, LON₃, and ALT₁ are the middle values. These values would be used by GNC software. The fact that they as a group do not come from one identifiable LRU (i.e., do not constitute a homogeneous data set) is not of consequence in the general case.

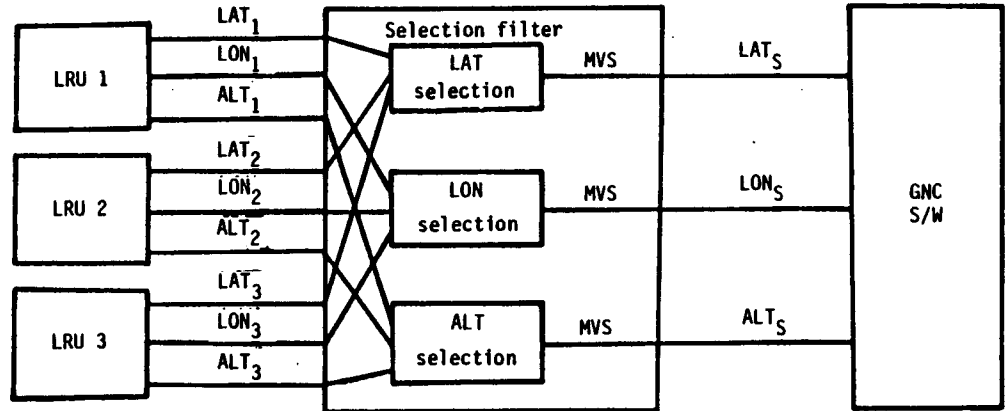


Figure 3-4.- MVS selection mode.

3.2.4 Four-LRU Case. The selection filter for the four-LRU systems use a quad-midvalue select (QMVS) scheme consisting of five MVS filters for each parameter as shown in figure 3-6. The QMVS logic will select the signal having either the second- or third-highest absolute value. The specific selection is based on the difference between these two signals. The purpose of the I-loaded constant is protection from selecting a fail-to-null signal.

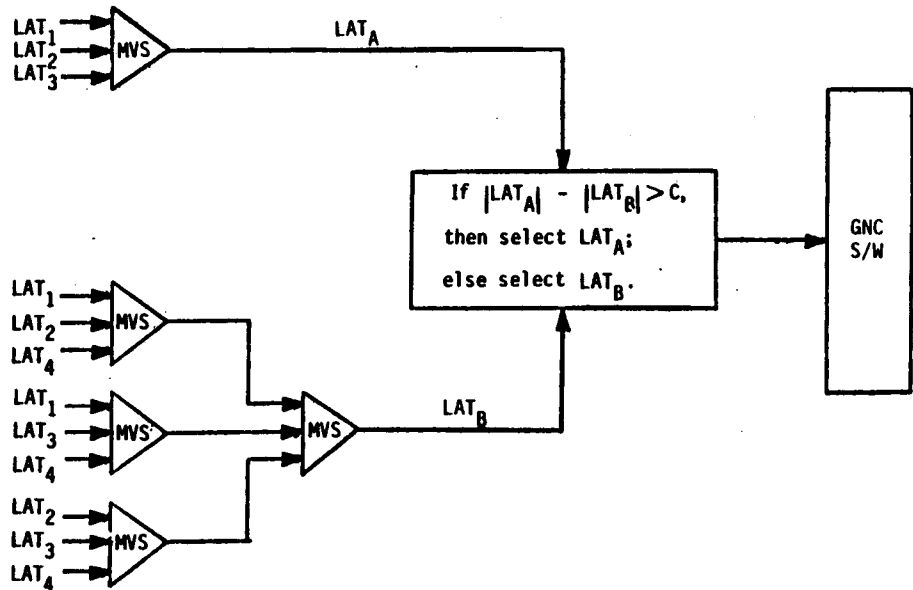


Figure 3-5.- QMVS selection mode.

3.2.5 Candidates. Up to now it has been assumed that the LRUs whose data are being selected are good, i.e.; present and accurate. Actually, the selection filters operate only on data from LRUs that are candidates.

To be a selection filter candidate, an LRU must

1. Not have a commfault flag set
2. Not have loss of lock if it is a radio frequency LRU
3. Not be failed
4. Not be manually deselected by the crew

Each of these conditions will set a bit which disqualifies the LRU as a selection filter candidate:

1. Commfault is set by systems software (FCOS) as described earlier.
2. Loss of lock is recognized by the LRU itself which sets the bit.
3. Failures are detected by the other part of RM-FDIR and their detection will be described shortly.
4. Manual deselection is accomplished by using CRT display item numbers.

If any of the four conditions is not met, the selection filter downmodes. That is, if there are three LRUs and the selection filter has been using the three-LRU case, it will downmode to the two-LRU case. It will use the two-LRU case selection filter on the data from the two candidate LRUs. All parameters from the noncandidate LRU will be disregarded.

Selection filter downmoding is said to be "on an LRU basis". This means that the selection filter level is the same for all parameters of an LRU. If, in the three-LRU case, ALT₂ is declared failed, the selection process for all three parameters - LAT, LON, ALT - downmodes to the two-LRU or average case using data from LRU 1 and LRU 3.

Selection filter upmoding occurs when

1. Loss of lock no longer exists.
2. The commfault flag is reset by FCOS after a systems software reconfiguration.
3. The crew manually re-selects previously failed or deselected LRU.

3.2.6 No-LRU Case. It is possible to downmode a selection filter successively until it has no candidates. This is called the no-available-sensor mode (NAS). In this case it continues to output the last values it output when it had a candidate.

A software flag called the "data good" flag is set in the selection filter to warn users of data when data is suspect. This flag is useful only in the no- and two-LRU cases. The flag is always set valid for four-, three-, or one-LRU processing. It is always set invalid for the no-LRU case to indicate data is bad or questionable.

In the two-LRU case, a tracking test is performed on each parameter. The difference between the two values is compared to a limit. If the difference exceeds the limit the data good flag is set invalid.

If $|LAT_1 - LAT_2| \geq LIMIT_{LAT}$, $DGLAT = 1$

If $|LON_1 - LON_2| \geq LIMIT_{LON}$, $DGLON = 1$

If $|ALT_1 - ALT_2| \geq LIMIT_{ALT}$, $DGALT = 1$

When the data good flag is set, it is up to the user of the data - guidance, navigation, or flight control - to look at the flag and not use the data if the flag is set invalid. Not every LRU has a data good flag and not every user looks at it.

The data good flag is dynamic. It is set or reset whenever the tracking test outcome changes. It provides protection while FDIR is isolating a failure, when FDIR cannot isolate a failure, and when the bad data are transient.

3.3.5 Candidates. To be a candidate for FDIR a parameter must

1. Not have a commfault
2. Not have loss of lock
3. Not be failed (as declared by FDIR earlier)
4. Not be deselected manually (via CRT display item numbers)

Implications of the above are that FDIR downmodes itself for any of these conditions. This means that the LRU cannot be declared failed and it cannot be a selection filter candidate. Additionally, FDIR downmodes itself if it declares a parameter failed. It usually downmodes on an LRU basis; however, some systems downmode on a parameter basis. Downmoding on a parameter basis allows FDIR to use the good parameters from a partially faulty LRU in comparisons with parameters from the other LRUs. Neither FDIR nor the selection filter downmodes for a dilemma.

Some final points about analog data RM:

1. If in dilemma, FDIR continues making the comparison and checking BITE to try to resolve the dilemma. The dilemma is reset and the system downmoded if one of the parameters is declared failed or one of the parameters has a commfault or loss of lock. The dilemma is not reset if the comparison subsequently becomes good. The data good flag generated in the selection filter changes dynamically with the tracking test outcome. A transient failure causing a dilemma could be recovered from by deselecting and reselecting one of the LRUs.
2. The limit used in the comparisons may be a constant or it may be computed.
3. The foregoing discussion was based on the general case, which does not hold for any particular type of GNC hardware. Differences are mentioned in the following hardware/software descriptions.

3.4 GNC HARDWARE RM AND FDIR

Most of the GNC hardware devices have redundancy levels between 2 and 4, and most of them have something unique associated with their RM and FDIR schemes. This section gives a brief discussion of those schemes for each type of hardware.

3.4.1 IMU. IMU RM is somewhat nonstandard. Attitude and velocity RM are performed separately, although an IMU failure declared by one results in downmoding of both on an LRU basis. The following is a summary of the schemes.

1. Velocity. Candidates are disqualified for RM failure, commfault, manual deselection, or IMU slewing.
 - A. Selection filter. Selection is made from transformed M50 accumulated velocity components. The selection of these components follows the standard 3-LRU scheme.
 - B. FDIR
 - 3 Level. The velocity component changes over the last cycle are transformed back from M50 to cluster coordinates of one IMU (i.e., skew is removed). Pairwise differences are formed and the differences for three components of ΔV for 2 LRUs are called an error vector. The error vector magnitudes are squared and compared to a threshold tolerance. The threshold is a constant dependent on the major mode. If two of the error vector magnitudes exceed the tolerance, the IMU common to them is declared failed.
 - 2 Level. Same as 3 level except there is only one pairwise-differenced error vector. If its squared magnitude exceeds the threshold, BITE is used to try to resolve the dilemma. The IMU with BITE is declared failed. If neither BITE is set, parity equations are used to associate the error with an accelerometer in one of the IMU's. Because the IMU alignments are skewed with respect to each other, an accelerometer bias in an IMU appears along only one axis in the bad IMU but the bias is resolved into accelerations along all three axes in the other IMU. The IMU with errors along one axis only is declared failed. Note that this assumes single accelerometer failures are most probable. If BITE is set on both IMUs FDIR does not try to resolve the dilemma.
 - 1 Level. In OPS 2 or 8 the IMU will downmode to the zero level for BITE.
 - 0 Level. None

2. Attitude. Candidates are disqualified for RM failure, commfault, manual deselection, or IMU slewing.

A. Selection filter. A homogeneous data set is selected. That is, an IMU is selected and all three gimbal angles and the identification (ID) of that IMU are output.

- 3 Level. The theoretical skewed relationship and actual skewed relationship between two IMUs are used to calculate a relative misalignment error vector which is squared. The IMU not a contributor to the largest squared relative misalignment error vector is selected.
- 2 Level. The previously-selected IMU is selected if it is one of the two candidates. If it is not, the lowest-numbered of the two candidate IMUs is selected. The squared relative misalignment error vector is compared to a computed threshold. If it is greater, the data good flag is set bad.
- 0 and 1 Levels. Standard.

B. FDIR

- 3 Level. Each of the three squared relative misalignment error vectors is compared to a threshold. If two of them exceed the threshold, the IMU common to the two is declared failed.
- 2 Level. The squared relative misalignment error vector is compared to a threshold. If it exceeds the threshold, then BITE is used to try to resolve the dilemma. If BITE does not resolve the dilemma, parity equations are used to associate the error vector with a gyro in one of the IMUs. The IMU with misalignment in one axis only is declared failed, taking advantage of the skewed alignments.
- 1 Level. In OPS 2 and 8 the IMU will downmode to the zero level for BITE.
- 0 Level. None.

3.4.2 Star Tracker. There is no RM associated with the star trackers, since they point different directions and cannot be tracking the same star at the same time.

3.4.3 COAS. There is only one COAS and thus no RM.

3.4.4 TACAN. The RM scheme assumes at least two TACANs are selected to the same channel. RM may request a channel change in the GPC mode based on range loss of lock. Downmoding is on a parameter basis. Loss of lock causes temporary downmoding.

1. Selection filter. Standard for the case of three LRUs. Bearing is inhibited if the Orbiter is in the cone of confusion.

2. FDIR. Standard. Self test is requested by FDIR to try to resolve a dilemma only if in the GPC mode. If one TACAN fails self test it is declared failed. If neither fails self test the dilemma is unresolved. During self test (18 seconds) the last valid value is output by the selection filter.

3.4.5 Air Data System

1. RM Candidates. Standard, plus the additional criterion that a probe must be deployed before either of its associated ADTAs can be considered selection filter or FDIR candidates. Refer to figure 3-6 showing the ADTA RM scheme while reading the following material.

2. Selection Filter. There are 3 selection filters associated with the air data system: a left selection filter, a right selection filter, and a center selection filter. The left filter uses data from ADTAs 1 and 3 which operate from the left probe, and the right filter uses data from ADTAs 2 and 4, which operate from the right probe. The center filter uses the outputs from the left and right filters. All three filters use standard 2-level RM selection.

3. FDIR. ADTA FDIR has seven comparators shown in the RM diagram as C₁ through C₇. C₁, C₂, and C₃ comparisons are made continuously as long as they have two inputs. Any miscompare exceeding threshold limits for these three comparators results in the data good flag set bad and three consecutive miscompares results in a dilemma. FDIR cannot resolve dilemmas at C₃ so crew action is required. FDIR attempts to resolve dilemmas at C₁ by using comparators C₄ and C₅. Miscompares at C₄ fail ADTA 1 and miscompares at C₅ fail ADTA 3. Comparators C₆ and C₇ are similarly used by C₂ to resolve dilemmas.

Comparator thresholds for C_1 and C_2 are different constants for each pressure measurement. Comparator thresholds for $C_3, C_4, C_5, C_6,$ and C_7 are computed to be a percent of the incoming pressure data. This percent varies with Mach number and sideslip.

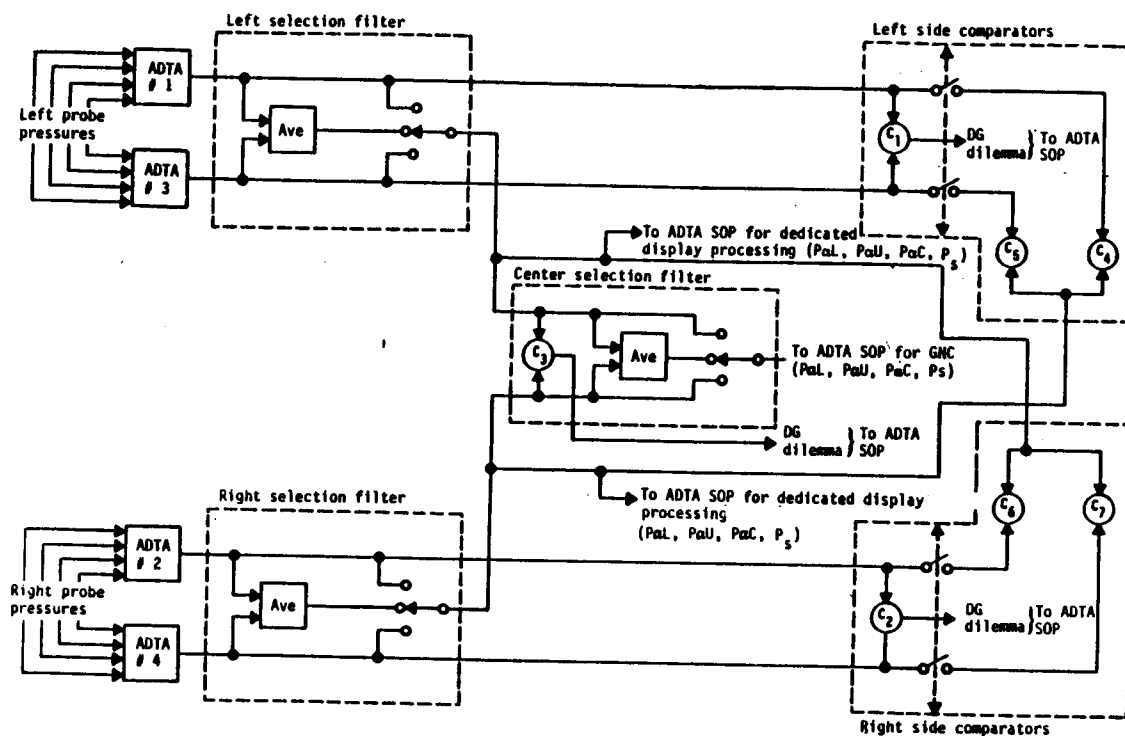


Figure 3-6.- ADTA RM.

3.4.6 MLS. The MLS uses standard three-LRU RM and FDIR schemes.

3.4.7 RA. The radar altimeter RM processing is available but presently not used. This available software is only a selection filter for navigation purposes and does not include any FDIR.

3.4.8 RR. There is only one rendezvous radar and thus no RM.

3.4.9 AA and RGA

1. Selection Filter. Standard QMVS scheme.

2. FDIR. Standard 4-level FDIR scheme for four LRUs. Downmoding is on an LRU basis.

3.4.10 SRB RGA

1. Selection Filter. The selection filter actually consists of two selection filters - one for the left side and one for the right side. The outputs of these two filters are averaged in flight control to give the selected pitch and yaw rates. The output of the left filter is used as a selection filter candidate by the right filter whenever one or more of the right LRUs become unusable. The output of the right filter is used in the same way by the left filter.

2. FDIR. FDIR does three-LRU processing on each side until one LRU becomes unusable. At that time FDIR terminates processing on that side and the selection filter uses MVS regardless of subsequent failures. The selection filters on each side will downmode further for commfaults.

3.4.11 Controllers

1. RHC. Standard for three-parameter case. Selection filter and FDIR processes are done on the three transducers of an RHC axis. For candidate criteria purposes, an LRU is the set of three like-numbered transducers. For example, the pitch channel 1, roll channel 1, and yaw channel 1 transducers of the left RHC constitute an LRU. Downmoding is on an LRU basis. Dilemmas are not resolved since there is no BITE. There is a data good flag.

2. THC. The THC RM is on switch contacts which is a discrete RM scheme discussed in subsequent sections.

3. RPTA. Standard for the three-LRU case where an LRU is a transducer. Dilemmas are not resolved since there is no BITE. There is a data good flag.

4. SBTC. Standard for the three-LRU case where an LRU is a transducer. Dilemmas are not resolved since there is no BITE. There is a data good flag.

5. Body Flap and Trim Switches. These are discrete RM systems which will be discussed in subsequent sections.

3.4.12 ASA/Actuators

1. Aerosurface position feedbacks and delta pressure feedbacks. Standard four-LRU QMVS selection filter with 3-level FDIR. Downmoding is on a channel basis.
2. Body flap position feedback. Standard four-LRU QMVS selection filter with 4-level FDIR.
3. Body flap command RM. A unique flight software scheme detects body flap motion different from that commanded by flight control, attempts to stop uncommanded movement, and alerts the crew to what the body flap is doing.

When BF CMD RM detects uncommanded body flap movement, it goes through the following three phases:

- Hold 2. BF CMD RM commands channel 2 to attempt to stop the uncommanded motion. If this command succeeds, it will continue to be commanded and an SM alert will be annunciated. If it does not succeed in a specified number of seconds, the command is removed and the scheme goes to HOLD 1.
- Hold 1. BF CMD RM commands channel 1 in the same way it did for Hold 2. If this is unsuccessful in a specified number of seconds, the command is removed and the scheme goes to CYCLE.
- Cycle. BF CMD RM annunciates an SM alert telling the crew that Hold 2 and Hold 1 were unsuccessful and that flight control is issuing commands to keep the body flap within the position deadband.

When BF CMD RM detects that the body flap is not moving in the direction commanded by flight control, it annunciates a body flap failure.

3.5 DISCRETE DATA SELECTION FILTER

The selection filter for discrete data has several differences from analog data selection. The method of selection is different, there is no loss of lock possible, there is no data good flag since there are no tracking tests, and downmoding is on a switch position (not entire switch) basis.

Certain features of the discrete data selection are similar to the concepts discussed for the analog data selection. These features include downmoding for commfault, failure or crew deselection, upmoding for software reinitialization of commfault flags, and selection from acceptable candidates.

Selection is made from candidate contacts for each position of the switch. To be a candidate a switch contact must

1. Not have a commfault flag set
2. Not be declared failed by FDIR
3. Not be deselected manually by the crew

The discrete output for a switch closed is 1 and for a switch open is 0. This discussion also use a 1 and 0 for contacts closed and open.

3.5.1 No-Contact Case. The discrete output is 0 since no contact is made.

3.5.2 One-Contact Case. The value of the only contact 1 or 0 is output.

3.5.3 Two-Contact Case. The selected value is the logical AND of the two contacts. In order for a 1 to be output, both contacts must be 1.

3.5.4 Three-Contact Case. The selected value is the majority vote of the three contacts. That is, if two or three of the contacts are made, a 1 is output; otherwise, 0 is output.

3 Contacts			Output
0	0	0	0
1	0	0	0
1	1	0	1
1	1	1	1

Suppose there is a switch with two positions, HIGH and LOW, and each position has three contacts. If the switch were placed in the HIGH position, the contacts and selection filter outputs would be as shown below.

Position	Contact			Selection	
	1	2	3	filter	output
HIGH	1	1	1		1
LOW	0	0	0		0

(using majority vote)

If contact 3 of the HIGH position had a commfault, the selected HIGH position would be determined by ANDing the two contacts and the selected LOW position by majority vote. Placing the switch in the LOW position would yield:

Position	Contact			Selection	
	1	2	3	filter	output
HIGH	0	0	COMM FAULT		0
LOW	1	1	1		1

(using AND)
(using majority vote)

If contact 1 of the HIGH position failed OPEN and the switch were placed in HIGH, the following would result.

Position	Contact			Selection	
	1	2	3	filter	output
HIGH	0	1	COMM FAULT		0
LOW	0	0	0		0

3.6 DISCRETE DATA FDIR

Candidates are contacts

1. Not failed

2. Not having commfault

3. Not manually deselected

3.6.1 No-Contact or One-Contact Case. No FDIR.

3.6.2 Two-Contact Case. If the two contacts disagree, a dilemma is declared. Generally, there is no annunciation to the crew of this situation. FDIR cannot resolve the dilemma.

3.6.3 Three-Contact Case. If one contact disagrees with the other two, it is declared failed. This is implemented by comparing each contact value to the selected value and declaring failed the contact which is different. FDIR downmodes itself and the selection filter. There is no annunciation of the failure to the crew. Downmoding is on a switch position contact basis.

3.7 SWITCHES

There are a number of switches not associated with GNC hardware but which reconfigure GNC software. These switches are too numerous and their functions too diverse to describe individually in an overview. Similarly the power sources, number of contacts, and data paths cannot be generalized. Three types of software processing may be done on the switch positions and contacts: redundancy management, station selection, and default position processing. In some documentation all three types of processing are called RM.

3.7.1 Redundancy Management. RM for switches with redundant contacts follows the general scheme described in the previous section.

3.7.2 Station Selection. In some cases there are multiple switches which perform the same function in software. The only difference is the location and hence the crewmember who uses it. After the selection filter has selected a discrete for each position of the switch, the station selection process selects which of the redundant switch positions to send to the GNC software. The station selection process is a logical OR of the values; i.e., if any switch has that position selected, the switch position command is sent to GNC software.

Take as an example the ATT REF pushbutton (PB) used to take star sighting marks described in the COAS unit earlier. There are three ATT REF PBs: one at the commander's (CDR) station, one at the pilots (PLT) station, and one at the mission specialist's (MS) station. Assume the CDR takes a mark using his ATT REF PB and the others do not. The contact status and selection filter output is shown below.

	Contact 1	Contact 2	Selected contact	Station selection
CDR	1	1	1	
PLT	0	0	0	1
MS	0	0	0	

The logical OR of 1, 0, 0 is 1 so the COAS SOP would be notified that a mark had been taken.

3.7.3 Default Position Processing. Some switches do not have redundant contacts processed by software. For example, data from either radar altimeter may be displayed on an AVVI. This is done by positioning a switch to either 1 or 2. Software receives only one contact for each position. Because the switch must be in one of the positions and cannot be in both positions, one position should be 1 and the other 0 so software knows which radar altitude to display. Default position processing detects failures when both positions are 1 or both positions are 0. If the software detects a failure (including commfault) it uses the switch default position (also called green dot position) which for the RA case is 1 for the right AVVI and 2 for the left AVVI. The failure is annunciated with a fault message and SM alert. The message is DISPLAY SW X where X is L for left, R for right, or A for aft. The failure annunciation notifies the crew that the data displayed may not be that reflected by the switch position. The safe thing to do in this case is for the crew to place the switch in the green dot position. The same processing may be applied to three-position switches. The default position is used and the failure is annunciated if:

1. No switch position is received (three 0's).
2. More than one switch position is received (two or three 1's).

This processing is called GN&C SWITCH RM in documentation, but it is a misnomer, as there is no redundancy involved.

SECTION 4 SOFTWARE CONTROLS/DISPLAYS/FAILURE ANNUNCIATION

4.1 CRT DISPLAYS

This section contains the displays which have controls and information on them concerning the hardware discussed in section 2. This section give a brief overview discussion of the parts of the displays dealing with this hardware. Fault messages and nondisplay failure annunciations will be discussed later in section 4.

4.1.1 GNC SYS SUMM 1. (See figure 4-1.) The SURF section of this display shows the aerosurface positions (POS) and hinge moments (MOM). The positions are in degrees for all except the speedbrake and body flap which are in percent of total deflection. The hinge moment is in percent of maximum. The field below the BDY FLP POS is the body flap RM status; blank, HOLD 2, HOLD 1, CYCLE, or FAIL. The POS status column will display an | or | anytime the position of the aerosurface exceeds the software limits. The MOM status column displays a | anytime the hinge moment exceeds software limits.

The CNTLR, FCS CH, and NAV sections contain status indicators which will display an M for commfaults and an | for failures. A ? is also displayed for dilemmas in any of these positions except for the FCS CH, ACC, and RGA LRUs, which cannot be in dilemma.

4.1.3 IMU ALIGN. (Figure 4-3.) This display allows the crew to monitor and control the on-orbit IMU hardware and software modes, including alignments. The status of each IMU is in the upper left corner next to each number. The status can be M, I, ? or blank. The STAT row can be blank or can display BITE. The TEMP row displays OK, LO, or HI for the temperature status of each IMU. The STBY row provides controls to select the standby mode of operation. The OPER row allows selection of the operation mode. The DES row allows for selection or deselection of each IMU. The acceleration of each IMU in M50 coordinates is displayed in the ACC section, and attitude information is displayed in the ANG section. The align section of this display is used to perform different types of IMU alignments.

```

XXXX/021/      IMU ALIGN      XX X 000/HH:MM:SS
IMU              000/HH:MM:SS
  1S   2S   3S
STAT XXXX XXXX XXXX      ALIGN XXX
TEMP  XX  XX  XX      IMU  1 10X  STAR ALIGN 13X
STBY  1X  2X  3X      2 11X  14 IMU/IMU  I
OPER  4X  5X  6X      3 12X  MATRIX    15X
DES   7X  8X  9X

                                EXEC 16X
                                TERM 17

ACC
  1      2      3
X  XX.XX  XX.XX  XX.XX
V  XX.XX  XX.XX  XX.XX
Z  XX.XX  XX.XX  XX.XX

ANG
  1      2      3
X  XXX.XX  XXX.XX  XXX.XX
V  XXX.XX  XXX.XX  XXX.XX
Z  XXX.XX  XXX.XX  XXX.XX
AX  XXX.XX  XXX.XX  XXX.XX
AY  XXX.XX  XXX.XX  XXX.XX
AZ  XXX.XX  XXX.XX  XXX.XX

                                NAV AV THRESH
                                10 XXXXXX
                                MM READ 10X

                                (XX)

```

Figure 4-3.- IMU Align.

4.1.4 S TRK/COAS CNTL. (Figure 4-4.) This display provides the crew a means of monitoring and controlling star tracker and COAS operations. The S TRK CNTL section allows the crew to control the mode of operation of the star trackers. Information about stars tracked is displayed in the S TABLE. The S TRK area allows you to select specific stars to track and displays information about stars as they are being tracked. Shutter status and control are also in this area. The COAS section allows control while using the COAS to sight stars.

```

XXXX/022/      S TRK/COAS CNTL  XX X 000/HH:MM:SS
                                000/HH:MM:SS
S TRK CNTL -V -Z      S TABLE  1  2  3
SELF-TEST  1X  2X      TRK ID   XX  XX  XX
STAR TRK   3X  4X      A MIN    XX  XX  XX
TGT TRK    5X  6X      ANG DIF  XXX.X XXX.X XXX.X
BREAK TRK  7  8      ERR      X.XX X.XX X.XX
TERM/IDLE  9X 10X     SEL       17X 18X 19X
                                S TABLE CLR 20

S TRK      -V      -Z      COAS
REQD ID    11  12  13  14  15  16  17  18  19  20
TRK ID     21  22  23  24  25  26  27  28  29  30
S PRES     31  32  33  34  35  36  37  38  39  40
AANG       41  42  43  44  45  46  47  48  49  50
THOLD      51  52  53  54  55  56  57  58  59  60
SHUTTER    61  62  63  64  65  66  67  68  69  70
MAB OP     71  72  73  74  75  76  77  78  79  80
STATUS     81  82  83  84  85  86  87  88  89  90

                                REQD ID    21  22  23  24  25  26  27  28  29  30
                                ADEC X     XX.X
                                V           XX.X
                                SIGHT MODE  22X
                                ACCEPT      23
                                CAL MODE    24X
                                DES          25X
                                POS  +X 26X -Z 27X
                                ARIAS  X.XX  X.XX
                                UPDATE  28  29

                                (XX)

```

Figure 4-4.- S TRK/COAS CNTL.

4.1.5 HORIZ SIT. (Figure 4-5.) The bottom table on this display provides information and control of ADTA and TACAN data used for navigation. The right-hand side of this table provides for each TACAN channel, range, and bearing. The TACANs can be individual deselected or reselected and the status of each parameter is shown as blank, M, |, or ?.

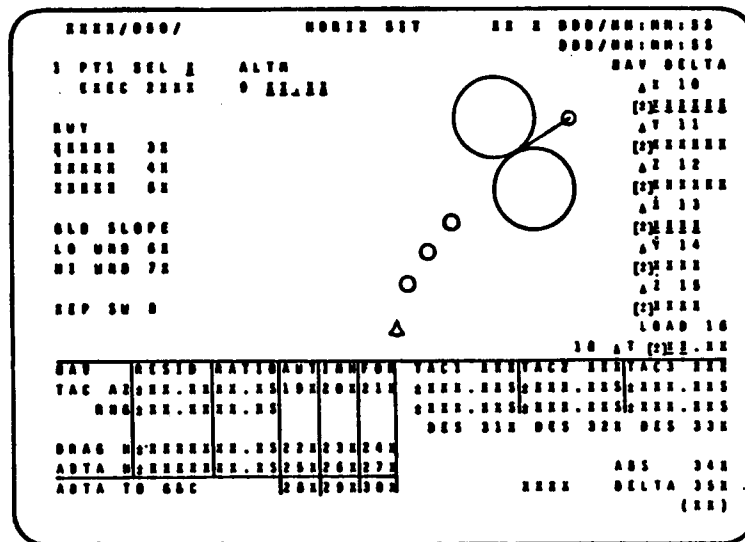


Figure 4-5.- HORIZ SIT.

4.1.6 RM ORBIT. (Figure 4-6.) This display provides the crew the means to monitor and control the selection filters for the translational and rotational hand controllers. The THC section shows transducer-sensed commands by a + or - and the status of each transducer by an M, |, ?, or blank. The THC channels can also be deselected or reselected.

The RHC section shows the RHC transducer sensed commands as a percent of full deflection. This percent is followed by an R or L for roll and yaw commands and a D or U for pitch commands. The status of each transducer is also shown with an M, |, ?, or blank. The RHC channels can also be deselected or reselected.

The last item on this display provides the crew with the capability to manually inhibit or enable GNC switch redundancy management. When inhibited, switch FDIR is prevented from declaring failed or removing GNC switch contacts (including THC) from the selection filters.

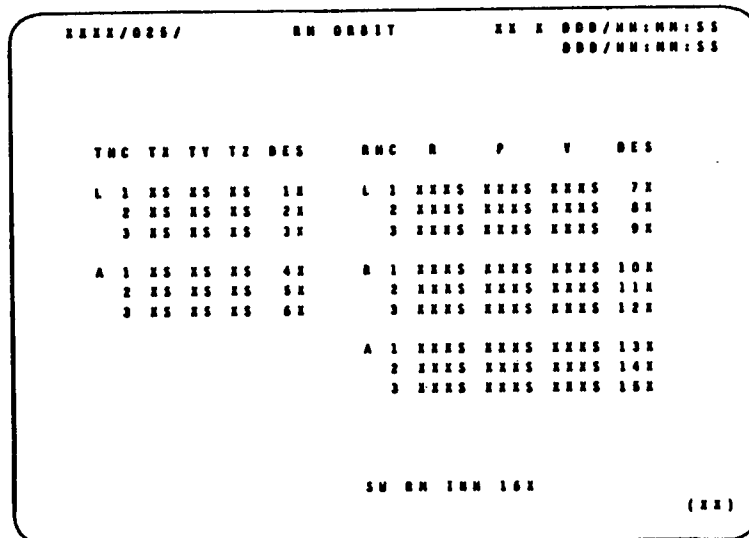


Figure 4-6.- RM Orbit.

4.2 ON-ORBIT FCS CHECKOUT

Toward the end of a flight but prior to deorbit, the Shuttle crews follow a checklist procedure to check out the flight control system (FCS). This procedure includes checking out all the GNC hardware discussed in this workbook. This checkout is done in OPS 8 which has a unique set of displays for that purpose.

The FCS checkout procedure is contained in the Orbit Operations Checklist of the Flight Data File. The FCS checkout procedure is self-explanatory and easy to follow. A brief description of each procedure follows.

4.2.1 Sensor Test. Refer to figure 4-7 for the sensor test display.

The sensor test for MLS, TAC, RA, and AA first tells you what to do in case of a failure during the test and tells you what equipment to configure before the test starts. Then, on Spec 40, you check for proper status before the test starts. You configure for free drift and check the AA limits. Next, you start the self-test with an ITEM 13 EXEC. The test works by presenting pre-set values to the equipment and then displaying the measured output. You compare the output to the values in the limit table to determine if there is a failure. The test is complete when all asterisks on the display blank. Item 14 is then executed to stop the test.

```

XXXX/040/          SENSOR TEST  XX X 000/HH:MM:SS
                                000/HH:MM:SS

      DES STAT  RRG      AZ      EL
MLS 1  1X XXXX  XX.XS  2X.XXS  X.XXS
      2  2X XXXX  XX.XS  2X.XXS  X.XXS
      3  3X XXXX  XX.XS  2X.XXS  X.XXS

      DES STAT  RRG      AZ
TAC 1  4X XXXX  X.XS  XXX.XXS
      2  5X XXXX  X.XS  XXX.XXS
      3  6X XXXX  X.XS  XXX.XXS

      DES STAT  ALT
RA   1  7X XXXX  XXXXS
      2  8X XXXX  XXXXS

      DES          V          Z
AA  1  9X          2XX.XS  2XX.XS
      2 10X          2XX.XS  2XX.XS
      3 11X          2XX.XS  2XX.XS
      4 12X          2XX.XS  2XX.XS

          START 13X  STOP 14X

                                (XX)

```

Figure 4-7.- Sensor test display.

4.2.2 RGA/ADTA/RCS. Refer to figure 4-8 for the RGA/ADTA/RCS display.

The format for the RGA, ADTA sensor test is similar to the previous test. After checking for proper starting status and configuring for free drift, you perform the RGA high and low test and the ADTA high and low test with an ITEM 9 EXEC or ITEM 10 EXEC, respectively. Each test takes about 10 seconds. If you get an ADTA BITE during the test, there follows a procedure to find out the exact failure. Finally there is a table to check the test values. Once the test is complete, you do an ITEM 11 EXEC to stop the test.

```

XXXX/041/          RGA/ADTA/RCS  XX X 000/HH:MM:SS
                                000/HH:MM:SS

      DES STAT      R          P          V
R1  1X XXXX  2XX.XS  2XX.XS  2XX.XS
R2  2X XXXX  2XX.XS  2XX.XS  2XX.XS
R3  3X XXXX  2XX.XS  2XX.XS  2XX.XS
      4  4X XXXX  2XX.XS  2XX.XS  2XX.XS

      DES STAT      PS          PaC          PaU          PaL          T
A1  5X XXXX  XX.XXS  XX.XXS  XX.XXS  XX.XXS  XX.XXS
R3  6X XXXX  XX.XXS  XX.XXS  XX.XXS  XX.XXS  XX.XXS
T2  7X XXXX  XX.XXS  XX.XXS  XX.XXS  XX.XXS  XX.XXS
A4  8X XXXX  XX.XXS  XX.XXS  XX.XXS  XX.XXS  XX.XXS

TEST  N1 9X  LOW 10X  STOP 11X

                                RCS DRIVERS

      RJD7  12X  TEST EXEC 16X  TEST XXXX
      RJD8-A 13X  CONTINUE 16X  JET  XXX
      -B 14X  STOP 17

                                (XX)

```

Figure 4-8.- RGA/ADTA/RCS display.

4.2.3 Switch/Surface. Refer to figure 4-9 for surface switch test body flap display.

In the FCS/MODE/CH switch test, all positions of the FCS channel and mode switches are tested and the results are viewed on SPEC 42. In addition, the speedbrake and pushbuttons on the eyebrow panels are tested. Failed open contacts make the switch unusable. Failed closed contacts are deselected by executing items 7 through 18 as appropriate. The surface test consists of verifying that no single feedback exceeds the average of the other three feedbacks by more than specified limits and then deselection any feedback that does with items 19 through 22 as appropriate.

XXXX/042/		SWITCH/SURF						XX X 000/HH:MM:SS		000/HH:MM:SS	
		ORIDE	DES		ORIDE	DES					
	1	XS	1X	3	XS	7X					
	2	XS	2X	2	XS	8X					
FCS	3	XS	3X	3	XS	9X					
CH	2	XS	4X	4	XS	10X					
	1	XS	5X	2	XS	11X					
	3	XS	6X	3	XS	12X					
		P			R/Y			SPD BRK	DDY FLP	DES	
		AUT	CSS	GAIN	AUT	CSS	GAIN	AUT	MAN	AUT	
FCS	L	XS	XS	XS	XS	XS	XS	XS	XS	XS	
	2	XS	XS	XS	XS	XS	XS	XS	XS	XS	
	3	XS	XS	XS	XS	XS	XS	XS	XS	XS	
MODE	R	XS	XS	XS	XS	XS	XS	XS	XS	XS	
	1	XS	XS	XS	XS	XS	XS	XS	XS	XS	
	2	XS	XS	XS	XS	XS	XS	XS	XS	XS	
	3	XS	XS	XS	XS	XS	XS	XS	XS	XS	
		L 00	L 10	R 10	R 00	RUD	DDY	FLP			
SURF	1	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	
	2	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	
	3	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	
	4	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	XXX.XS	

Figure 4-9.- Switch surface display.

4.2.4 Controllers. Refer to figure 4-10 for the controller and switch test display.

The controller and switch test involves moving the different controllers and switches and then checking SPEC 43 for failed contacts and proper deflection angles. The THC and RHC contacts are first deselected to prevent jets from firing when you move the controller. There are specific rules for deselection failed contacts depending on the number of contacts failed and the specific controller.

XXXX/043/		CONTROLLERS						XX X 000/HH:MM:SS		000/HH:MM:SS	
		THC	Y	Z	DES						
	1	XS	XS	XS	1X						
	2	XS	XS	XS	2X						
THC	3	XS	XS	XS	3X						
	1	XS	XS	XS	4X						
	2	XS	XS	XS	5X						
	3	XS	XS	XS	6X						
		SPD			DDY FLP						
		L 1			R 1						
		L 2			R 2						
		L 3			R 3						
		L 4			R 4						
		L 5			R 5						
		L 6			R 6						
		L 7			R 7						
		L 8			R 8						
		L 9			R 9						
		L 10			R 10						
		L 11			R 11						
		L 12			R 12						
		L 13			R 13						
		L 14			R 14						
		L 15			R 15						
		L 16			R 16						
		L 17			R 17						
		L 18			R 18						
		L 19			R 19						
		L 20			R 20						
		L 21			R 21						
		L 22			R 22						
		L 23			R 23						
		L 24			R 24						
		L 25			R 25						
		L 26			R 26						
		L 27			R 27						
		L 28			R 28						
		L 29			R 29						
		L 30			R 30						
		L 31			R 31						
		L 32			R 32						
		L 33			R 33						
		L 34			R 34						
		L 35			R 35						
		L 36			R 36						
		L 37			R 37						
		L 38			R 38						
		L 39			R 39						
		L 40			R 40						

Figure 4-10.- Controllers display.

4.3 FAILURE ANNUNCIATION

Whenever RM declares a failure, it is annunciated by a class 2 or a class 3 alarm. The indications of these alarms are as follows:

4.3.1 Class 3

1. SM alert light and tone
2. ↓ (failure) or ? (dilemma) on one or more CRT displays containing status information on the failed LRU
3. Fault message on any active CRT

4.3.2 Class 2

1. Master alarm lights and tone
2. ↓ (failure) or ? (dilemma) on one or more CRT displays containing status information on the failed LRU
3. Fault message on any active CRT
4. Backup caution and warning light on the C/W matrix on panel F7

The alarm class, fault message, and CRT containing status information for each LRU discussed in section 2 are given in table 4-1. Switches are not included.

TABLE 4-1. -ALARM CLASS SUMMARY

LRU	Failure	Alarm class	Fault message	CRT status information
IMU	Failure	3	RM FAIL IMU	GNC SYS SUMM 1 OVERRIDE IMU ALIGN
	Dilemma	2	RM DLMA IMU	
STAR TRACKER	Self-test	3	G22 STAR TRKR	S TRK/COAS CNTL
COAS	N/A			
TACAN	Failure	3	RM FAIL TAC	GNC SYS SUMM 1 HORIZ SIT
	Dilemma	3	RM DLMA TAC	
ADTA	Failure	3	RM FAIL ADTA	GNC SYS SUMM 1 OVERRIDE
	Dilemma	2	RM DLMA ADTA	
MLS	Failure	3	RM FAIL MLS	GNC SYS SUMM 1
	Dilemma	3	RM DLMA MLS	
RA	Failure	None	None	None
RR	Failure	None	None	None
AA	Failure	3	RM FAIL ACC	GNC SYS SUMM 1
RGA	Failure	3	RM FAIL RGA	GNC SYS SUMM 1
SRB RGA	Failure	None	None	None
RHC	Failure	3	RHC L (R, A)	GNC SYS SUMM 1 RM ORBIT
	Dilemma	2	RHC L (R, A)	
THC	Failure	3	SBTC/THC L (A)	GNC SYS SUMM 1 RM ORBIT
	Dilemma	3	SBTC/THC L (A)	
RPTA	Failure	None	None	None
	Dilemma	None	None	
SBTC	Failure	3	SBTC/THC L (R)	GNC SYS SUMM 1
	Dilemma	3	SBTC/THC L (R)	
FCS	Channel bypass	2	FCS CH 1 (2, 3, 4)	
	Elevon limits exceeded	2	FCS SAT POS	
	Hinge moments exceeded	2	FCS SAT MDM	
Body flap	Command failures	3	BDY FLP HOLD BDY FLP CYCL BDY FLP FAIL	GNC SYS SUMM 1

SECTION 5 BACKUP FLIGHT SYSTEM

5.1 INTRODUCTION

The Backup Flight System (BFS) is a software system designed to fly the Shuttle in case the Primary Avionics Software System (PASS) for some reason is unable to do so. The BFS does not use all of the sensors used by PASS, and the RM schemes are generally simpler. This section will briefly describe the differences between the PASS and BFS for each type of hardware. Note that the BFS does not have any FDIR. It downmodes for BITE or commfault in some cases. The BFS software does not support the star trackers, COAS, MLS, RA, or RR hardware.

The BFS software uses hardware interface programs (HIPs) to process the hardware data it receives. These software modules perform essentially the same processing done by the SOPs for the PASS.

5.2 IMU

The BFS assumes the IMUs are in the operate mode. Therefore, the BFS does not support any IMU initialization, alignment, or calibration. Calibration data and other IMU information are obtained from the PASS before flight, at transition to MM 301, and at other times when needed. Accelerometer and resolver data are processed by the BFS both before and after it is engaged.

Assuming all three IMUs are good, the BFS uses the same method as PASS to select velocity and attitude data. If a BITE or commfault is detected on one IMU, the BFS will downmode to the lowest numbered of the two remaining IMUs. It should be noted that the BFS does not detect PASS RM failed IMUs, so a PASS-failed IMU has to be deselected in the BFS or it will continue to be used. For a second BITE or commfault, the BFS will downmode to the remaining IMU.

The BFS does not upmode for recovered IMUs as the PASS does. Once a failure has occurred, the BFS selects one IMU for data, and it will not return to using the original three good IMU selection method. When IMUs are reselected, the BFS logic establishes an order for use in case of additional BITEs or commfaults. This logic scheme basically is that the last IMU failed will be the first one used should the currently used IMU fail.

For a failed IMU (BITE or commfault), the BFS will drive a SM light and tone on the BFS GNC SYS SUMM 1 CRT display, and the fault message RM FAIL IMU on a BFS CRT. Note that the BFS cannot have a dilemma. If all three IMUs are commfaulted or if BITE failed, the last available IMU will continue to be used.

5.3 TACAN

At least two TACAN channels must be locked on to the same station unless two of the three are commfaulted. The range and bearing data is processed separately and a midvalue selection scheme is used at the 3 level. At the 2 level the lowest numbered good unit is used and at the 1 level the only good unit is used.

The BFS will process TACAN data both before and after engage. A commfault will give a class 3 alarm, | on the GNC SYS SUMM 1 display, and the fault message RM FAIL TAC. The TACAN can be deselected or reselected on the BFS HORIZ SIT CRT display.

5.4 ADTA

The BFS processes data from both probes if both are deployed. Data is used from the lowest numbered ADTA considered good on each side. If data are available from both probes, the data are averaged. If a commfault or BITE is detected on the ADTA being used, the BFS defaults to the remaining ADTA on that side. If both ADTAs are failed on a side, processing continues using a single ADTA from the good side.

The BFS will process data both before and after engage. Commfault or BITE will drive a class 3 alarm with | on the BFS GNC SYS SUMM 1 display and an RM FAIL ADTA fault message on a BFS CRT. There is no select/deselect capability for the ADTAs by the BFS other than a manual stowing of the probe.

5.5 FLIGHT CONTROL SENSORS

The four AAs and four RGAs receive identical processing in the BFS software. The data from the three lowest numbered LRUs are processed and a single value is output based on a midvalue selection scheme. The number 4 LRU is not used or acknowledged by the BFS. The BFS does not perform any commfault analysis or validity checks and thus does not recognize failures of any kind. The midvalue select scheme of the outputs from the three lowest numbered LRUs is used no matter what state they are in. The three RGAs on each SRB also follow this scheme.

5.6 CONTROLLERS

5.6.1 RHC. Only the left and right RHCs are supported by the BFS. The command output is based strictly on a midvalue selection scheme. The BFS does not support commfaults, validity checks, or

selection/deselection of RHC channels. The panel trim switches are honored by BFS flight control from both the left and right sides; however, only the left RHC roll and pitch trims are functional.

5.6.2 THC. Only the left THC is processed by the BFS. The output command is based on a simple majority (2 of 3). The BFS does not perform any commfault or validity checks, nor is it possible to deselect or select a THC channel.

5.6.3 SBTC. Both the left and right SBTCs are supported by the BFS; however, only the speedbrake function is processed. Output commands are based on a simple midvalue selection scheme for the three channels. In the event that both SBTCs are engaged and active simultaneously, the commander's SBTC takes precedence. Again, no commfaults or failures are recognized.

5.6.4 RPTA. Only the left RPTA is supported by the BFS. The left RPTA receives signals from both the left and right rudder pedal because the right pedal is mechanically linked to the left. The output command is based on a simple midvalue selection scheme independent of channel status.

5.7 EFFECTORS

The BFS processes the commands and feedback data from only the three lowest numbered channels for the elevons, rudder/speedbrake, and body flap. The selection in every case is a simple midvalue selection scheme. No commfaults, failures, or select/deselect data are recognized by the BFS.

APPENDIX A ACRONYM LIST

AA	accelerometer assembly	LRU	line replaceable unit
ADI	attitude director indicator	LS	limit switches
ADS	air data system	M	Mach number
ADTA	air data transducer assembly	MDM	multiplexer/demultiplexer
ALT	altitude	MLS	microwave landing system
AMI	alpha/Mach indicator	MS	mission specialist
ASA	aerosurface servo amplifiers/actuators	MSBLS	microwave scanning beam landing system
ATT REF	attitude reference	MVS	midvalue select
AVVI	altitude/vertical velocity indicator	NAS	no available signal
BITE	built-in test equipment	NAV	navigation
CDR	commander	OPS	operational sequence
CG	center of gravity	PASS	primary avionics software system
CMDS	commands	PB	pushbutton
COAS	crew optical alignment sight	PLT	pilot
C&W	caution & warning	q	dynamic pressure
D	distance	QMVS	quad-midvalue select
DDU	display driver unit	RA	radar altimeter
DME	distance measuring equipment	RCS	reaction control system
EAS	equivalent airspeed	RGA	rate gyro assembly
FA	flight-critical aft	RHC	rotational hand controller
FCOS	flight computer operating system	RM	redundancy management
FCS	flight control system	RPC	remote power controller
FDIR	fault detection, identification, and reconfiguration	RPTA	rudder pedal transducer assembly
FF	flight-critical forward	RR	rendezvous radar
GNC	guidance, navigation, and control	SBTC	speedbrake/thrust controller
GPC	general purpose computer	SEL/DES	select/deselect
h	altitude	SMRD	spin motor run detector
h	altitude rate	SOP	subsystem operating program
HSI	horizontal situation indicator	SPI	surface position indicator
H/W	hardware	SRB	solid rocket booster
ID	identification	SRB/ET	solid rocket booster/external tank
IMU	inertial measurement unit	ST	star tracker
KSC	John F. Kennedy Space Center	S/W	software
LAT	latitude	T	time
LON	longitude	TACAN	tactical air navigation
LOS	line of sight	TAS	true airspeed
		THC	translational hand controller
		V	velocity
		VORTAC	VHF omnirange tactical air navigation