

WORLD SPACEFLIGHT NEWS SPECIAL REPORT:

# GNC Overview Workbook

## GNC OV 2102

### Advanced Training Series

For Training Purposes only  
Training Division  
Flight Training Branch

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## SECTION 1

### INTRODUCTION

One of the major blocks of software loaded in the Orbiter's onboard General Purpose Computers (GPC's) is the Guidance, Navigation, and Flight Control (GNC) software. The GNC major function software, as it is called, consists of all the software necessary to perform guidance, navigation, and flight control, as well as software used in the crew interface with GNC-related displays and controls. The GNC software also includes processing which makes data acquired through external sensors available to the appropriate guidance, navigation, or flight control programs, and converting outputs from these programs into proper commands to be sent to the aerosurfaces, main engines, etc.

#### 1.1 WORKBOOK GOALS AND SCOPE

This workbook will familiarize students with concepts used in the guidance, navigation, and flight control schemes and describe how they are used during the different flight regimes. It will provide an overview of the data available to the crew concerning the GNC software (CRT displays, dedicated displays, etc) and the crew monitoring task during different phases. This workbook also includes a review of the highlights from the CSI 2102 workbook, an overview of the dedicated displays (detailed descriptions are available in the DED DISP 2102 workbook), and an overview of basic flight data file organization which should facilitate discussions given later in the workbook concerning activities in each flight phase. For information concerning the software associated with processing data from sensors and commands to effectors, see the GNC HW/SW Overview 2102 workbook.

#### 1.2 STUDENT PREREQUISITES

Before reading this workbook, the student should have some familiarity with the basic operation of the Orbiter. This level of knowledge can be acquired by reading the SSV FAM 1107 workbook, which will familiarize the student with a typical mission profile, the onboard equipment, and many of the commonly used acronyms (SRB, ET, MPS, RTLS, DPS, IMU, TAC, etc). Students should also be familiar with the cockpit layout and cockpit panel nomenclature; i.e., L-left, R-right, O-overhead, C-center, F-forward, A-aft. The CSI 2102 workbook is also a prerequisite which should ensure that the student understands the terminology used in the crew interface with the software.

#### 1.3 NOTES TO STUDENTS

The cockpit panel drawings referenced in the text are included at the back of this book.

Since this workbook gives only an overview of the GNC CRT displays, students interested in detailed descriptions of CRT displays might find it useful to have a copy of the DPS DICTIONARY available while reading this text.

## SECTION 2

### BASIC CONCEPTS OF GUIDANCE, NAVIGATION AND FLIGHT CONTROL

#### 2.1 NAVIGATION SOFTWARE

The primary purpose of navigation software is to maintain an accurate estimate of the Orbiter's position and velocity with respect to time. This is called the Orbiter state vector. This information is made available to the guidance software, to the ground controllers for monitoring of the onboard system accuracy, and to the crew. For the crew, such navigation data is translated into more useful forms such as airspeed, altitude, altitude rate, etc., which then appear on the dedicated displays and on the GNC CRT displays. Depending upon flight phase and upon which navigation sensors are available, a variety of schemes are used by the navigation software to propagate and correct the Orbiter's state vector. For example, Inertial Measurement Unit (IMU) data is available throughout the mission and provides information about vehicle accelerations along 3 axes as well as the Orbiter inertial attitude. Navigation uses a fast numerical scheme called Super G to propagate the state vector. This uses the IMU-sensed accelerations, if above a specified "noise" threshold, and a simple gravity model to integrate the state vector forward in time using the equations of motion. In cases where the IMU-sensed accelerations are below the "noise" threshold, such as during an orbit coast mode, a drag model of the Orbiter, as well as a more sophisticated gravity model, is used by the software instead of IMU-sensed acceleration. Any errors in the IMU data or the inherent errors in the gravity and drag models cause a gradual degradation of the onboard state vector. To correct it, the ground sends updates based upon very accurate ground tracking to correct the onboard state vector. These state vector updates are planned to occur every 2-13 hours during orbit. This capability is available during time frames in which the ground is capable of sending uplink commands to the Orbiter (i.e., over an S-band site or in contact with the Tracking and Data Relay Satellite (TDRS) system.)

During the entry phase of the flight, additional navigation sensors provide the accuracy necessary to bring the Orbiter to a pinpoint landing. These include the Tactical Air Navigation (TACAN) system, which provides slant range and magnetic bearing to various fixed stations around the landing site; the Air Data System (ADS), which provides barometric altitude to navigation (angle of attack, Mach number, equivalent airspeed, true airspeed, dynamic pressure and altitude rate are also derived from air data but are not used by navigation); and the Microwave Landing System (MLS), which provides slant range, azimuth, and elevation angles. Another available "sensor" is drag altitude, which consists of a software model of the Earth's atmosphere and which correlates the IMU-measured drag acceleration to altitude. It then passes this altitude estimate to navigation. The capability to update the state vector with TCAN data during the orbit phase of flight is to be added at some time in the future.

When rendezvous navigation is active, the software also propagates a target state vector. Based upon the assumption that the target state vector can be accurately computed, navigation can improve its estimate of the Orbiter state vector with data from one of three external sensors which may be available. These

include the Rendezvous Radar (RR) which provides range, range rate, azimuth and elevation angles between the Orbiter and target, the Star Tracker (ST) which provides horizontal and vertical angles to the target, and the Crew Optical Alignment Sight (COAS) which also provides horizontal and vertical angles to the target.

As illustrated in figure 2-1, these navigation sensors (TACAN, ADTA, MLS, drag altitude), along with the IMU's, are important in maintaining the accuracy of the Orbiter's state vector. Each of the TACAN, ADTA and MLS systems is redundant. Students may obtain further information concerning the software required to convert the raw data from each sensor into a usable form and the software necessary to determine which of the redundant units has the best data (called redundancy management software) by reading the GNC HW/SW OVERVIEW 2102 workbook and the detailed workbooks for each sensor.

Navigation initially propagates the state vector based only on IMU-sensed acceleration or modeled drag acceleration. During entry and rendezvous navigation, data from the nav sensors may be used to improve the propagated state vector. To do this, navigation computes the expected range of error possible for the navigation state based upon which external sensors are available. This range of error (some call it a "cloud of uncertainty") in matrix form the covariance matrix) changes with time and is propagated along with the state vector. When an external sensor, such as TACAN, becomes available, a check is made to see if that data lies within the current expected range of error prior to its being used by navigation. Controls on one of the CRT displays govern whether the data is incorporated in navigation. This scheme is part of what is called Kalman filtering and is used to prevent navigation from being polluted by bad sensor data.

During dynamic mission phases, guidance and flight control need the state vector and attitude data often. In addition, dedicated displays driven with navigation parameters need to be updated more often during dynamic mission phases. However, the navigation software at a fairly slow rate, typically once every 3.84 seconds. To supply data more quickly, User Parameter Processing (UPP) software is used to calculate a state vector at a higher rate between navigation cycles using a simplified state propagation scheme based upon the last state vector computed by navigation. It then updates its state vector with the standard navigation-computed state vector every navigation cycle. The data from this faster propagation is then made available to guidance, flight control, and to the crew displays.

2.2 GUIDANCE SOFTWARE

The guidance software computes commands required to move the vehicle from its current state to a desired state specified by a set of target conditions. In each flight phase, a different kind of target state is specified and a different technique is used to compute the necessary commands which are then sent to flight control. For example, during first-stage flight (launch through SRB separation), guidance uses an open loop scheme which computes commands based only upon a pre-mission determined table of required roll, pitch, and yaw attitudes referenced to relative velocity. On the other hand, during second-stage operations (from SRB separation through MECO), guidance uses a closed loop, interactive scheme which targets for the proper MECO conditions - correct velocity, radius from Earth center, flight path angle, and orbital inclination. This scheme, along with the guidance routines used for the insertion/deorbit burns, the powered RTLS phase, and for on-orbit burns, was developed by Draper Labs at MIT. They are called Powered Explicit Guidance (PEG) routines, and each has an assigned number. The various guidance routines will be discussed in more detail in each of the phase-related GNC sections.

In making its computations, guidance uses the current Orbiter state vector (velocity, position, time tag) from navigation, and during the entry phases, some ADTA parameters. The required steering commands (rotational and translational angles, rates and accelerations) are then sent to the flight control software to be converted into the actual commands sent to appropriate aerosurfaces, Reaction Control System (RCS) jets, main engines or Orbital Maneuvering Subsystem (OMS) engines. (See figure 2-2.)

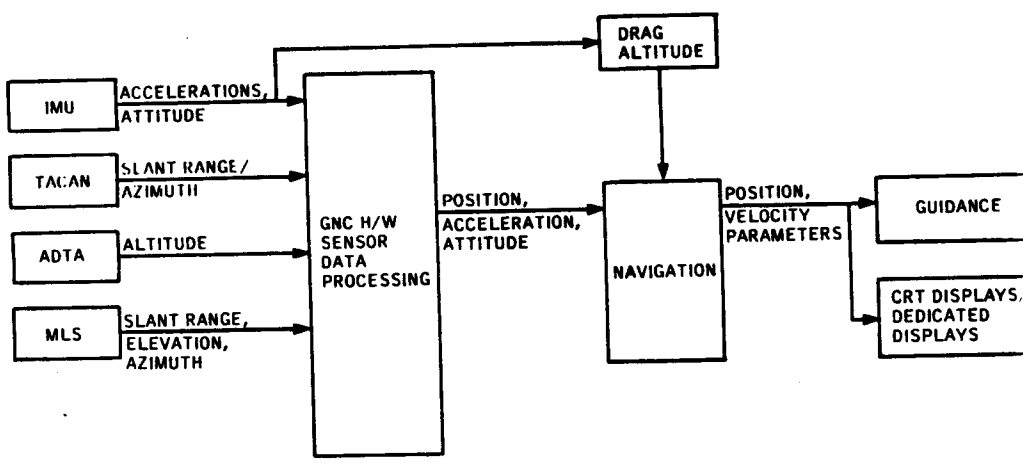


Figure 2-1. Navigation Interfaces

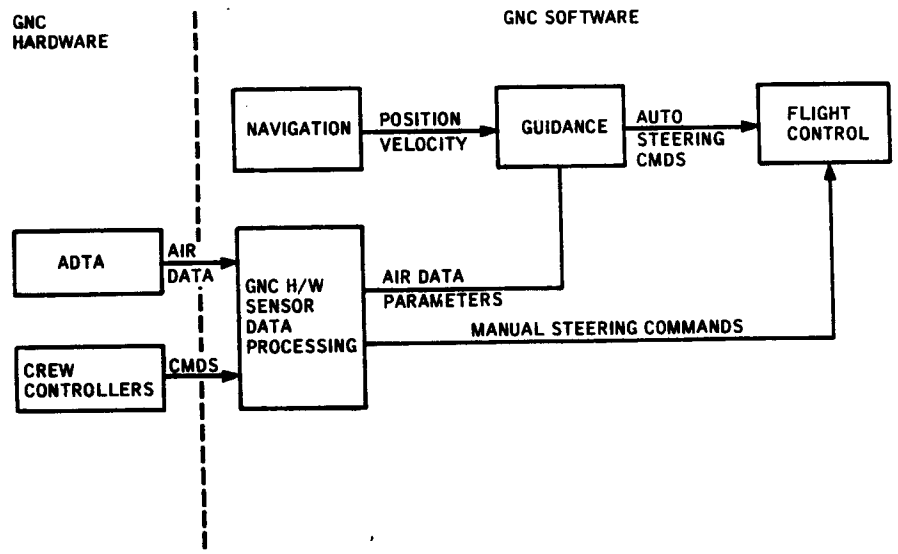


Figure 2-2. Guidance Interfaces

## 2.3 FLIGHT CONTROL SOFTWARE

The purpose of the flight control software, illustrated in figure 2-3, is to maintain vehicle control by accomplishing the steering commands sent to it by the guidance software or initiated by the crew via hand controller/switch inputs. Vehicle control is maintained through the firing, throttling, and/or gimballing (positioning) of onboard engines (MPS, OMS, RCS) and, during atmospheric flight, by the deflection of aerosurfaces. When flight control uses the guidance-computed steering commands, it is in AUTO. When crew inputs are used, flight control is in Control Stick Steering (CSS) or MANUAL. Crew input (via the FCS Mode pushbuttons, the RHC, the DAP panel, etc.) determines whether AUTO or CSS is selected. The decision is typically made on an individual axis basis (pitch, roll/yaw).

Flight Control receives steering commands from guidance or crew inputs which are defined in terms of attitudes, rates and accelerations. One of flight control's jobs, then, is to convert these commands to Orbiter body coordinates, smooth these over time and to limit them in terms of rate and accelerations so that necessary vehicle constraints are met. The attitude commands are compared to the actual attitude (IMU derived) and the error is computed. These attitude errors and rates are sent to another subset of flight control called the Digital Autopilot (DAP). The vehicle attitude, attitude errors and rates are displayed on the Attitude Director Indicator (ADI) which is one of the dedicated displays. The DAP uses the attitude errors and rate commands along with the feedback from external sensors to compute the necessary aerosurface and engine commands. During the ascent, entry, and RTLS phases (OPS 1, 3, and 6), normal and lateral acceleration data are supplied by the Accelerometer Assemblies (AA's). In addition, roll, pitch, and yaw body rates are supplied by the Rate Gyro Assemblies (RGA's). Angle of attack, mach number, and dynamic pressure can be supplied to Flight Control during the latter part of entry from the Air Data System. During the on-orbit phase (OPS 2,) flight control uses only IMU data for acceleration and attitude information.

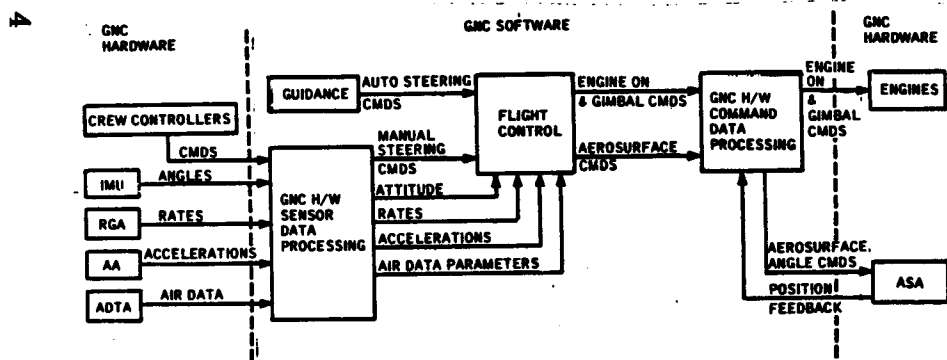


Figure 2-3. Flight Control Interfaces

## 2.4 COMMONLY USED COORDINATE SYSTEMS

Several coordinate systems are used by the GNC software and dedicated displays to define the Orbiter's position, velocity, and attitude. For example, to uniquely define the Orbiter's state vector, an inertial (i.e. non-rotating, non-accelerating) coordinate system must be used. The one selected is called the Mean of 1950 inertial system and is discussed in detail in a later section. For other purposes, the vehicle's attitude, position, or velocity may be expressed using a different system, such as the Runway Centered Coordinate system. Four of the most commonly referenced systems are the Body Axis, M50 Inertial, Local Vertical/Local Horizontal (LVLH) and Runway Coordinate systems. These are defined as follows:

**2.4.1 Body Axis Coordinate System.** This coordinate system, illustrated in figure 2-4, has its origin at the Orbiter's center of gravity, and is oriented as follows:

- X axis is parallel to the Xs axis of the Orbiter structural body (Orbiter + ET + SRB's); the X<sub>0</sub> axis is in the Orbiter plane of symmetry, 400 inches below the payload bay centerline. Positive is toward the nose
- Z axis is parallel to the Orbiter plane of symmetry and is perpendicular to the X axis; positive down with respect to the Orbiter fuselage
- the Y axis completes the right-handed orthogonal system.

Note that pitch is a rotation about the Y-axis, roll is a rotation about the X-axis, and yaw is rotation about the Z-axis.

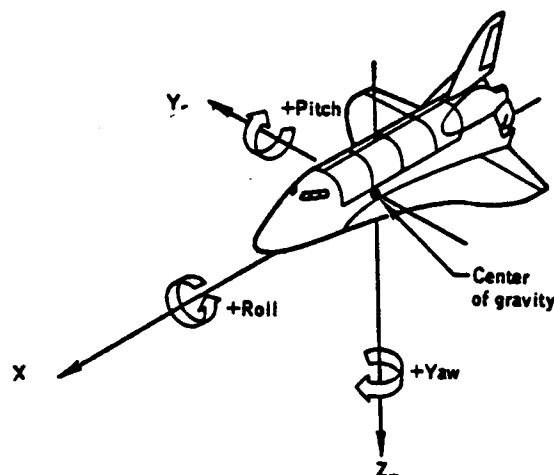


Figure 2-4. Body Axis Coordinate System

2.4.2 M50 inertial coordinate system. This coordinate system has its origin at the center of the Earth, and its axes are fixed relative to the stars. The Inertial Earth-centered coordinate system, illustrated in figure 2-5, has its axes defined as follows:

- +Z axis is directed towards the Earth's North Pole at epoch (the Earth's mean rotational axis at epoch)
- +X axis is pointed at the first point of Aries (the Earth's mean vernal equinox at epoch)
- +Y axis completes the right-hand system.

Because the epoch of this system is the beginning of the Besselian year 1950, and the +X axis points toward the first point of Aries, this reference frame is often referred to as the Mean of 50 (M50) frame.

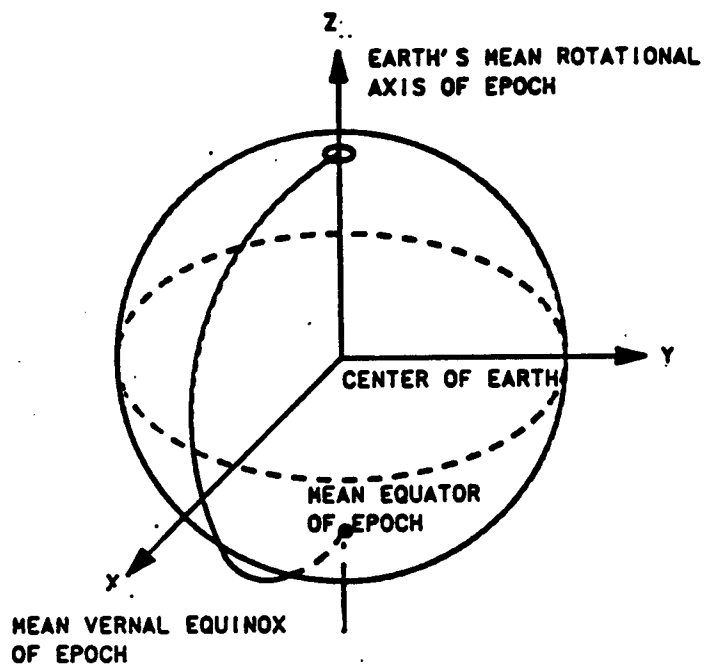


Figure 2-5. M50 Inertial Coordinate System

2.4.3 LVLH coordinate system. The Local Vertical/Local Horizontal (LVLH) reference frame, illustrated in figure 2-6, is a rotating reference coordinate system that instantaneously (referenced with respect to time) defines the attitude of the Orbiter. These coordinate system axes are defined as follows:

- +Z axis is pointed at the center of the Earth
- +Y axis is pointed opposite to the momentum vector (H) of the vehicle
- +X axis is positioned to complete the right-hand system.

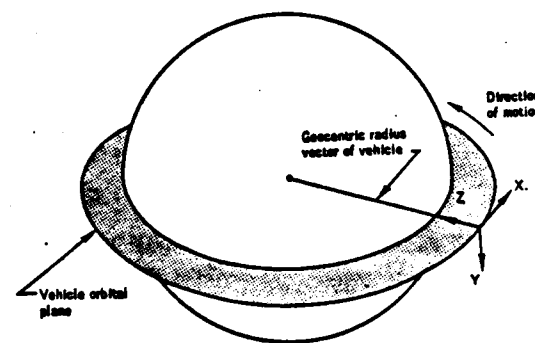


Figure 2-6. LVLH Coordinate System

2.4.4 Runway coordinate system. This coordinate system, illustrated in figure 2-7, has its origin on the runway center at the approach threshold.

Its axes are defined as follows:

- Z axis normal to the ellipsoid model of the Earth through the runway centerline at the approach threshold and positive toward the center of the Earth
- X axis perpendicular to the Z axis and lies in a plane containing the Z axis and the runway centerline and is positive in the direction of landing
- Y axis completes the right-handed system.

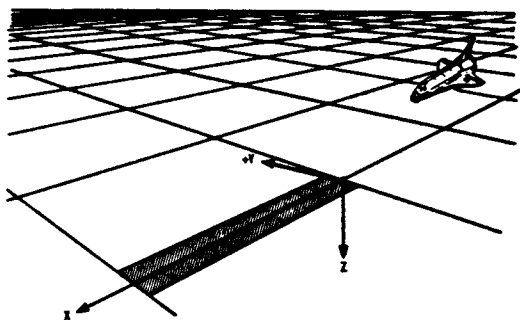


Figure 2-7. Runway Coordinate System.

Note: To express the current vehicle attitude with respect to one of the last three coordinate systems, imagine a vehicle initially oriented with its X, Y and Z body axes superimposed on the axes of the desired coordinate system. The unique Euler sequence in terms of pitch (rotation about the Y axis), yaw (rotation about the Z axis) and roll (rotation about the X axis) required to maneuver to the current vehicle attitude is the Orbiter attitude with respect to the desired coordinate system.

## SECTION 3

### CREW SOFTWARE INTERFACE REVIEW

As discussed in the CSI 2102 workbook, the onboard GNC software consists of various types of software programs required during the five different flight phases. The five major flight phases, or Operational Sequences (OPS), seen during flight are as follows:

OPS 1	Ascent through orbital insertion
OPS 2	On-orbit operations
OPS 8	On-orbit vehicle checkout
OPS 3	Deorbit through landing
OPS 6	Return to Launch Site (RTLS) abort

With the exception of OPS 1 and 6, which reside simultaneously in the GPC's, the other OPS are only available independently. The OPS are subdivided further into Major Modes (MM) which are accessed sequentially based upon some event in flight (such as Solid Rocket Booster (SRB) separation) or through crew inputs ("OPS 105 PRO"). Each major mode is associated with a CRT display (called the OPS display) which gives the crew the most important information necessary to monitor and/or control the vehicle during that phase. Other displays, called SPEC's, may also be monitored. Going from one OPS to the next is called an OPS transition, and going from one major mode to another is called a major mode transition.

The CSI 2102 workbook also mentioned that two major versions of flight software may be loaded in the onboard computers - the Primary Avionics Software System (PASS) or the Backup Flight System Software (BFS). The PASS is a more sophisticated system than the BFS and provides redundancy by being loaded into multiple computers. Because a large amount of code is required to execute the PASS programs, only subsets of it (the OPS we discussed earlier) may be loaded at a given time. The Mass Memory Units (MMU's) are used to load new OPS periodically. The BFS, on the other hand, is a more simplistic system which is completely loaded, preflight, into one GPC. The PASS and BFS systems were developed by separate organizations (IBM and ROCKWELL, respectively) with the intention that if the PASS failed due to a generic flight software problem, the BFS would not have the same problem and could control the vehicle. In addition, there are other GPC and hardware failure modes which could require the BFS for vehicle control. Forcing the BFS to take control of the vehicle is called "engaging" the BFS. In order to be capable of "engagement," the BFS maintains readiness by operating in a state in which it monitors (tracks) the PASS by monitoring the same inputs from avionics sensors (with the exception of MLS data), aer-surfaces, jets, etc., as those received by the PASS. The BFS can also drive CRT displays. In some cases the crew actively monitors both PASS and BFS CRT displays to verify the health of both systems. Also, some data is only available for crew monitoring on the BFS CRT displays. This overview workbook deals primarily with the PASS GNC activities. The subtle differences in the two systems will be discussed in the detailed workbooks.

## SECTION 4

### DEDICATED DISPLAY OVERVIEW

The GNC Dedicated Displays include meters and lights displaying vehicle attitude, velocity, acceleration, and other parameters related to vehicle performance. These provide the crew with the information required to fly the vehicle manually or to monitor the automatic flight control system performance. In addition, these displays also allow the crew to monitor Reaction Control System (RCS) activity, the current flight control system mode, and entry events. The data on the dedicated displays may be generated by the navigation or by flight control software, as well as more directly from one of the navigation sensors (ADTA, TAC, MLS, RA). The dedicated displays are located in front of the Commander's (CDR's) and Pilot's (PLT's) seats and on an aft panel below one of the aft windows. The following meters and lights are considered dedicated displays and can be found on the panel drawings included in appendix A:

Attitude Director Indicator (ADI)	Panel F6, F8, A1
Horizontal Situation Indicator (HSI)	Panel F6, F8
Alpha/Mach Indicator (AMI)	Panel F6, F8
Altitude/Vertical Velocity Indicator (AVVI)	Panel F6, F8
Surface Position Indicator (SPI)	Panel F7
Flight Control System Mode Status Lights	Panel F2, F4
Reaction Control System Activity Lights	Panel F6
G-meter	Panel F7
Heads-up Display (HUD)	Forward Windows

The crew has some control over the source for the data used to drive the dedicated displays via switches on F6, F8 and A6. Not all of the dedicated displays are available in every OPS or major mode. Their availability is related to the requirements of each flight phase. In the following paragraphs, each type of dedicated display will be briefly described.

#### 4.1 ATTITUDE DIRECTOR INDICATOR (ADI)

The ADI, illustrated in figure 4-1, is used to display

- Orbiter pitch/yaw/roll attitude via a gimballed ball
- Attitude errors via three needles
- Attitude rates via three pointers.

Three ADI's are installed in the Orbiter - one each in the CDR, PLT, and aft stations (panels F6, F8, and A1, respectively). The two forward ADI's are supported throughout the mission, while the aft ADI is active only during orbit operations.

The Orbiter attitude is displayed on the gimballed ball in terms of a pitch, yaw, roll sequence and is derived from one of three reference frames selected by the crew via the ADI ATTITUDE source switch. A separate switch is used for each ADI located on Panel F6 for the CDR's ADI, Panel F8 for the PLT's ADI, and Panel A6 for the aft ADI. The three switch positions are INERTL, LVLH and REF. The position selected depends upon the task the crew is required to

perform. Basically, the INRTL position allows the crew to view the Orbiter's attitude with respect to an Earth-centered inertial (non-rotating) reference frame useful in locating stars. The LVLH position shows the Orbiter's attitude with respect to an Orbiter-centered rotating reference frame (Section 2.4), which also allows the crew to see the vehicle's attitude with respect to the Earth. The REF position is primarily used to see the Orbiter's attitude with respect to an inertial reference frame defined at the instant the crew depresses the ATT REF pushbutton. This is useful when flying back to a previous attitude.

The ADI error needles are driven by flight control outputs which show the difference between the required and current vehicle attitude. These needles are "fly-to," meaning that the crew must maneuver in the direction of the needle in order to null the error. For example, if the pitch error needle points down, the crew must manually pitch down to null the pitch attitude error. These errors are displayed with respect to the Orbiter body axis coordinate system and thus are independent of which reference frame is selected for ADI attitude display. The amount of needle deflection indicates the number of degrees of attitude error and is dependent upon the ADI ERROR switch position - HIGH, MED, LOW. This switch allows the crew to set the sensitivity of the needle. There is an ADI ERROR switch associated with each ADI - panel F6 for the CDR's ADI, panel F8 for the PLT's ADI and panel A6 for the aft ADI.

The ADI rate pointers show the crew the vehicle body rotational rate. The scale to which these are driven is dependent upon the ADI RATE Switch (Panel F6, F8 and A6). These pointers also are "fly-to" in the sense that the rotation hand controller (RHC) must be moved in the same direction as the pointer to null a rate. Rate pointers are independent of the ADI ATTITUDE source switch.

A Sense Switch, located on Panel A6, allows the crew to use the aft ADI, RHC, and THC a -X or -Z control axis sense. These two options make the ADI and hand controllers correspond to the visual data out the aft (-X) or overhead (-Z) windows.

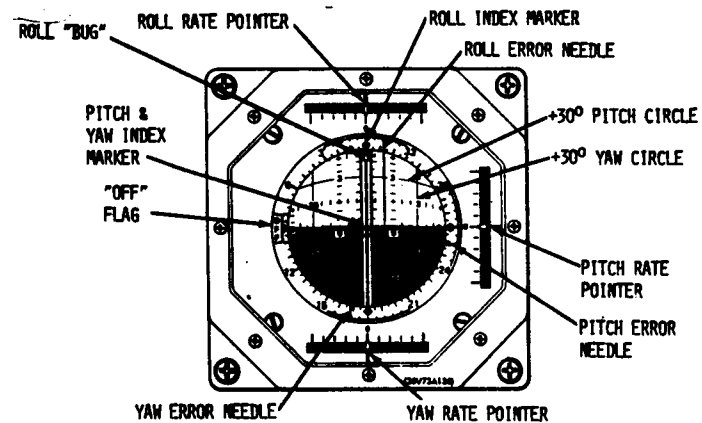


Figure 4-1. Attitude Director Indicator

#### 4.2 HORIZONTAL SITUATION INDICATOR (HSI)

The HSI provides the crew with a pictorial view of the vehicle position with respect to various navigation points and a visual perspective of certain GNC parameters such as directions, distances, and course/glideslope deviation (see figure 4-2). The crew uses this information to control and/or monitor vehicle performance. There are two HSI's in the cockpit, one each in the commander and pilot stations. The HSI's are supported during the entry/landing phases and, with some limitations, in the Ascent/RTLS phases. The HSI displays the following information:

- magnetic heading (compass card)
- selected course
- course deviation
- glideslope deviation
- primary/secondary bearing
- primary/secondary range

These indicators are driven differently during different flight phases. The crew has some control over their meaning using the HSI SELECT MODE switch, a three-position switch (ENTRY/TAEM/A/L) that allows the crew to see the most meaningful parameters during a particular flight phase. The HSI SELECT SOURCE switch allows the crew to see the HSI data as driven by a specific TACAN or MLS, or to monitor the output of the navigation software which is based on IMU, DRAG ALTITUDE, TAC, ADTA, and/or MLS data.

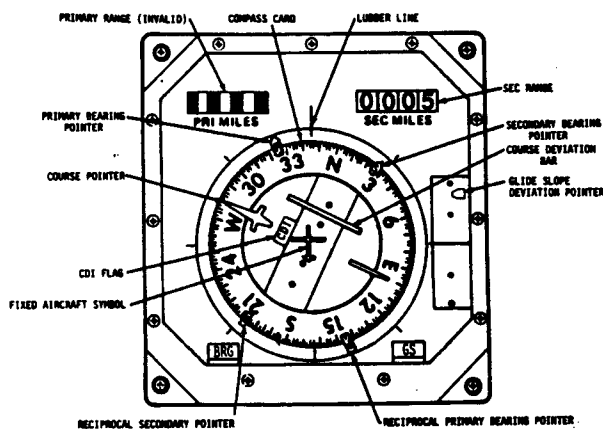


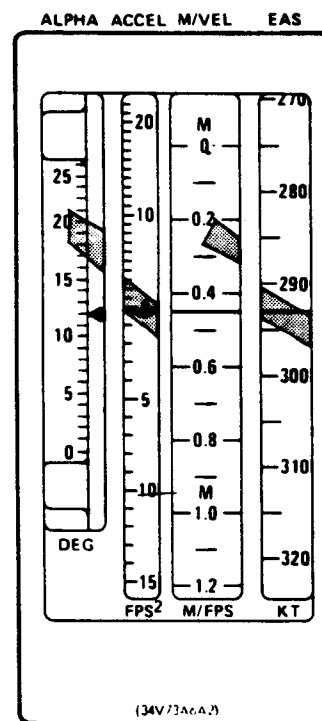
Figure 4-2. HSI

#### 4.3 ALPHA/MACH INDICATOR (AMI)

The AMI consists of four tape meters displaying the following parameters:

- angle of attack (ALPHA)
- vehicle acceleration (ACCEL)
- vehicle velocity (M/VEL)
- equivalent airspeed (EAS).

There are two AMI's - one on the CDR's side and the other on the PLT's side. The AMI's are active during ascent and the entry/landing phase. During ascent, the data on the display comes from navigation. During entry, the AMI data source is either navigation or the air data system (left probe or right probe) depending upon the position of the AIR DATA SOURCE SELECT switch. The crew can use this switch to compare the left and right air data systems and to see how air data compares to navigation computations prior to its incorporation into navigation. The AMI is illustrated in figure 4-3.



- ALPHA -18 TO +60 DEGREES
- ACCEL -50 TO +100 FT/SEC<sup>2</sup>
- M/VEL 0-4;  
4 K TO 27 KFT/SEC
- EAS 0-500 KNOTS

Figure 4-3. AMI

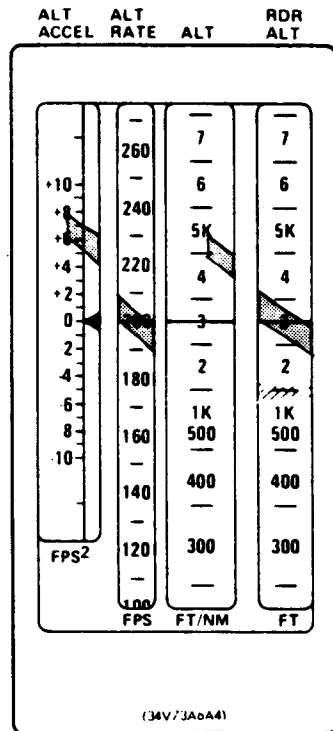


#### 4.4 ALTITUDE/VERTICAL VELOCITY INDICATOR (AVVI)

The AVVI displays the following flight parameters:

- vertical acceleration (ALT ACCEL)
- vertical velocity (ALT RATE)
- barometric altitude (ALT)
- radar altitude (RDR ALT)

There are two AVVI's, one on the CDR's side and the other on the PLT's side. Only the ALT and ALT RATE tapes are active during ascent, whereas all four tapes become active during entry. The RDR ALT tape is not driven until the radar altimeters (RA's) lock on, at approximately 5000 feet. During entry, the AVVI data source depends on the AIR DATA SOURCE SELECT switch and the RADAR ALTM switch. As with the AMI, the AIR DATA SOURCE SELECT switch allows the ALT ACCEL, ALT RATE and ALT tapes to be driven using data from the left or right air data system or navigation data. Normally, NAV is selected. The AVVI is illustrated in figure 4-4.



- ALT ACCEL -13.3 TO +13.3 FT/SEC<sup>2</sup>
- ALT RATE -2940 TO +2940 FT/SEC
- ALT -1100 FT TO 165 NM
- RDR ALT 0 TO 9 KFT

Figure 4-4. AVVI

#### 4.5 SURFACE POSITION INDICATOR (SPI)

The SPI displays the position of the vehicle control surfaces and the commanded speedbrake position and includes the following:

- elevon position
- body flap position
- rudder position
- aileron position
- speedbrake actual and commanded positions.

The only SPI installed in the vehicle is located in the center of Panel F7 (See figure 4-5). The SPI is active during entry and during the entry portion of an RTLS.

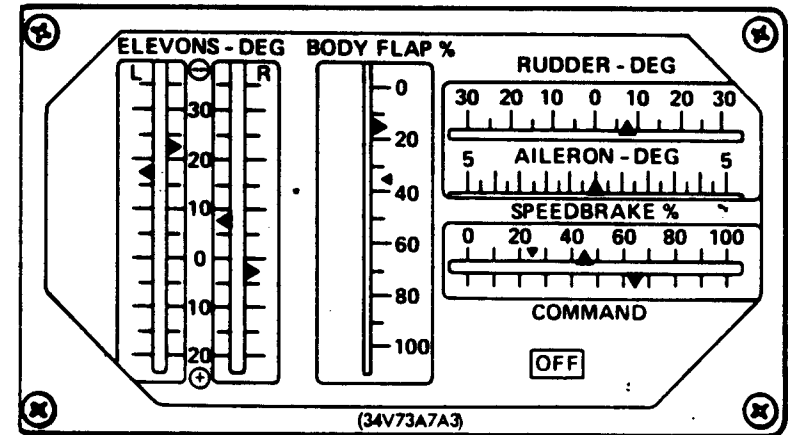


Figure 4-5. SPI

#### 4.6 FLIGHT CONTROL SYSTEM (FCS) MODE PUSH BUTTON INDICATORS (PBI'S)

The FCS Mode PBI's transmit automatic versus manual moding requests to flight control and reflect the current state of the flight control system from lift-off thru MECO and during entry (starting at 500K ft). The FCS Mode PBI's are used to command and reflect the status of the following:

- PITCH Control Mode (CSS or AUTO)
- ROLL/YAW Control Mode (CSS or AUTO)
- Speed Brake/MPS Throttle (SPD BK/THROT) Control Mode (AUTO or MAN)
- BODY FLAP Control Mode (AUTO or MAN).

There are two sets of FCS Mode PBI's, one for the CDR and one for the PLT. The FCS PBI's are illustrated in figure 4-6.

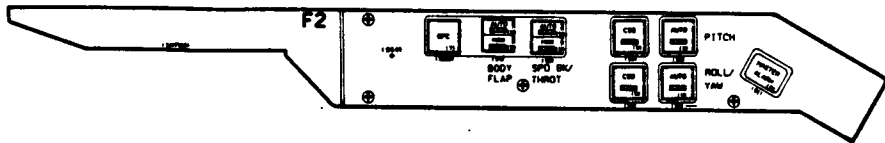
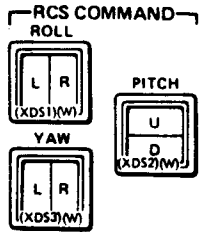


Figure 4-6. FCS Mode PBI's (Panel F2)

#### 4.7 REACTION CONTROL SYSTEM (RCS) ACTIVITY LIGHTS



The RCS Activity Lights are located on panel F6 and are used primarily to indicate RCS jet commands by axis and direction. Secondly, they may be used to indicate when more than two yaw jets are commanded and when the elevon or aileron drive rate is saturated, which may indicate possible flight control problems. These lights are active during OPS 2 and 3 and during the glide portion of an RTLS abort. There are three types of lights: ROLL L-R, YAW L-R, and PITCH U-D.

#### 4.8 G-METER

The G-meter is a self-contained accelerometer and display unit mounted on Panel F7. It senses linear acceleration along the Z-body (normal) axis of the vehicle. It is illustrated in figure 4-7. It is especially useful in flying contingency aborts and the final phase of entry.

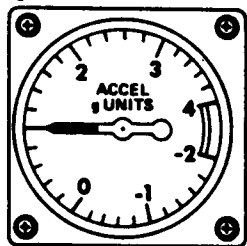


Figure 4-7. G-meter

#### 4.9 HEADS-UP DISPLAY (HUD)

The Heads-Up Display (figure 4-8) is an optical mini-processor system which provides sufficient pilot cues to allow the final phase of entry, particularly final approach, to be flown with minimal transitions from eyes out the window (HEADS-UP) to eyes on the dedicated display instruments (HEAD-DOWN). There are two HUD's, one for the CDR and one for the PLT. A HUD consists of an electronics unit that receives and processes data to be displayed and a pilot display unit mounted on the Orbiter glare shield in front of a forward window. The HUD displays the same data also displayed on several instruments including the ADI, SPI, AMI and AVVI. The types of data and graphics displayed on the HUD vary with the phase of entry. The HUD also provides event cueing and crew alerts such as MLS data loss, CSS control in all axes, or body flap not at trail.

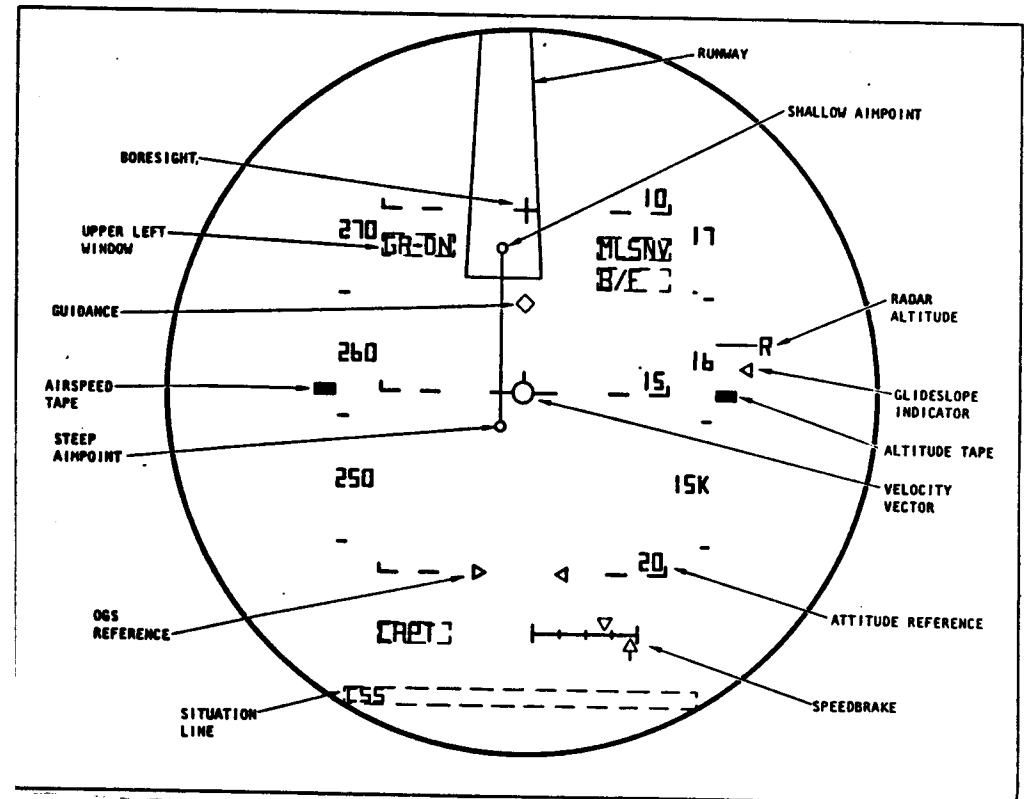


Figure 4-8. Heads up display

SECTION 5

FLIGHT DATA FILE (FDF) ORGANIZATION

The onboard FDF contains books, cards, and charts that include instructions for equipment operation, troubleshooting procedures for system failures, and time-sequential procedures. Some of the checklist procedures are printed on velcro-backed cards placed in designated locations around the cockpit. These are called cue cards and are used for time-critical procedures (such as for Ascent and Entry) as well as for standard operating procedures. Figure 5-1 shows a summary of the major FDF books typically used during a mission.

	Nominal Procedures	Malfunction Procedures
Pre-launch	Ascent C/L	
Launch thru MECO	Cue Cards - Ascent Flip Book	Cue Cards
MECO thru OMS 22	Ascent C/L, Cue Cards for Burns	Ascent Pocket C/L
MET 00:50 thru 4:00	Post Insertion C/L	Orbit Pocket C/L STS Malfunction Procedures
MET 4:00 up to D/O - 3:30	Crew Activity Plan Orbit OPS C/L Payload Related C/L Cue Cards	Orbit Pocket C/L STS Malfunction Procedures
D/O-3:30 to D/O Burn	Deorbit Prep C/L	Orbit Pocket C/L Entry Pocket C/L STS Malfunction Procedures
D/O Burn	Cue Cards	Entry Pocket C/L
Post Burn to Entry Interface	Entry C/L	Entry Pocket C/L
Entry Interface to Roll-Out	Cue Cards	Cue Cards
Post Roll-Out	Entry C/L	

Figure 5-1. Typical FDF used in a Mission

Figure 5-2 shows an example of the type of procedures seen on ascent.

ASCENT PROCEDURES	
-:30	16mm Camr (two) - ON **
R180	LVLH**
0.6M	✓ Pc - 68%
1.4M	✓ Pc - 100%
2:00	✓ Pc < 50' (backup 2:15) (✓ rates < 5,2,2) * 'SEP INH' + 5 sec - MAN SRB SEP *
2:07	✓ TMECO * NOT STABLE (10 sec) * * NO COMM - CSS & MAN THROT * FLASH EVAP CNTLR PRI A - ON ** TOP EVAP HTR DUCT - A** HI LOAD DUCT HTR - A** O2 TK1,2 HTRS B (two) - AUTO** H2 TK1,2 HTRS B (two) - AUTO**
3:00	✓ EVAP OUT (T < 60)
MECO	✓ 25,670 * If Man Throttle (3 eng) * * Man Shutdn at 25,550 *
ET SEP	(MECO+18) * 'SEP INH' & rates < .7, .7, .7: * * OPS 104 PRO, ET SEP - MAN * * Assume Feedline Fail: * * ASC C/L, DELAYED OMS 1 * ✓ ATT within 100° of SEP ATT * If not - delete +X xlation and * * delay maneuvers until TIG -1:19 *
MM104+2s	+X xlation for 11 sec ** ✓ TGTS

Figure 5-2. Ascent Procedures

## SECTION 6

## ASCENT GNC OPERATIONS

Figure 5-3 shows an example of the type of malfunction procedures seen on cue cards.

The Ascent portion of flight includes the final pre-launch activities (terminal count); the launch of the Shuttle using the SRB's and main engines (First Stage); SRB separation; continuation of powered flight using the main engines; MECO/ET separation; and the insertion burns, made using the OMS engines, which place the Shuttle in the desired orbit.

If main engine or certain other Shuttle system failures occur which jeopardize the flight, the crew may be required to exercise an ascent abort. Depending upon the time of the main engine failure or the type/severity of the system problem, the crew will execute one of the following:

- Return to Launch Site (RTL) - the vehicle is pitched around using the main engines and glides to a landing at the launch site
- Transatlantic Abort Landing (TAL) - the vehicle continues across the Atlantic ocean and is landed at a contingency landing site such as Dakar, Senegal or Rota, Spain
- Abort-Once-Around (AOA) - the vehicle continues around the Earth and is landed at Edwards Air Force Base, Northrup Air Strip at White Sands New Mexico, or the Kennedy Space Center
- Abort-To-Orbit (ATO) - the vehicle is inserted in a lower-than-planned orbit.
- Contingency abort - the vehicle is ditched in the ocean.

The RTL, TAL, AOA, and ATO aborts are referred to as "intact" as the vehicle is recoverable.

In this section of the workbook, each subphase of the nominal (non-abort) ascent profile will be discussed, including a description of all major GNC events, GNC software functions, and the major GNC crew interface during that subphase. Each ascent abort mode will be discussed at the end of the section.

GNC		
<b>FCS CH (1st)</b> 1. FCS CH - OVRD,AUTO <b>FCS CH (2nd)</b> 1. Good FCS CHs - OVRD 2. Failed FCS CH - OVRD,AUTO If recovered: 3. Overridden FCS CHs - AUTO  <b>RHC FAIL (1st)</b> If BFS engaged: 1. Aff FLT CNTLR PWR - OFF <b>RHC DLMA</b> 1. Aff FLT CNTLR PWR - OFF <b>SBTC FAIL/DLMA</b> 1. Use other SBTC <b>ENTRY ROLL MODE SW FAIL</b> 1. [GSI] ITEM 21 - AUTO SEL <b>DISPLAY SW L(R)</b> 1. L(R) Disp Sws - green dot	<b>IMU FAIL</b> 1. [BFS GSI] - desel failed IMU <b>IMU DLMA</b> [GSI] 1. PWC, des bad IMU (PASS & BFS) Post ET SEP, if still dlma & 'RGA' for 15 sec: 2. Des lower # IMU with '?' <b>ADTA FAIL*</b> [GSI] 1. If ADTA data good: rese <b>ADTA DLMA*</b> [GSI] 1. If failed ADTA identified: desel or if failed Probe - STOW If transient failure: desel, rese 2. If 3 ADTAs failed: ADTA to G&C - INH *BFS Engaged: if failed ADTA candidate for selection - stow Probe and inhibit to G&C	<b>TACAN FAIL</b> [GSI] 1. If failure verified: desel in BFS 2. If transient failure: desel,rese <b>TACAN DLMA</b> [GSI] 1. If failed TACAN identified (HSI/CRT) and failure is Range: Failed TACAN - OFF 2. If transient failure: desel,rese <b>BODY FLAP FAIL/CYCLE</b> (GRTL/304/305) 1. BDY FLP - MAN Do following for CH X = 1(2,3) until CH isolated: 2. HYD MN PUMP PRESS X - LO If fail clears or cycling stops: 3. FCS CH X - OFF 4. HYD MN PUMP PRESS X - NORM 5. BDY FLP - AUTO >> If fail remains or runaway: 6. HYD MN PUMP PRESS X - NORM 7. Rtn to step 2 for next CH  0V099/MULTI/BAS

Figure 5-3. Typical Example of Cue Card Malfunction Procedures

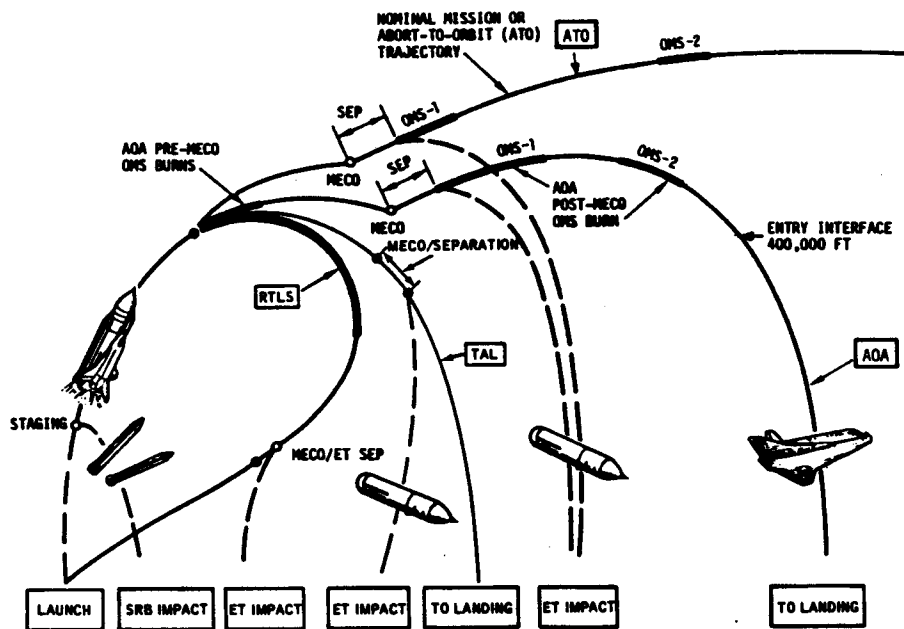


Figure 6-1. Abort & Nominal Mission Profile

## 6.1 TERMINAL COUNT

**6.1.1 Terminal Count Description.** The Terminal Count phase extends from T-20 minutes (the "T" refers to the lift-off time) through SRB ignition. During this timeframe, the GNC OPS 1 software is loaded into the onboard computers (GPC's) and verified by the ground. In addition, last minute launch preparations are made. A planned 10-minute "HOLD" occurs at T-9 minutes. Of interest is that the main engines are lighted at T-7 seconds and the SRB's are ignited 2.75 seconds after all three main engines reach 90% thrust.

**6.1.2 Terminal Count GNC Software Functions.** During Terminal Count, the GNC software is in Major Mode 101. Guidance and flight control software are not active. Navigation software is initialized at T-8 seconds when the onboard state vector is initialized and state vector propagation begins.

**6.1.3 Terminal Count GNC Crew Interface.** During Terminal Count, the crew follows procedures in the Ascent Checklist, including the following GNC-related tasks:

- Verify that the CRT display, HORIZ SIT, is correctly configured for ascent and for a landing in case an RTLS is required. This display includes altimeter setting, runway selection, and an item allowing performance of certain automatic ascent maneuvers used to test the vehicle
- Receive a weather update
- Monitor the main engine start at T-7 seconds by watching the Main Propulsion System (MPS) chamber pressure (Pc) indicators on Panel F7.

The crew also is required to perform tasks related to other onboard systems. During this phase, no information is driven on the PASS MM 101 CRT display, "LAUNCH TRAJ." The only data available on the BFS MM 101 display, "LAUNCH TRAJ 1", is the MPS throttle command, in percent. See figures 6-2 and 6-3.

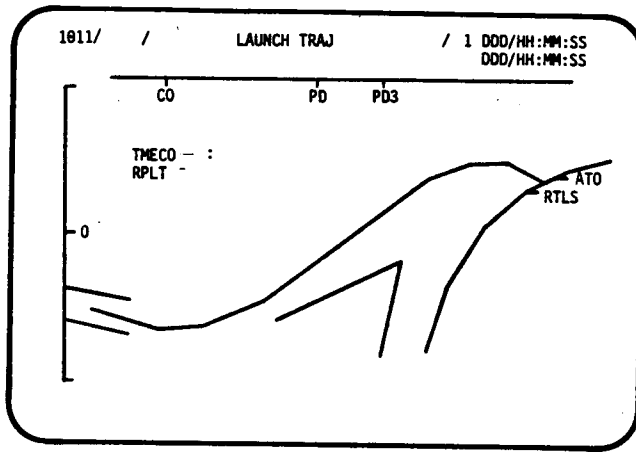


Figure 6-2. PASS "LAUNCH TRAJ" display

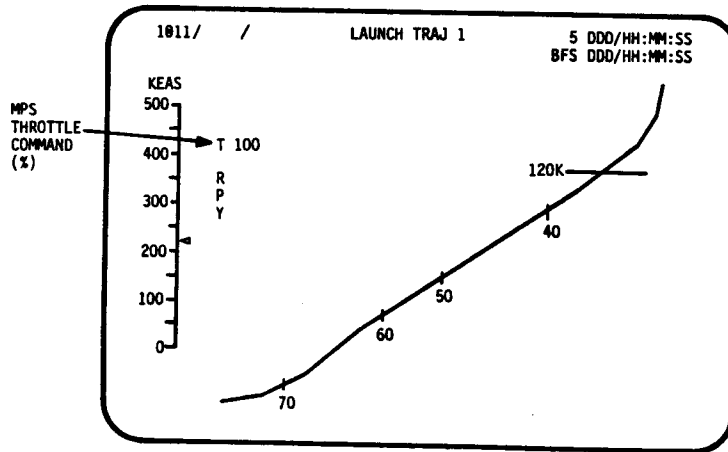
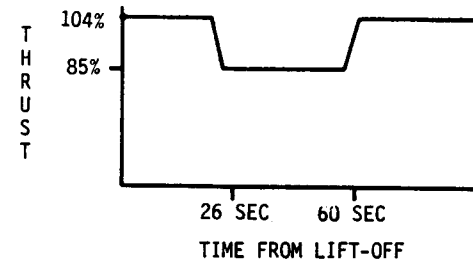


Figure 6-3. BFS "LAUNCH TRAJ 1" display

## 6.2 FIRST STAGE

**6.2.1 First Stage Description.** First stage ascent extends from SRB ignition through SRB separation, also called SRB staging. The sequence of major GNC events proceeds as follows: the vehicle lifts off the pad .3 seconds after the SRB's ignite. It then rises vertically in attitude hold until the SRB nozzles clear the tower lightning rod. The vehicle then begins a combined roll, pitch, yaw maneuver (the "roll maneuver" or "pitchover") which places the vehicle head down, with wings level and aligned with the launch azimuth. This orientation, along with other factors, results in more favorable aerodynamic loads during the regions of high dynamic pressure. In addition, it allows the crew to use the ground as a visual reference. The vehicle steadily accelerates through the atmosphere. Due to concerns about the structural load on the vehicle, especially during the first 90 seconds of flight, the flight control system provides "load relief" by making adjustments to its commands to reduce vehicle loads at the expense of maintaining a precise trajectory profile. In addition, a special schedule of elevon position with respect to velocity is followed to protect the elevons from excessive loads. In order to keep the dynamic pressure ( $\bar{q}$ ) on the vehicle below a specified level (on the order of 580 psf), the main engines are throttled down at about T+26 sec and are throttled back up to thrust at about T+60 seconds. The throttle values vary between flights. As an example, on STS6, the engines were throttled from 104% to 85% and then back up to 104%. If the main engine thrust were graphed with respect to time, it would look something like this:



Due to the appearance of this graph, the term "thrust bucket" evolved. Maximum dynamic pressure occurs shortly after throttle up. The next major event is SRB separation, which occurs 6 seconds after the SRB separation sequence software detects both SRB chamber pressures ( $P_c$ ) below 50 psi within 4.3 seconds of one another and detects vehicle rates within specified limits. These checks ensure that neither SRB is burning at separation and that the vehicle and SRB's will not recontact the Orbiter after separation. SRB separation occurs at about 2 minutes after launch. At SRB separation, first stage is complete and the software automatically modes to Major Mode 103.

**6.2.2 First Stage GNC Software Functions.** First-stage guidance is active from SRB ignition through SRB separation plus 4 seconds. In this stage, guidance uses a pre-flight planned ("canned") table of roll, pitch, and yaw attitudes referenced to relative velocity. There are canned tables defined for the cases of three good engines, as well as for left, center, or right engine failed cases. In addition to sending commands to flight control to obtain proper vehicle attitudes, guidance also sends commands to the MPS throttle per a pre-flight defined (I-loaded) throttle schedule which is a function of relative velocity unless the crew has taken manual control of the throttle. Steering of the vehicle during first stage is accomplished by gimbaling the SRB nozzles.

Navigation during first stage propagates the vehicle state vector using the Super-G integration scheme using IMU data and a gravity model.

If in the AUTO mode, flight control during first stage uses commands sent to it from guidance. If the crew has selected Control Stick Steering (CSS), commands are input through the RHC. Feedback from external sensors (Rate Gyro Assemblies and Accelerometers) is used to generate commands to reposition the SRB nozzles and main engine gimbals. Flight control sends the current vehicle attitude rates and the errors between current and desired attitudes to the ADI for display to the crew. As mentioned earlier, flight control also performs load relief by orienting the vehicle to reduce normal and lateral accelerations and by commanding the elevons to alleviate hinge moment loads per the pre-mission-defined elevon schedule. If the crew does select CSS, the resultant control mode is discrete rate/attitude hold. In this mode, when the RHC is moved, a specified attitude rate is commanded as long as the stick is out of its neutral, detent position. When the stick is in detent, attitude hold commands are generated by flight control. In addition, load relief is not performed by the software when the crew selects CSS.

**6.2.3 First Stage GNC Crew Interface.** During first stage, no GNC-related crew actions are planned unless a failure occurs. Following launch, the crew uses the Ascent "Flip Book" for general procedures. To ensure that the auto flight control system is maintaining the expected ascent profile, they can scan the variety of cue cards placed next to the dedicated displays. For example, the crew can verify that the vehicle is at the correct pitch attitude (via the ADI) and altitude rate (via the AVVI) at each of five designated times during the first stage. Using the Ascent Procedures page in the Ascent Flip Book, the crew can monitor that the main engines correctly throttle down and up. They also can ensure that the "Pc < 50" message prior to SRB SEP correctly appears on the MM 102 CRT display (ASCENT TRAJ) and that SRB separation occurs on time. Manual intervention by the crew is required if these events are not automatically accomplished. The crew is also responsible for monitoring engine performance. During first stage, only limited information is available to the crew on the PASS and BFS MM 102 displays. See figures 6-4 and 6-5.

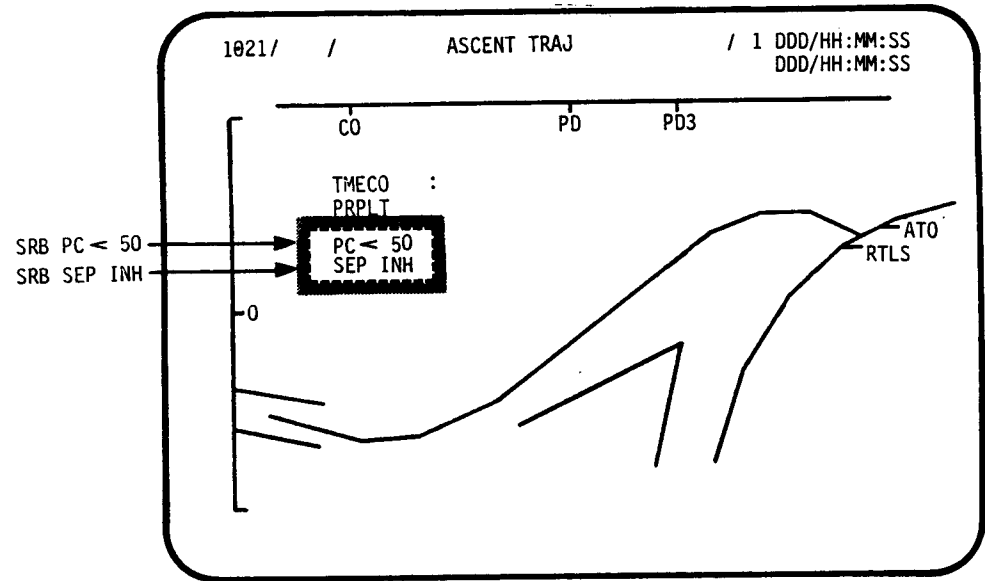


Figure 6-4. PASS ASCENT TRAJ Display in MM 102

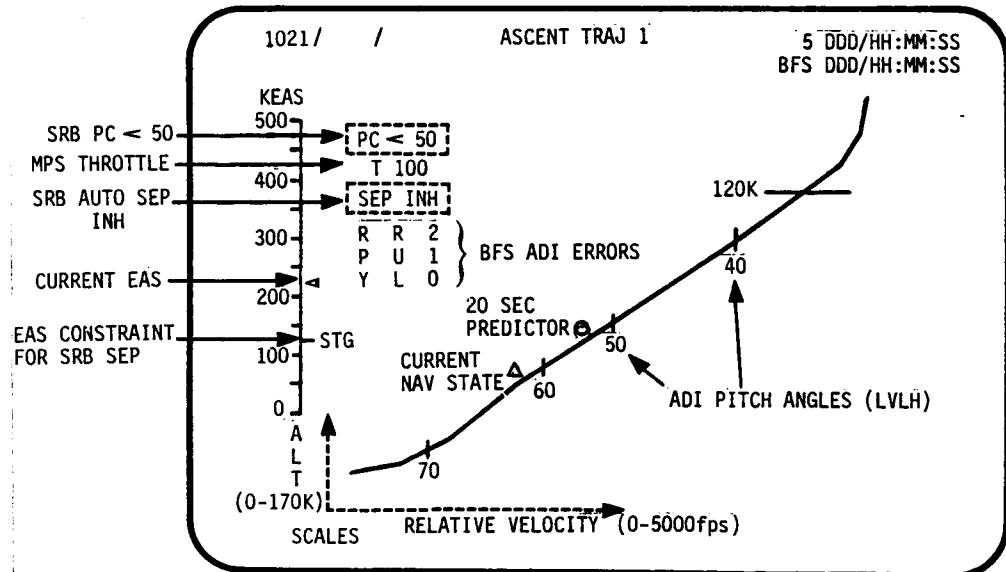


Figure 6-5. BFS ASCENT TRAJ 1 Display

## 6.3 SECOND STAGE

**6.3.1 Second Stage Description.** Second stage ascent begins at SRB separation and extends through MECO and ET separation. The GNC software is in Major Mode 103 at this time, calculating the required main engine steering commands in order to achieve preflight-defined MECO conditions. One of the objectives of the MECO constraints is to ensure that the separated ET lands in an unpopulated area, usually the Indian Ocean. During this phase, the acceleration on the vehicle reaches 3 g's at approximately 7:40 Mission Elapsed Time (MET). The main engines are then throttled down to maintain 3 g's and stay within structural and physiological constraints. About 7 seconds before MECO, the main engines begin throttling down in preparation for MECO, which occurs at  $\approx$  8:34 MET. At MECO, the vehicle is maintained in attitude hold using the RCS jets until ET separation, which occurs 18 seconds after MECO. During this timeframe, the ET SEP sequence software sends the necessary commands to unlatch and retract the ET umbilical, monitors engine preclude closure, etc. After the necessary events occur, the ET SEP sequence checks vehicle rates and the status of certain valves and MDM's. If these are satisfactory, the ET is separated from the vehicle. If not, the crew must follow some special procedures before manually commanding ET separation. After ET SEP, a 4 fps -Z translation is initiated using RCS jets to ensure that the Orbiter and ET do not recontact. When this is complete, the software automatically transitions to Major Mode 104.

**6.3.2 Second-Stage GNC Software Functions.** Second-stage guidance uses a cyclic, closed loop scheme to calculate the necessary commands to take the mated vehicle to a specified set of target MECO conditions. These conditions include cut-off velocity, radius from the Earth, flight path angle, orbital inclination and longitude of the ascending node. The name of the targeting scheme is PEG 1. Guidance also governs the main engine throttle commands such that 3 g's are not exceeded. The predicted time of MECO (TMECO) is calculated and displayed to the crew on the ASCENT TRAJ display. Following SRB SEP, it may take the PEG 1 guidance algorithm several cycles to converge and for TMECO to become stable. Forty seconds prior to MECO, guidance begins targeting only for the desired cutoff velocity, ignoring position constraints.

Second-stage navigation continues as it did during first stage; i.e., it propagates a state vector using the super-G integration scheme.

Second-stage flight control continues through MECO as it did during first stage with the exception that there are no SRB's. After SRB SEP, attitude hold is commanded until guidance converges. At MECO, a different digital autopilot called the transition (or TRANS) DAP, becomes active. It initially sends attitude hold commands. ET SEP is automatically commanded 18 seconds after MECO (Assuming no ET SEP constraints have been violated). The DAP immediately sets commands to fire the Orbiter -Z RCS jets resulting in the Orbiter translating in the -Z direction (away from the ET). When -4 fps  $\Delta V_z$  is reached, the fire commands are removed. The TRANS DAP is used from MECO through Major Modes 104, 105, 106, 301, 302, and 303.

**6.3.3 Second-Stage GNC Crew Interface.** During second stage, it is the crew's responsibility to monitor the onboard systems to ensure that the major GNC events occur correctly and on time. These events include closed loop guidance convergence, 3-g throttling, MECO, ET Sep, and the -Z translation following ET Sep. To monitor these events, the crew uses the Ascent Procedures in the Ascent Flip Book, the ascent GNC cue cards, the dedicated displays, the MAIN ENGINE STATUS lights on Panel F7, the PASS ASCENT TRAJ display and the BFS ASCENT TRAJ 2 display. Figures 6-6 and 6-7 describe the information available on these two displays during second stage.

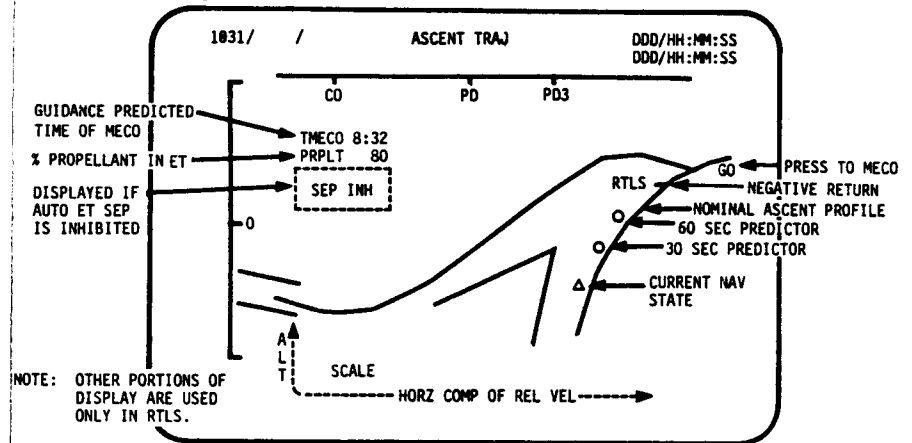


Figure 6-6. PASS ASCENT TRAJ Display in MM 103

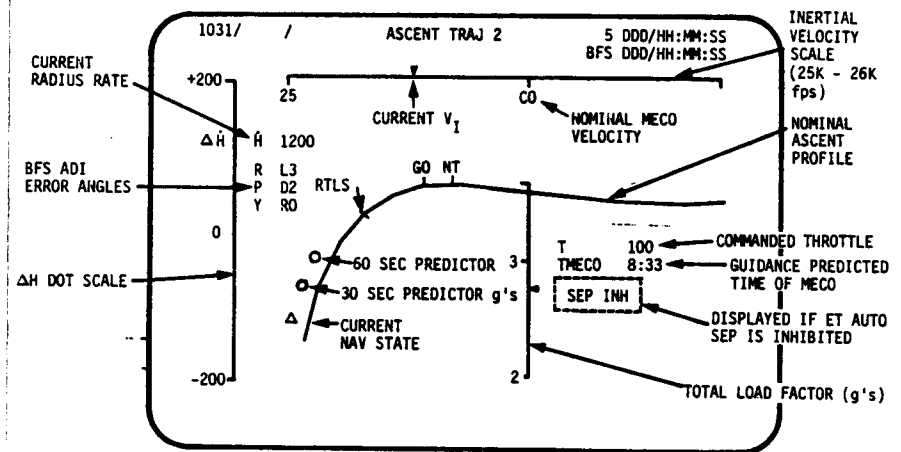


Figure 6-7. BFS ASCENT TRAJ 2 Display



The crew can monitor for guidance convergence by noting if the guidance-computed time of MECO (TMECO) is stabilized on the ASCENT TRAJ display. If not, the crew is required to take manual control of the vehicle. They can also ensure that 3 g's are not exceeded via the BFS ASCENT TRAJ 2 display as well as the ACCEL tape on the AMI. The crew can monitor MECO velocity on the BFS ASCENT TRAJ2 display as well as on the M/VEL tape on the AMI. MECO is detected by the illumination of three red MAIN ENGINE STATUS lights and by the MPS Pc meters on panel F7 going to zero. The crew's first indication of an underspeed is if the actual MECO velocity is much below the planned velocity. The crew then watches for ET Sep. When this occurs, the MPS status lights extinguish. If an ET Sep Inhibit is encountered, the crew must take the appropriate actions given in the checklists. The crew will know that the -Z translation following ET Sep is complete if the software automatically transitions to Major Mode 104. After this is complete, the crew may be required to translate in the +X direction (using the Translational Hand Controller) for 11 seconds to allow the ET camera to photograph the ET.

At specified points during the second stage, Mission Control will make "voice calls" to the crew indicating their status with respect to aborts. For example, the Negative Return call indicates that it is too late to select an RTLS.

#### 6.4 ORBIT INSERTION - OMS1/OMS2 BURNS

6.4.1 Orbit Insertion Description. Following MECO, the Orbiter is at approximately 80 n.m. altitude with an inertial velocity of  $\approx 25,660$  fps (the altitude and velocity vary between missions). This puts the Orbiter in a sub-orbital trajectory so that the ET will re-enter following separation. In order to boost the Orbiter to a viable orbit which will not degrade appreciably during the mission and which will satisfy mission objectives, two propulsive burns are made with the OMS engines. The first burn is referred to as OMS 1 and boosts the Orbiter to the desired apogee. The second burn is called OMS 2 and typically circularizes the orbit. The optimal orbital altitude, (the altitude which satisfies mission and payload goals), is determined before launch. During flight, however, problems such as main engine and SRB performance loss, OMS systems problems such as propellant leaks or certain electrical power system failures may prevent the vehicle from achieving optimal orbit. In these cases, the OMS-1 and 2 burns will be changed to compensate for the failure by selecting a delayed OMS 1, AOA, or ATO abort option. The ascent abort options will be discussed later.

The main events which occur during the orbit insertion phase include the following:

- Execution of the OMS 1 burn, typically two minutes past MECO
- An MPS propellant dump which begins during OMS 1. After the dump is complete, the main engine bells are positioned for entry
- APU shutdown after OMS 1 and after the main engine bells are stowed
- MPS power down and "vacuum inerting" to ensure all traces of MPS propellant are vented into space. Also, the ET umbilical doors are closed

- Execution of the OMS 2 burn, typically at 45 minutes MET
- Post burn reconfiguration which prepares the vehicle for orbit activities.

6.4.2 Orbit Insertion GNC Software Functions. During the orbit insertion phase, the guidance software is used to target the OMS 1 and OMS 2 burns. The primary targeting scheme used is called PEG 4. This is a cyclic, closed loop scheme which computes the required delta velocity ( $\Delta V$ ) in order to achieve a target specified by a time of ignition (TIG), a position relative to the Earth and at which the Orbiter will have a flight path angle equal to zero. Usually, the target point is at the apogee of the post-burn orbit, or else the resultant orbit is circular. In addition, the software is constrained to optimize fuel usage. For the OMS 1 burn, it also must ensure that the burn attitude at ignition is near the MECO attitude. The PEG 4 targeting scheme can compensate for the loss of one OMS engine or an RCS downmode during the burn if it detects the OMS loss. Prior to and during the burn, guidance computes the remaining burn time (TGO) and total remaining delta velocity (VGO).

Another guidance targeting option available for use during OMS 1 and 2 is the "external  $\Delta V$  mode called PEG 7. In this mode, the burn is specified by a required TIG and a delta velocity defined in the local vertical coordinate system at ignition ( $\Delta V_X$ ,  $\Delta V_Y$ ,  $\Delta V_Z$ ). Commands are then generated to thrust along the original thrust vector until the specified  $\Delta V$  is achieved. There is no target position or velocity feedback during this burn.

Navigation during the orbit insertion phase is identical in processing to that used in mainstage whenever the IMU sensed acceleration is greater than the noise threshold. If the IMU-sensed acceleration drops below the threshold, navigation ignores the IMU data and propagates the state vector using an atmospheric drag model of the Orbiter.

Insertion flight control is accomplished using the Transition Digital Auto-pilot (TRANS DAP) which becomes effective after MECO and until transition to OPS 2. The TRANS DAP uses commands from guidance for automatic maneuvers to burn attitude using the RCS jets. During the OMS 1 and 2 burns it uses the OMS engines and RCS jets as required. It also receives commands from the crew via the forward THC and RHC's. The TRANS DAP takes these commands and converts them into the appropriate RCS commands. It then monitors the resultant attitude and attitude rates and sends the necessary commands to achieve the targeted attitude and attitude rate within pre-mission specified deadbands.

6.4.3 Orbit Insertion Crew Interface. During the orbit insertion phase, the crew can modify vehicle performance using the onboard CRT displays, the ADI, and various crew station switches and meters. The CRT displays primarily used during this timeframe are OMS 1 MNVR EXEC and OMS 2 MNVR EXEC. (See figure 6-8.) The background format of these displays is identical to that used for the abort OMS 1 and 2 burns and for the deorbit burns - only the titles change. Certain portions of the displays are not active in all flight phases, however. In OPS 1, the crew uses these displays primarily to set up the targets, maneuver to the required attitude, and execute the OMS 1 or OMS 2 burn using two, one, or no OMS engines (RCS only). The data shown on the display will indicate whether there has been a performance loss requiring an abort (ATO, AOA). This determination is made by the ground controllers but is backed up by the crew using information on these displays and the OMS 1 or OMS 2 TARGET cue card.

Prior to the burn, the crew goes to the OMS 1 (or OMS 2) BURN CUE CARD (See figure 6-9) which reminds them of any necessary switch reconfigurations and which informs them of required actions in the event of an OMS system failure.

18

1041/	OMS 1	MNVR EXEC	1 000/00/09:47
OMS BOTH 1X			000/00:00:49
L 2	BURN ATT		EXEC
R 3	24 R 135	ΔVTOT	245.1
RCS SEL 4	25 P 279	TGO	02:34
5 TV ROLL 180	26 Y 344	VGO X + 242.14	
TRIM LOAD	MNVR 27	Y + 32.05	
6 P (-) 0.4	REI 4005	Z + 20.79	
7 LY (-) 5.7	TTA 15:01		
8 RY (+) 5.7	GMBL	HA HP	
9 WT 244313	L R	TGT 160 + 51	
10 TIG	P +0.4 + 0.4	CUR 82 - 1	
Q/Q:10:12.1	Y -5.7 + 5.7		
TGT PEG 4			
14 C1			
15 C2 (±)0.0000	PRI 28* 29*	35 ABORT TGT -0	
16 HT 185.453	SEC 30 31		
17 T 186.202	OFF 32 33	FWD RCS	
18 PRPLT()		ARM 36	
TGT PEG 7	GMBL CK 34	DUMP 37	
19 AVX (L)---		OFF 38*	
20 AVY (L)---		SURF DRIVE	
21 AVZ (L)---		ON 39	
LOAD 22/TIMER 23		OFF 40*	

Figure 6-8. OMS 1 MNVR Display

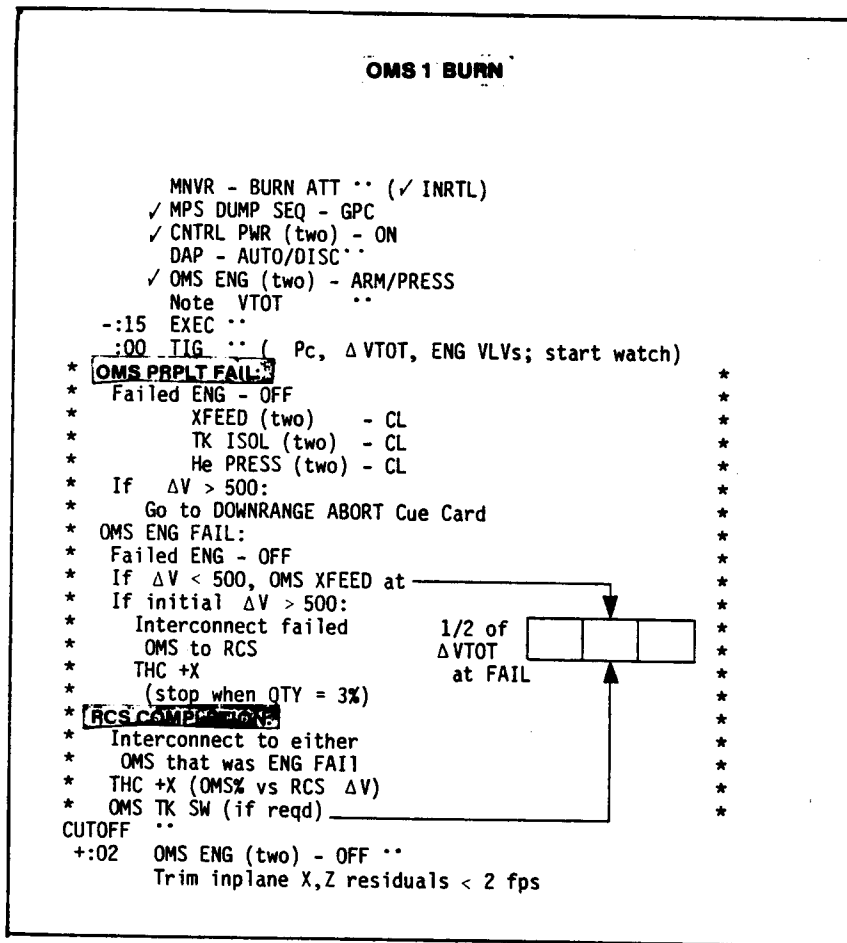


Figure 6-9  
OMS Burn Cue Card

During the burn the crew is responsible for monitoring the OMS system via cockpit meters (such as the OMS PC meter on F7) and other CRT displays. The crew also monitors the burn progress on the MNVR EXEC display. When the burn is complete, the crew turns the OMS engines and trims any velocity residuals to less than 2 fps using the RCS system.

The crew interfaces with the TRANS DAP via the forward hand controllers (THC, RHC), indirectly through entries to the OMS 1 (2) MNVR EXEC display (for OMS burn execution, auto maneuvers to burn attitude, etc.) and via the DAP panel pushbuttons on panel C3 (See figure 6-10).

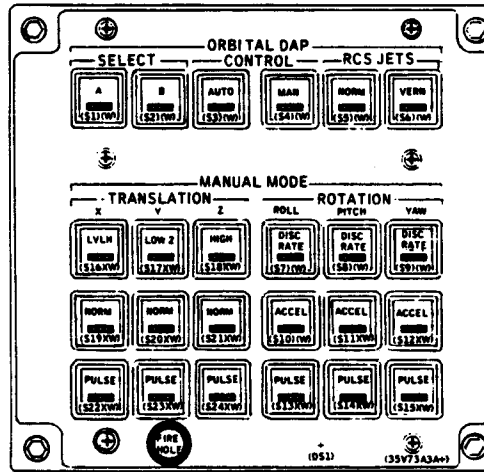


Figure 6-10. DAP Panel

For rotations, the crew can select either AUTO or MANUAL control via the DAP panel pb's or by moving the RHC. The capability exists in MANUAL to rotate in any axis in either a "PULSE" mode, in which each RHC deflection results in a single burst of jet fire, or in a "DISC" (discrete rate) mode, in which RHC deflection results in a specified rate being commanded in that axis for the entire time the RHC is deflected. It is also possible to go to a FREE DRIFT mode in which no RCS jets are fired, or to an ATTITUDE HOLD mode in which the DAP sends commands to maintain the current attitude with null rates within pre-mission-specified deadbands. Also, if the RHC is deflected beyond a certain point, continuous RCS jet firings will result.

In translation, movement of the THC results in continuous jet firings. Note that there are many subtleties in the TRANS DAP which are beyond the scope of this workbook and which will be discussed in the IOD G&C 2102 workbook.

## 6.5 ASCENT ABORT OVERVIEW

The preceding pages described the events, crew interface, and GNC software processing involved in a nominal ascent. However, if failures result in a major loss of performance (SSME fail, OMS engines fail, etc.), and the desired orbit cannot be safely achieved, or if a major system problem (cabin leak, loss of 2 freon loops for cooling) occurs which makes it necessary to land as quickly as possible, the mission is aborted. Several abort modes have been developed to optimize the situation with regard to Orbiter, crew, and payload recovery as well as for mission success. Mission Control is prime for calling these aborts, as they have a more precise knowledge of the Orbiter's state vector (through the use of sophisticated tracking equipment and ground computer resources) than the crew can obtain from the onboard navigation system. Before MECO, mission control makes periodic calls to the crew to tell them which abort mode is (or is not) available. For example, mission control will call "NEGATIVE RETURN" when it becomes impossible to achieve the RTLS abort mode. If ground communications are lost, the crew has onboard methods such as cue cards, dedicated displays, and GNC CRT display information to determine the current abort region (See figure 6-11).

NO COMM MODE BOUNDARIES	
2 ENG TAL	- 6800 (Inertial Velocity)
NEG RETURN (100)	- 8200
PRESS TO MECO (100)	- 11900
SE DAKAR (107)	- 12500
SE DAKAR (100)	- 13800
SE PRESS TO MECO	- 17700
LAST DAKAR TAL -	
2 or 3 engine (65)	- 19000
1 engine (100)	- 20000
LAST MANUAL DAKAR	- 23600

STS-5/ASC/BAS 1

Figure 6-11. Cue Card Showing Abort Region Boundaries

The onboard determination of abort mode is impaired to some degree because it depends on onboard navigation which may be in error and because checklist data is based on preflight-determined vehicle performance and atmospheric conditions.

Mission control also has more information than the crew about the Orbiter's onboard systems health, although the crew does have cue cards and displays that allow them to determine what type of abort would be necessary for certain failures. (See figure 6-12)

SYS FLIGHT RULES		
	RTLS/TAL	MAN DAKAR
<b>OMS</b>	- 2 He TKS 1 OX & 1 FU TKS (diff pods) 2 OX of 2 FU TKS (diff pods)	Qty < OMS 1
<b>RCS</b>	- 2 AFT He/PRPLT TKS (Same PRPLT)	Qty > AOA
<b>2 FREON LOOPS</b>		
<b>CABIN LEAK</b>	- Bkup dP/dT > .5	Bkup dP/dT > .96

Figure 6-12. Cue Card Showing Abort Criteria for Pre-MECO Systems Failures

The necessary pre-MECO abort mode (e.g., RTLS, TAL) is selected by the crew using the Abort Rotary Switch and pushbutton located on panel F6.

Post MECO, mission control is still prime for calling an abort due to loss of performance or system problems, although the crew also bears the responsibility of determining the necessary abort mode for cases in which there is a loss of ground communication. To do this, the crew uses its knowledge of the onboard systems health as well as performance information obtained via the OMS 1 MNVR EXEC display (the  $\Delta V$  TOT required for OMS 1 and the targeted perigee altitude) to determine the required abort mode as indicated on the OMS 1 (2) TARGETING CUE CARD (See figure 6-13).

FAILURE	OMS 1 TARGET	OMS 2
DELAY DO NOT DELAY BURN IF CUR HA $\leq$ 70 OMS - 1 He TK 1 PRPLT TK 2 OMS ENG OMS - 2 H2 TKS (I'nect to RCS) EPS - MNA & B MNA & C ET SEP	DELAYED ATO: TIG 20:33 H <sub>r</sub> 111 $\theta$ <sub>r</sub> 235	N/R
FEEDLINE DISC VLV	DELAYED NOM: TIG 20:33 H <sub>r</sub> 169 $\theta$ <sub>r</sub> 240	NOM
OMS - 2 He TKS - 1 OX & 1 FU TK diff PODS - 2 OX or 2 FU TK diff PODS	DELAYED AOA-S: TIG 20:33 H <sub>r</sub> 100 $\theta$ <sub>r</sub> 165	AOA-S
ON TIME 2 APU/HYD 2 FREON LOOPS 2 H2 LOOPS 3 AV BAY FANS (BAYS 1,2) CABIN LEAK (BKUP dP/dT > .12)	NOMINAL TIG MECO+2 H <sub>r</sub> 165 $\theta$ <sub>r</sub> 167	AOA/ AOA-S

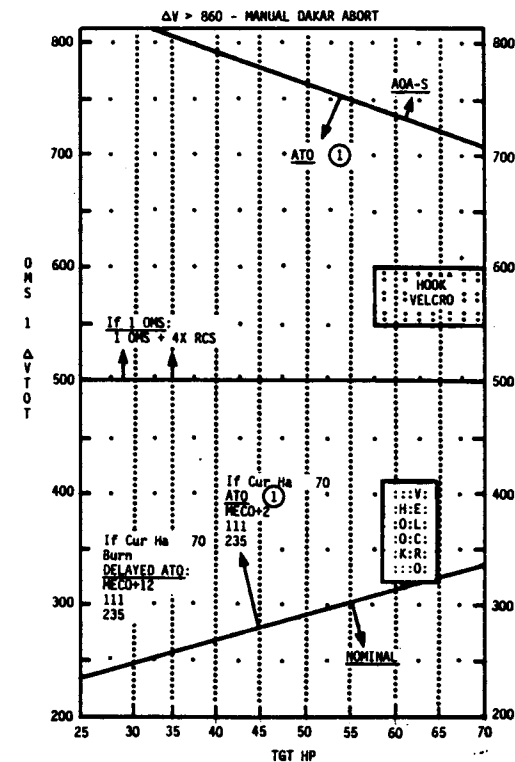


Figure 6-13. OMS 1 Targeting Cue Card

The crew selects the abort target for the OMS 1 and 2 burns one of three ways - using the Abort Rotary switch and pushbutton, entering a special ID to the OMS 1 (2) MNVR EXEC display, or by manually typing in the required parameters to the MNVR EXEC display; i.e., TIG,  $H_t$ ,  $\theta_t$ , as listed on the cue card and in the ASCENT CHECKLIST.

Which abort mode is selected depends upon the cause and timing of the failure causing the abort and which mode is safest or improves mission success. If the problem is a main engine failure, the crew/MCC selects the "best" option available at the time the engine failed. Figure 6-14 shows the typical options.

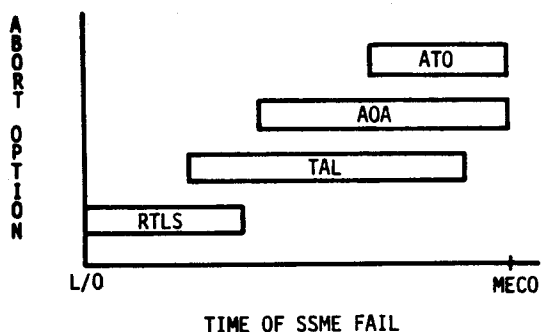


Figure 6-14. Typical Abort Mode Options

If the problem is a systems failure which jeopardizes the vehicle, the fastest abort mode which results in the earliest vehicle landing is chosen. RTLS and TAL are the quickest options ( 35 minutes) whereas an AOA requires approximately 90 minutes. Which of these is selected depends upon the time of the failure. See figure 6-15 to see which abort modes are available with three good main engines.

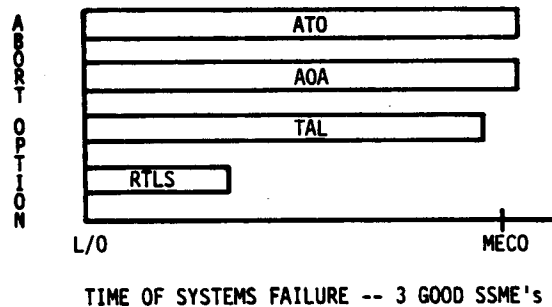


Figure 6-15. Abort Mode Options with 3 Good SSME's

Other combinations of failures make the abort selection process more difficult. The following paragraphs describe the planned ascent abort modes, their causes, and significant events involved in their execution.

6.5.1 Abort to Orbit (ATO). An ATO is an abort mode used to boost the Orbiter to a safe orbital altitude in cases where performance has been lost and it is impossible to reach the planned orbital attitude. If a main engine fails in a region which results in a MECO underspeed, the MCC will determine that an abort mode is necessary and will inform the crew. In addition, the crew can verify, via the nominal OMS burn solutions on the OMS 1 MNVR EXEC display and via the OMS 1 TARGETING CUE CARD, that an ATO is required. To achieve this, they load the ATO OMS 1 targets on the OMS 1 MNVR EXEC display and burn them instead of the nominal targets. Similarly, they load the ATO OMS 2 burn targets and use them for OMS 2. This results in the Orbiter being placed in a 105 x 105 nm circular orbit.

Another cause for an ATO is loss of OMS performance due to various failures, such as loss of 2 OMS engines, loss of 1 OMS propellant tank, or loss of MNA and B. In these cases, and in cases where the ET does not separate, the OMS 1 burn is delayed about 10 minutes and the ATO OMS 1 targets are used. This results in an approximately 105 x 80 nm orbit, which is considered a safe orbit for 24 hours. Thus, an OMS 2 burn is not immediately necessary. The delayed ATO is accomplished by loading the delayed ATO targets in the OMS 1 MNVR EXEC display and burning only the OMS 1 burn. If an underspeed exists at MECO such that an abort is required (ATO or "worse"), the OMS 1 burn must not be delayed. An underspeed results in apogee moving close to the post MECO Orbiter position. Delaying OMS 1 would result in the burn being performed after apogee, which requires increased delta velocity and thus more propellant.

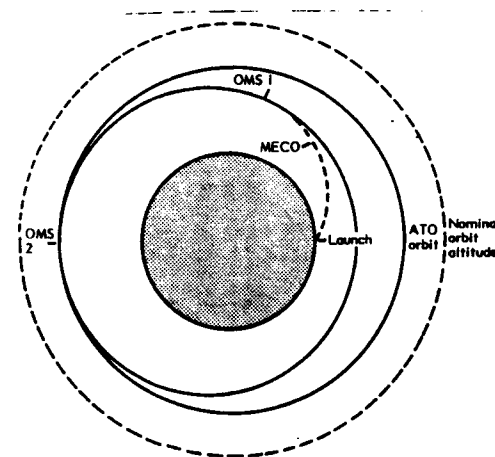


Figure 6-16. ATO Abort Option

6.5.2 Abort Once Around (AOA). The AOA abort mode is used in cases in which vehicle performance has been lost to such an extent that either it is impossible to achieve a viable orbit or not enough OMS propellant is available to accomplish the OMS 1, OMS 2 and deorbit burns. In addition, an AOA is used in cases in which a major systems problem (cabin leak, loss of cooling) makes it necessary to land quickly. In the AOA abort mode, one OMS burn is made to adjust the post-MECO orbit such that a second OMS burn will result in the vehicle deorbiting and landing at the AOA landing site (Northrup, Edwards, KSC). Thus, an AOA results in the Orbiter circling the Earth once and landing approximately 90 minutes after lift-off.

Several different options are available in performing an AOA. The OMS 1 targets are selected based upon whether the AOA was caused by a MECO underspeed, a systems problem, or an OMS/RCS performance problem. The OMS 2 targets are selected one of two ways depending on whether a MECO underspeed exists and its magnitude. (Recall that the AOA OMS 2 burn is really a deorbit burn). One set of targets results in a trajectory more steep than the other as the vehicle enters the atmosphere (entry interface) and is thus referred to as a "steep AOA." This is a more normal trajectory and stays well within the vehicle thermal limits once the atmosphere is penetrated. It does require more delta velocity ( $\Delta V$ ) and thus more propellant for the deorbit burn. Thus, in cases where the MECO underspeed is severe or if both OMS He tanks have failed, the "shallow AOA" targets are used, resulting in a more shallow trajectory at entry interface and placing the vehicle closer to the skipout boundary and vehicle thermal limits.

The crew determines that an AOA is required by ground call and by checking the OMS 1 target solution against the OMS TARGETING cue card. Depending upon what kind of AOA is required, the crew loads the required OMS 1 targets and executes the burn. They then mode the software to OPS 3 and load the appropriate OMS 2 (deorbit) targets. After the burn is executed, the crew flies to a landing at the replanned site much as they would for a nominal entry.

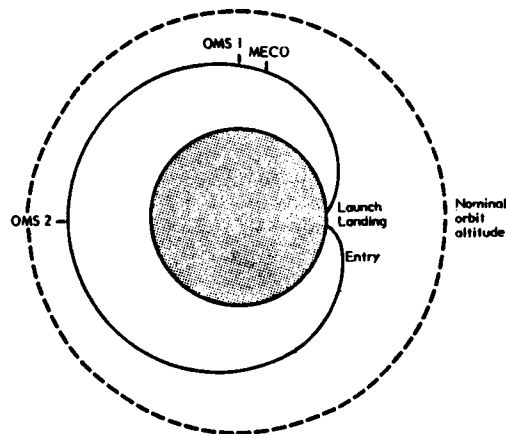


Figure 6-17. AOA Abort Option

6.5.3 Transatlantic Abort Mode (TAL). The Transatlantic Abort Landing mode (TAL) was developed during the early OFT flights to improve the options available in cases where:

1. An engine failure after the last RTLS opportunity but before the first time that an AOA can be accomplished with only two main engine.
2. Two engines fail before it is possible to achieve an AOA with only one SSME.
3. A major Orbiter systems failure occurs after the last RTLS opportunity, making it imperative to land as quickly as possible; i.e., a large cabin leak or cooling system failure.

In a TAL abort, the vehicle continues on a ballistic trajectory across the Atlantic Ocean to land at a predetermined runway. Landing occurs approximately 45 minutes after launch. The landing site is selected near the nominal ascent ground track of the Orbiter in order to make the most efficient use of SSME propellant. The landing site also must have the necessary runway length, weather conditions, and U.S. State Department approval. The two landing sites which currently have been identified are Rota, Spain, and Dakar, Senegal (on the west coast of Africa).

The onboard software is designed to accomplish a TAL automatically under most circumstances. To select the TAL abort mode the crew must place the ABORT ROTARY switch in the "TAL/AOA" position on panel F6 and depress the ABORT push-button prior to MECO. (Depressing it post-MECO selects the AOA abort mode). The crew will then see the title on the "ASCENT TRAJ" display change to "TAL TRAJ". The TAL software begins sending commands to steer the vehicle toward the plane of the landing site. It also rolls the vehicle heads up pre-MECO and sends commands to begin an OMS propellant dump (through the OMS engines and the RCS Jets). This dump is necessary to increase vehicle performance (by decreasing weight), to place the c.g. in the proper place for vehicle control, and to decrease the vehicle's landing weight. The software also targets for a specified MECO configuration that will ensure that the vehicle has the proper energy state required to glide to the runway. The crew must mode the software to OPS 3 after ET separation and set up the entry display for a TAL landing site. From that point on, the TAL is handled similarly to a nominal entry.

There are cases in which a TAL is desired but in which it is not possible or desirable to use the automatic TAL software. For example, it is only possible to select an automatic TAL pre-MECO. Therefore, if 3 engines fail, an automatic TAL cannot be selected. For these cases, a "manual" TAL capability may be possible. In this mode, the crew must take manual control of the vehicle and exercise special procedures in order to make it to the landing site without violating any vehicle constraints.

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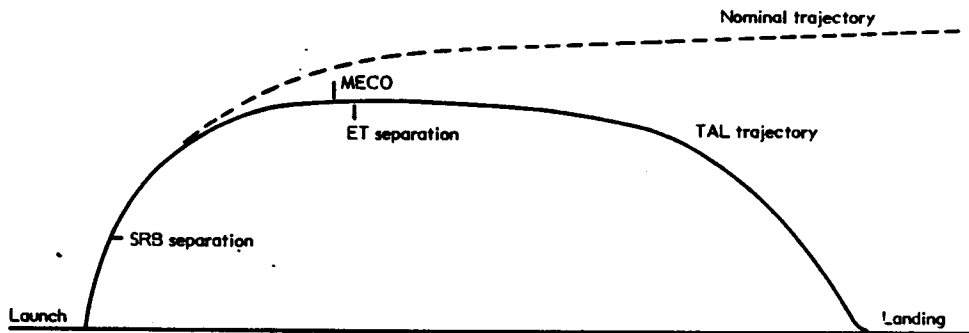


Figure 6-18. TAL Abort Option

6.5.4 Return to Launch Site (RTLS). The Return to Launch Site abort mode is designed to allow the return of the Orbiter, crew, and payload to the launch site, KSC, approximately 25 minutes after lift-off. The RTLS profile is designed to accommodate the loss of thrust from one engine between lift-off and  $\approx 4$  m 20s, at which time not enough MPS propellant remains to return to the launch site. An RTLS is also used in cases in which a time-critical failure occurs which requires the earliest possible landing, such as a cabin leak or major cooling system failure. Another cause for an RTLS is the complete loss of performance in the OMS or RCS systems. In the last two situations, all three engines are still burning which leads to the term 3-engine RTLS.

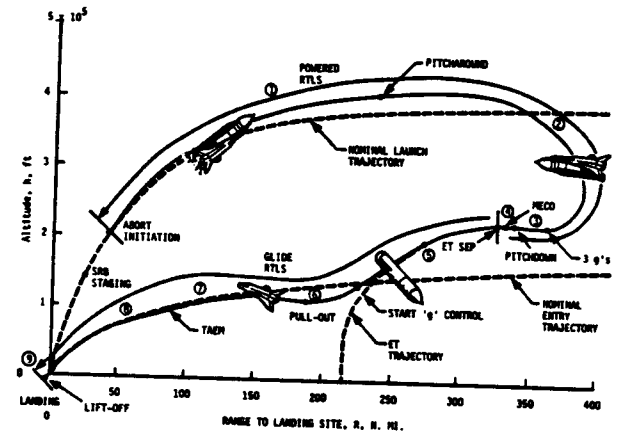


Figure 6-19. RTLS Flight Phases

An RTLS can be considered to be composed of 3 stages - a powered stage during which the main engines are still burning, an ET separation phase, and the glide phase during which the Orbiter glides to a landing at KSC. The powered RTLS phase, often called PRTLS, begins with the crew selection of the RTLS abort which is done after SRB separation. The crew selects the abort mode by positioning the abort rotary switch on panel F6 to "RTLS" and depressing the "ABORT PB". The time at which the RTLS is selected is dependent upon the reason for the abort. For example, a 3 engine RTLS is selected at the last time,  $\approx 3:45$  MET whereas an RTLS selected due to an engine out at lift-off is selected at the earliest time,  $\approx 2:20$  MET (after SRB SEP). Upon selection of the RTLS, the ASC TRAJ display title changes to RTLS TRAJ and several portions of the display relate to RTLS become active. Also, the software major mode changes from 103 to 601. After RTLS is selected, the vehicle continues downrange to dissipate excess MPS propellant. The goal is to leave only enough MPS propellant to be able to turn the vehicle around, fly back towards KSC and achieve the proper MECO conditions such that the vehicle can glide to KSC after ET separation. During

the outbound phase, a pitch-around maneuver is initiated (the time depends in part upon the time of the engine failure) to orient the Orbiter/ET configuration to a "heads up" attitude, pointing toward the launch site. At this time, the vehicle is still moving away from the launch site but the main engines are now thrusting to null the outbound velocity and begin moving the vehicle towards the launch site. In addition, excess OMS and RCS propellant are "dumped" by continuous OMS and RCS jet firings to improve the Orbiter weight and c.g. for the glide phase and landing. The guidance software computes the necessary MPS throttle commands to keep the vehicle under 3 g's and to reach the desired MECO point with less than 2% excess propellant remaining in the ET. At MECO -20 seconds, a pitch-down maneuver (called Powered Pitch Down) takes the mated vehicle to the required ET separation attitude and pitch rate. After MECO is commanded, the ET separation sequence begins, including a -Z translation after ET SEP which ensures that the Orbiter does not recontact the ET and that the Orbiter has achieved the necessary pitch attitude to begin the glide phase of the RTLS.

After the -Z translation maneuver is complete, the software goes to MM602 and the glide phase of the RTLS (called GRTLS) begins. First, guidance sends commands to stabilize the trajectory after ET separation and adjust the Orbiter's angle of attack for range control. From then on, the RTLS is handled similarly to a normal entry.

## SECTION 7

### ORBIT GNC OPERATIONS

The major GNC-related orbit tasks include achieving the proper position, velocity, and attitude necessary to accomplish the mission's objectives. To do this, the GNC software maintains an accurate state vector, targets, and accomplish maneuvers to specified attitudes and positions, and points a specified Orbiter body vector at a target. These activities are planned with several constraints in mind, including fuel consumption, vehicle thermal limits, payload requirements, and rendezvous/proximity operations considerations.

The GNC software for the majority of on-orbit operations is referred to as OPS 2, which is further subdivided into Major Mode 201 (in which the majority of attitude and pointing operations occur) and Major Mode 202 (in which OMS translations are targeted and executed). GNC software is also used for the Flight Control System checkout prior to deorbit (OPS 8). In this configuration, the crew checks out the NAVAID systems, the dedicated displays, the RCS jets, the aerosurfaces, and the hand controllers.

#### 7.1 ORBIT NAVIGATION

The navigation software available in OPS 2 has several important features. As before, it propagates the Orbiter state vector using the Super-G integration scheme. During coasting flight, the software uses a model of atmospheric drag acceleration to propagate the state vector. If translational thrusts are anticipated, the crew can set a flag for navigation to use IMU-sensed acceleration if above a noise threshold value. When this flag has been set via an item (AVG G) on an orbit display (REL NAV), the crew can monitor the thrust which is sensed (figure 7-1).

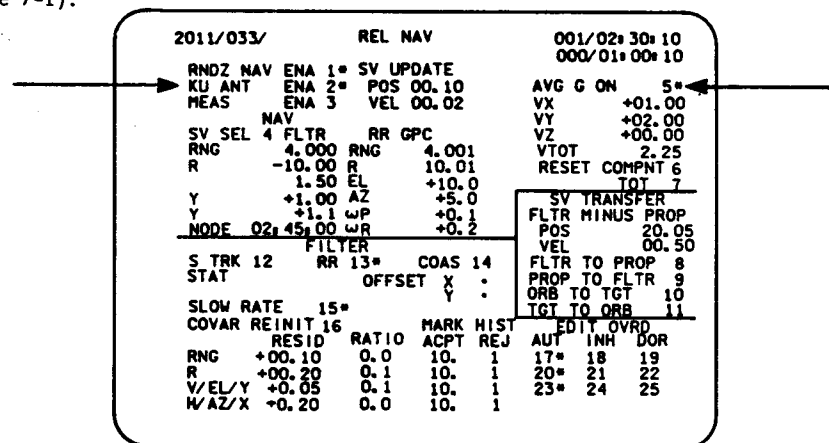


Figure 7-1. Rel Nav Display



Another navigation option which may be available on-orbit is called Rendezvous Navigation. When this option is enabled by a crew input on the REL NAV display, the software maintains a target state vector as well as the Orbiter state vector. In this mode, it is possible for navigation to use external sensor data from the Star Tracker, COAS, or rendezvous radar (based upon reasonableness tests) in computing the Orbiter target state vector. This is based upon the assumption that the target state vector is accurate.

On orbit, the accuracy of the Orbiter state vector depends upon the accuracy of the IMU's and the accuracy of the modeled drag acceleration. As there is currently no method on board to independently compute corrections to the state vector, periodic updates are sent from the MCC to correct any errors that develop with the on-board state vector. In the future, a system is planned in which TACAN data may be used on orbit to update the Orbiter state vector.

Another feature available in navigation in OPS 2 is the Landing Site Update Function. This feature allows the crew to select different runways to be used in the entry guidance computations. Crew interface with this capability is via the Universal Pointing (UNIV PTG) display (figure 7-2).

## 7.2 ORBIT GUIDANCE

The on-orbit software has several features. One is called Universal Pointing is used to compute attitude changes required to "point" a specified Orbiter body axis at a specified target, to maneuver to a predetermined attitude, to rotate the Orbiter about a specified body axis, or to maintain attitude hold. The crew interacts with this software via the UNIV PTG display in Major Mode 201 (Figure 7-2). The complete capabilities of the Universal Pointing software are available only in Major Mode 201, although a subset of these are available in Major Mode 202 and OPS 8.

Another guidance feature is PEG 7, or external  $\Delta V$ , targeting for OMS or RCS burns. This targeting scheme is identical to that used in OPS 1 and 3 (section 6.4.2). In this mode, guidance sends the commands to flight control to execute a burn specified by an ignition time (TIG) and delta velocities in the local vertical coordinate system at ignition. Commands continue to be generated until the original  $\Delta V$  requirement is met. This option is available in Major Mode 202 via the ORBIT MNVR EXEC display (Figure 7-3).

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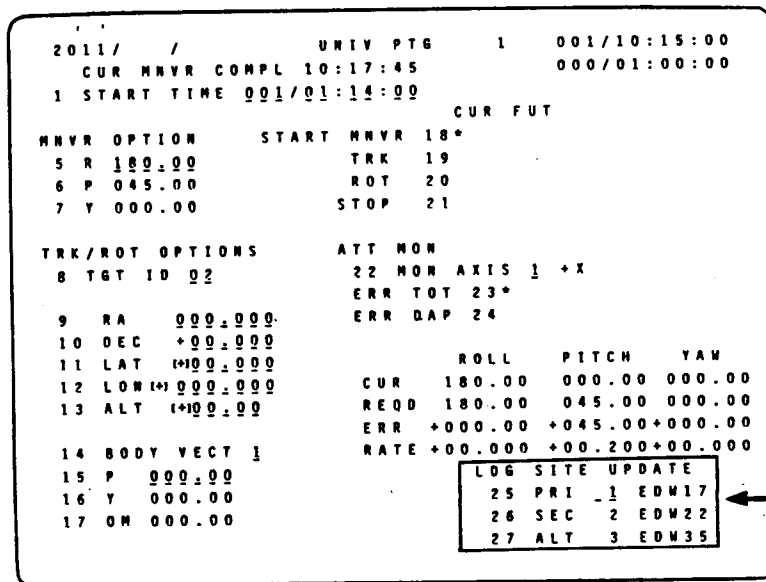


Figure 7-2. UNIV PTG Display

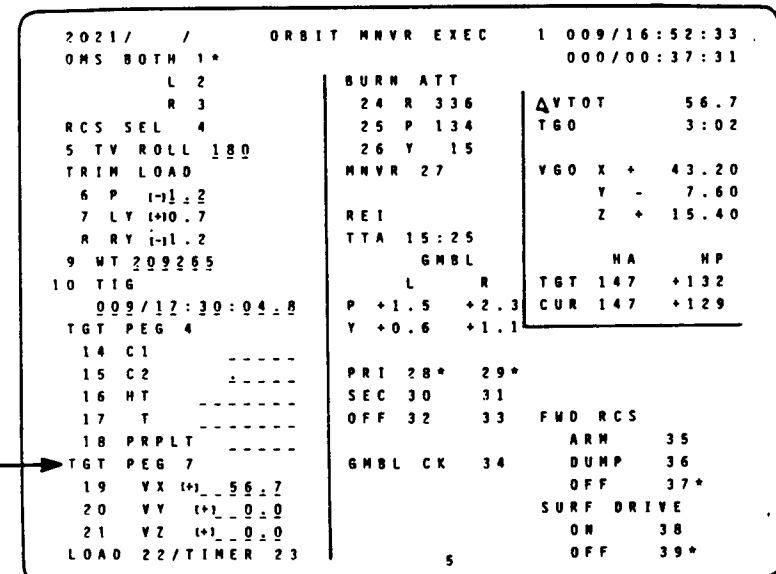


Figure 7-3. ORBIT MNVR EXEC Display

The third guidance capability is an on-orbit targeting scheme which is used to compute the burn parameters required to move the Orbiter to a specified target offset position in a given amount of time. This capability is used to do on-board targeting of rendezvous maneuvers. This feature is enabled via the ORBIT TARGETING CRT display (Figure 7-4). The actual burn is still accomplished via the ORBIT MNVR EXEC display.

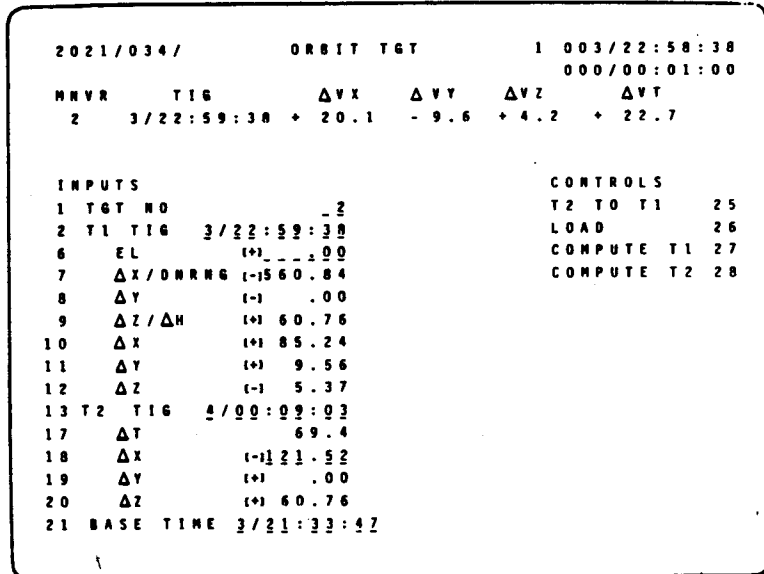


Figure 7-4. ORBIT TGT Display

### 7.3 ORBIT FLIGHT CONTROL

The orbit flight control software includes an RCS DAP, an OMS Thrust Vector Control (TVC) DAP, a module to calculate vehicle attitude called an attitude processor, as well as logic to govern which DAP is selected. The attitudes calculated by the Attitude Processor are displayed on the ADI along with another crew display, UNIV PTG (Figure 7-2) which is available in Major Mode 201. The vehicle attitude is used by the digital autopilot to determine attitude and rate errors.

The RCS DAP is used in OPS 2 at all times except when an OMS burn is in progress. This DAP controls vehicle attitudes and rates using RCS jet fire commands. Either the larger, primary jets or the less powerful vernier jets are used for rotational maneuvers, depending on whether NORM or VERN is selected on the panel C3 ORBITAL DAP panel. (Figure 7-5). Whether NORM or VERN is selected depends upon fuel consumption considerations and how quickly the vehicle needs to be maneuvered to satisfy a mission objective.

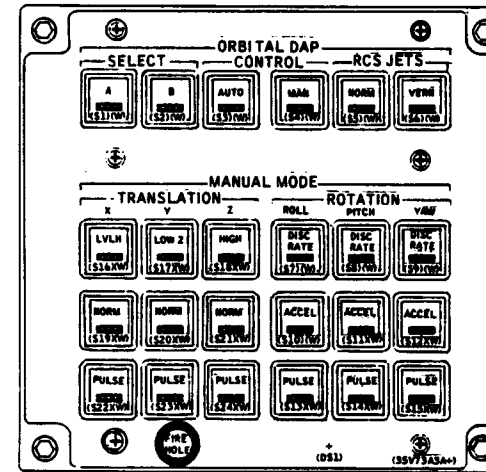


Figure 7-5. Orbital DAP Panel

The rotation rates and deadbands, translation rate, and certain other options for the DAP are changeable by the crew during the orbit phase using the DAP CONFIG display (figure 7-6). The crew can load the DAP with these options two ways at a time. One set is accessed by depressing the DAP A pushbutton on the ORBITAL DAP panel and the other by depressing the DAP B pushbutton. For convenience, each planned DAP configuration is given a number and is referenced with that number and which DAP A or B, is used to access it. For example, on STS 7, the checklist contained the necessary item inputs to set up DAP A1 through A16 and B1 through B12. See figure 7-7. Typically, the DAP A configurations will have larger deadbands and higher rates than the DAP B configurations. The wide deadbands are used to minimize fuel usage while the tight deadbands allow more precision in executing maneuvers or in holding attitude. The Crew Activity Plan (CAP) specifies which DAP configurations should be loaded at each time in the mission.

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2011/020/		DAP CONFIG		1 000/HH:MM:SS		000/HH:MM:SS	
TRANSLATION		A	B				
PULSE	1	0.10	15	0.02			
ROTATION							
DISC RATE	NORM 2	0.200	16	0.500			
	VERN 3	0.200	17	0.200			
PULSE	NORM 4	00.10	18	0.04	INERTIAS		
	VERN 5	0.010	19	0.002	29 IX	0.874	
COMP	NORM 6	.00	20	.000	30 IY	6.800	
	VERN 7	.000	21	.000	31 IZ	7.070	
DEADBAND							
ATT	NORM 8	5.00	22	3.00			
	VERN 9	1.000	23	1.000			
RATE	NORM10	0.20	24	0.20			
	VERN11	.020	25	.020			
JET OPT	P12	1	26	1			
	Y13	1	27	1			
CNTL ACCEL	14	0	28	0			

(XX)

Figure 7-6. DAP CONFIG Display

DAP	PURPOSE	DAP	PURPOSE
A1	- NOMINAL -ZLV IMU ALIGN TAIL ONLY CONTROL BACKOUT PRE/POST PAM DEPLOY OMS BURN TAIL-TO-SUN	B1	- NOMINAL PRCS IMU ALIGN TOP-TO-SUN TAIL ONLY CONTROL BACKOUT OMS & RCS BURNS
A2	- PTC	B2	- PRCS COAS CAL LOSS OF VERNIERS
A3	- RCS HOT FIRE	B3	- COAS CAL
A4	- PRCS GG	B4	- PAM DEPLOY
A5	- VRCS GG	B5	- LOSS OF VERNIERS
A6	- LOSS OF VERNIERS	B6	- LR SK
A7	- IUS STAR SCAN	B7	- SPAS GYRO DRIFT
A8	- IUS DEPLOY	B8	- PROX OPS PLUME
A9	- LSS VIBRATION DAMPING	B9	- PROX OPS
A10	- RMS/PRCS INTERACTION	B10	- YAW MNVR
A11	- PROX OPS PLUME	B11	- SPAS DEPLOY
A12	- SPAS GYRO DRIFT	B12	- (Blank)
A13	- PROX OPS		
A14	- RMS LOADED		
A15	- LR SK		
A16	- (Blank)		

STS-7/ORB/BAS

STS-7/ORB/BAS

DAP A1		A1	
PURPOSE:		0.1 0	1
-ZLV			TRANSLATION PULSE
IMU ALIGN		0.2 0 0	2
TAIL ONLY CONTROL BACKOUT		0.2 0 0	3
PRE/POST PAM DEPLOY		0.1 0	4
OMS BURN		0.0 1 0	5
TAIL-TO-SUN		0 0 0	6
		0 0 0	7
		5.0 0	8
		1.0 0 0	9
		0.2 0	10
		0 2 0	11
		1	12
		1	13
		0	14

STS-7/ORB/BAS

STS-7/ORB/BAS

Figure 7-7. STS 7 DAP Configurations Example

## SECTION 8

### DEORBIT GNC OPERATIONS

The RCS DAP has both an automatic and manual rotation mode. Which is used depends upon crew selection of the AUTO or MAN pushbuttons on the ORBITAL DAP panel. The manual mode is also accessed when the RHC is moved out of its detent (neutral) position. In both the automatic and manual modes, the rotation rate is controlled by the selection of DAP A or B and the information loaded in the DAP CONFIG display. In addition, in AUTO, the DAP determines the required attitude to be achieved from Universal Pointing. It then computes the RCS jet fire commands necessary to achieve these requirements within the current set of deadbands. In the manual rotation mode, the RCS DAP converts crew inputs with any of the three RHC's to RCS jet fire commands depending upon whether PULSE, DIS-CRETE RATE or ACCEL is selected on the ORBITAL DAP panel. Simplistically, in PULSE, a single burst of jet fire is produced with each RHC deflection. The resultant rotational rate is specified on the DAP CONFIG display. In DISC RATE, jet firings continue to be made as long as the RHC is out of detent in order to maintain the rotational rate specified on the DAP CONFIG display. In ACCEL, continuous jet firings are made as long as the RHC is out of detent.

Another manual RCS DAP mode, LVLH, is used to maintain the current attitude with respect to the rotating LVLH reference frame. It is selected via the LVLH pushbutton on the ORBITAL DAP panel.

The RCS DAP has only a manual translation capability, which is executed via the forward or aft THC. Only the primary RCS jets are used. Deflections of the THC result in RCS jet firings based upon which translation DAP mode pushbutton is selected on the ORBITAL DAP panel. In PULSE, a single burst of jet fire results; in NORM there are continuous jet firings with a specified subset of the available jets; in HIGH all upfiring jets fire continuously in a Z translation, and in LOW a special technique accomplishes a Z translation using the forward and aft firing RCS jets in order not to directly fire towards a target (this avoids "plume impingement" and contamination of a target payload).

The OMS TVC DAP is available when an OMS burn is executed in Major Mode 202 via the ORBIT MNVR EXEC display. The TVC DAP uses the guidance-generated V requirements and converts these into the appropriate OMS gimbal commands to achieve this target assuming AUTO is selected on the ORBITAL DAP panel. It generates the OMS fire commands, the OMS shutdown commands and, if necessary due to OMS engine failure, required RCS commands to maintain attitude control. If MANUAL is selected, the TVC DAP uses inputs from the RHC to control attitude during the burn.

As with the TRANS DAP, there are many subtleties in the operation of the Orbit DAP which are covered in the I/O/D G & C 2102 Workbook.

#### 8.1 DEORBIT PHASE DESCRIPTION

In order to return home, the Orbiter must be decelerated sufficiently such that when its orbit intercepts the atmosphere, the vehicle can maintain control and glide to the landing site. For the nominal end-of-mission, a retrograde burn of  $\approx 2.5$  minute duration is planned at the appropriate point in the vehicle's trajectory. Nominally, this burn is executed with 2 OMS engines, although under failure conditions it may be executed using only one OMS engine or solely via RCS jets. The deorbit phase of the mission includes the deorbit burn preparations including the load of burn targets and maneuver to burn attitude, the execution and monitoring of the burn, post-burn reconfiguration, and a coast mode until the atmosphere (and thus dynamic pressure build-up) is reached at  $\approx 400,000$  ft. This is called entry interface (EI). Certain failure conditions prior to or during the burn may cause the burn to be terminated and/or delayed.

The deorbit and entry flight software is called OPS 3. Major Mode 301 is a deorbit coast mode in which deorbit targets can be loaded, although the burn cannot be executed in this mode. Major Mode 302 is the mode necessary to execute the burn. After the burn, the software is transitioned to another coast mode, Major Mode 303, which is used to prepare for penetration into the atmosphere.

#### 8.2 DEORBIT GNC SOFTWARE FUNCTIONS

During the deorbit phase, navigation once again uses the Super-G algorithm to propagate the Orbiter state vector based upon a drag model or upon IMU data if sensed vehicle accelerations are above a specified threshold. During OPS 3, navigation maintains and propagates three Orbiter state vectors - each based upon a different IMU. From these three state vectors, a single Orbiter state vector is calculated using a mid-value selection process. This single state vector, then, is passed on for use by guidance, flight control, dedicated display, and CRT display software. Three separate state vectors are propagated to protect the onboard software from problems due to two IMU data failures. In such a case, once the bad IMU's are detected and deselected, the state vector associated with the remaining good IMU will not have been polluted. This three-state vector system is only used during OPS, since that phase is most critical with respect to navigation errors and their effects on vehicle control and an accurate landing. Another feature available during this phase is that the software begins computing a statistical estimate of the error in the state vector propagation which will be used later in flight when external sensor data is available. Also, in this phase it is possible for the crew or MCC to input a "delta state vector" to correct navigation.

Guidance during this phase is similar to that used in the Orbit Insertion (OMS 1/2) phase. Once again, the PEG 4 scheme is used to target the deorbit burn and guide the vehicle during the burn although the required conditions are different. The deorbit burn targets are for the proper conditions for entry interface, including altitude, position with respect to the Earth and thus the land-

ing site, and satisfaction of certain velocity/flight path angle constraints. Together, these ensure that the vehicle can glide to the landing site within thermal limits. These deorbit burn targets are specified preflight for a nominal mission, although it is also possible for the ground to uplink changes or for the crew to recompute them using an onboard hand-held calculator program. It is also possible to specify OMS fuel to be "wasted" (burned out of plane) during the burn in order to establish an acceptable Orbiter center of gravity (c.g.) for entry. The crew is responsible for loading these targets on the DEORB MNVR EXEC display (figure 8-1). Guidance then computes the necessary vehicle attitude to be established prior to the burn and displays it to the crew.

As in OPS 1, it is also possible to load an external  $\Delta V$  (PEG 7) target, although this option is not normally used.

Flight control during the deorbit phase is similar to that used during orbit insertion - the TRANS DAP is once again in effect. See section 6.4.2.

### 8.3 DEORBIT GNC CREW INTERFACE

Crew interface with the guidance, navigation, and flight control software during the deorbit phase is accomplished via CRT display inputs, RHC/THC maneuvers, and ADI monitoring. The major CRT display used during this phase is the DEORB MNVR EXEC display (Figure 8-1). In Major Modes 301 and 303, the title becomes DEORB MNVR COAST whereas in Major Mode 302 the title is DEORB MNVR EXEC. This display is identical in format to the OMS1 MNVR EXEC and ORBIT MNVR EXEC displays, although there are some differences in its capabilities between OPS 1, 2 and 3. It is used to set up and target the OMS burn, to specify fuel to be wasted during the burn, to display the required burn attitude, to initiate an automatic maneuver to that attitude, and to monitor the progress of the burn.

3021/ DEORB MNVR EXEC		1 005/00:41:55	
OMS BURN 1*			000/00:15:00
L 2	BURN ATT		
R 3	24 R 122	ΔVTOT	0279.1
RCS SEL 4	25 P 327	TGO	02.23
5 TVROLL 122	26 Y 340	V00 1 + 0260.32	
TRIM LOAD	MNVR 27	Z	-00
6 P 122.1	REI 4005	T	+076.45
7 LY 122.7	TFF 23:04	SA	HP
8 RT 122.7	GNBL	TGT 149	-003
9 NT 122222		CUR 149	149
10 T16			
11 122.122.122.2	P 20.05 ± 0.05		
12 PEG 4	V 20.05 ± 0.05		
13 C1 12222			
14 C2 122.122.1	PRI 20° 20° 25	ABORT TGT	--
15 NY 122.122	SEC 30 31		
170T 122.122	OFF 32 33	FWD RCS	
18 PRPLT11		ARM 36	
TGT PEG 7	GNBL CE 34	BURP 37	
19 ΔV1		OFF 38°	
20 ΔV2		SURF DRIVE	
21 ΔV3		ON 39	
LOAD 22/TIMER 23		OFF 40°	

Figure 8-1. DEORB MNVR EXEC

Another CRT display available during this phase is the HORIZ SIT display (Figure 8-2). During the deorbit preparation phase, the crew may verify that the display is ready for use during entry (correct runway selection, altimeter setting, etc.) although its other capabilities are not utilized until after entry interface.

3051/050/ HORIZ SIT		005/01:50:00	
PT1 INH2	ALTM	000/00:20:00	
INDEX	9 29.92	NAV DELTA	
		ΔX 10	
		ΔY 11	
RWY CTR L		ΔZ 12	
EDW 22 3		ΔX 13	
EDW 11 4		ΔY 14	
NOR 17 5		ΔZ 15	
		LOAD 16	
TAEM TGT		.18 ΔT	
G&N OVHD 6			
HSI			
NEP 7			
AIM NOM 8			
NAV	RESID	RATIO	AUTINHFOR
TAC AZ	0.06	0.1	19 20 21
RNG	0.50	0.3	
			TAC111 187.09
			TAC211 187.03
			TAC311 187.01
			DES 31 115.00
			DES 32 115.04
			DES 33 115.50
DRAG H	70	22 23 24	
ADTA H	102	25 26 27	
ADTA TO G&C	20 29 30	HU037L*	38R
			ABS 34
			DELTA 35

Figure 8-2. HORIZ SIT Display

The crew's task during this phase includes entering the correct burn targets in the DEORB MNVR EXEC display and maneuvering to burn attitude either in AUTO or manually using the RHC. The burn itself is typically executed in AUTO such that the crew's task is to monitor the burn progress in terms of velocities gained and OMS system performance. Figure 8-3 is a sample DEORB BURN cue card.

In cases where OMS system (engine, propellant tank, data path) failures have occurred, the crew must be prepared to reconfigure the system to ensure that the burn can safely continue to completion, that sufficient RCS propellant remains for entry and the Orbiter c.g. stays within limits. To do this, the crew uses a series of cue cards which aid them in determining the type of problem (DEORB BURN MONITOR cue card - Figure 8-4). In addition, they have cue cards available which list the required steps to reconfigure the system in order to continue the burn (DEORB BURN cue card - Figure 8-3).

```

:::HOOK:::      DEORBIT      :::HOOK:::
:::VELCRO:::    BURN          :::VELCRO:::
:::OVS099/DEORB/BAS
OVS099/DEORB/BAS

✓MM302 ** /OMS BOTH      CNTRL PWR (two) - OM*
Enter TGO + 5 sec
✓TRIM: P +0.0, LY -5.7, RY +5.7
✓DAP - AUTO(PASS)/DISC ADI - LVLH/HI/MED
L,R OMS He PRESS/VAP ISOL A (two) - OP, then GPC**
B (two) - GPC**

TIG-2 OMS ENG (two) - ARM/PRESS
-:15 EXEC ** (NO EXEC > TIG +  :  )
:00 TIG** (/Pc, ΔVTOT, ENG VLVs; start watch)
* If no OMS ignition, APUs - SHUT DN

* OMS PRPLT FAIL:
* HP >  L or  R (STOP):
* OMS ENG (two) - OFF, APUs - SHUT DN
* Affected OMS TK ISOL (two) - CL
* HP <  L or  R (CONTINUE):
* Failed OMS ENG - OFF
* ITEM 18 +0 EXEC
* Affected OMS He PRESS (two) - CL
* TK ISOL (two) - CL
* When Pc < 80%, 2nd OMS ENG - OFF
* Complete RCS (PRPLT FAIL)
* OMS ENG FAIL (CONTINUE BURN):
* Failed ENG - OFF
* OMS XFEED at
*  XL  XR 1/2 of ΔVTOT
* RCS COMPLETION: at FAIL 
* ANY OMS PRPLT FAIL:
* If HP >  , safe HP: APUs - SHUT DN
*  , safe HP: I'cnect to
* good OMS PRPLT
* THX +X (/ OMS % vs RCS V)
* AFT RCS RECONFIG
* THX +X to TGT HP or TOT AFT QTY 1  %
* If CUR HP < PRI HP  , FRCS to PRI Site
* If CUR HP > PRI HP  and > B/U HP ]
* FRCS TO B/U Site
* If CUR HP > B/U HP  , THX +X to
* B/U HP 
* or TOT AFT QTY 2  ; FRCS to B/U Site
* MNVR to -X ATT (pitch up at 30°/sec to VGOz =
* THX -X (+)1/4 ΔVTOT)
* BOTH OMS ENG FAIL:
* OMS ENG - OFF
* Interconnect OMS to RCS
* THX +X (/ OMS % vs RCS Burn Time)
* OMS TK SW
* THX +X (/ OMS % vs RCS Burn Time)

CUTOFF**
+:02 OMS ENG(s) - OFF** (If > 3 IMU, at  :  )
* AFT RCS RECONFIG if INTERCONNECT
Trim X, Z residuals 2 fps (< 0.5 fps if shallow)

```

Figure 8-3. DEORB BURN Cue Card

DEORBIT BURN MONITOR		HOOK:::OVS099/ VELCRO:::DEORB/ BAS
OMS Pe LOW OMS + and ENG VLV 1 or 2 70%		OMS ENG FAIL
OMS + and ENG VLV 1 and 2 > 70% and FU and OX TK P > 185 185		OMS PRPLT FAIL Burn to Pc=72%, then OMS PRPLT FAIL
No OMS +		Sensor Fail
OMS TEMP FU IN P > 232 < 197 P 198-231		OMS ENG FAIL OMS PRPLT FAIL SENSOR FAIL
OMS OX/FU TK P OX & FU LOW OX LOW FU LOW OX & FU HIGH		He PRESS/VAP (two) - OP Burn to OMS +, then OMS PRPLT FAIL He PRESS/VAP ISOL (two) - CL Burn to OMS +, then OMS PRPLT FAIL Cycle He to maint OMS TK P 234-288
OMS GMBL 1st FAIL 2nd FAIL		Select SEC GMBL If control problems or high RCS fuel usage: Affected OMS ENG - OFF XFEED if ΔVTOT > 20
PASS SET FAIL OR IMU DLMA OR 2 MIN DC BUSES	Safe HP: HP > <input type="checkbox"/> HP < <input type="checkbox"/>	Stop burn, APUs - SHUT DN Continue burn (if RGA, desellower # IMU)
GPCI(4)		GPCI(4) MODE - STBY, HALT FFI(4) - OFF, ON L(R) OMS Gimbals - 901 SEC L(R) OMS Gimbals - 901 SEC
PAI(4)		

Figure 8-4. Deorbit Burn Monitor Cue Card

## SECTION 9

### ENTRY GNC OPERATIONS

#### 9.1 ENTRY PHASE DESCRIPTION

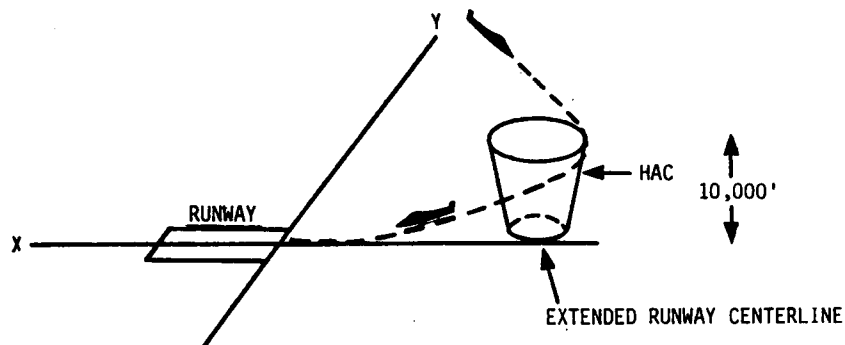
The entry phase of the flight begins approximately five minutes prior to entry interface which is defined as occurring at 400,000 feet. At EI-5, the Orbiter has an approximate altitude of 557,000 feet with a velocity of 25,400 fps. The goal of the guidance, navigation, and flight control software is to guide and control the vehicle from this state (in which aerodynamic forces are not yet felt) through the atmosphere to a precise landing at the designated runway. All of this must be accomplished without exceeding thermal or structural limits on the vehicle. The entry phase of flight is subdivided into three separate phases because the software requirements for each are different:

**ENTRY:** Extends from EI-5 to the point at which the vehicle is traveling at 2500 fps ( $\approx 83,000$  ft altitude). This is Major Mode 304.

**Terminal Area Energy Management (TAEM):** Begins at 2500 fps and extends to the approach/landing capture zone, which is defined when the vehicle is on glide-slope, on airspeed, and on runway centerline. This occurs at below 10,000. TAEM is the first part of Major Mode 305.

**Approach/Landing (A/L):** Begins at the approach/landing capture zone and extends through the receipt of the weight on nose gear signal after touchdown. This completes Major Mode 305.

During the ENTRY subphase, the primary objective is to dissipate the vehicle's energy as it comes into the atmosphere so that it does not burn up, stays within structural limits, and arrives at the TAEM interface with the altitude and range to the runway necessary for a landing. This is accomplished by adjusting the vehicle's drag acceleration using bank commands relative to the vehicle's velocity. During TAEM, as the name implies, the goal is to manage the vehicle's energy while the orbiter travels about an imaginary arc along what is called the heading alignment cone (HAC) which lines up the vehicle on the runway centerline. A HAC is an imaginary cone which, when projected on the earth, lies tangent to the extended runway centerline, as shown below:



The vehicle normally intercepts and is tangent to the HAC at an altitude of 10,000 feet. There are actually two HAC's, one on each side of the runway. Normally, the software is set up to fly the vehicle around the HAC on the opposite side of the extended runway centerline. This is called the overhead (OVHD) approach. (See figure 9-1.) Then, if the vehicle is low on energy, it can be flagged to acquire the HAC on the same side of the runway to save energy. This is called the straight in (STRT) approach.

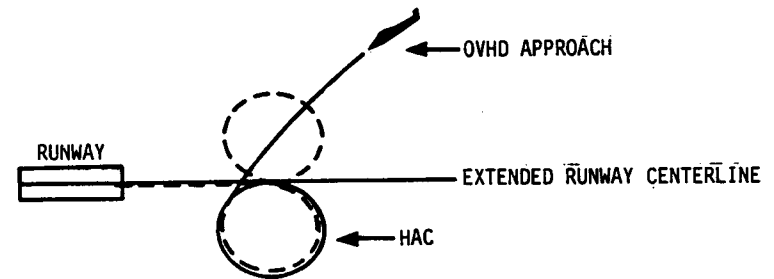


Figure 9-1. Overhead Approach

As the vehicle comes around the HAC, it should be lined up on the runway and established at the proper flight path angle and airspeed to begin a steep  $19^\circ$  glideslope to the runway. (Depending upon vehicle weight, a  $17^\circ$  glideslope may be established instead.) Following a final flare maneuver, the vehicle lands on its main gear. Typical landing parameters would be a sink rate  $< 2$  fps (up to 9 fps is acceptable depending upon vehicle weight) and EAS = 195 kts. As the airspeed drops below 165 kts, the vehicle begins derotation in preparation for nosegear slapdown.

#### 9.2 ENTRY GNC SOFTWARE FUNCTIONS

**9.2.1 Entry Guidance.** Guidance during the entry portion of flight performs different tasks during each of the ENTRY, TAEM, and A/L subphases. During the ENTRY subphase, guidance attempts to keep the vehicle on a trajectory which provides protection against overtemp, over  $q$  (dynamic pressure), and excessive

normal acceleration ( $N_z$ ) limits. To do this, it sends commands to flight control to guide the vehicle through a tight corridor limited on one side by altitude and velocity requirements for ranging (in order to make the runway) and vehicle control, and, on the other side, by the thermal,  $g$  and  $N_z$  constraints. Ranging is accomplished by adjusting drag acceleration to velocity such that the vehicle stays in that corridor. Drag acceleration can be adjusted in primarily two ways - by modifying angle of attack ( $\alpha$ ) which changes the vehicle's cross sectional area with respect to the airstream, or by adjusting the vehicle bank angle, which affects lift and thus the vehicle's sink rate into denser atmosphere which in turn affects drag. Using angle of attack as the primary means of controlling drag results in violation of thermal constraints on the vehicle. Thus, the vehicle's bank angle is used as the primary method of controlling drag, and thus ranging, during this phase. The vehicle's angle of attack is kept at a high value (40 degrees) during most of this phase to protect the upper surfaces from extreme heat. It is modulated at certain times when necessary to "tweak" the system and is ramped down to a new value at the end of this phase for vehicle controllability. Using bank angle to adjust drag acceleration results in the vehicle turning off course. Therefore, at certain times, the vehicle must be rolled back toward the runway. This is called a roll reversal and is commanded as a function of azimuth error from the runway. The ground track during this phase, then, results in a series of S-turns.

Guidance during TAEM generates the necessary commands to enable the vehicle to achieve the proper A/L conditions by following an energy/weight versus range profile. It does this as follows (reference figure 9-2).

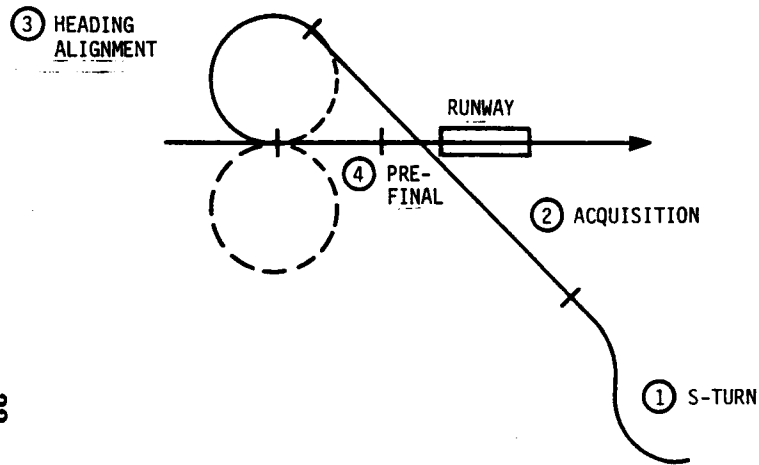


Figure 9-2. TAEM Guidance Subphases

- 1 S-Turns: If the vehicle is high on energy, it dissipates the excess by turning away from the HAC until the proper energy conditions are met.

- 2 HAC acquisition: Turns the vehicle toward the targeted tangent point on the HAC and flies to this point.
- 3 Heading alignment: Flies the vehicle around the HAC to the point at which the HAC is tangent to the runway centerline - this is called the Nominal Entry Point (NEP).
- 4 Prefinal: From the NEP, flies the vehicle down the runway centerline until the proper A/L conditions (airspeed, altitude, distance off centerline) are met.

These are accomplished via speedbrake, roll, and  $N_z$  commands to flight control. Within the above phases, TAEM guidance makes the necessary adjustments to account for various energy conditions.

Approach/landing guidance, then, picks up at about 10,000 feet when the vehicle is on airspeed ( $\approx 285$ ) kts, near the runway centerline and is near the proper altitude and flight path angle for the outer glideslope ( $19^\circ$  or  $17^\circ$ ). Guidance sends commands to keep the vehicle tracking the runway centerline and on the steep glideslope until  $\approx 1750$  ft, at which point the vehicle begins a pre-flare maneuver to put it on a shallow glideslope ( $1.5^\circ$ ). Final flare is then begun at  $\approx 80$  ft, reducing the sink rate to near zero for touchdown. At this point, guidance generates commands directing the vehicle on the runway. See figure 9-3.

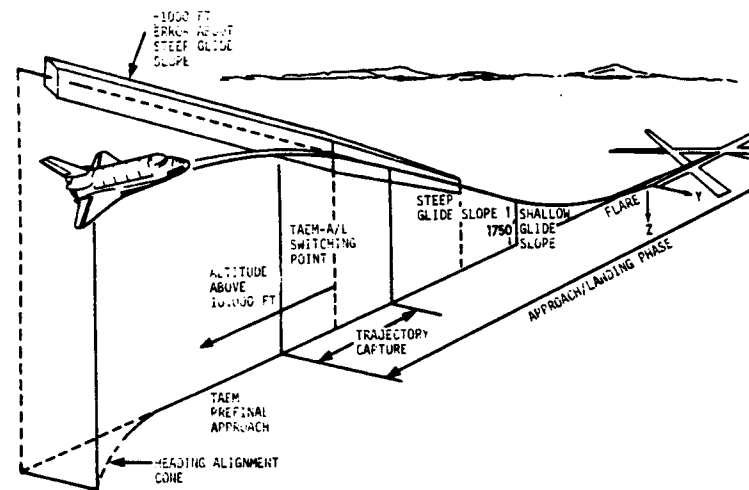


Figure 9-3. A/L Guidance Subphases



9.2.2 Entry Flight Control. Entry flight control is accomplished using the aerojet DAP, which generates effector and RCS jet commands to control and stabilize the vehicle during its descent from orbit. The aerojet DAP is a 3-axis rate command feedback control system which uses commands from guidance if in AUTO, or from a crew hand controller if in CSS. Depending upon the type of command and the flight phase, these result in fire commands to the RCS or deflection commands to the aerosurfaces. During exoatmospheric flight or flight during which certain aerosurfaces are rendered inefficient due to adverse aerodynamics, a combination of aerosurface and RCS jet commands are used. For example, only RCS jet commands are used to control pitch prior to dynamic pressure, ( $\bar{q}$ ) of 2. Between  $\bar{q} = 2$  and 20 lb/ft<sup>2</sup>, both elevator and jet commands are used, and, after  $\bar{q} = 20$ , the pitch jets are turned off as they are no longer effective. As another example, at Mach > 4, the angle of attack profile flown has the vertical tail virtually blanked so that the rudder is almost useless. To compensate for this, the flight control system uses yaw jets and the yaw due to aileron (differential elevon) movement to keep the vehicle sideslip within limits.

To control the vehicle longitudinally, commands are sent to a combination of pitch jets, elevons, body flap and speedbrake. The elevons are the primary tool to satisfy pitch requirements. However, there are other considerations with respect to elevon deflection including thermal protection of the main engines at high Mach numbers, the fact that the elevons function as ailerons for roll/yaw control at certain Mach numbers, and that the elevons need to be kept down to stay within hinge moment constraints. With these factors in mind, the elevons are kept primarily on a predetermined schedule dependent upon Mach number. When a pitch rate is required, the elevons will be used to establish the commanded pitch rate, but then the body flap will act as a trim tab to offload the elevons. To ensure that the body flap does not become saturated, the speedbrake is used to create a positive nose-up pitching moment after Mach 10. (Before this, the speedbrake stays closed to avoid heating problems). Between Mach 10 and Mach 4 it stays full open. After Mach 4, the rudder becomes effective, but fully deployed speedbrakes allow only 50% rudder effectiveness. Thus, the speedbrake is closed to 65% as a compromise between rudder effectiveness and nose-up moment generation. The speedbrake is also used to control range and L/D below Mach 1 and it is fully deployed at main gear touchdown to lessen the impact rate of nosegear slapdown.

A combination of yaw jets, aileron, and rudder deflections are used to control the vehicle laterally. (Aileron, here, means the use of the elevons differentially to produce a roll or yaw). As mentioned before, at Mach > 4, roll/yaw is controlled via RCS jet and aileron commands. When the rudder becomes effective below Mach 4, it is used in conjunction with the yaw jets for yaw control. At Mach 0.9, the yaw jets are ineffective and are turned off.

9.2.3 Entry Navigation. In the entry phase, Navigation software functions as it did during the deorbit phase (3 state vectors computed using Super-G algorithm corresponding to each IMU) with the exception that additional external sensor data is sequentially incorporated. This data is used to provide the accuracy necessary to bring the Orbiter to a pinpoint landing and to some extent to maintain vehicle control. The Tactical Air Navigation (TACAN) system provides slant range and magnetic bearing to various fixed stations around the

landing site and becomes available at  $\approx 156,000'$ . It is used until 1500' above ground level, at which point it is not used due to ground reflection problems. The Air Data System (ADS), which includes two Air Data Transducer Assemblies (ADTA's) attached to a probe on the left side of the vehicle and two ADTA's attached to a probe on the right side of the vehicle, provides pressures from which angle of attack ( $\alpha$ ), Mach number (M), equivalent airspeed EAS, true airspeed (TAS), dynamic pressure ( $\bar{q}$ ), barometric altitude (H) and altitude rate (H) are computed. Only barometric altitude is used by navigation. The other parameters are used by guidance and flight control as well as for display to the crew. The probes are normally deployed around Mach 3.8. The Microwave Landing System (MLS) very precisely determines slant range, azimuth, and elevation to the ground stations located along the side of the landing runway. When landing at runways with MLS ground stations, MLS data becomes available at 20,000 feet for processing by navigation. See figure 9-4.

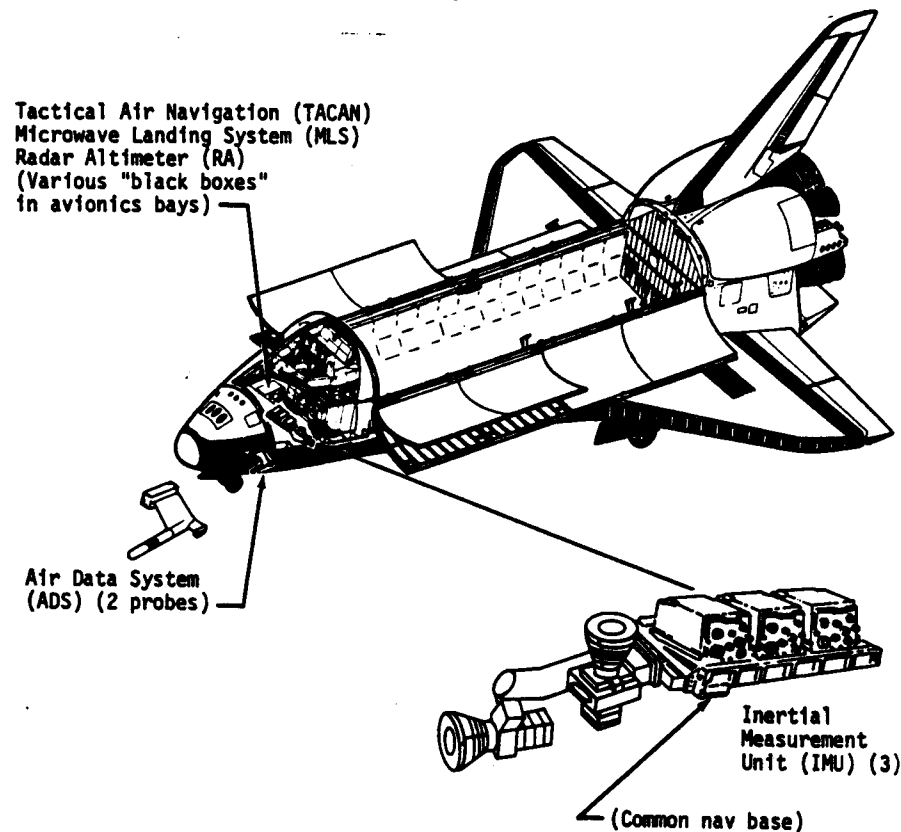


Figure 9-4. Orbiter Entry Navaid

One other tool used by navigation is a "software sensor" called drag altitude, which uses a model of the atmosphere to correlate the drag acceleration measured by the IMU's to altitude. This measurement, then, is only as good as the atmospheric model on which it is based. This model is not perfect. However, it has been determined through testing and analysis that the drag altitude data is important in keeping downrange and altitude errors bounded during the blackout portion of entry (from  $\approx 265,000$  to  $\approx 162,000'$ ). During this timeframe the ground is unable to uplink state vector corrections to the Orbiter and TACAN data is not available due to the ionization of the atmosphere around the vehicle caused by excessive heat generation.

As mentioned earlier, navigation also maintains a statistical estimate of the expected error in the state vector. This is called a covariance matrix and is propagated along with the state vector. When an external sensor, such as TACAN, becomes available to the navigation software, a check is made to see if the data lies within the current expected range of error. Crew controls are provided on an onboard CRT display "HORIZ SIT" (Figure 9-5) to force the software to accept or inhibit the external sensor data regardless of whether it lies within this expected range. There is also a control on the display which may be selected to allow the software to use the external sensor data to update its state vector as long as the data lies within the expected range. This scheme is part of what is called Kalman filtering and is used to prevent navigation from being polluted by bad sensor data. The navigation parameters computed as a result of this filter can be seen by the crew on the dedicated displays when NAV is selected as the instrument's data source.

34

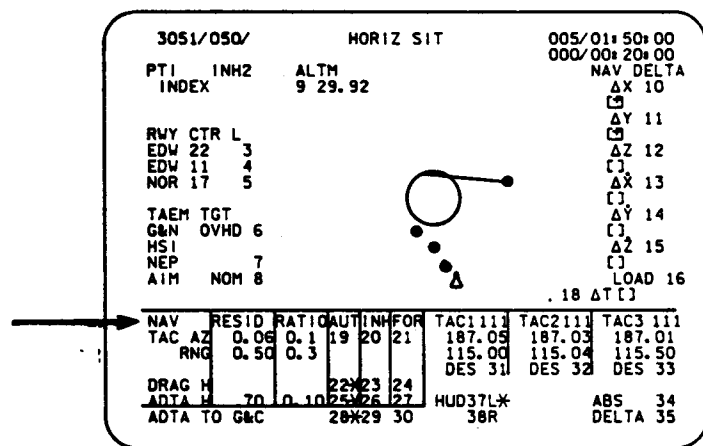


Figure 9-5. HORIZ SIT Display During MM 305.

### 9.3 Entry GNC Crew Interface

During the entry phase of flight, the crew uses CRT displays, dedicated displays, certain panel switches (such as the PANEL TRIM switches), the Rotational Hand Controller (RHC), and the Speedbrake Thrust Controller to control the vehicle. Approximately five minutes prior to entry interface, the crew modes the software to Major Mode 304. During this major mode, which lasts until TAEM interface, five CRT displays sequentially become available. These displays are used to monitor autoguidance and the vehicle's trajectory compared to the planned entry profile. The five displays are identical with the exception of the central plot which shows vehicle velocity versus range or energy/weight versus range with a changing scale as the vehicle approaches the landing site. (See figures 9-6, 9-7, 9-8, 9-9, 9-10). This central plot also includes static background lines which allow the crew to monitor the vehicle's progression compared to planned entry profiles.

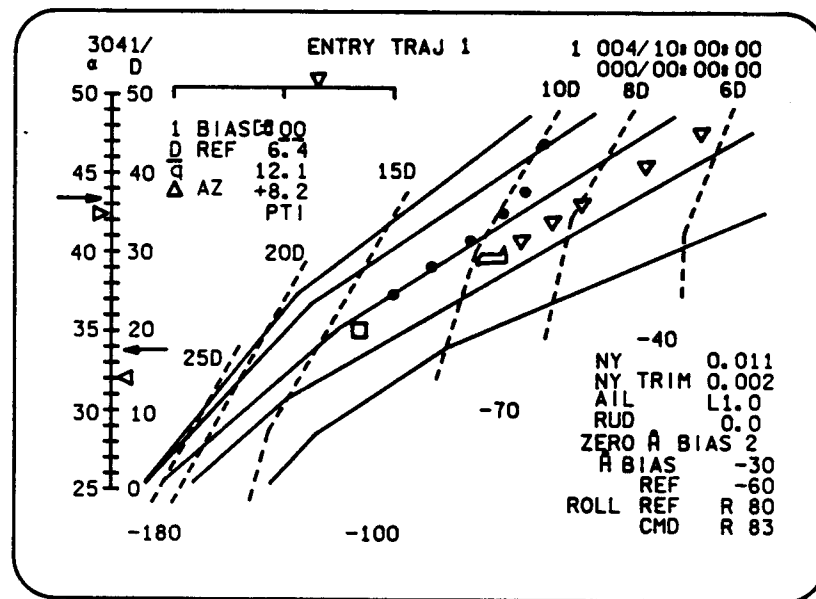


Figure 9-6. ENTRY TRAJ 1 Display

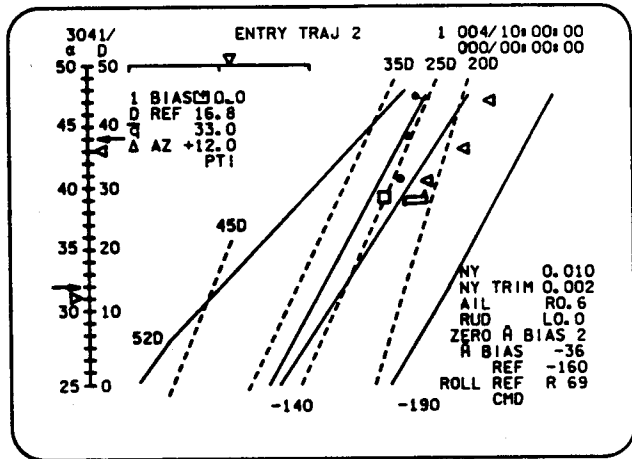


Figure 9-7. ENTRY TRAJ 2 Display

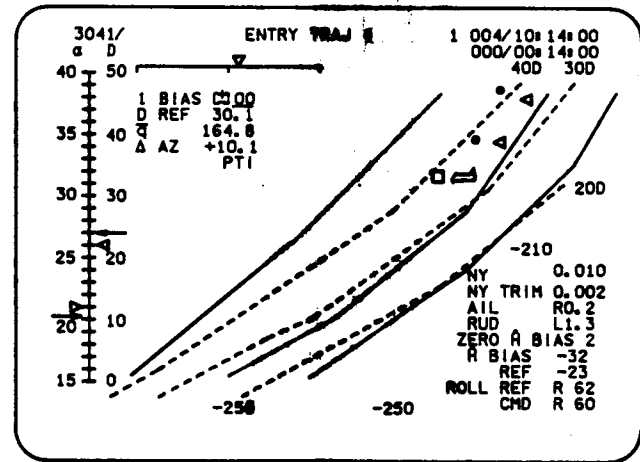


Figure 9-9. ENTRY TRAJ 4 Display

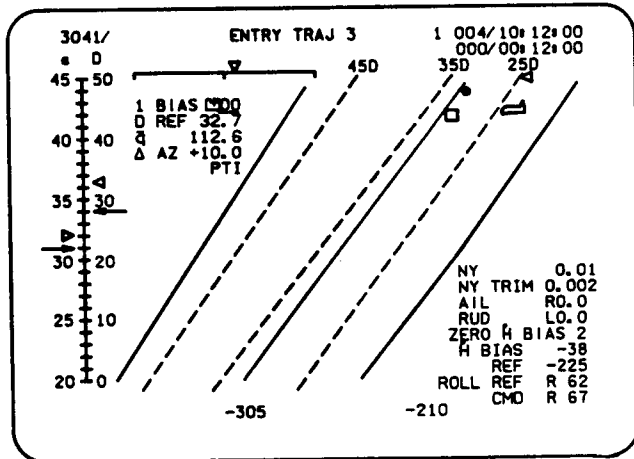


Figure 9-8. ENTRY TRAJ 3 Display

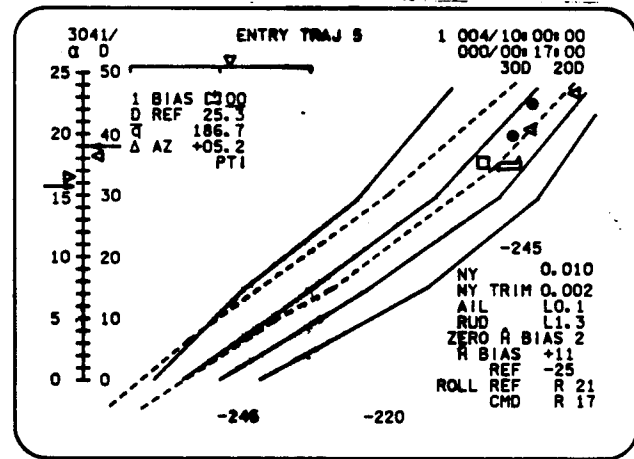


Figure 9-10. ENTRY TRAJ 5 Display

Once TAEM interface is reached, the software automatically transitions to Major Mode 305. The CRT display VERT SIT 1 then becomes available (Figure 9-11). This display includes a central plot of vehicle altitude with respect to range. This plot has three background lines which represent the nominal altitude versus range profile, a  $\bar{q}$  limit in guidance profile, and a maximum L/D (lift over drag) profile. At 30,000 ft. altitude, the scale and titles on the display change to VERT SIT 2. (Figure 9-12) This display is then used through landing. When the A/L interface conditions are met, a flashing 'A/L' is displayed on the VERT SIT 2 display.

Another prime CRT display used during entry is the HORIZ SIT display which was partially described in Section 9.2.3. In addition to providing insight and control over navigation parameters, this display provides the crew with spacecraft position and heading information once the Orbiter altitude decreases below 200,000 ft. (See figure 9-5).

The ENTRY TRAJ, VERT SIT, and HORIZ SIT CRT displays, then, are used by the crew to monitor the GN software. They can also be used by the crew to determine whether a manual takeover is required.

During Entry, the crew uses a series of cue cards found in the ENTRY FLIP BOOK. These cue cards remind the crew of their required activities and of the significant entry events (e.g., pitch RCS jets inactive at  $\bar{q} = 20$ ). In addition, they include pads on which the crew can write information voiced up from the ground.

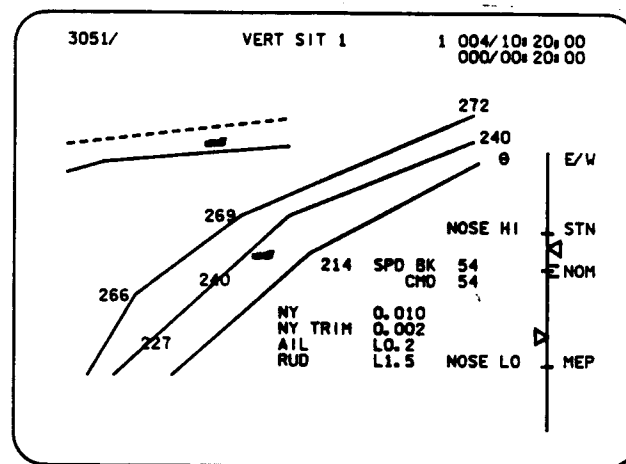


Figure 9-11. VERT SIT 1 Display

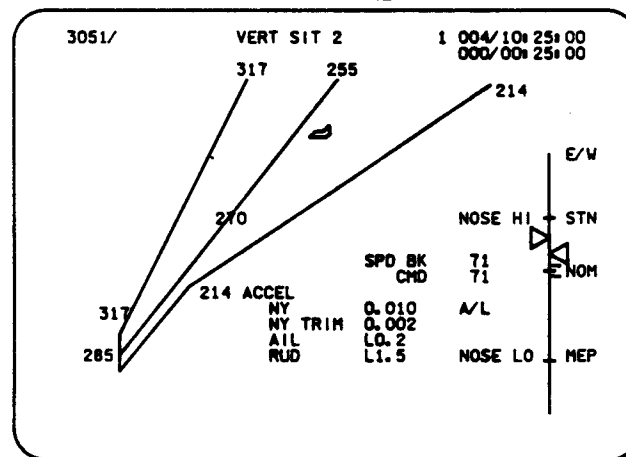


Figure 9-12. VERT SIT 2 Display

APPENDIX B

ACRONYMS AND ABBREVIATIONS

a angle of attack  
 ADS Air Data System  
 ADTA Air Data Transducer Assembly  
 AGL Above Ground Level  
 AMI Alpha/Mach Indicator  
 AOA Abort Once Around  
 ATO Abort-To-Orbit  
 AVVI Altitude/Vertical Velocity Indicator  
 A/L Approach/Landing  
  
 BFS Backup Flight System  
  
 CDR Commander  
 C.G. Center of gravity  
 C/L Checklist  
 COAS Crew Optical Alignment Sight  
 CRT Cathode Ray Tube  
 CSS Control Stick Steering  
  
 DAP Digital Autopilot  
 DPS Data Processing System  
  
 EAS Equivalent Airspeed  
 ET External Tank  
  
 FCS Flight Control System  
 FDF Flight Data File  
  
 GNC Guidance, Navigation, and Flight Control  
 GPC General Purpose Computers  
 GRTLS Glide Return to Launch Site  
  
 H Altitude  
 H, HDOT Altitude Rate  
 HAC Heading Alignment Cone  
 HSI Horizontal Situation Indicator  
 HUD Heads-Up Display  
  
 IMU Inertial Measurement Unit  
  
 KSC Kennedy Space Center  
 Kts Knots  
  
 LVLH Local Vertical/Local Horizontal  
  
 M Mach number  
 M50 Mean of 1950

ACRONYMS AND ABBREVIATIONS (Continued)

MCC Mission Control Center  
 MDM Multiplexer/Demultiplexer  
 MECO Main Engine Cutoff  
 MET Mission Elapsed Time  
 MLS Microwave Landing System  
 MM Major Mode  
 MPS Main Propulsion System  
  
 N<sub>z</sub> Normal Acceleration  
 NEP Nominal Entry Point  
  
 OMS Orbital Maneuvering Subsystem  
 OPS Operational Sequences  
 OVHD Overhead  
  
 PASS Primary Avionics Software System  
 PEG Powered Explicit Guidance  
 PLT Pilot  
  
 q̄, QBAR Dynamic Pressure  
  
 RA Radar Altimeter  
 RCS Reaction Control System  
 RTLS Return-to-Launch-Site  
  
 SPI Surface Position Indicator  
 SRB Solid Rocket Boosters  
 SSME Space Shuttle Main Engines  
  
 TACAN, TAC Tactical Air Navigation System  
 TAEM Terminal Area Energy Management  
 TAL Transatlantic Abort Landing  
 TAS True Airspeed  
 TDRS Tracking and Data Relay Satellite  
 THC Translation Hand Controller  
 TIG Time of Ignition  
 TMECO Time of Main Engine Cutoff  
 TVC Thrust Vector Control  
  
 V Velocity