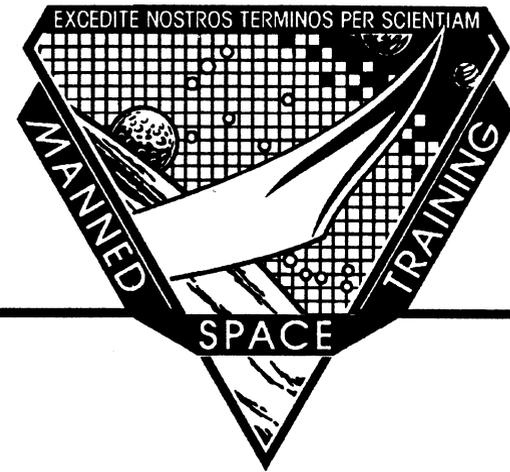


Ascent Guidance and Flight Control Training Manual

ASC G&C 2102



Mission Operations Directorate
Space Flight Training Division

Current as of January 1995

Originally published
January 1993



National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas

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ASC G&C 2102

January 1993

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Foreword

The content of this document was provided by the Control Propulsion Section, Flight Training Branch, Mission Operations Directorate (MOD), Lyndon B. Johnson Space Center (JSC), National Aeronautics and Space Administration (NASA). Technical support was provided by the Rockwell Space Operations Company (RSOC) for the Space Transportation System Operations Contract (STSOC). This document was produced through the MOD Training Center, Flight Training Branch, Training Division, MOD, JSC, with the assistance of the Integrated Documentation Support Department, Omniplan Corporation, STSOC. For additions, deletions, or revisions to the technical content of this document, fill out a JSC form 318 and mail to the document coordinator, Robin Lewis/DG6, or contact the subject matter expert, David S. Matthews, at 483-2520/DG66.

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A Training Materials Evaluation is included at the end of this document. Inputs on this sheet will be used to evaluate the lesson material. You do not need to sign the sheet.

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Section 1: Overview

In order for the space shuttle to achieve the desired orbit and maintain vehicle control, either automatically or manually, an abundance of information is required. This information is provided by the vehicle sensors and the onboard general purpose computers (GPCs) via software. This software is known as applications software and is divided into application programs. Guidance, navigation, and flight control are examples of applications programs. These programs are designed to manage the information from the vehicle sensors and the GPCs.

Navigation software provides current vehicle position and velocity (state vector) to guidance. This is necessary because guidance must "know" where the vehicle is in order to stay on course or to correct back to the desired course.

Guidance software determines thrust magnitude and direction (thrust vector) and duration required to achieve a desired target orbit from a known current orbit (state vector) provided by navigation software. Guidance must achieve the desired target by staying within prescribed vehicle and trajectory limits. Structural and thermal constraints on the vehicle define these limits.

Flight control software converts the guidance-computed thrust vector into effector commands in order to direct thrust in the required direction to steer the vehicle to the target. The effectors utilized are the main engine thrust vector control (TVC) actuators and solid rocket booster (SRB) TVC actuators. Figure 1-1 is a very simplified diagram showing the interface between guidance, navigation, and control (GN&C).

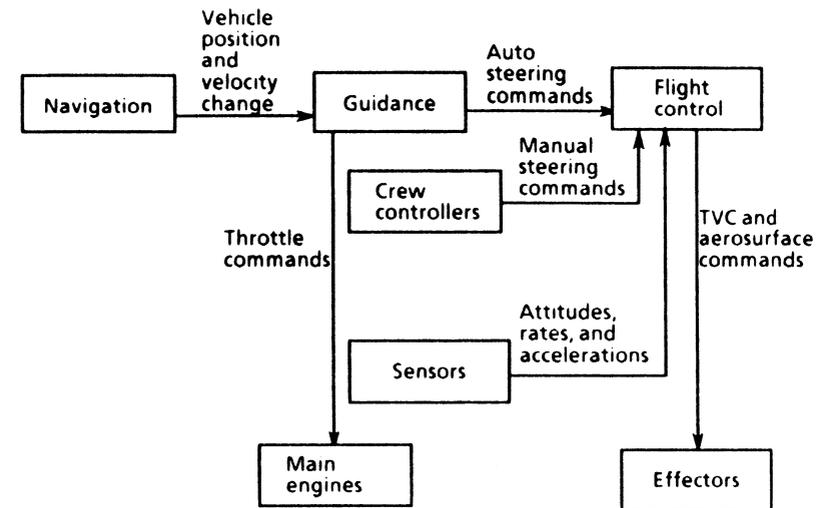


Figure 1-1.– GN&C interface.

The ascent flight phase begins at lift-off and ends at insertion into a circular or elliptical orbit. It consists of four subphases:

- First stage
- Second stage
- Orbital maneuvering system 1 (OMS-1) burn
- OMS-2 burn

This book concentrates primarily on the first-stage and second-stage subphases. Guidance software is discussed first, then flight control software is discussed for each subphase.

Section 2: First-Stage Guidance

Objectives

After reading this section, the student should be able to do the following:

- Explain which subtask achieves the first-stage trajectory
- Describe which subtask corrects vehicle performance as a result of a failed main engine
- Define the purpose of adaptive first-stage guidance

Ascent first-stage guidance is initiated at the SRB ignition command (major mode (MM) 102 transition) and is terminated at the SRB separation command. The first-stage trajectory is designed to perform four functions:

- Not to exceed maximum dynamic pressure ($\max \bar{q}$) for the mated vehicle structure under nominal and perturbed (winds, shears, flutter, and buffet) conditions
- Maintain maximum vehicle performance
- Avoid contact with the launch tower
- Avoid recontact with jettisoned SRBs

In order to meet these constraints, guidance in first stage works open loop based on an I-loaded table that is a function of Earth relative velocity (V_{rel}).

This table calculates the vehicle attitude based on V_{rel} . Stated another way, as a specific V_{rel} is attained, guidance will command the vehicle to a new attitude. These attitudes (Euler angles) are defined in relation to the boost reference coordinate system, as illustrated in figure 2-1. Basically stated, this system has its origin at the launch site at a specific time, which makes it inertial. The order of body rotations is yaw, pitch, and roll. Since first-stage guidance only commands an attitude and receives no feedback as to whether or not the target is being achieved, guidance is said to be "open loop."

The first-stage trajectory is divided into two phases.

- Vertical rise phase
- Tilt phase

This trajectory is shown in figure 2-2. During the vertical rise phase, the launch pad attitude is commanded until an I-loaded V_{rel} sufficient to ensure that launch tower clearance is achieved. Then the tilt maneuver (roll program) orients the vehicle to a heads-down attitude required to generate a negative q -alpha, which in turn alleviates structural loading. Other advantages with this attitude are performance gain, decreased abort maneuver complexity, improved S-band look angles, and crew view of the horizon. The tilt maneuver is also required to start gaining downrange velocity to achieve the main engine cutoff (MECO) target in second stage.

Origin: Derived from the center of the Earth

Orientation: The Z-axis is parallel to the gravity gradient which passes through the launch site and is positive in the direction of gravity. It is fixed at the time of launch.

The X-axis lies along the launch site meridian and is positive northward.

The Y-axis completes a standard right-handed system and is positive eastward.

The X - Y plane is normal to the launch site gravity gradient vector.

Characteristics: Inertial, right-handed, cartesian

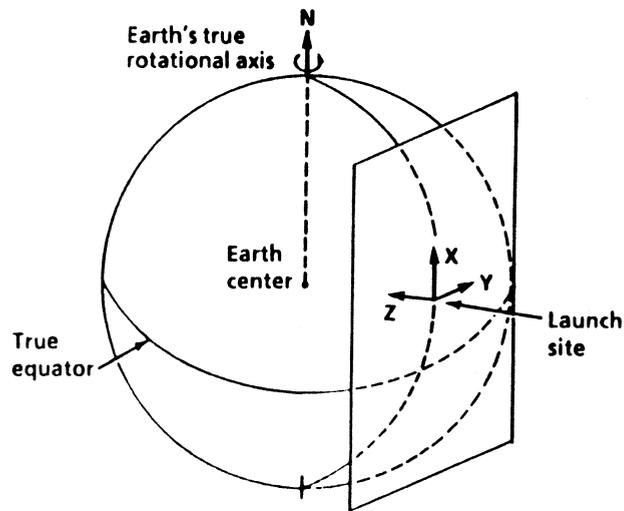


Figure 2-1.- Boost reference coordinate system.

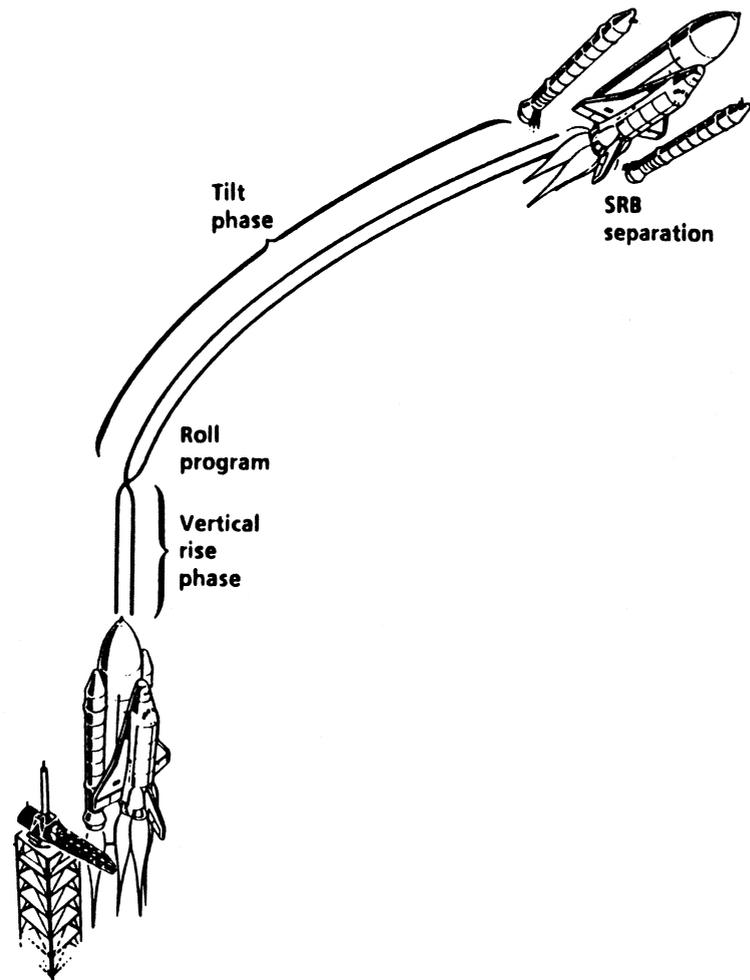


Figure 2-2.- First-stage trajectory.

Guidance will also issue main engine throttle commands to limit the max \dot{q} and to achieve the required performance during atmospheric flight. The throttle commands are also open loop in nature, with the commands being issued as a function of V_{rel} . Figure 2-3 illustrates a typical throttle profile. These commands, as are the vehicle attitudes, are I-loaded prelaunch and are flight specific.

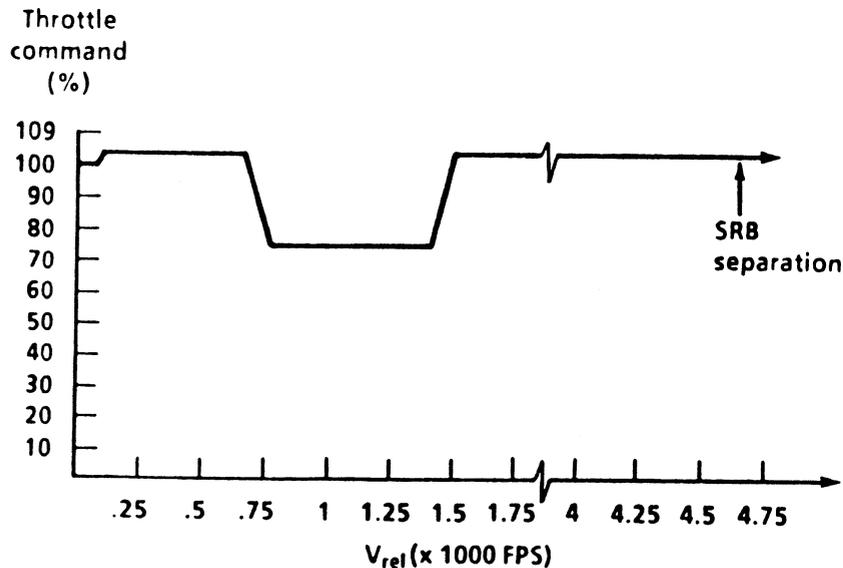


Figure 2-3.- Throttle profile.

In order for first-stage guidance to perform these functions, four subtasks are required.

- First-stage guidance input subtask
- Boost guidance subtask
- Boost throttling subtask
- Space shuttle main engine (SSME)-out safing subtask

Figure 2-4 is a block diagram of ascent first-stage guidance.

To facilitate understanding of first-stage guidance, each of these subtasks will be discussed in detail.

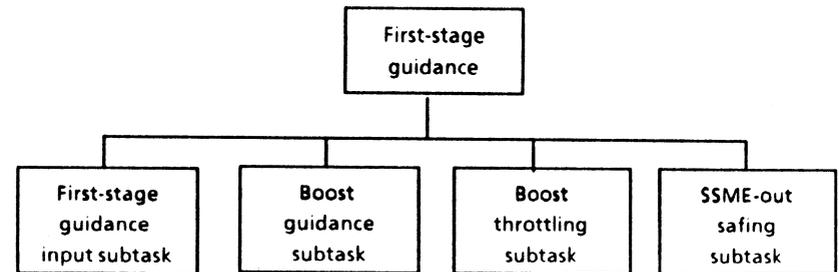


Figure 2-4.- First-stage guidance.

First-Stage Guidance Input Subtask

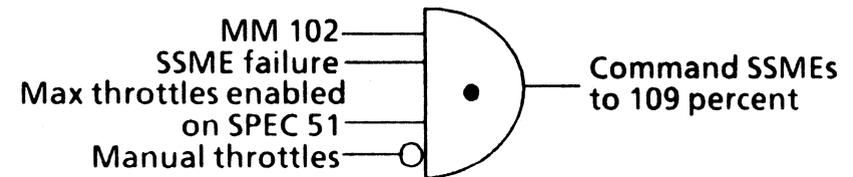
The first-stage guidance input subtask calculates various parameters to be used by the other software subtasks within first-stage guidance, as well as users in second stage and powered return to launch site (PRTL) guidance. These parameters must be calculated because other inputs to guidance require them to determine their solution. For example, to determine how much propellant the SSMEs are burning, one must know how many active engines are operating before any further calculations are made. This subtask performs four functions.

- Calculates the number of thrusting SSMEs
- Calculates the orbiter mass to be used in second stage or PRTL
- Saves V_{rel} and altitude rate (\dot{H}) for return to launch site (RTL)
- Enables maximum throttles

The first parameter calculated is the number of thrusting engines. This is required for guidance to know its performance so that later on, guidance can calculate SSME burn duration and thrust magnitude. Next, the orbiter mass is calculated; this calculation is used in second stage or PRTL to determine vehicle performance increase as a result of burning main propulsion system (MPS)

propellant. The third function is to save the V_{rel} and \dot{H} at the end of the last guidance cycle of MM 102. These parameters are required by the pitch command subtask within PRTL guidance in order to calculate an alternate fuel dissipation thrust angle depending on the magnitude of V_{rel} at the MM 102 transition. Section 3 will cover this in greater detail. The last function enables and maintains maximum throttles when requested via SPEC 51, OVER-RIDE DISPLAY. Maximum throttle (known as max throttles) is a throttle setting higher than the nominal value of 104 percent, to the highest setting available of 109 percent. As of this date, max throttles is an uncertified flight mode but, if needed in an emergency, is available; however, overall engine useful life is reduced.

First-stage max throttle logic:



Boost Guidance Subtask

The boost guidance subtask performs five functions.

- Commands attitude hold at lift-off
- Determines nominal roll, pitch, and yaw attitude profile
- Determines attitude profile for single and multiple SSME failures
- Adaptive first-stage guidance
- Calculates the Mean of 1950 (M50)-to-commanded-body quaternion

The first function is performed during the vertical rise phase (pre-roll program). This is required to prevent contact with the launch tower. Guidance commands the space shuttle attitude to the attitude of the launch tower until it reaches a specified V_{rel} magnitude which ensures that launch tower clearance is attained. This V_{rel} is mission dependent and is I-loaded prior to launch.

The boost guidance subtask contains an I-loaded table of roll, pitch, and yaw attitudes for a nominal ascent. After the vertical rise phase, guidance will command the vehicle to these attitudes based on V_{rel} . Linear interpolation is performed between each point to smooth the attitude commands over the trajectory such that step inputs are avoided. Table 2-1 shows 30 spaces for the yaw and pitch profiles along with 30 spaces for V_{rel} . There are 15 roll values, all of which are 180° , which utilize the first 15 velocity spaces.

Table 2-1.—Attitude table.

Flight phase	V_{rel} (fps)	Yaw angle	Pitch angle	Roll angle
Vertical rise	0	0°	90°	0°
TILT	127	89.7°	70.5°	180°
	341	89.7	70.1	180
	414	89.9	70.0	180
	493	89.7	69.7	180
	571	89.5	69.4	180
	678	89.6	68.6	180
	747	89.4	67.8	180
	846	89.4	66.2	180
	932	89.2	64.4	180
	1068	89.0	62.1	180
	1149	89.0	61.1	180
	1304	88.9	59.0	180
	1382	88.9	58.2	180
	1463	88.7	57.8	180
	1550	88.5	57.1	-
	1693	88.1	54.7	-
	1797	88.0	52.9	-
	2027	88.5	48.7	-
	2154	88.6	46.4	-
	2286	88.7	43.0	-
	2353	88.7	42.0	-
	2421	88.8	41.6	-
	2746	88.8	38.6	-
	3019	88.8	36.3	-
	3157	88.9	35.3	-
	3350	88.9	33.9	-
	3573	88.9	32.4	-
	3950	88.9	29.8	-
	5000	88.9	22.6	-

Should an SSME fail during first stage, a pitch bias is added to the nominal pitch table value to loft the trajectory slightly higher than the nominal profile. This action is taken to aid the RTLS abort maneuver. Figure 2-5 shows the trajectory lofting which is necessary because of the open loop nature of this guidance scheme. Without it, the trajectory would be depressed due to the loss of performance with the engine failure. If this were to happen, the vehicle would exceed the design loading constraints because of the relatively lower altitude (higher dynamic pressure, q -alpha and q -beta). Also to prevent tower contact, this bias is not added until tower clear, which is based on V_{rel} .

The trajectory would be depressed even further in contingency abort situations where more than one SSME fails. To reduce this effect, logic exists in first-stage guidance which calculates a new pitch command to loft the trajectory further than what is done for a single failure. This logic is known as autoloft and is used in conjunction with scant manual intervention in order for the vehicle to remain controllable. Without this logic, the open loop guidance scheme would be trying to achieve the nominal profile with insufficient performance capability.

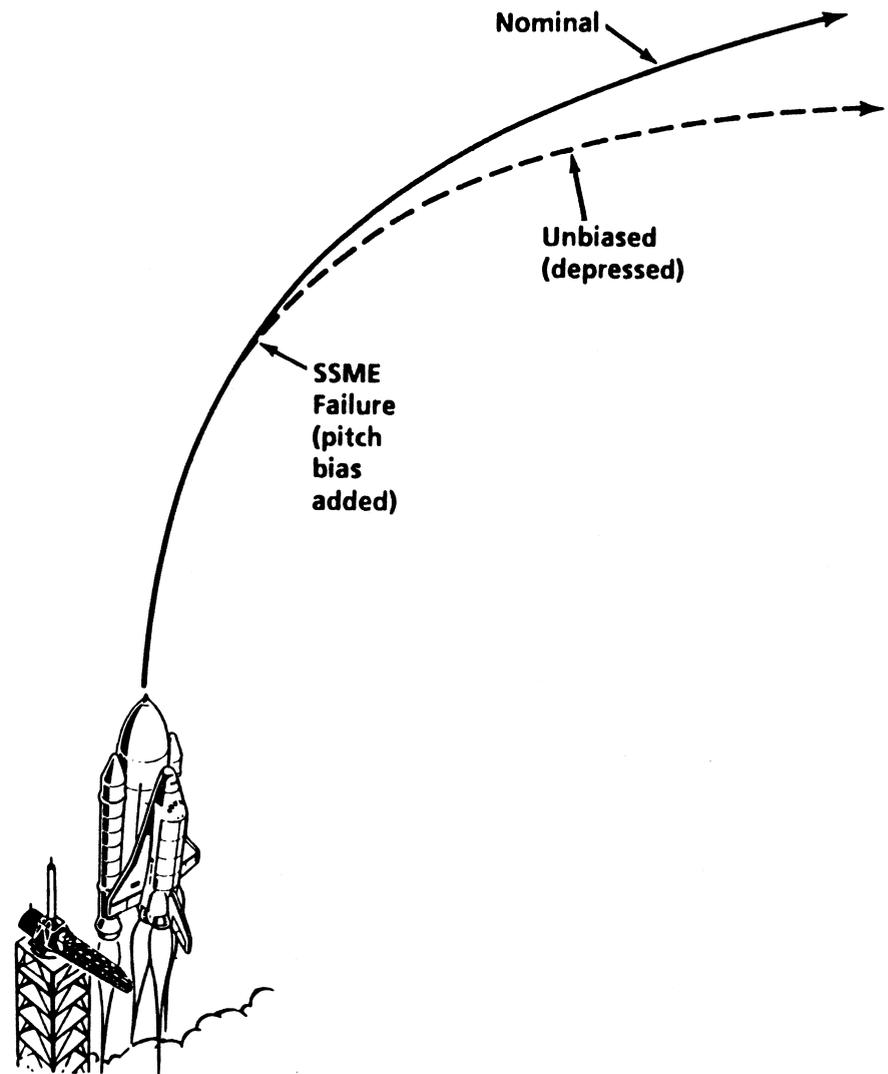


Figure 2-5.- Trajectory lofting.

The fourth function within this subtask is adaptive first-stage guidance. Adaptive first-stage guidance determines when SRB thrust performance is off-nominal. When this happens, it will automatically adjust the pitch and throttle commands during load relief to correct for this performance dispersion. Off-nominal SRB performance can be high, caused by hot SRBs, or low, caused by cold SRBs. Barring any engine failure(s) prior to load relief (the region of max \dot{q} where the vehicle attempts to relieve structural loading by adjusting SSME power level, elevon schedule, and trajectory), only SRB dispersions will significantly change the velocity/time history. Adaptive guidance computes the elapsed time since lift-off, at which a specific V_{rel} is achieved, and compares it to an I-loaded reference time to determine if a time difference (Δt) exists. If a Δt exists, the SRB performance is off-nominal. If the Δt is >0 , the SRB performance is low. If the Δt is <0 , the SRB performance is high. There are three reasons for correcting for performance variations.

- To prevent violating max \dot{q} constraints
- To regain lost performance
- To prevent unnecessary abort for small variations in SRB performance

Note in figure 2-6 that if the SRB performance is low, adaptive guidance will pitch the vehicle down slightly with reference to the natural horizon. Operationally, this is very difficult to notice; it is mentioned here to gain insight into how the vehicle is responding. The objective of this maneuver is to maintain the nominal trajectory as much as possible. Since acceleration is lower with low SRB performance, the vehicle will be further downrange when a specific V_{rel} is achieved. Therefore, to prevent a lofted trajectory, the nose is lowered. This also increases the downrange velocity so second-stage guidance, which is closed loop, does not have as big a deficit to work with to obtain the MECO target. Also in this situation, guidance attempts to regain the lost performance by adjusting the thrust bucket. When SRB performance is low, the SSMEs will not be commanded to throttle down as far as they would have with nominal SRBs. In other words, the minimum throttle setting during the bucket will be higher. Figure 2-7 illustrates a nominal bucket and SRB dispersed buckets.

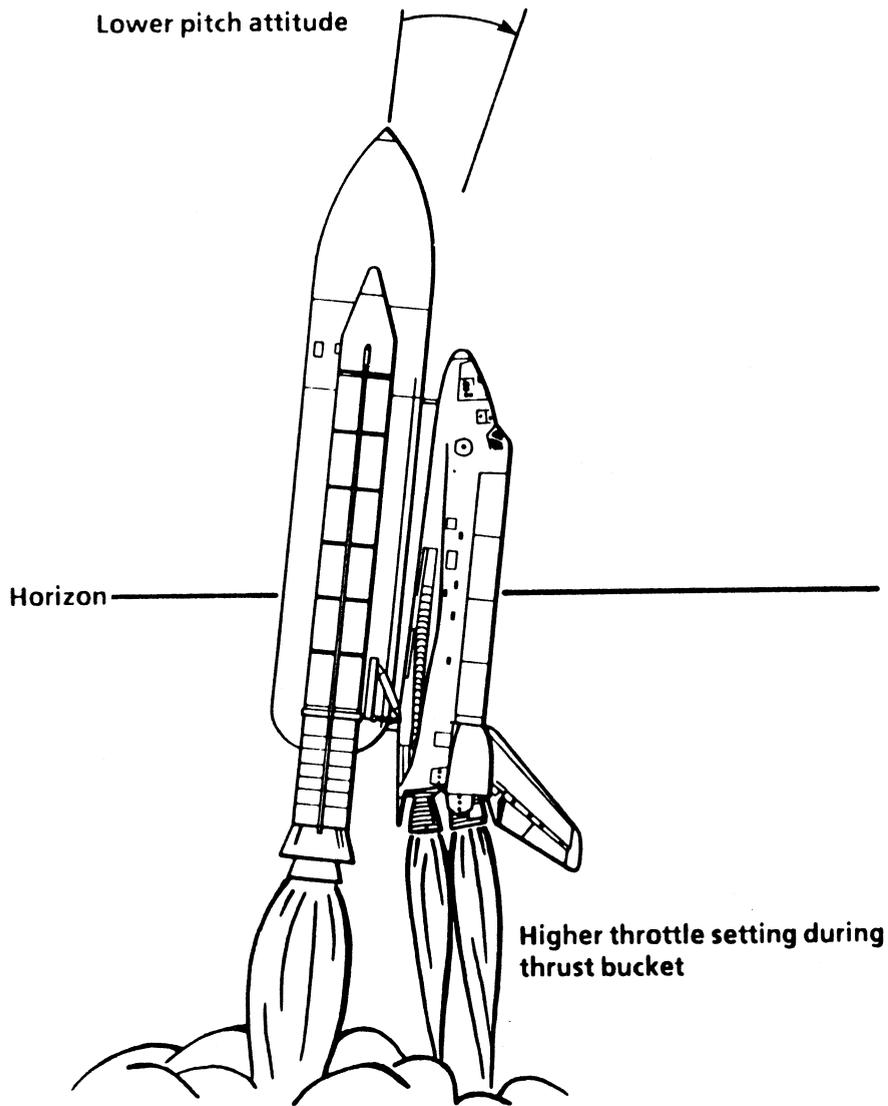


Figure 2-6.- Vehicle response to cold SRBs.

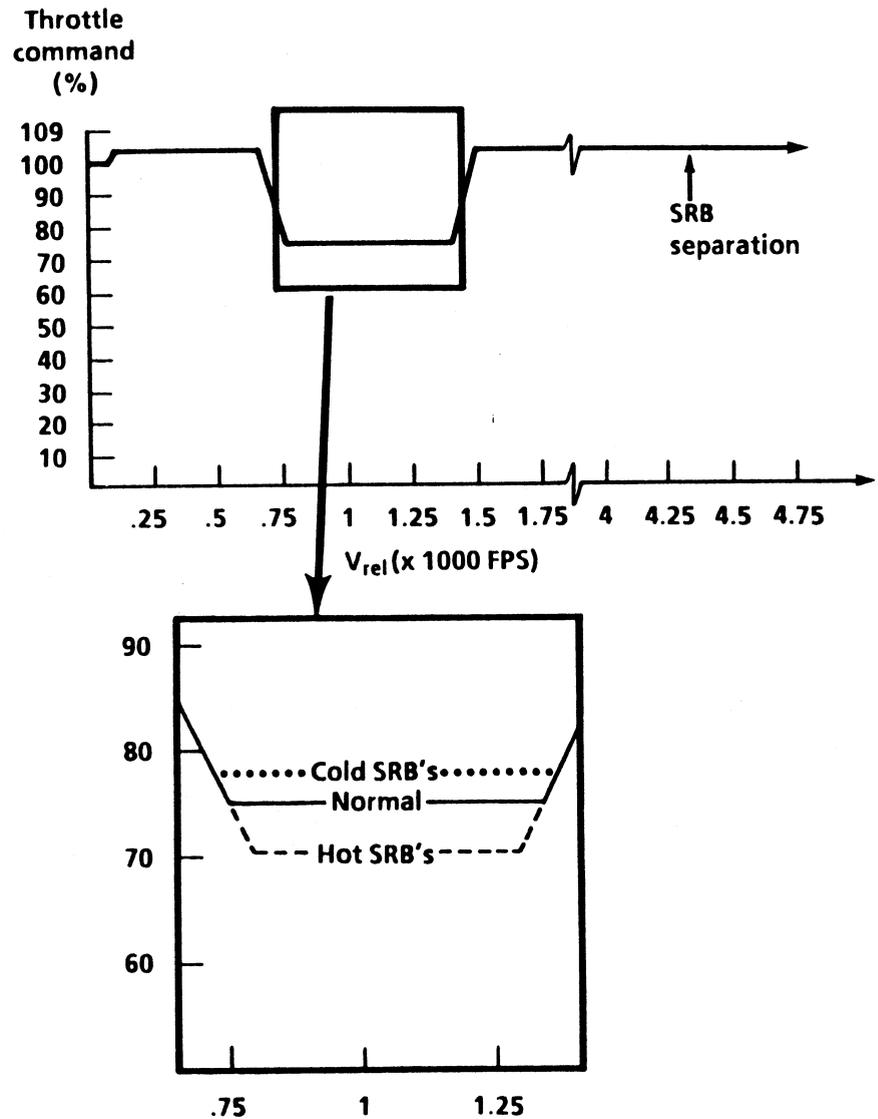


Figure 2-7.- Thrust bucket throttle commands.

As shown in figure 2-8, the SRB performance is high. In this case, the vehicle will pitch up slightly (again, difficult to notice) with reference to the natural horizon, and the SSMEs will throttle down further than nominal during the bucket. The objective of this maneuver is to prevent exceeding the max \dot{q} on the vehicle by throttling the SSMEs to a lower (65 percent minimum) setting. The pitchup maneuver attempts to maintain the nominal trajectory. With high SRB performance, the acceleration will be higher and the vehicle further uprange when a specific V_{rel} is achieved. Therefore, the nose is raised slightly to prevent a depressed trajectory.

All corrections for off-nominal performance will be performed during load relief and applied as a function of load relief. The vehicle pitch correction will be ramped out by the completion of load relief, and the nominal I-loaded table for attitude and SSME throttle commands will be resumed.

Should any SSME(s) fail prior to load relief, the SSME-out safing subtask will "kick in," and no throttle adjustments will be performed for off-nominal performance. In this case, there will be no bucket in order to achieve the highest possible performance. If any SSME(s) fails during load relief, load relief and adaptive guidance will be terminated and guidance will mode to the SSME(s)-out case. Remember that a pitch correction will be made in this situation to loft the trajectory.

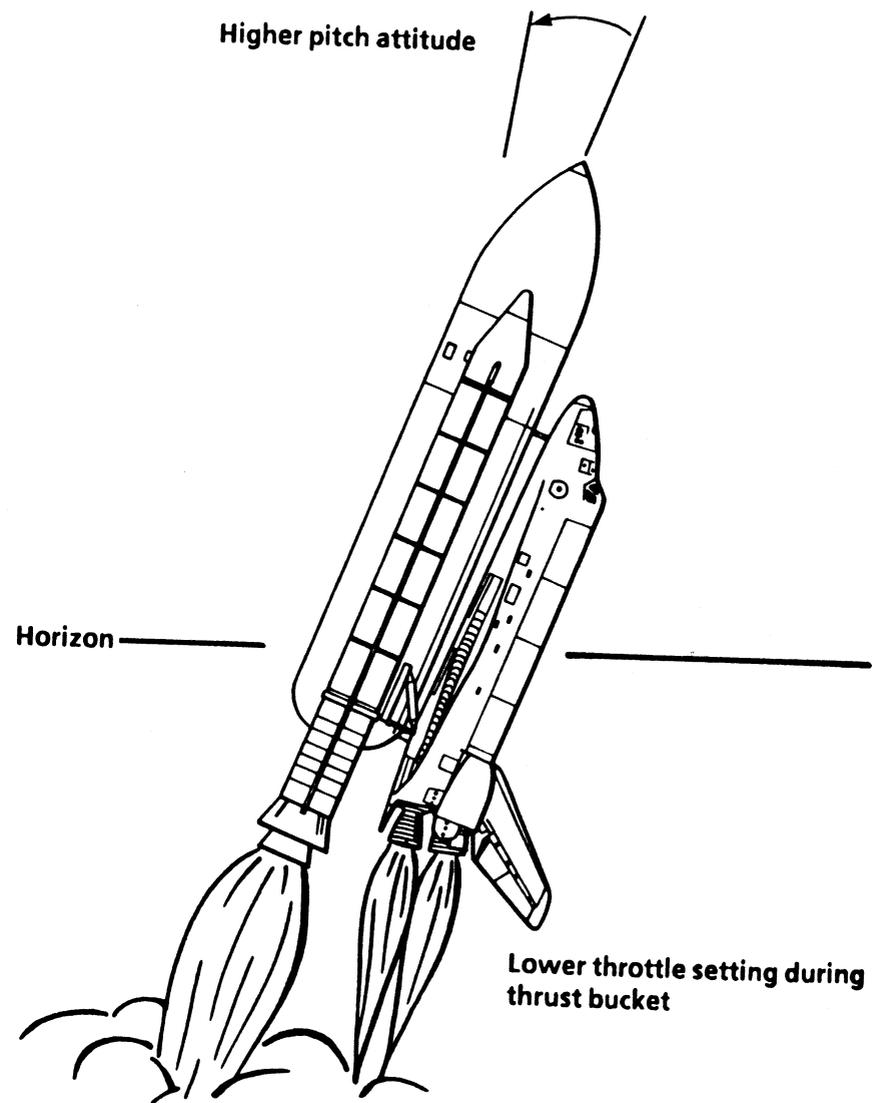


Figure 2-8.- Vehicle response to hot SRBs.

The last function within the boost guidance subtask converts the shuttle attitude commands from the M50 reference frame to the shuttle body reference frame. This conversion is performed using a quaternion. All a quaternion does is transform one reference frame to another (for additional information on quaternions, consult the quaternions supplemental workbook, Quat-S 2102). This conversion is required in order for guidance and control steering (G/C steer) interface to calculate the body attitude errors. Body attitude errors are what flight control uses to steer the vehicle in the correct direction. G/C steer is a software module that will be discussed in the flight control section (section 5).

Boost Throttling Subtask

This subtask provides first-stage SSME throttle commands as a function of V_{rel} to gain the desired vehicle performance and to limit the maximum value of dynamic pressure ($\max \dot{q}$). This is accomplished via an I-loaded table of throttle commands, table 2-2. There are a maximum of four throttle command change points within this software. Essentially, this is the subtask which creates the thrust bucket. The thrust bucket (figure 2-7) gets its name from the shape of the line when the throttle commands are plotted with respect to V_{rel} . This bucket limits the $\max \dot{q}$ on the vehicle by throttling down the SSMEs during load relief. The bucket will not occur if manual throttling is in progress or if any SSME(s) fails. Should an SSME fail, the remaining engines will be commanded to the full power level (I-load, 104 percent) by the SSME-out safing subtask.

Table 2-2.– Typical throttle command table.

Change point no.	V_{rel} (ft/sec)	Throttle command (%)
1	60	104
2	650	104
3	792	65
4	1304	104

Under nominal conditions the launch processing system (LPS) will command the main engines to 100 percent. Vehicle acoustic and structural constraints limit the LPS from commanding a higher power level on the launch pad. The 100-percent throttle command is maintained until an I-loaded V_{rel} is achieved. Normally, after this velocity is reached, the main engines will be commanded up to 104 percent of rated power level (RPL) (or any power level loaded into this subtask). The throttle setting will remain at this value until a specified V_{rel} is reached, at which time the main engines will go into the thrust bucket to limit $\max \dot{q}$. When a specified V_{rel} is attained that ensures the $\max \dot{q}$ region has passed, the main engines will be commanded back to full power.

Table 2-2 is an example of a one-stage bucket because the main engines are commanded less than 104 percent only once. If it is determined preflight that the designed trajectory causes max \dot{q} exceedances when a single-stage bucket is used, a two-stage bucket will be required to properly limit the dynamic pressure. As can be seen in table 2-3, a two-stage bucket is one in which the main engines are throttled down twice. The first stage is created when the engines throttle down at change point two to an intermediate value between 104 and 65 percent. Then, at change point three, they enter the second stage and are commanded even lower (65-percent minimum).

Regardless of the type of thrust bucket, it is this table that adaptive guidance modifies for off-nominal SRB performance. Adaptive guidance not only adjusts the throttle commands but also the change points as well. For cold SRBs, the main engines are commanded higher than the table value at change point three. For hot SRBs, the engine throttle commands are lower than the table value at change point two.

Table 2-3.- Throttle command table for a two-stage bucket.

Change point no.	V_{rel} (ft/sec)	Throttle command (%)
1	60	104
2	650	89
3	792	65
4	1304	104

SSME-Out Safing Subtask

This subtask is used by ascent first-stage guidance, second-stage guidance, and PRTL guidance when an SSME fails. This subtask monitors SSME fail flags from the SSME operations sequence software to determine when an engine has failed. The purpose of the SSME-out safing subtask is to command a two-engine power level equivalent to three engines or to full-power level. This is accomplished in first-stage guidance simply by commanding the operating SSMEs to full-power level (104 percent). However, in second-stage or PRTL guidance, a two-engine power level equivalent to three engines will be commanded by using the least of either full-power level or 1.5 times the last throttle command. For example, let's see what happens when three SSMEs have been manually commanded to 65 percent, then auto throttles resumed, and an SSME subsequently fails.

Three engines at 65 percent

$$\begin{array}{r} 65 \\ \times 3 \\ \hline 195\% = \text{three-engine power level} \end{array}$$

Next, an SSME fails

$$\begin{array}{r} 65\% \text{ last throttle command} \\ \times 1.5 \\ \hline 97.5\% = \text{new throttle command per engine} \\ \text{(97.5 percent < 104 percent)} \\ \times 2 \text{ engines} \\ \hline 195\% = \text{two-engine equivalent power level} \end{array}$$

Notice that maximum vehicle performance is maintained without exceeding structural flight constraints.

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Questions

1. Which subtask achieves the first-stage trajectory?
 - A. Boost guidance subtask
 - B. Guidance input subtask
 - C. Boost throttling subtask
 - D. SSME-out safing subtask
2. Which subtask issues nominal SSME throttle commands?
 - A. Boost guidance subtask
 - B. Guidance input subtask
 - C. Boost throttling subtask
 - D. SSME-out safing subtask
3. Which subtask corrects vehicle performance as a result of a failed main engine?
 - A. Boost guidance subtask
 - B. SSME-out safing subtask
 - C. Guidance input subtask
 - D. All of the above
 - E. A and B above
4. What is the purpose of adaptive first-stage guidance?
 - A. Corrects low vehicle performance resulting from an SSME failure
 - B. Enables maximum throttles
 - C. Corrects vehicle performance variations caused by SRB thrust dispersions
 - D. Stores V_{rel} and H for an RTLS abort
5. Under normal conditions, how are the SSME throttle commands issued from guidance?
 - A. Guidance does not issue throttle commands; this function is handled by the SSME operations sequence
 - B. By an open loop command schedule based on mission elapsed time
 - C. By a closed loop command schedule based on altitude
 - D. By an I-loaded throttle command table that is a function of V_{rel}
6. Why must the vehicle pitch profile be biased if an SSME fails?
 - A. To prevent a depressed trajectory resulting from an engine failure in an open loop guidance scheme
 - B. To prevent a lofted trajectory resulting from an engine failure in a closed loop guidance scheme
 - C. To prevent a depressed trajectory resulting from an engine failure in a closed loop guidance scheme
 - D. None of the above

7. How does the vehicle respond to cold SRBs?
- A. It will pitch to a more vertical attitude and throttle lower during the thrust bucket.
 - B. It will pitch to a less vertical attitude and throttle higher during the thrust bucket.
 - C. The throttle setting during the thrust bucket will always be full power.
 - D. B and C above.
 - E. None of the above.
8. How is guidance informed of an SSME failure?
- A. The SSME-out subtask sends an engine failure flag to guidance.
 - B. Guidance monitors engine chamber pressure indications.
 - C. Guidance recognizes a major deviation from the planned trajectory.
 - D. Engine fail flags are sent to first-stage guidance from the SSME operations sequence.
 - E. All of the above.
9. Assuming auto throttles, under what circumstances will the main engines be commanded to 109 percent?
- A. B, C, and D below.
 - B. MM 102 is in progress.
 - C. An SSME has failed.
 - D. When maximum throttles have been enabled by the crew on SPEC 51.
 - E. None of the above.
10. Why does the shuttle fly downrange in a heads down attitude?
- A. To ensure MPS propellant flow to the SSMEs.
 - B. To relieve structural loading during the max \dot{q} region.
 - C. This attitude is dictated by the launch pad orientation.
 - D. Decreased complexity with the SRB separation maneuver.

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Answers

1. A. The trajectory flown is created by the I-loaded attitude table contained within the boost guidance subtask (page 2-5).
2. C. The SSME throttle command table is contained within the boost throttling subtask (page 2-10).
3. E. Vehicle performance is corrected by a pitch bias added in the boost guidance subtask and SSME power level adjustments within the SSME-out safing subtask (pages 2-6 and 2-11).
4. C. Adaptive first-stage guidance detects SRB thrust dispersions and adjusts vehicle pitch and throttle profiles to restore nominal ascent performance (page 2-7).
5. D. Throttle command and relative velocity data are loaded into the flight software preflight. Relative velocity magnitude determines when a new command will be issued (page 2-10).
6. A. Without a pitch bias, the vehicle would attempt to fly the nominal trajectory profile with insufficient thrust (page 2-6).

7. B. Mission elapsed time is compared to an I-loaded reference time which determines the low SRB performance. Lost performance is regained by increasing downrange performance and the minimum throttle setting during thrust bucket (page 2-7).
8. D. SSME performance is monitored by the SSME operations sequence software, which informs guidance software of a failure with an indicator known as a fail flag (page 2-11).
9. A. All three conditions must be met in order for first-stage guidance to command the main engines up to 109 percent (page 2-4).
10. B. Given the performance available, a heads-down attitude generates the negative angle of attack required to safely fly through the maximum dynamic pressure region (page 2-1).

Section 3: Ascent Second-Stage Guidance

Objectives

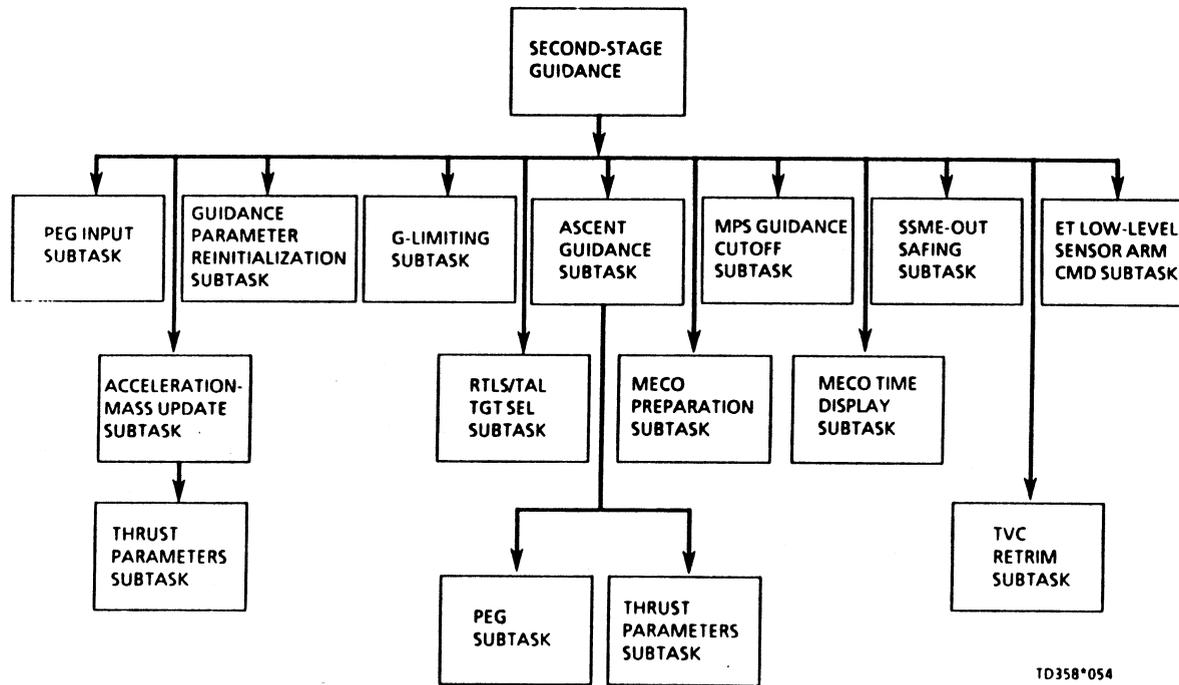
After reading this section, the student should be able to do the following:

- Explain what type of guidance scheme is used for second-stage flight
- Describe powered explicit guidance
- List the three steering parameters PEG calculates
- Explain the linear tangent guidance

Second-stage guidance is completely different from first stage. Now, guidance has gone closed loop, which means that guidance will compute each cycle where the vehicle

should be in the sky to hit the I-loaded MECO target. The vehicle "knows" its current position, via navigation, and where the desired position, the MECO target, is located. This difference determines in what direction and how long the vehicle is steered in a given direction to meet the MECO target conditions. If there are any performance variations left over from first stage, this guidance scheme will be able to correct it, provided the errors are not too large.

Second-stage guidance is initiated at the SRB separation command and is terminated at MECO confirm. If an abort to orbit (ATO) or transoceanic landing (TAL) is declared, second-stage guidance is used to hit these MECO target conditions as well. Figure 3-1 is a block diagram illustrating the elements making up second-stage guidance.



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Figure 3-1.- Second-stage guidance.

Powered Explicit Guidance

The type of guidance used for second stage is known as powered explicit guidance (PEG). Four independent PEG algorithms were developed to handle all phases of shuttle exoatmospheric flight. The four algorithms are

- PEG 1 - Used for nominal, ATO, or TAL ascents.
- PEG 4 - Used for nominal, abort-once-around (AOA)/ ATO, and deorbit OMS maneuvers. Intended for use so the cutoff velocity passes through a specific target position with a specified vertical and horizontal component of the velocity at the target.
- PEG 5 - Used for RTLS abort guidance.
- PEG 7 - Used for on-orbit maneuvers when accuracy in achieving the target conditions is not required. This guides the vehicle through a constant inertial thrust attitude to achieve a specific velocity change.

Only PEG 1 and PEG 5 will be discussed in further detail since these are the only guidance schemes involved in second-stage flight.

The algorithm known as PEG 1, usually shortened to PEG, is the scheme used for a nominal ascent. The objective of PEG is to generate commands to place the vehicle in a desired position with a specific velocity and to get there with minimal fuel usage. This position and velocity are called the target, and the target for PEG is the MECO target. This target is determined from a set of I-loaded (flight-specific) target parameters. Looking at figure 3-2, the parameters defining the MECO target are as follows:

- Inertial velocity (feet/second)
- Flightpath angle (radians)
- Radius from Earth center (feet)
- Desired orbital plane

Although the capability exists to take the external tank (ET) into orbit, the MECO targets are selected to ensure correct placement of the ET footprint in relation to populated areas (land masses). The inertial velocity is chosen to fall short of orbit insertion but close enough for the OMS to insert the orbiter into orbit. This keeps the ET out of orbit so its reentry can be controlled. The flight-path angle is the angle between the velocity vector and local horizontal. It merely specifies the ratio of vertical to horizontal velocity component at MECO. The radius is specified from Earth center in feet, which really specifies the MECO altitude. The desired orbital plane (IY) is defined by the longitude of the ascending node and the inclination. The orbital plane, also known as trajectory plane or insertion plane, is dictated by mission objectives. Typical MECO targets are shown in table 3-1.

Table 3-1.- MECO targets.

Target	Inertial velocity (ft/sec)	Flight-path angle (deg)	Altitude (x1000 ft)	Apogee (n.mi)	Perigee (n.mi)
Standard insertion	25725	0.675	360	85	0
Direct insertion	25871	0.792	360	160	35
ATO	25871	0.792	360	Variable	Variable
TAL	22500-25000	0.139	360	N/A	N/A

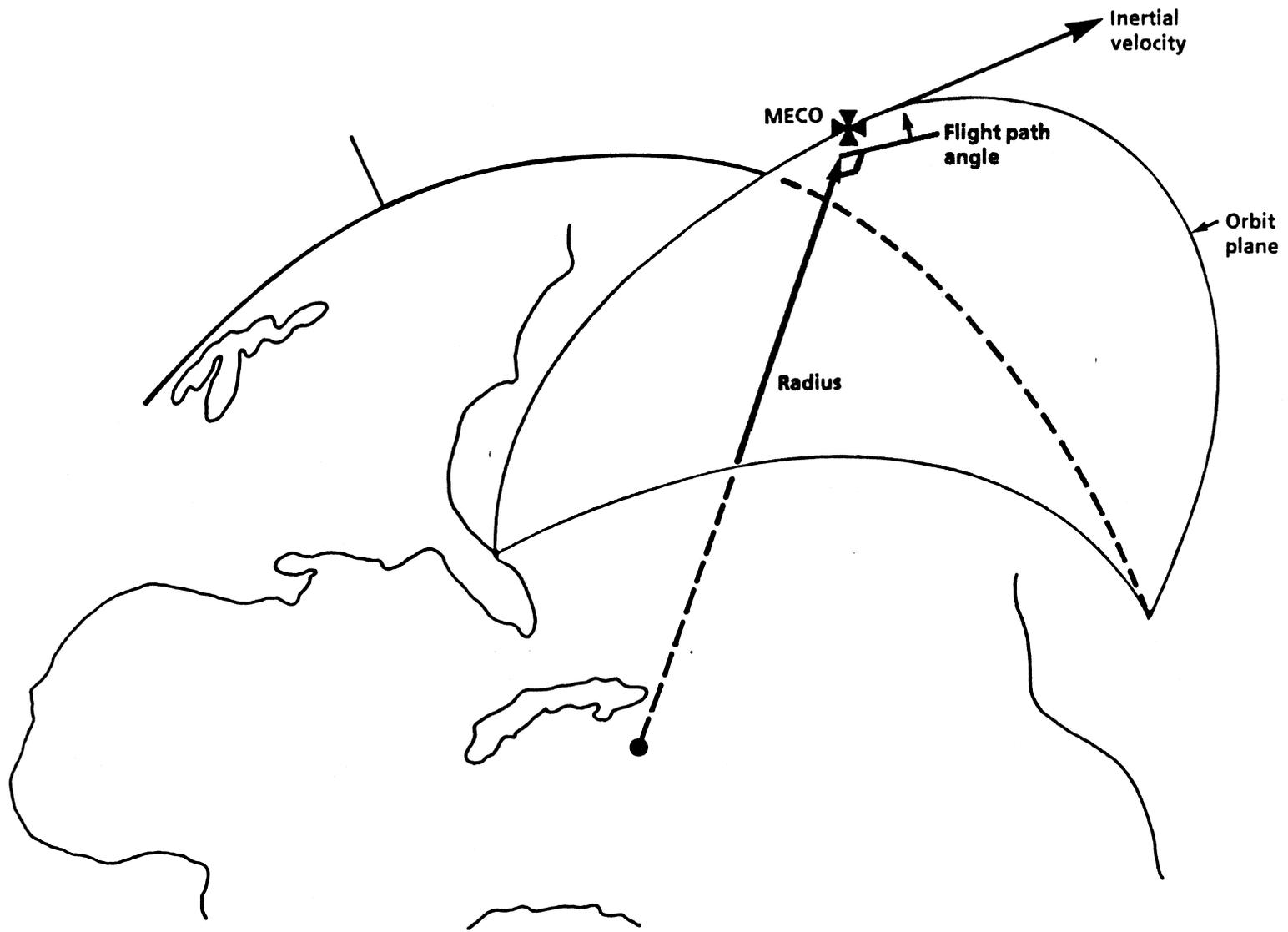
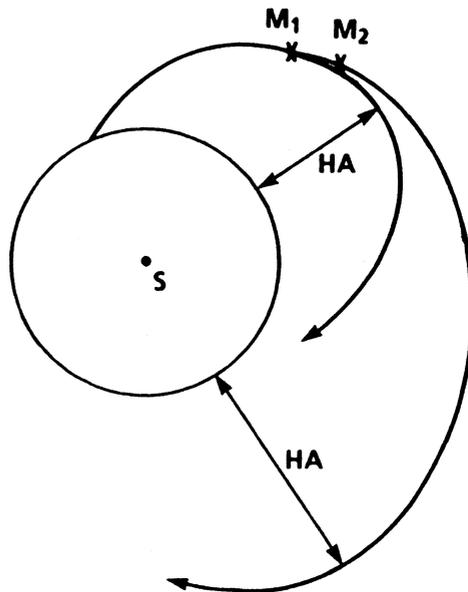


Figure 3-2.- MECO target parameters.

Standard Insertion Vs. Direct Insertion

A standard insertion is one in which the MECO targets are chosen to require an OMS-1 maneuver. With a direct insertion ascent, the main engines are burned slightly longer to achieve a higher inertial velocity. As illustrated in figure 3-3, this creates a higher post-MECO apogee (apogee altitude, HA) such that an OMS-1 maneuver is not required. In this case, the OMS-2 maneuver is used for orbital insertion/circularization. There are many programmatic decisions which determine whether a particular flight will have a standard or direct insertion.



M₁ = Standard insertion MECO
M₂ = Direct insertion MECO

Figure 3-3.– Standard vs. direct insertion.

Some of the aspects considered are mission objectives, ascent performance, ET impact, orbital altitude, etc. In general, a direct insertion will be considered for orbital altitudes of 160 nautical miles or higher; otherwise, the flight will have a standard insertion ascent.

The big advantage to direct insertion is that higher orbital altitudes are possible because more OMS propellant is saved for orbital maneuvers. Note in figure 3-4 that direct insertion also aids in downrange and AOA abort capabilities because the higher velocity results in a greater ballistic arc trajectory.

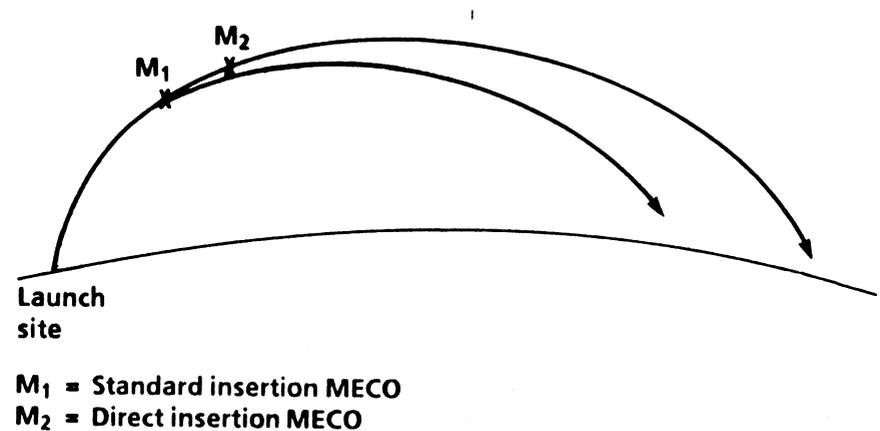


Figure 3-4.– Downrange abort capability.

The PEG Algorithm

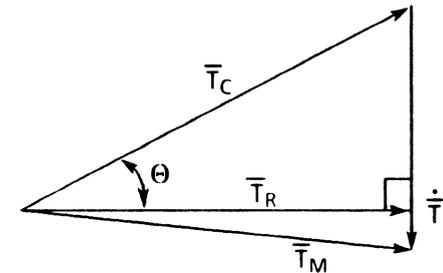
The PEG algorithm is an iterative process which first computes steering parameters and, using these parameters, predicts the vehicle state at cutoff. Subsequently, PEG will correct its prediction by providing desired thrust direction and thrust turning rate information to the flight control system (FCS). To "guide" the vehicle in a fuel optimal manner, the following steering parameters must be computed:

- Reference thrust vector
- Thrust turning rate vector
- Reference time

The purpose of each steering parameter is outlined below.

- Reference thrust vector - Provides a starting point or reference point for computing changes in thrust direction. It is a unit vector in the direction of the velocity to be gained (V_{GO}), indicating the constant thrust direction which would achieve the desired velocity.
- Thrust turning rate vector - The vector indicating the direction and rate at which the thrust direction must turn to achieve the desired position. This vector is perpendicular to the reference thrust vector.
- Reference time - The calculated time such that midway through the burn, the average thrust direction is parallel to the reference thrust vector, which is parallel to V_{GO} . This will achieve the desired position and velocity at MECO. Essentially, it is calculated by taking the burn time remaining until MECO (time to go (T_{GO})) and dividing it by 2.

A fuel optimum trajectory is achieved when the tangent of the thrust attitude angle varies linearly as a function of the flight time. The thrust attitude angle is the angle formed between the current thrust direction and the reference thrust vector. All linear tangent guidance does is to make the total change in thrust direction occur over the total maneuver time, T_{GO} . This is required in order to achieve the MECO position. The thrust attitude angle and the thrust steering law is shown in figure 3-5. To thoroughly understand this figure, one must also look at figure 3-6.



Where:

- Θ = Thrust attitude angle
- \bar{T}_C = Current thrust vector
- \bar{T}_M = Thrust vector at MECO
- \bar{T}_R = Reference thrust vector
- $\dot{\bar{T}}$ = Thrust turning rate vector

Note:

- $\Theta \approx 25^\circ$ at SRB staging
- $\Theta \approx -2^\circ$ at MECO command

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Figure 3-5.- Thrust steering law.

It is important to note in figure 3-6 that the second-stage maneuver is generally a fly-down maneuver necessitated from the lofting performed in first stage. Lofting is required to reduce aerodynamic drag as rapidly as possible so downrange velocity can be gained. Therefore, a large positive vertical velocity (H) exists at SRB staging, resulting in the velocity vector shown in figure 3-6. This vector must be rotated downward such that it is pointed in the required direction at MECO. To accomplish this, the current thrust vector is "aimed" low to arrest the H and increase velocity in the downrange direction. The resulting velocity vector is rotated more downward, as illustrated at the maneuver midpoint. However, the velocity vector is still not in the direction required for

MECO, and further steerdown is necessary. Notice that, just prior to MECO, the velocity vector is nearly parallel to and has almost the same magnitude as the MECO vector (MECO target). Notice, also, that the thrust attitude angle has decreased linearly with time such that at MECO the velocity vector has rotated into the direction and velocity required for MECO. Now, mentally superimpose the average (over the entire maneuver) reference thrust vector between the current thrust vector at SRB staging and the thrust vector just prior to MECO; this image should look like the thrust steering law illustrated in figure 3-5. Figure 3-6 illustrates the second-stage maneuver in the radial direction. Crossrange steering is performed the same way.

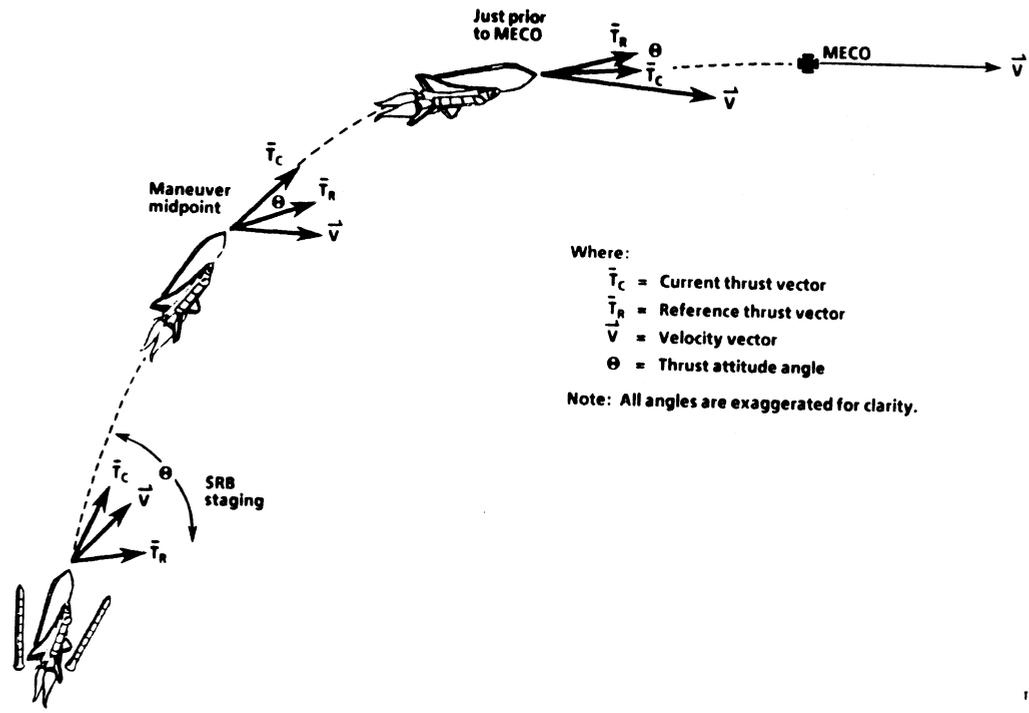


Figure 3-6.- Second-stage maneuver (not to scale).

The PEG Task

Since PEG is iterative in design, it calculates the steering parameters for the entire maneuver at one instant in time and updates them on each successive pass. PEG executes these computations once every 1.92 seconds. Each cycle through the PEG calculations is known as a guidance cycle. Therefore, approximately once every 2 seconds these steering parameters are updated to flight control (G/C steer) to steer the vehicle to the target. Unfortunately, this is not all there is to PEG. On each pass, PEG must construct the information required to compute the steering parameters. The process by which PEG constructs this information and builds upon it to compute these parameters will be discussed next. We will discuss each subtask within PEG in the order in which it is executed in the software. The reader is advised to read through page 3-25 before studying the details of any subsection.

Position Magnitude Subtask

This subtask calculates the magnitude of the current vehicle position vector to be used later on in the PEG calculations. This is accomplished simply by taking the absolute value of the position vector to come up with the magnitude.

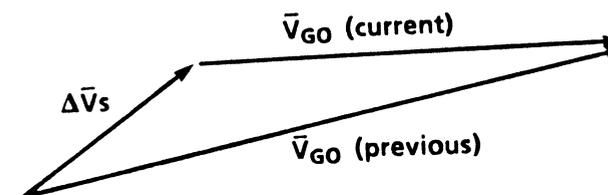
Initialization Subtask

On the first pass only, this subtask initializes dependent parameters. Since PEG is a cyclic process that uses the last cycle outputs as inputs to the current cycle, certain parameters must be initialized in order for it to start the calculations necessary for program execution. Examples of the

parameters that are required to be initialized are T_{GO} , V_{GO} , software position constraint switches, desired cutoff vectors, etc. Once this subtask is entered, a software switch is set to off which prevents reentry into this subtask.

V_{GO} Update Subtask

This subtask updates the V_{GO} vector because the steering parameters and the burn are dependent on this vector. Since the vehicle has undergone thrust acceleration during the last cycle, the V_{GO} vector can be updated by subtracting the velocity change over the last guidance cycle. Figure 3-7 shows how this velocity vector is calculated.



$$\bar{V}_{GO} \text{ (current)} = \bar{V}_{GO} \text{ (previous)} - \Delta \bar{V}'s$$

where $\Delta \bar{V}'s$ = Velocity vector change accumulated since last guidance cycle

Figure 3-7.- V_{GO} update calculation.

Remember, the velocity change information comes from the inertial measurement units (IMUs). Once \bar{V}_{GO} is known, the burn T_{GO} and the thrust integrals can be evaluated.

T_{GO} Subtask

The purpose of this subtask is to calculate the total burn T_{GO} until the end of the maneuver, MECO. T_{GO} is a necessary parameter required for several calculations within PEG. These calculations are the thrust integrals and the range to be gained (R_{GO}) subtasks. T_{GO} is also used to predict the thrust cutoff time, which is used for display purposes as well. The thrust cutoff time is calculated by the following equation:

$$\text{Thrust cutoff time} = \text{current time} + T_{GO}$$

The T_{GO} computations are complicated by the fact that PEG assumes that two distinct phases will occur. Three phases actually exist, but the first, extending from SRB separation to T-fail, is bypassed by an I-load of zero. However, a discussion of T-fail is necessary to understand how T_{GO} until MECO is calculated. T-fail is an I-loaded hypothetical mission elapsed time (MET) at which PEG assumes an SSME will fail. This time corresponds to the earliest time an AOA abort is possible. Early in the second-stage trajectory, it is desirable to prevent an abort gap between the latest RTLS and earliest AOA capability, should an SSME fail. In order to prevent or reduce an abort gap, a higher than normal trajectory must be flown. This is known as abort shaping or lofting. Lofting is required only when the flight performance is critical, as in the case of a heavyweight vehicle. The engine failure assumption at T-fail causes PEG to loft the trajectory such that the MECO target can be reached using only two SSMEs. In other words, PEG plans for a loss of performance at T-fail. The resulting lofted trajectory achieves AOA capability earlier in the flight profile because of the potential energy gain.

At T-fail, PEG stops assuming that an engine failure will occur and the remainder of second stage will assume three active SSMEs. Barring any engine failures, PEG will eventually fly out the loft because of the downrange performance gained. T_{GO} for this phase is calculated with the following equation:

$$T_{GO_1} = T\text{-fail} - \text{present MET}$$

where

$$T_{GO_1} = \text{Burn time remaining until T-fail is reached (sec)}$$

As mentioned earlier, for most flights T-fail is I-loaded to zero MET, thus causing the first phase of the T_{GO} calculation to be bypassed. This can be done because increased confidence in the TAL abort covers the gap between last RTLS and earliest AOA. The two phases remaining are constant thrust and constant acceleration. Figure 3-8 illustrates these phases.

The constant thrust phase, as the name implies, means a constant power level is maintained on all three SSMEs to obtain the required performance. The constant acceleration phase (3g) is achieved by commanding the SSMEs to a lower power level that maintains a constant acceleration on the vehicle. This phase is required to prevent exceeding the structural limitation of 3.5g on the airframe (spaceframe?).

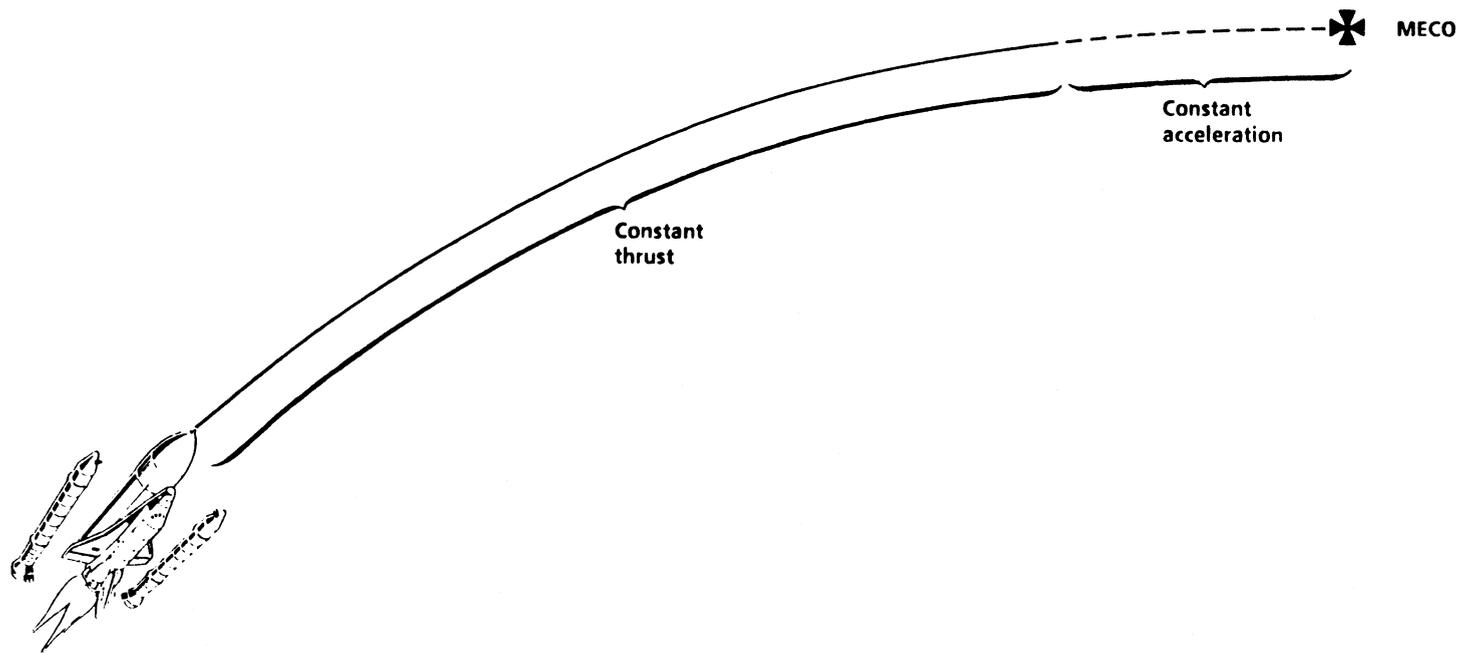


Figure 3-8.— Second-stage flight phases.

The constant thrust phase extends from the SRB separation command until the constant acceleration phase (≈ 1 minute prior to MECO) is entered. Within the constant thrust phase is a simpler and more favorable method to loft the trajectory for abort shaping. This method uses a pseudo MECO target. A pseudo MECO target is a "fake" target that is used until an I-loaded time is reached. When this occurs, the pseudo targets are replaced with the nominal MECO targets. The primary difference between the pseudo and nominal MECO targets is the targeted radius. Figure 3-9 shows that the pseudo MECO radius is higher than nominal to achieve the required trajectory lofting. T_{GO} for the constant thrust phase is calculated independently of the MECO targets and is a function of when the constant acceleration phase is reached. T_{GO} is calculated using the current V_{GO} , exhaust velocity (VEX) of the thrusting SSMEs, current acceleration (estimated), and overall vehicle mass. The following equation shows how T_{GO} is calculated for the constant thrust phase.

$$T_{GO_2} = \frac{M}{\dot{M}} - \frac{VEX}{CA}$$

where T_{GO_2} = Burn time remaining until the constant acceleration phase is reached

M = Current vehicle mass (slugs)

\dot{M} = Total vehicle mass flow rate (slugs/sec)

VEX = Exhaust velocity (function of the number of thrusting SSMEs, fps)

CA = Constant acceleration (3g, fps²)

Basically, all this equation represents is that acceleration increases as the overall vehicle mass decreases due to the burning of propellant (MPS, OMS, and reaction control system (RCS)). From this equation, whenever T_{GO} is less than or equal to a small positive constant, the constant acceleration guidance phase has been reached.

T_{GO} for the constant acceleration phase is strictly a function of the remaining V_{GO} and how fast the vehicle is getting there (≈ 96 ft/sec² = 3g). Therefore, T_{GO} can be expressed by the following equation:

$$T_{GO_3} = \frac{V_{GO}}{CA}$$

where T_{GO_3} = Burn time remaining for the constant acceleration phase

V_{GO} = Velocity to be gained (fps)

CA = Constant acceleration (fps²)

There are times when the constant acceleration phase will not be entered before MECO. This is caused by a heavy-weight vehicle and/or degraded SSME performance (i.e., engine failure). Also, if an engine fails during the constant acceleration phase, the acceleration would decrease and the constant thrust phase would be reentered. Finally, the T_{GO} until MECO is obtained by adding the T_{GO} for each phase together

$$T_{GO_1} + T_{GO_2} + T_{GO_3}$$

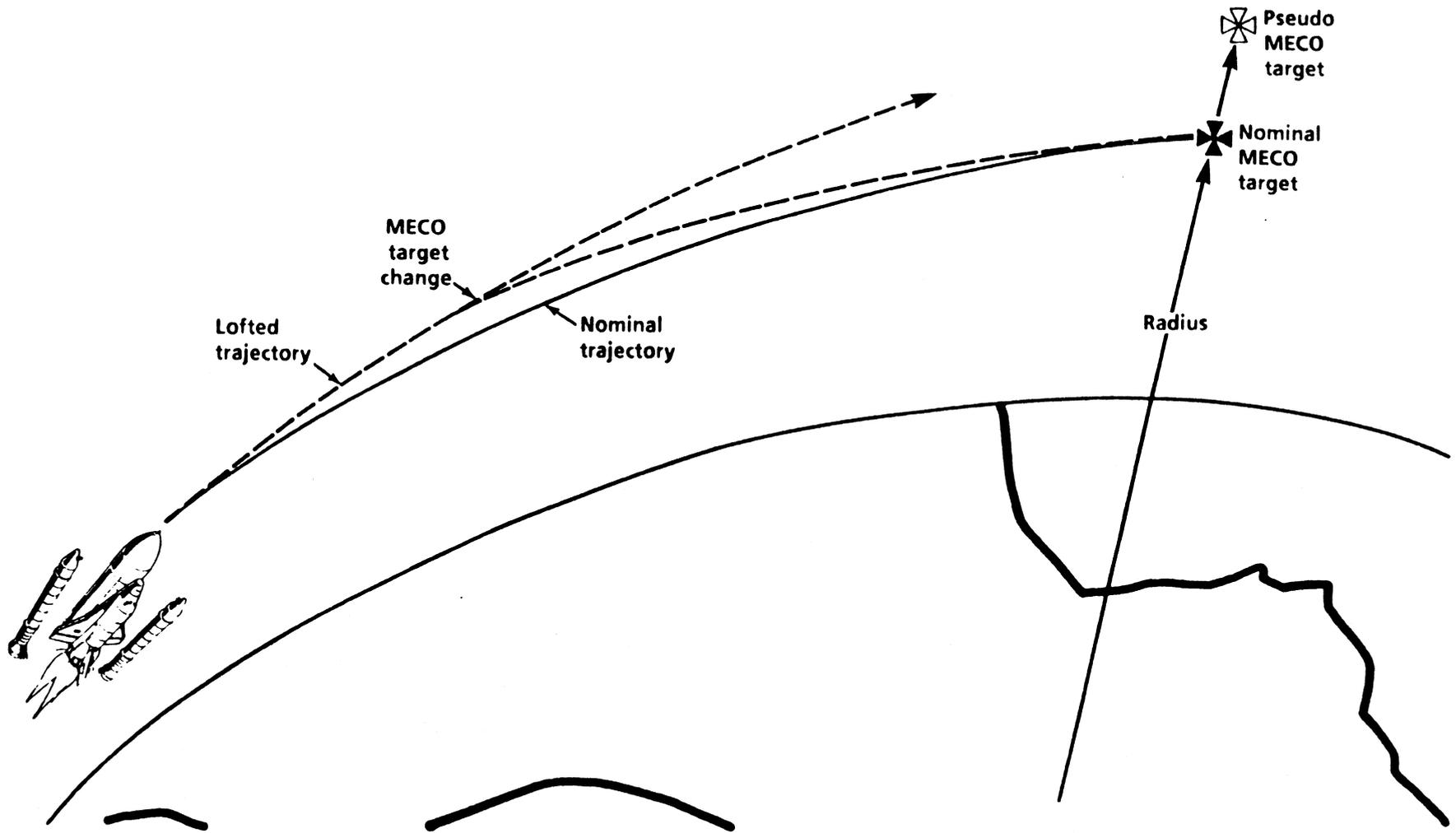


Figure 3-9.- Pseudo MECO target.

Thrust Integrals Subtask

In order for PEG to steer the vehicle (using the linear tangent guidance law, figure 3-5), the thrust magnitude must be evaluated over the burn time, T_{GO} . That is exactly what this subtask does. It evaluates various time integrals of force over mass, which is the equation of motion for the linear tangent steering law; i.e., how much force is moving how much mass over what period of time to move the vehicle from A to B.

The calculations for these integrals require the burn times for each phase (from the T_{GO} subtask) and basic thrust parameters provided by guidance tasks external to PEG (from the thrust parameters subtask). Two sets of thrust integral equations are evaluated. One set is used for the constant thrust phase and the other is used for the constant acceleration phase. Next, these two integrals are summed to give the total integral for the maneuver required to reach MECO. They are then used to compute the following:

- The velocity and position change due to thrust
- R_{GO}
- Thrust turning rate vector

The final computation within this subtask is the time associated with the reference thrust vector. Recall that the reference time is chosen so that the average thrust direction is parallel to the reference thrust vector in order to achieve the required position and velocity. This time is essentially $T_{GO}/2$ (average). In reality, this time is not an average of T_{GO} . The reference time is a little larger than the average because the thrust is not constant throughout the entire maneuver.

Reference Thrust Vector Subtask

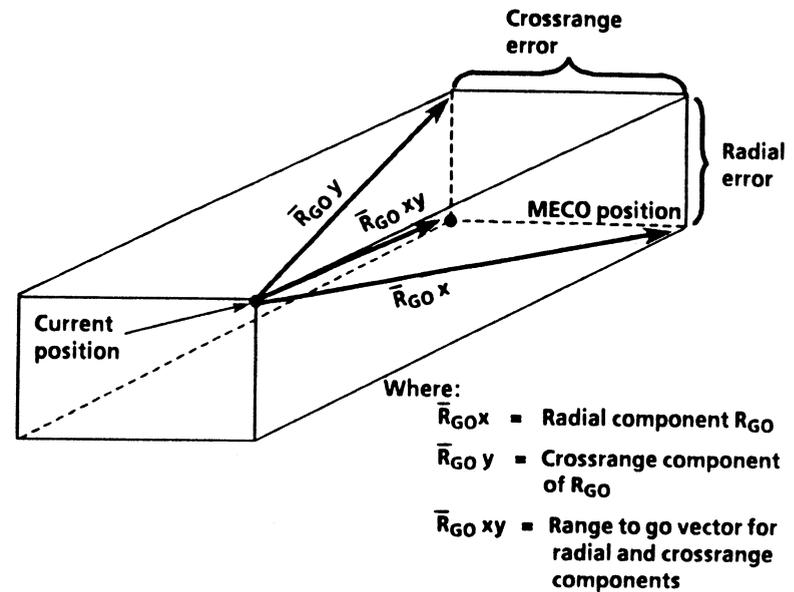
This subtask simply determines the reference thrust vector for all guidance modes. Since this vector is a unit vector, it is rather easily calculated by the following method:

$$\text{Reference thrust vector} = \frac{\text{Velocity to be gained vector}}{\text{Magnitude of } V_{GO} \text{ vector}}$$

If the magnitude of V_{GO} is zero, PEG is exited.

RGO Subtask

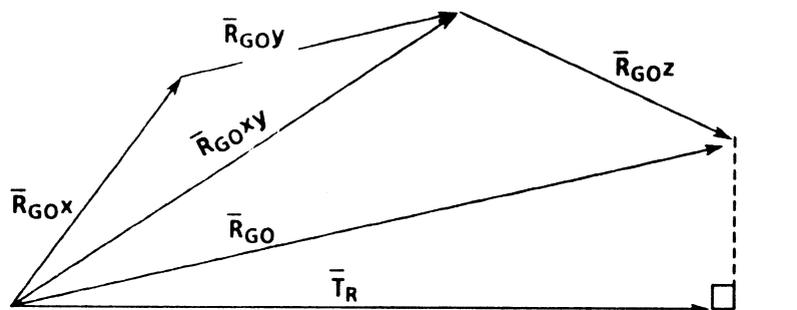
This subtask is required in order to determine the thrust turning rate vector. Since this vector must steer the vehicle to the desired MECO conditions, the commanded thrust direction vector requires an estimate of the position error at cutoff. As can be seen in figure 3-10, this error is what is "steered out" by the thrust turning rate vector before MECO so the desired position can be attained. This error estimate is computed from the thrust integrals, computed in thrust integrals subtask, and the R_{GO} vector. R_{GO} represents the range (position) change that must be accomplished by thrusting in order to reach the required position at MECO.



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Figure 3-10.— Calculation for radial and crossrange components of R_{GO} .

Figure 3-10 illustrates only how the radial and crossrange components of \bar{R}_{GO} are calculated. The downrange component is never constrained because it would be too fuel inefficient to hit a single point in the sky. However, the downrange component must be considered in the \bar{R}_{GO} calculation as well. Note in figure 3-11 that the downrange component of \bar{R}_{GO} is calculated as the \bar{R}_{GO} that is left over in the downrange direction as a result of adjusting the burn to meet the radial, crossrange, and reference thrust vector constraints.



Where
 \bar{R}_{GOz} = Downrange component of \bar{R}_{GO}

Figure 3-11.— \bar{R}_{GO} definition.

Once the current position vector and the desired cutoff position vector (\bar{R}_{GO}) are known, the turning rate vector can be computed from the resulting position error shown in figure 3-12. In summary, the turning rate vector is a function of \bar{R}_{GO} , reference thrust vector, and thrust integrals.

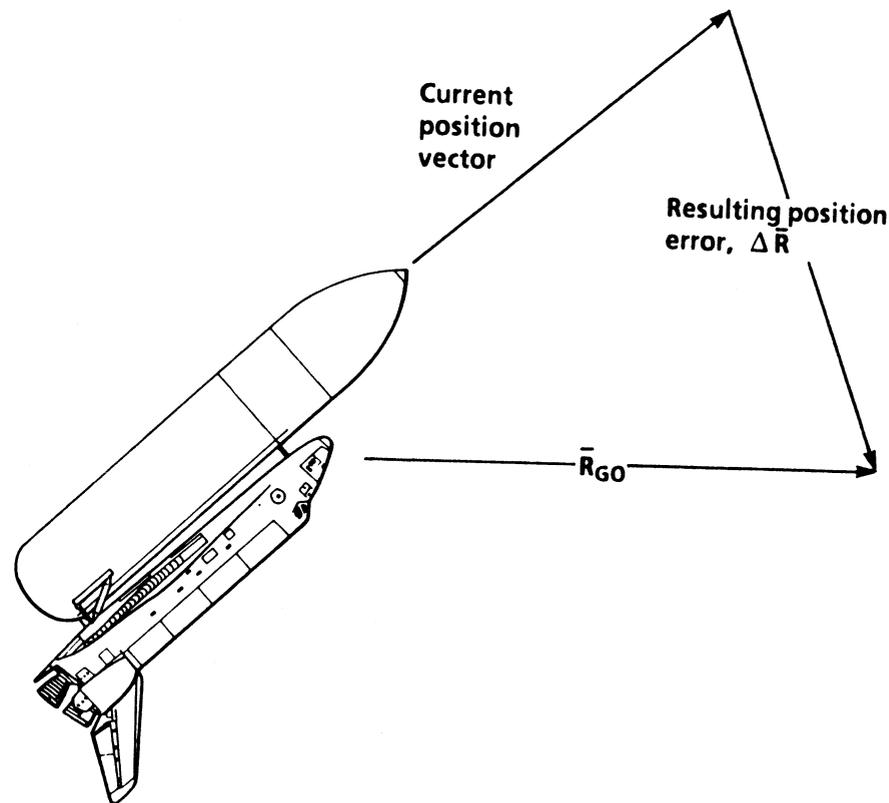


Figure 3-12.— Position error.

Thrust Turning Rate Vector Subtask

As you have probably noticed by now, as we go deeper into the PEG calculations, each subtask relies on information previously calculated. This subtask calculates the thrust turning rate vector based on previously calculated parameters. This vector is determined by using the position error that was calculated in the R_{GO} subtask and the T_{GO} , which was calculated in the T_{GO} subtask. Figure 3-13 illustrates this point. These parameters determine the direction and rate at which the thrust direction must rotate to achieve the MECO position.

If guidance calculates a large position error with respect to time, a large turning rate would be required. Of course, the converse would hold true as well. If the turning rate vector is too large, the vehicle will exceed its structural limitations. In this case, an I-loaded maximum turning rate is used and the reference thrust vector and R_{GO} are recomputed. In addition, the guidance convergence flag is set to off until the required turning rate vector becomes less than the maximum turning rate vector.

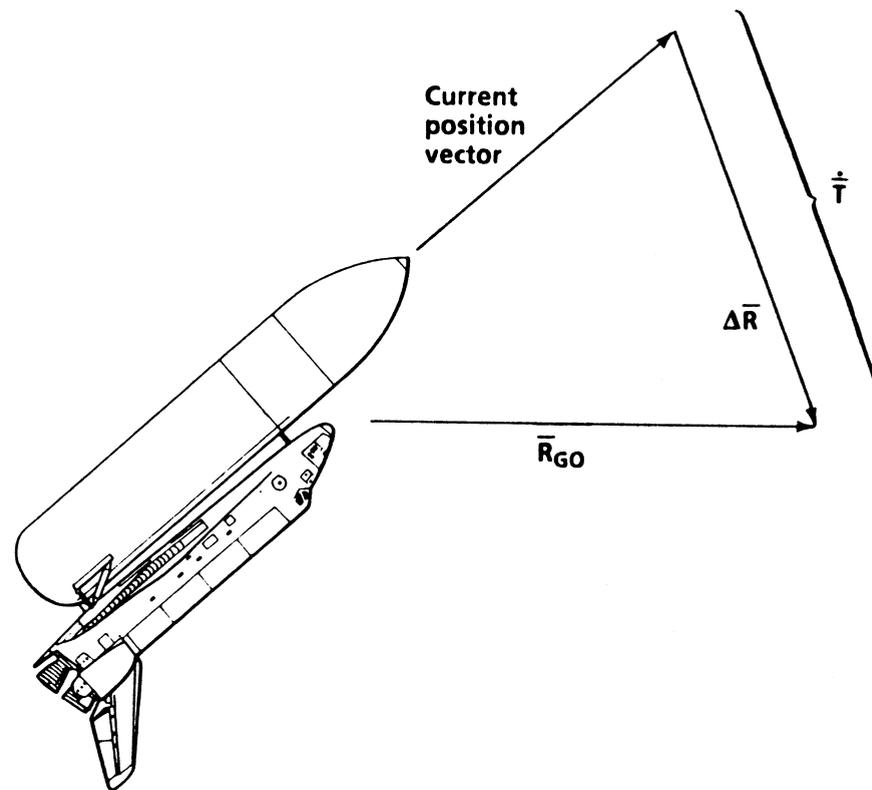


Figure 3-13.- Thrust turning rate vector.

Steering Inputs Update Subtask

This subtask provides to G/C steer the new steering parameters that have just been calculated. These parameters are updated only if the predicted cutoff time has not significantly changed since the previous guidance cycle. Failure to satisfy this condition implies that PEG is not converged. When this test is failed, G/C steer will retain the last valid commands received while PEG processing continues.

Burnout State Vector Prediction Subtask

Since PEG computes new values for the steering parameters based upon current conditions, errors can develop within its computations. To reduce these errors, PEG incorporates a predictor/corrector algorithm. This subtask is part of the prediction process. Its purpose is to predict the position and velocity at cutoff to be used in the corrector step to correct V_{GO} . This is accomplished by computing the thrust and gravity effects on position and velocity over the burn arc. This computation assumes that the current steering parameters are used for the rest of second stage. This is illustrated in figure 3-14. Do not confuse this subtask with the R_{GO} subtask. This subtask predicts position and velocity at cutoff, whereas the R_{GO} subtask calculates only position to be gained.

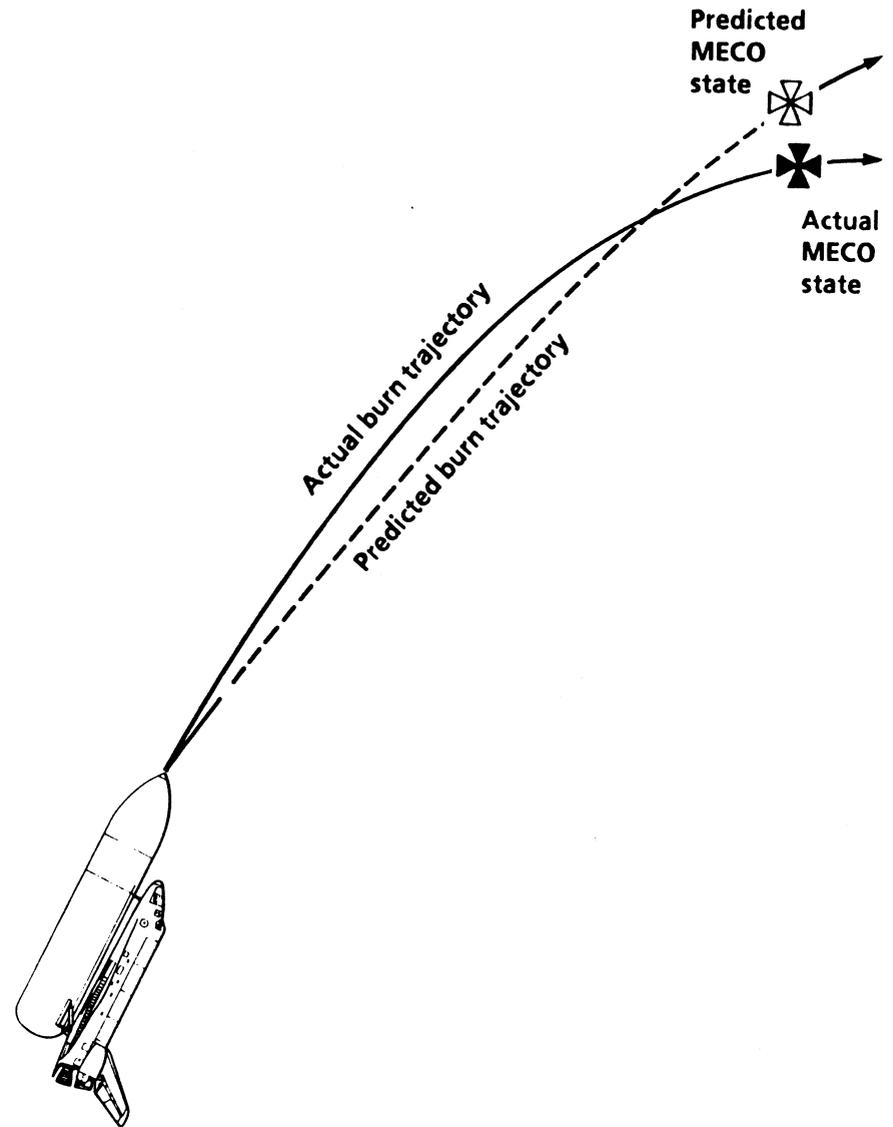


Figure 3-14.— Burnout state vector prediction.

Desired Orbit Plane Correction Subtask

This subtask is part of the corrector step in the PEG algorithm. The ascending node and inclination within the MECO targets define the orbit plane. The desired orbit plane is constrained by PEG until just prior to MECO, at

which time the orbit plane is released. As shown in figure 3-15, this subtask constrains the desired cutoff position vector to be in the desired plane. If, however, the orbit plane is unconstrained, the desired position will be in the predicted cutoff orbit plane, as calculated in the previous step.



Figure 3-15.— Orbit plane correction.

Desired Position Subtask

This subtask is also another step in the corrector part of PEG. This subtask corrects the errors in the radial direction back to the desired MECO position. Figure 3-16 illustrates this. This is true only if the radial component from Earth center is being constrained. If not, this subtask is not entered.

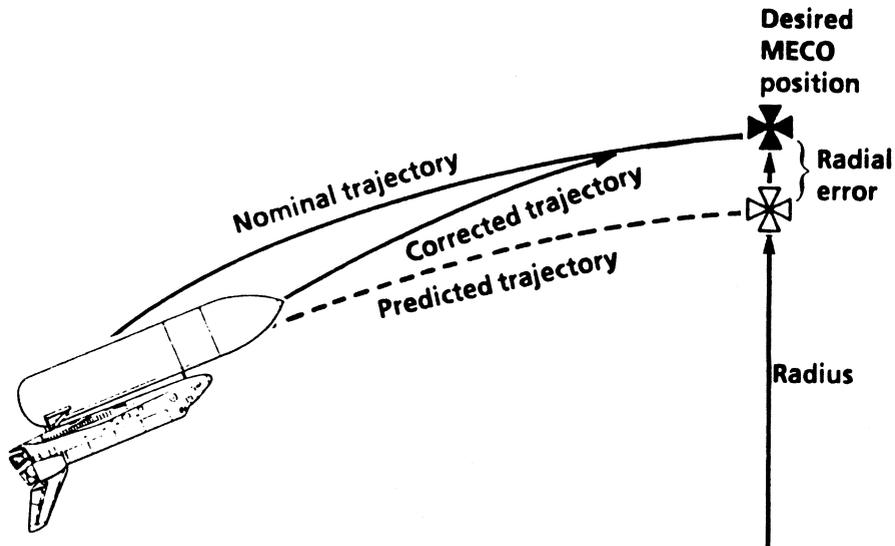


Figure 3-16.- Correction of radial error.

Desired Velocity Subtask

This subtask calculates the desired velocity vector. PEG already knows the desired velocity magnitude from the I-loaded MECO target, but not the direction. Referencing figure 3-17, the velocity vector is obtained from the velocity magnitude, flightpath angle, and orbital plane which yields direction. This velocity vector is required for the next step.

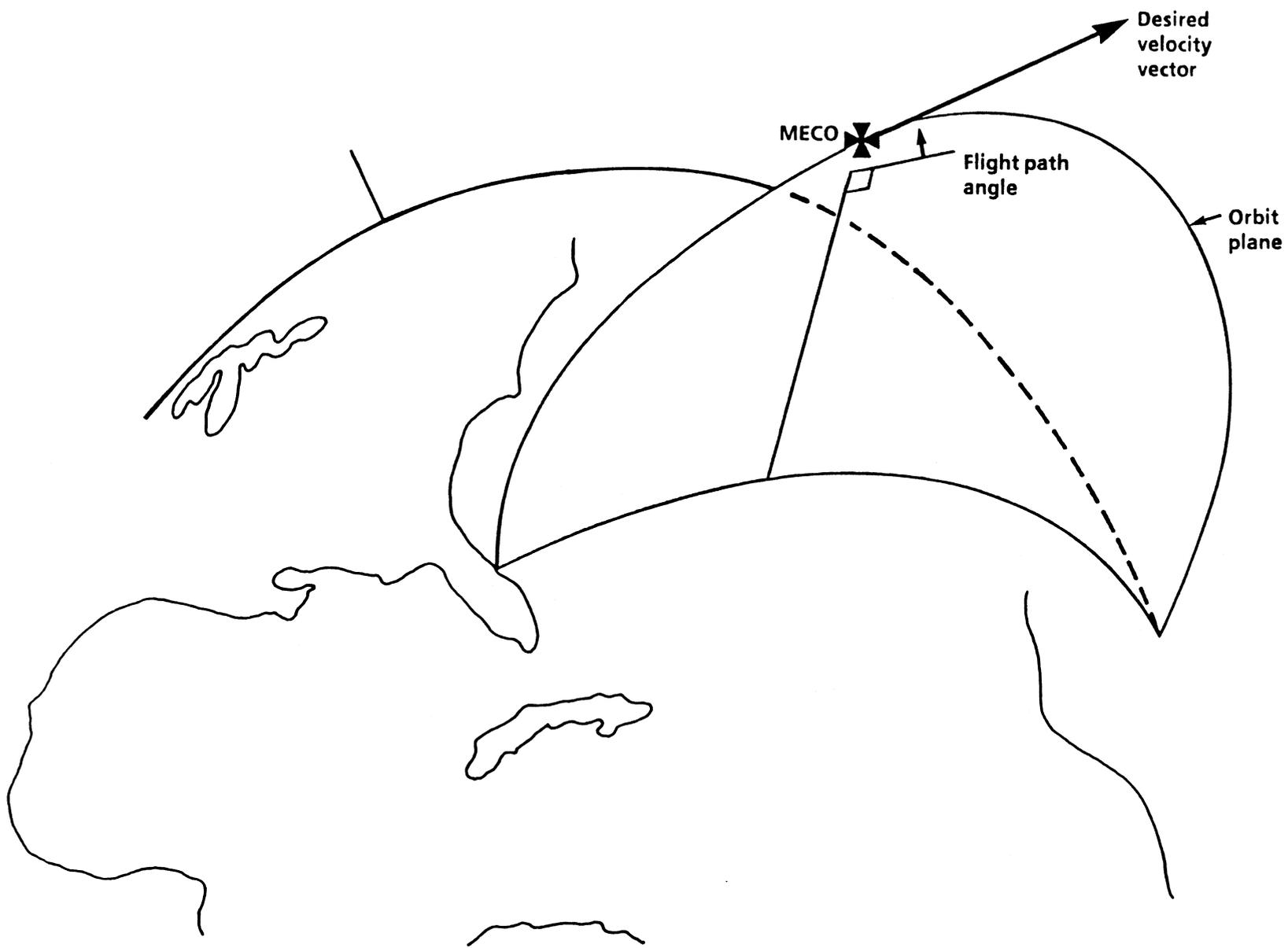
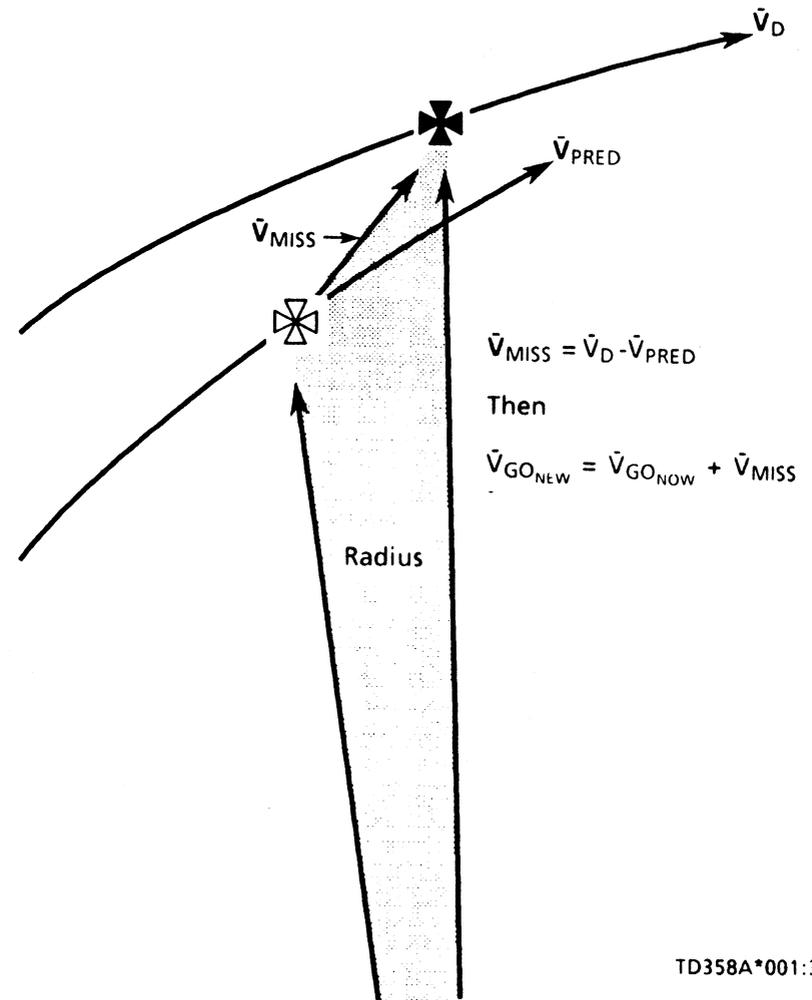


Figure 3-17.- Velocity vector definition.

VGO Correction Subtask

Using the current steering parameters, this subtask calculates a miss in the desired velocity resulting from a miss in the MECO target. This velocity miss is used to calculate a delta to the VGO vector so it may be corrected to the required VGO (figure 3-18). This corrected VGO is used by the VGO update subtask to start another guidance cycle and compute new steering parameters.



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Figure 3-18.- V_{MISS} computation.

Convergence Check Subtask

The purpose of this subtask is to determine if the V_{GO} is converged and to generate a flag indicating whether or not V_{GO} is converged. Guidance will go unconverged when the velocity miss is greater than 1 percent of the magnitude of the current velocity to be gained for five successive PEG cycles. This condition implies that the predicted state obtained from the previous steering parameters is not very close to the desired state.

If guidance does go unconverged, a discrete is sent to the attitude display indicator (ADI) processor to stow the ADI guidance error needles. This provides a cue to the crew to take control stick steering (CSS) and fly the ascent manually via cathode-ray tube (CRT) displays and flight data file (FDF). This is necessary because, when guidance goes unconverged, the vehicle will be in attitude hold utilizing the last guidance commands to flight control.

A first estimate of V_{GO} during the guidance cycle following the initiation of PEG is obtained. This first estimate will be in error by a substantial amount. For this reason, the first cycle steering parameters are not used for vehicle control. Instead, these parameters are used to start the predictor/corrector process. The corrected V_{GO} computed in the second cycle will be better than the initial estimate since the errors produced by the first cycle steering parameters have been reduced. As powered flight continues toward the MECO target, PEG will continue to increase in accuracy. Once again, during the period between PEG initiation and satisfying the convergence criterion, the vehicle will be in attitude hold, using the last valid attitude command received. It should take no longer than four passes through the PEG algorithm to achieve convergence. When convergence is achieved, a change in thrust attitude is to be expected since the attitude commanded by PEG will not, in general, be exactly the same as the attitude at PEG initiation.

Cutoff Position Constraint Release Subtask

This is the final function of the correction step. Its purpose is to release the position constraints on the radius and orbit plane near MECO for flight control stability. Basically, when T_{GO} is small, a small change in position error will provide large changes in the thrust turning rate vector, and overcontrolling will result. To prevent this problem, the position constraints are released 40 seconds prior to MECO when guidance is converged. When guidance is unconverged, these constraints are released 34 seconds prior to MECO.

External Subtasks

There are other subtasks external to PEG which are used to support the PEG calculations. They are necessary in order to provide PEG with information such as the number of active SSMEs, 3g throttling, when to terminate guidance, etc. These subtasks are as follows:

- TVC retrim subtask
- ET low-level sensor arm command subtask
- G-limiting subtask
- MPS guidance cutoff subtask
- MECO preparation subtask
- PEG input subtask
- SSME-out safing subtask
- Acceleration mass update subtask
- Thrust parameters subtask

TVC Retrim Subtask

When the vehicle mass first becomes less than a premission I-loaded mass ($\approx 500,000$ pounds), this subtask sends a retrim discrete to flight control. This discrete informs flight control to unparallel the main engines in the pitch axes. This is a course adjustment to prevent an out-of-trim condition, flight control saturation, and engine bell collision close to MECO. The autotrim integrator will take care of fine adjustments. Pitch paralleling and the autotrim integrator are discussed in greater detail in section 5.

ET Low-Level Sensor Arm Command Subtask

The ET low-level sensor arm command subtask enables the low-level MPS sensors to be armed when the current mass first becomes less than a premission I-loaded mass ($\approx 361,000$ pounds regardless of manual or auto throttles) or when two SSME's have failed. Arming these sensors based on mass is necessary to prevent an early MECO should two out of four low-level sensors fail early in the ascent. When two SSMEs have failed, the sensors are armed because the second engine failure is assumed to be catastrophic. In this situation, MPS propellant could be leaking overboard; therefore, guidance would not be able to accurately track the actual vehicle mass.

G-Limiting Subtask

This subtask comes into play approximately 1 minute prior to MECO when the vehicle reaches 3g acceleration. To prevent overstressing the vehicle, the SSMEs must be throttled down to maintain a constant acceleration of 3g. This subtask does just that. It determines when the 3g limit has been achieved based on measured vehicle acceleration from the IMUs. It then computes SSME throttle commands to limit the average acceleration to 3g. If an SSME fails during 3g throttling, the two remaining engines will be commanded back up to full power by the SSME-out safing subtask, provided guidance knows of the failure. If guidance does not see an engine as being failed, the remaining engines will stay at their last commanded throttle setting until 3g are again reached. Remember, this subtask only throttles down as a result of acceleration; it will not throttle engines back up.

MPS Guidance Cutoff Subtask

The purpose of this subtask is to compute a desired SSME cutoff time based on the desired cutoff velocity magnitude. The cutoff time calculation includes the predicted velocity change from the time the minimum throttle setting is commanded until cutoff and the predicted tailoff impulse from each active SSME. This is known as fine countdown or "fine count." For a nominal ascent, this time is 10 seconds, and for RTLS it is 6 seconds.

Two additional functions of this subtask are as follows:

- To command minimum throttle setting
- To command MECO

The minimum throttle setting is required to maintain an adequate net positive suction pressure for the SSMEs in preparation for a stable shutdown. If three engines are running, the minimum command is 65 percent; if less than three engines, the setting is 91 percent. This minimum setting is required for ≈ 6 seconds prior to shutdown.

The MECO command is issued when T_{GO} becomes less than two guidance cycles. This ensures an accurate guided cutoff. As soon as the MECO command is set, PEG is exited.

MECO Preparation Subtask

This subtask is performed from SRB separation until the MECO preparation discrete is set. The only thing this subtask does is set this discrete when T_{GO} is less than 10 seconds. This discrete is used by other software sequences for reconfiguration. For example, if an interconnected OMS dump were in progress close to MECO, the MECO preparation discrete would stop the dump and reconfigure the OMS/RCS valves back to straight feed.

PEG Input Subtask

This subtask provides PEG with required navigation and engine configuration information. The velocity vector and timetag associated with this vector are the navigation information. Obviously, in order to reach the MECO target, the number of active thrusting engines must be provided to PEG as well. This number includes both OMS and RCS engines as well as the SSMEs. The engine configuration must be known in order for PEG to keep track of vehicle mass and thrust calculations.

Thrust Parameters Subtask

The thrust parameters subtask computes the estimated thrust and vehicle performance parameters for the necessary operation of PEG. These parameters are total thrust force, total mass flow rate, and VEX. The estimated main engine thrust force is calculated as a function of the current throttle setting, the number of thrusting SSMEs, and an estimate of thrust force for a single main engine (I-load). The total thrust force is then calculated by adding the force of all thrusting engines, including the OMS and RCS engines.

Next, an estimate of the mass flow rate (which is really the weight of the propellant being burned off) is calculated. This, again, is a function of current SSME throttle command, number of thrusting engines, and an I-loaded estimate of mass flow rate for a single main engine. This estimate is added to the estimated flow rates for the OMS engines and RCS jets to derive the total mass flow rate. The total mass flow rate has to be calculated so that an estimate of the vehicle equivalent VEX can be derived. The VEX is primarily used by PEG to calculate T_{GO} and the thrust integrals. VEX is calculated by the following equation:

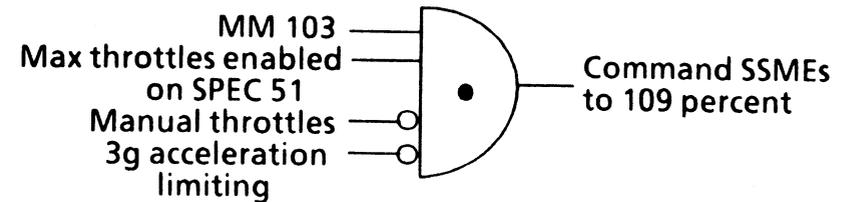
$$VEX = \frac{\text{Total thrust force (lbf)}}{\text{Total mass flow rate (slugs/sec)}}$$

Acceleration Mass Update Subtask

As the name implies, the purpose of this subtask is to provide PEG with estimates of the current vehicle mass and a smoothed value of thrust force which prevents spikes within the calculations. This subtask calls the thrust parameters subtask for estimates of thrust force, mass flow rate, and VEX in order to calculate vehicle mass and thrust to be used in the PEG algorithm.

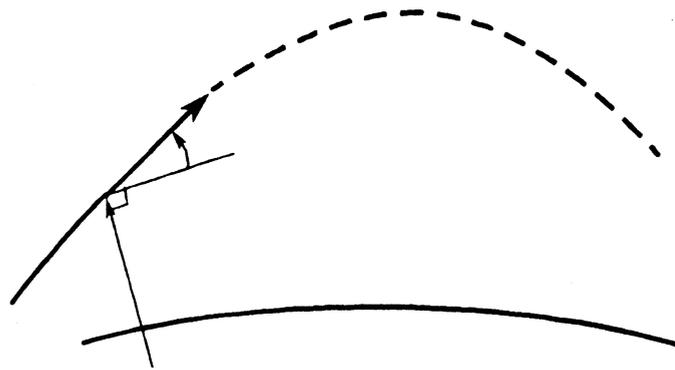
Second-Stage Maximum Throttles

Like first-stage, second-stage guidance has the capability to command maximum throttles; however, the logic is slightly different.

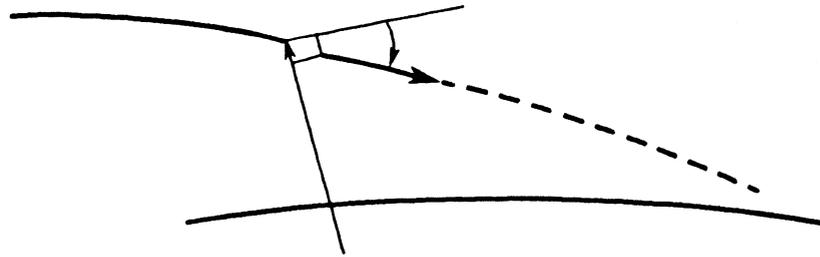


TAL Guidance

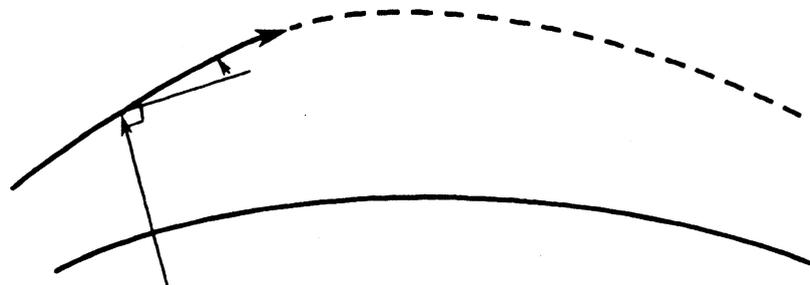
The PEG algorithm was designed to be flexible enough to handle ascent aborts; therefore, the same subtasks that are used for a nominal ascent are used for a TAL abort. The major change for TAL aborts is to redefine the MECO target from nominal. Once this is accomplished, PEG continues on as before. The RTLS/TAL target selection subtask performs this function. The parameters that are adjusted are the targeted orbit plane and flightpath angle. The targeted radius is the same as nominal, 360k feet. The TAL flightpath angle is shallower than normal. This is to allow for a shallower entry which will result in a smoother entry with less maneuvering, thus reducing thermal effects. Figure 3-19 illustrates this point. It can easily be seen that more maneuvering would be required to shallow out the trajectory if a large flightpath angle exists at MECO. A negative flightpath angle is undesirable because entry into the atmosphere would occur too soon after MECO, possibly before vehicle flight control has been gained in operational sequence (OPS) 3.



Resulting trajectory from large positive flightpath angle at MECO



Resulting trajectory from large negative flightpath angle at MECO



Nominal TAL flightpath angle resulting in shallower entry trajectory

Figure 3-19.- Effects of flightpath angle on TAL entry trajectory.

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The other MECO targets, such as the required orbit plane and inertial velocity, cannot be constrained by I-loads because they are functions of vehicle position with respect to the landing site. Obviously, a miss of the landing site is more critical than the miss in a desired orbit. The inertial velocity is calculated as a function of the surface range between the predicted cutoff and the TAL landing site (range/velocity (R/V) line). What this means is that the farther away from the landing site MECO is, the higher the cutoff velocity is in order to reach the runway. Based on a cutoff altitude of 360k feet, the range of MECO velocities is between 22.5k and 25k ft/sec (I-load). These are the minimum and maximum inertial velocities, respectively.

The ascent orbit plane determines the vehicle ground-track. For this reason, TAL landing sites are chosen to lie along the resulting groundtrack, as shown in figure 3-20. When a TAL abort is declared, the TAL orbit plane is set equal to the current orbit plane as specified by the current position and velocity vector. This will place all thrusting in plane to gain maximum performance in the downrange direction. If for some reason, the current crossrange to the TAL landing site exceeds the maximum vehicle crossrange capability, guidance will compute a new inertial aim point. This new target plane, as shown in figure 3-21, will result in a groundtrack with a crossrange that equals the maximum vehicle crossrange capability so the runway can be reached. Both the TAL velocity and orbit plane are calculated in the desired velocity subtask within PEG. Since the landing site is Earth fixed, the desired trajectory plane must be converted from Earth fixed to inertial coordinates. This is accomplished by the second-stage guidance parameter reinitialization subtask.

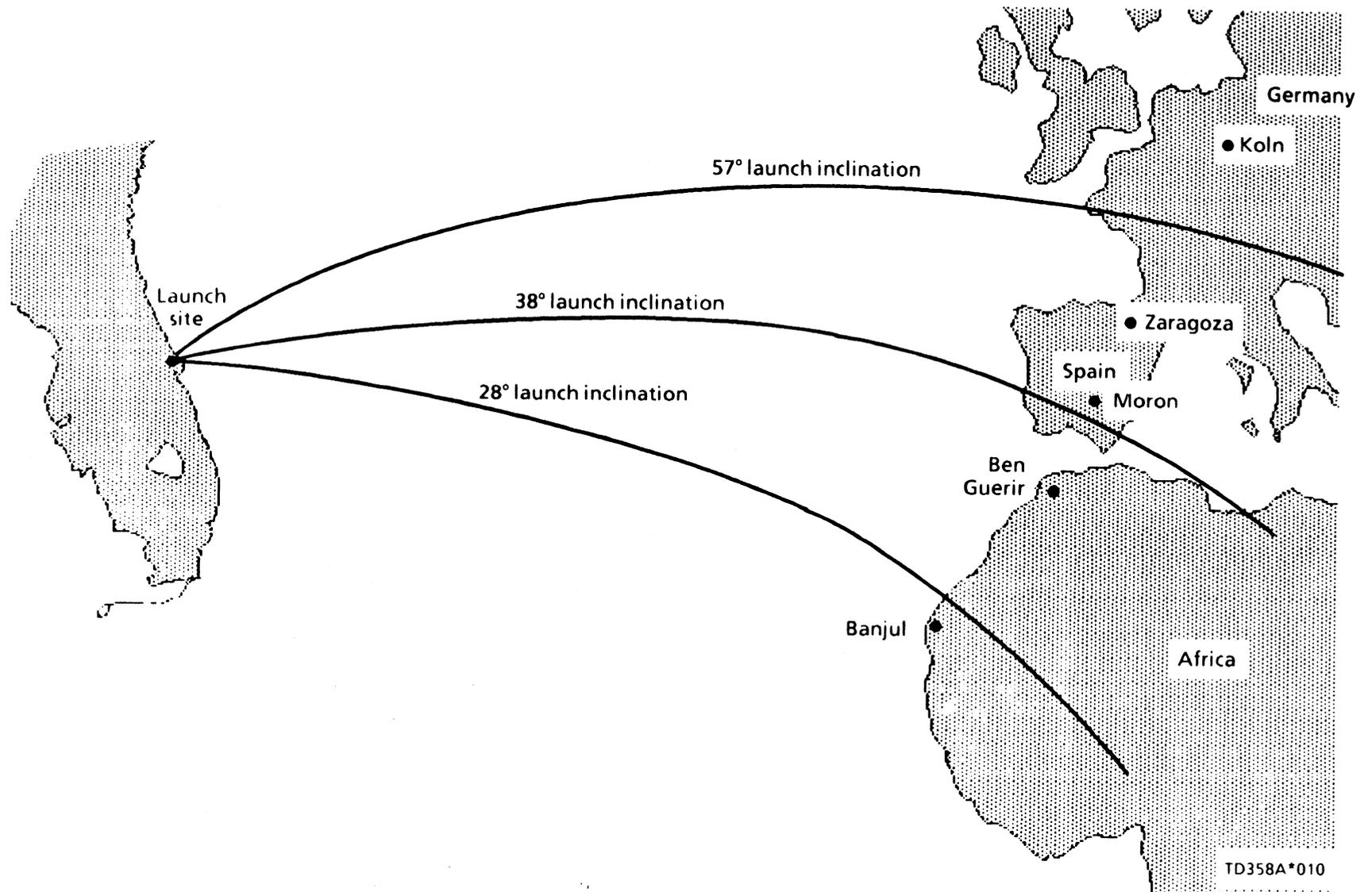


Figure 3-20.— The planned mission orbit plane determines groundtrack and optimum TAL site.

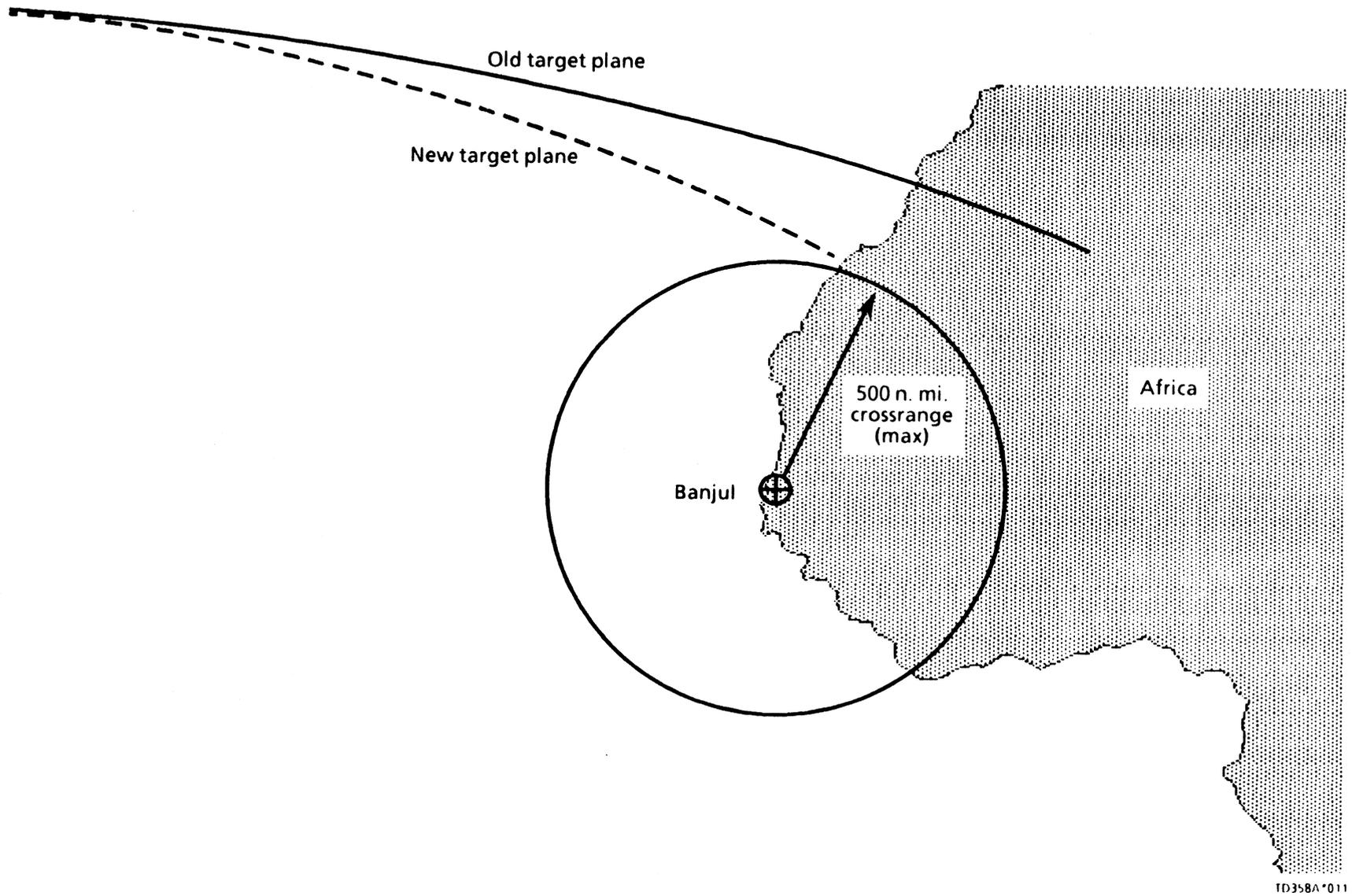


Figure 3-21.— Inertial aim point is changed to reduce crossrange to runway.

The last difference in TAL guidance is the commanded roll angle input to flight control. At a relative velocity of $\approx 14\text{k ft/sec}$, guidance will command a vehicle roll to a heads-up attitude. This is required because the TAL trajectory is lower than nominal, resulting in ET heating. The concern is setting off the range safety package because of the heating. The roll to heads up more evenly distributes the thermal load. This roll maneuver is inhibited when the inertial velocity is greater than $\approx 20\text{k ft/sec}$. The reason for this is to prevent MECO from occurring in the middle of a roll to heads up. If this were to occur, the ensuing ET separation sequence would have high vehicle rates to control.

ATO Guidance

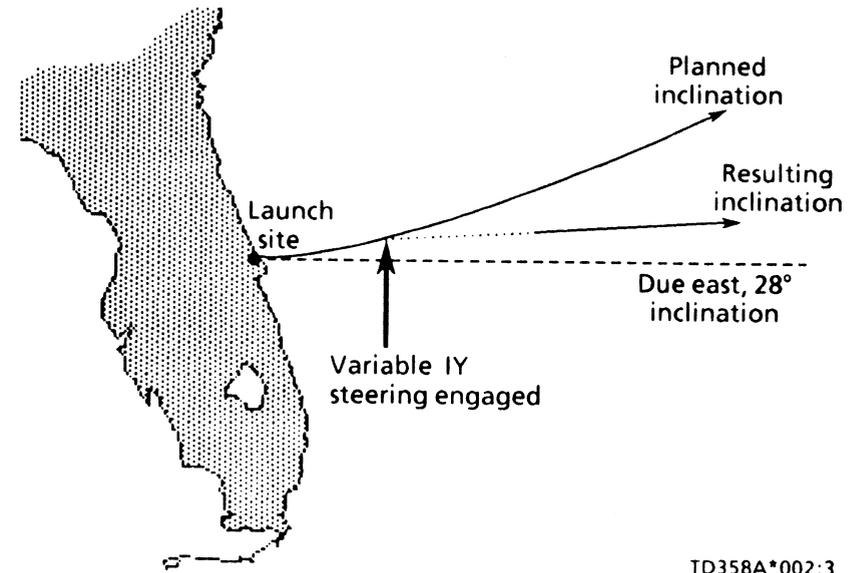
ATO selection will not always affect ascent second-stage guidance. Initiation of an ATO will be required only for flights in which OMS propellant must be dumped pre-MECO to decrease overall vehicle weight, which in turn increases performance. This is required in order to prevent disposing of the ET in populated areas. Otherwise, an ATO will not be selected pre-MECO. If an ATO is selected, guidance will perform the following functions as well as the functions performed during a nominal ascent:

- Select ATO MECO targets
- Initiate variable IY steering
- Initiate abort dump sequence

The ATO MECO target is an I-loaded target of altitude (radius), inertial velocity, and flightpath angle. For most flights, the ATO target is identical to nominal.

If an engine fails early enough in the ATO window, a low-level cutoff will most likely occur before the MECO velocity is reached due to the decrease in performance. After an ATO is selected, the ATO MECO target is selected within the software as a function of the relative velocity at main engine failure. There is a maximum I-loaded V_{rel} at which guidance will use the ATO MECO target. If the velocity at which the engine failure occurs is greater than or equal to this I-load, the abort MECO target is not selected. This engine-fail velocity is stored by the SSME-out safing subtask.

To achieve ATO/AOA abort capability as early as possible, a variable IY option is available which changes the targeted orbit plane at nominal MECO to the current orbit plane at the time of the engine failure (figure 3-22). This places all vehicle thrust in plane to optimize fuel usage and performance. This function is required only for orbit inclinations greater than 28° , where out-of-plane steering is done to achieve the required higher inclination. For more information on this subject, consult the ATO/AOA/TAL Training Manual.

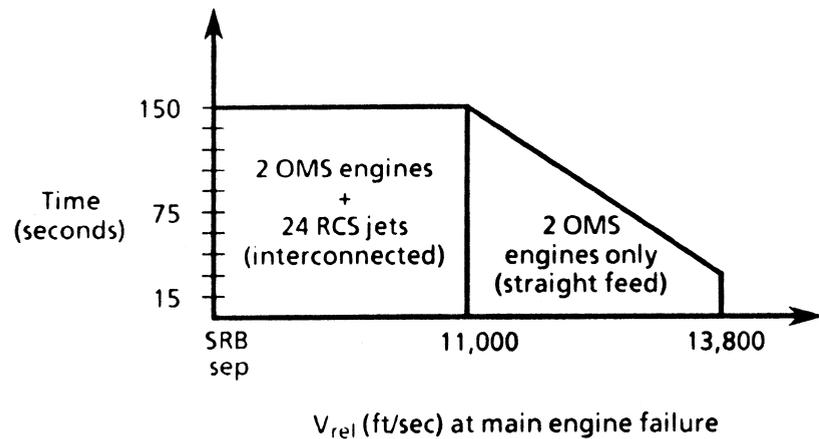


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Figure 3-22.– Result of variable IY steering.

The last difference in ATO guidance is to determine the pre-MECO OMS/RCS dump configuration. This is accomplished by the ATO/AOA targeting software within the guidance software. The actual dump is controlled by the abort control sequence, as is the case for all dumps. The ATO dump configuration is flight specific and changes from flight to flight. This is a timed dump; however, it may be fixed in duration, variable, or it may be a combination of the two, depending on the relative velocity at main engine failure. The dump may be interconnected or not, depending on what is loaded into the software. Figure 3-23 shows an example of a typical ATO dump.

It can be seen that an engine failure at or before V_{rel} 11,000 ft/sec will result in a fixed-duration interconnected dump. After this velocity, it is a variable-duration OMS-only dump. The two considerations for an ATO dump are ET footprint due to high inclination trajectory and increased performance by decreasing vehicle weight. The main reason for a variable ATO dump is to obtain a constant MECO underspeed velocity so the ΔV required for OMS-1 remains the same. A fixed ATO dump is utilized to dump large quantities of OMS propellant because a larger underspeed at MECO can be supported. It will also result in a variable underspeed velocity at MECO.



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AOA Guidance

Since there is no pre-MECO AOA abort selection, there is no guidance software devoted to AOA. ATO is selected, with a subsequent post-MECO downmode capability to AOA.

Figure 3-23.- Typical ATO dump configuration.

PRTLIS Guidance

Entry into OPS 6 automatically schedules PRTLIS guidance or PEG 5 guidance, as it is sometimes called. It is initiated at abort selection during second-stage flight and extends to the completion of the -Z translation post-MECO. This timeframe is referred to as MM 601. Should an RTLS abort be selected in first stage, MM 601 will not begin until the SRB separation command is sent. MM 601 can also be terminated by a crew-initiated transition to MM 602, which would terminate PRTLIS guidance.

PRTLIS guidance operates to achieve three flight phases.

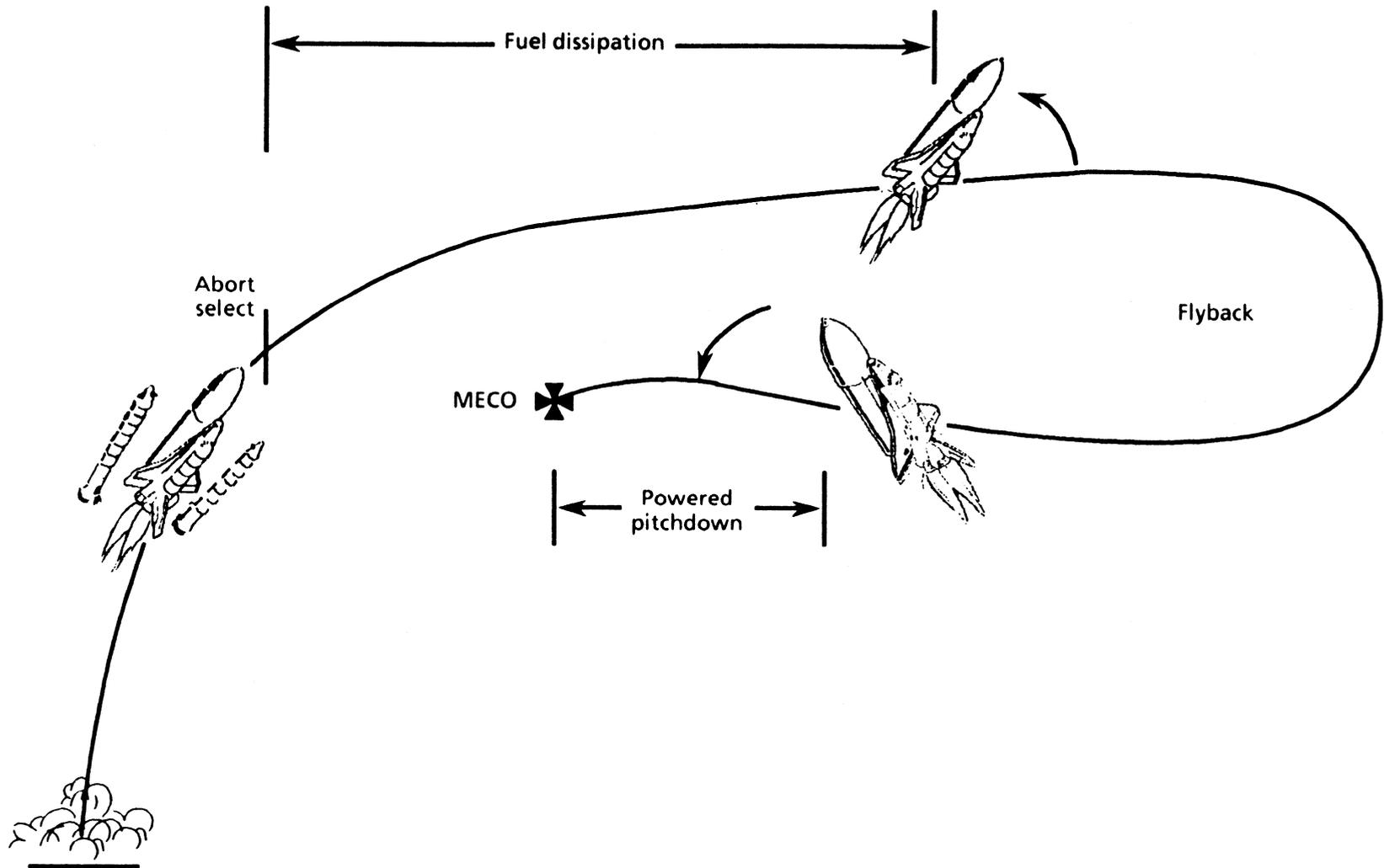
- Fuel dissipation
- Flyback
- Powered pitchdown (PPD)

Figure 3-24 illustrates these three phases.

The fuel dissipation phase continues the downrange flight at a fixed inertial thrust direction while cyclical computations are performed to determine the proper time to execute powered pitcharound (PPA). This phase is not executed in aborts near the negative return abort boundary, since PPA is begun immediately.

The flyback phase extends from PPA to targets designated at MECO minus 20 seconds. During this phase, throttling is performed for fuel depletion control.

The PPD phase begins when the vehicle is commanded to pitch down to the angle of attack (α) required for ET separation and ends at MECO.



TD358A*012

Figure 3-24. – Powered RTLS flight phases.

With these extra phases incorporated, PRTLs is more complex than a standard ascent or any of the other aborts. For this reason, additional subtasks are required to successfully accomplish the RTLs. Figure 3-25 is a functional block diagram of the PRTLs guidance software.

Note that this software uses the PEG algorithm and the same subtasks used for a nominal ascent as well as the additional subtasks for RTLs. Since PEG has already been discussed, only its differences with respect to the PRTLs guidance will be discussed. Then the additional subtasks external to PEG will be discussed.

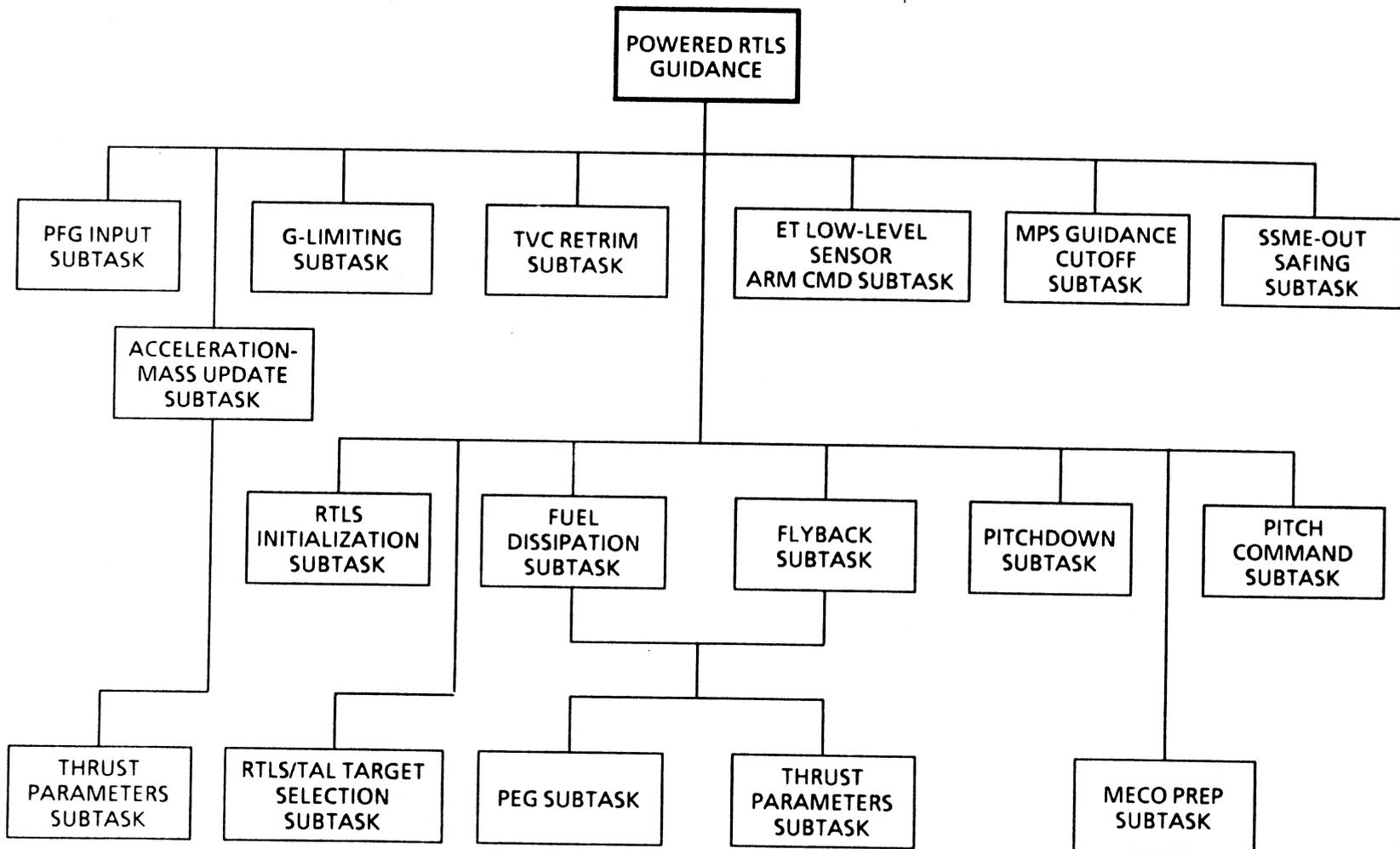


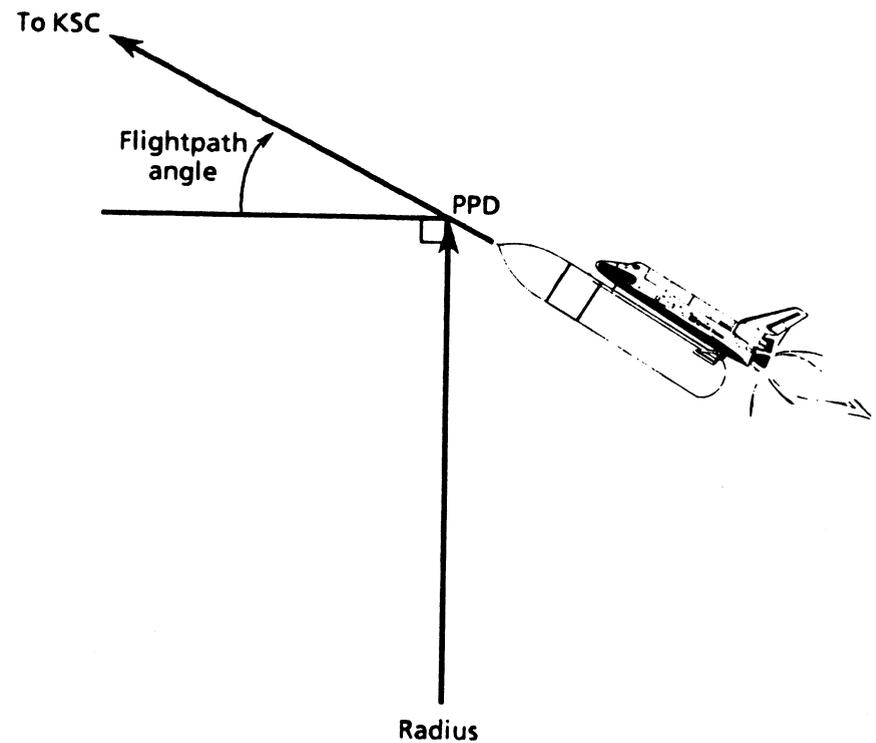
Figure 3-25.- Powered RTLs guidance software.

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After abort select, the PEG calculations for PRTLs are called by the fuel dissipation and the flyback subtasks external to PEG. These calculations are basically the same as a nominal ascent, with a few minor exceptions. These exceptions deal mostly with the PEG targets for MECO. The target for which PEG actually flies is PPD, at MECO minus 20 seconds. The PPD target is defined by the following parameters:

- I-loaded radius from Earth center (feet)
- Flightpath angle that is a function of the PPD angle and time to arrive at zero flightpath angle at MECO.
- A burnout heading such that the landing site is contained in the burnout plane.
- Desired MPS mass is approximately 2 percent at MECO.
- Desired V_{rel} that is a function of landing site range. (This is in reality MECO and not a PPD target parameter.)

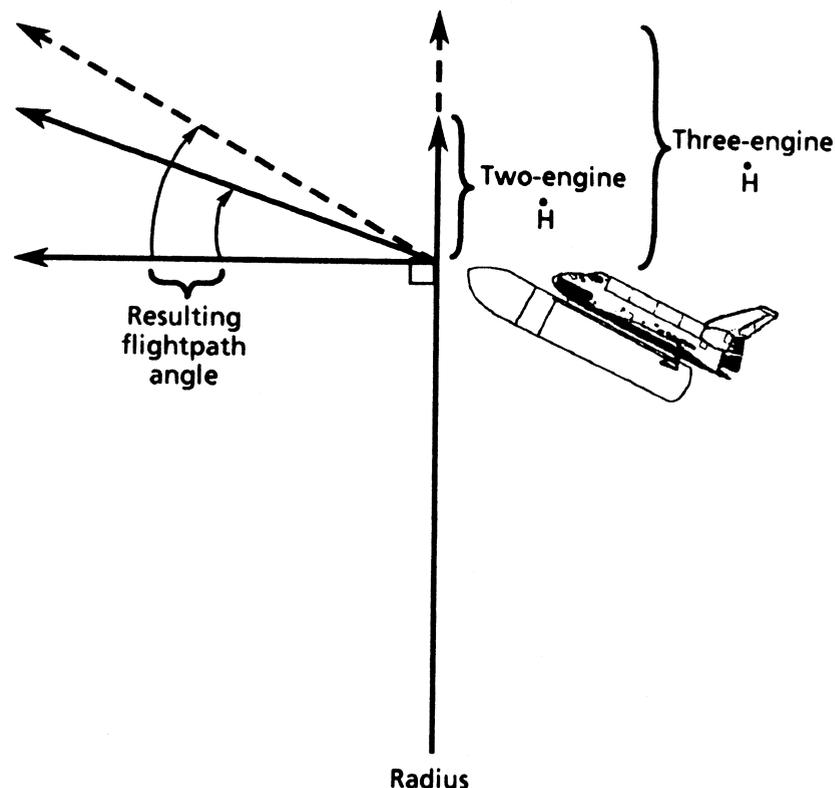
The PPD parameters are shown in figure 3-26. These parameters are obtained by additional subtasks within PEG that ordinarily are not called until an RTLS abort is declared.



TD358*005

Figure 3-26. – Powered pitchdown target parameters.

The desired radius is determined preflight via an I-load. Just as in the case of a nominal ascent, the radius will determine the altitude of MECO. The desired vehicle mass and flightpath angle are selected as functions of the active SSME configuration (two or three main engines operating). First, the desired mass is obtained by planning MECO to occur with 2-percent MPS propellant remaining within the ET. Safe ET separation is not guaranteed with more propellant remaining. One can see why the number of active main engines affects this; more engines burn more fuel in a given period of time. Second, the flightpath angle will be larger with three SSMEs operating because the additional thrust causes the \dot{H} to be larger. The larger the \dot{H} , the larger the flightpath angle at PPD, which causes the total angle for PPD to be larger, as shown in figure 3-27. This larger angle causes the time to arrive at zero flightpath angle at MECO to be greater.

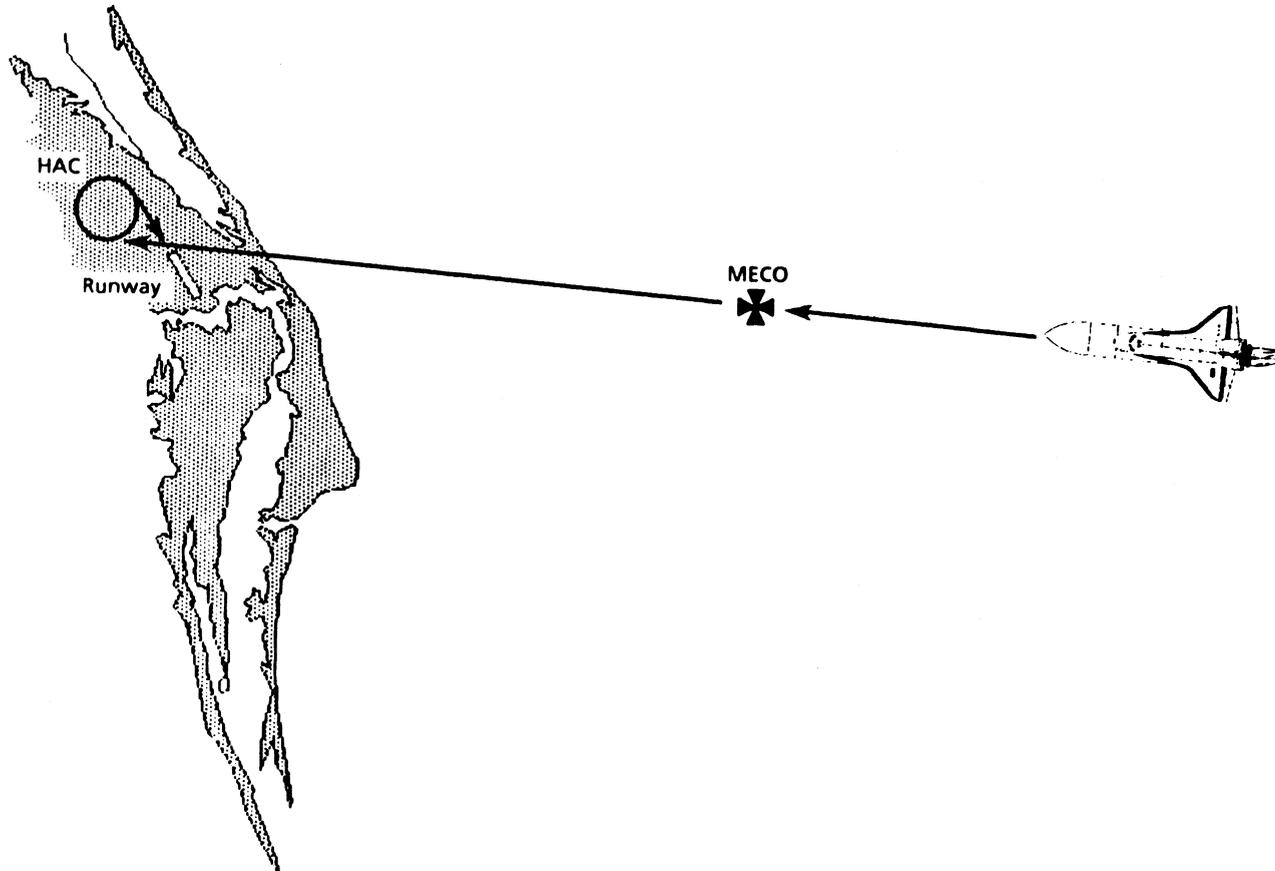


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Figure 3-27.— Effect of altitude rate on flightpath angle at PPD.

During the flyback phase, PEG calculates a vector that is used to specify the heading of the velocity vector at cutoff. This is necessary so that, when MECO occurs, the vehicle will be headed back to the landing site. In figure 3-28, it can be seen that PEG actually calculates a vector that points the vehicle to a point tangent to the heading alignment circle (HAC). This vector is analogous to an orbit plane.

The last difference in the PEG target calculations is the MECO velocity. The magnitude of this velocity vector is calculated as a function of the surface range between predicted cutoff and the landing site, just like the TAL MECO velocity. The greater the range, the higher the MECO velocity, and the converse is true as well. This is known as the range/velocity (R/V) line.



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Figure 3-28.- RTLS MECO vector.

Subtasks External to PEG for RTLS

To facilitate understanding, the software subtasks external to PEG will be discussed in chronological order rather than a data processing order. Only the additional subtasks unique to RTLS will be discussed.

RTLS Initialization Subtask. The RTLS initialization subtask is performed only once at the selection of the RTLS abort. It performs exactly what the name implies; it initializes guidance from OPS 1 to OPS 6 for the RTLS abort. In addition, it also performs the following:

- Inhibits the PEG steering commands
- Determines active main engine configuration
- Selects MECO target radius for RTLS
- Stores V_{rel} and H

The guidance steering commands are inhibited so the pitch command subtask can perform fuel dissipation. The steering commands from PEG will not be reenabled until the flyback phase is entered.

Since the RTLS SSME power level is designed for two active main engines (as opposed to three), this subtask must determine if a two-engine or three-engine RTLS is in progress. This is necessary to send the appropriate throttle command to the SSMEs. These commands are I-loads that are normally set at 104 percent for two engines and 69 percent for three engines. The lower throttle setting for three active SSMEs is equivalent to a two-SSME power level, thus satisfying the RTLS power level design. This is evident in the following example:

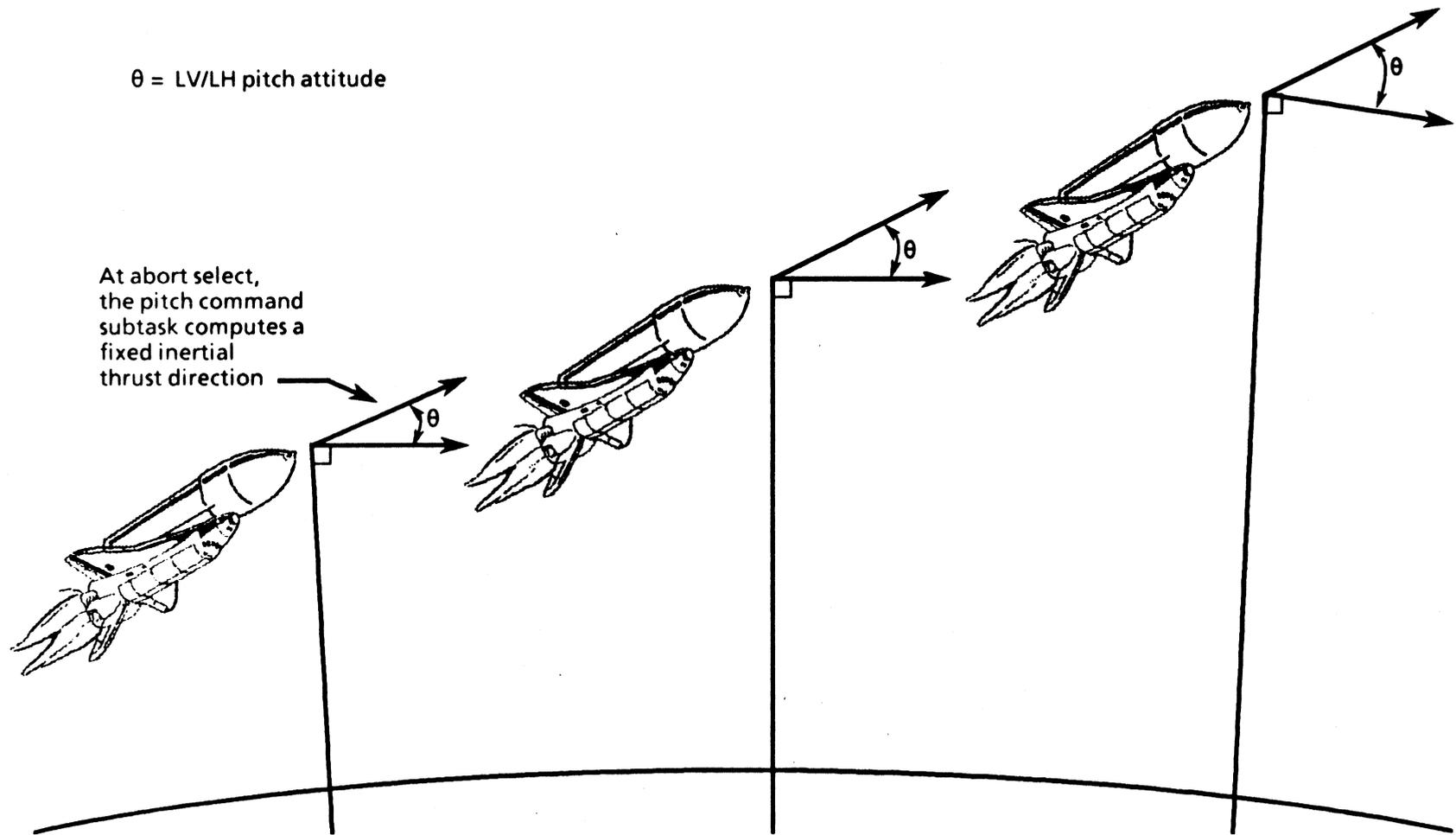
Three SSMEs at 69 percent = 207 percent

Two SSMEs at 104 percent = 208 percent

Also, a lower throttle setting aids in fuel dissipation because rocket engines run less efficiently than at nominal power levels. Of course, if manual throttling is being performed, the guidance command will not be sent to the SSMEs because the speed brake/thrust controller (SBTC) will override these commands. Upon entry into this subtask, the nominal MECO target radius is replaced with the RTLS target radius. This radius is an I-load based on expected vehicle performance and is approximately 220k feet.

The last function of the RTLS initialization subtask is performed during a three-engine RTLS. At abort select, the present V_{rel} and altitude rate must be stored to be used in the pitch command subtask for fuel depletion. Recall from section 2, when an SSME fails, the SSME-out safing subtask will store the V_{rel} and altitude rate.

Pitch Command Subtask. After OPS 6 guidance has been initialized, the pitch command subtask will compute a thrust direction for the fuel dissipation phase. This thrust direction is known as the fuel dissipation thrust angle. This is a fixed inertial thrust angle that is executed in the same plane in which abort initiation occurred. This angle is calculated as a linear function of the V_{rel} and H at the time of engine failure or abort select in the case of a three-engine RTLS. The local vertical/local horizontal (LVLH) pitch attitude at abort selection is higher for early aborts as compared to late aborts. The reason for this is to loft the vehicle to gain altitude. Since this is a fixed inertial thrust angle, the LVLH pitch attitude will increase with time as shown in figure 3-29. Once the fuel dissipation thrust angle is computed, the pitch command subtask is exited until PPA.

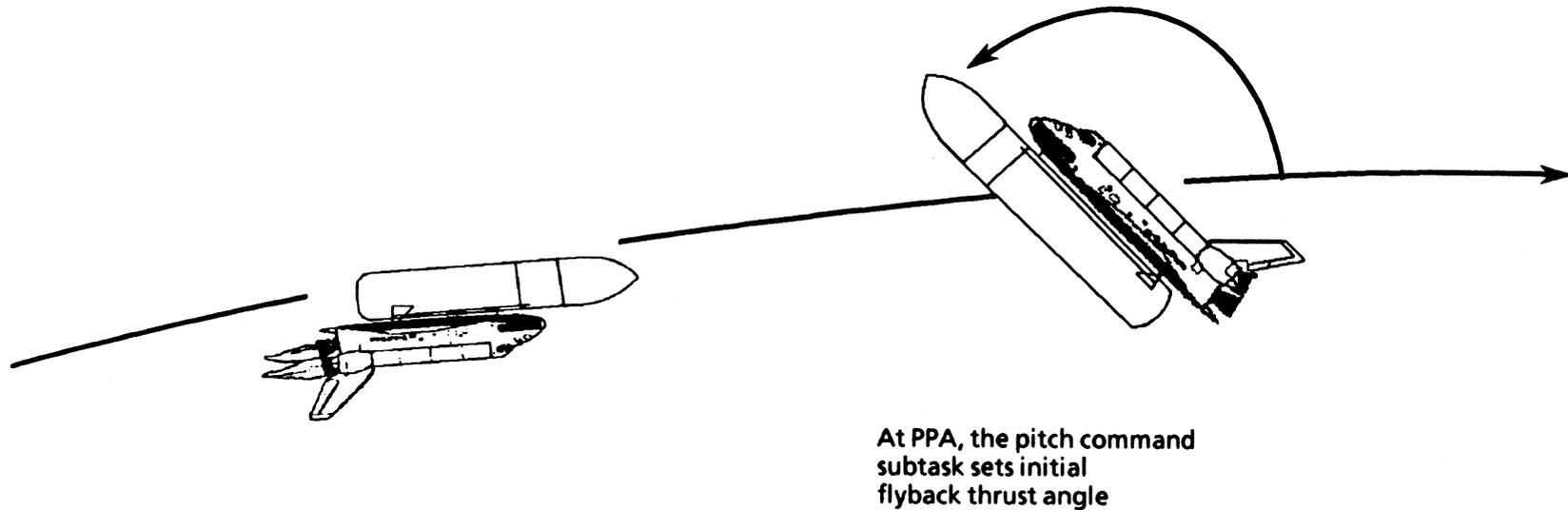


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Figure 3-29.- Fuel dissipation thrust attitude.

At PPA, the fuel dissipation subtask sets the pitch command subtask discrete back on. Then the pitch command subtask sets the initial flyback thrust angle to an I-loaded value so that PPA is begun, and the vehicle is commanded in plane to a wings-level, tail-up attitude.

This orients the mated vehicle to a heads-up attitude pointing back to the launch site. Figure 3-30 illustrates this maneuver. After the flyback thrust angle is commanded, this subtask is exited.



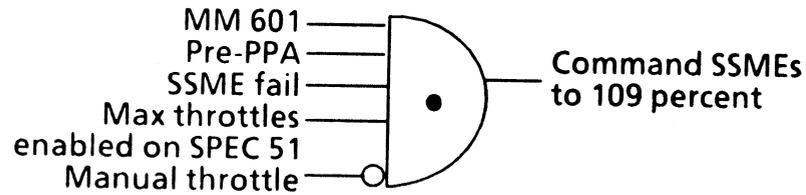
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Figure 3-30.— Powered pitcharound.

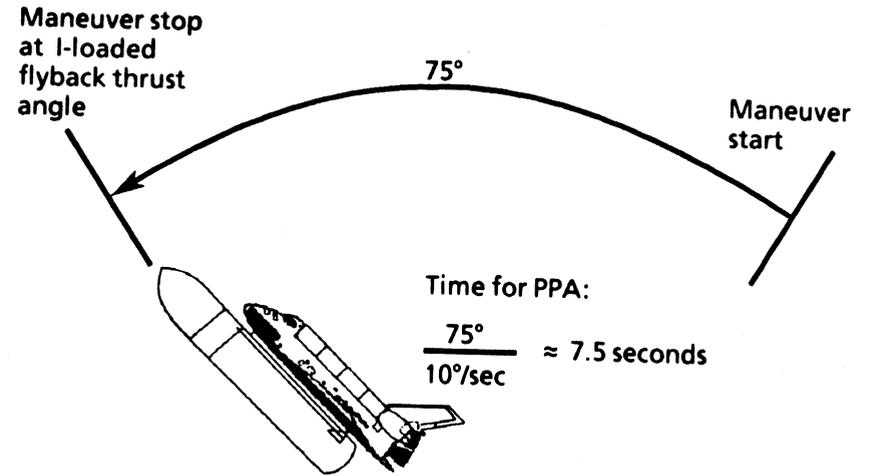
Fuel Dissipation Subtask. The fuel dissipation subtask is executed to burn off excess MPS propellant from the ET. Excessive OMS propellant is dumped during this time-frame as well. The OMS dump is performed for vehicle center of gravity (c.g.) management and to reduce the overall vehicle weight for landing. This subtask also lofts the mated vehicle to a suitable altitude for PPA. It is actually this subtask which calls the pitch command subtask at abort select to pitch the vehicle to the fuel dissipation thrust angle, which is held constant.

Remember that other subtasks are being performed in addition to these new subtasks for PRTL guidance (figure 3-25). With this in mind, the vehicle acceleration is obtained by executing the thrust parameters subtask. Using this acceleration, current vehicle position, and time it takes for PPA, the PEG subtask is called to obtain the magnitude of velocity to be gained by thrust to achieve the targets. This velocity is summed with the estimated velocity gained during PPA by thrust to predict the final mass at the PEG target. Remembering that PEG is a cyclic process that repeatedly performs these computations, when the predicted final mass is less than the desired final mass, PPA is begun. This final mass is biased to prevent a late PPA. The desired final mass ensures that just enough MPS propellant remains (≈ 2 percent at MECO) to achieve MECO. If an RTLS is selected near the negative return boundary, this subtask is not performed and PPA will occur immediately after abort select. The crew can also perform PPA manually with the use of the rotational hand controller (RHC). In any case, when PPA occurs, this subtask is exited and the RTLS flyback subtask is entered. The following secondary functions are performed by this subtask as well:

- Enable second-stage roll angle after an I-loaded V_{rel} is attained (180° , heads down, pre-PPA)
- Enable max throttle logic when requested pre-PPA



- Calculate time for PPA using ideal rates and accelerations expected for the total maneuver angle (figure 3-31).



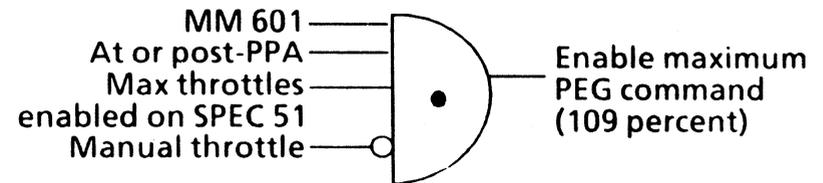
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Figure 3-31.– Time calculation for powered pitcharound.

RTLS Flyback Subtask. This subtask is entered at the start of PPA. It is at this point that the PEG steering discrete is set to on. This enables the vehicle to pitch around in plane at ≈ 10 deg/sec to orient the mated vehicle to a heads-up attitude pointing back to the launch site. Initially, PEG will steer the vehicle to an I-loaded flyback thrust angle to perform PPA as illustrated in figure 3-31. This angle is calculated by the pitch command subtask. Once PEG has converged in the turnaround, the calculated steering parameters (reference thrust vector, thrust turning rate vector, and reference time) are used for the flyback rather than the I-loaded angle. Once PEG has converged and steering is enabled, flyback guidance is closed-loop to the PPD target conditions and is explicitly targeted to altitude (radius), flightpath angle, and final mass. PEG will continue its cyclic process just as it did under a nominal ascent until the fine countdown discrete is set at PPD minus 6 seconds (I-load). If manual guidance is in progress, the fine countdown discrete is set at PPD minus 14 seconds rather than 6 seconds. Manual guidance may not track the trajectory accurately enough to ensure that T_{GO} is small enough to set fine count if I-loaded at 6 seconds.

This subtask also performs fuel depletion SSME throttling. This is necessary in order to waste fuel, if required, during flyback to minimize the amount of MPS propellant at MECO. The main engines are nominally commanded to a lower throttle setting than the normal of 104 percent. If during this phase, guidance commands 104 percent, maximum throttles should be enabled as soon as possible to enable a higher power level required to burn more efficiently towards the target.

Flyback max throttle logic:



If manual guidance and/or manual throttling is being performed, fuel depletion throttling will be terminated and never started again, even if guidance is returned to auto. In this situation, the PEG throttle is set and maintained at the current commanded engine throttle setting. As soon as the fine countdown discrete is set, this subtask is exited.

Powered Pitchdown Subtask. PPD occurs 20 seconds before MECO. This subtask takes the vehicle from a relatively high angle of attack to the required separation attitude with an angle of attack of minus 4° (I-load). This is accomplished by calculating the rotation angle required to achieve an alpha of -4° as shown in figure 3-32. Note in figure 3-33, this attitude uses the dynamic pressure to help separate from the ET.

If an SSME fails within ≈ 7 seconds after PPD initiation, the actual engine configuration may not match what guidance had calculated at PPD. Left unchecked, the subsequent dynamic pressure at ET separation would be higher than normal.

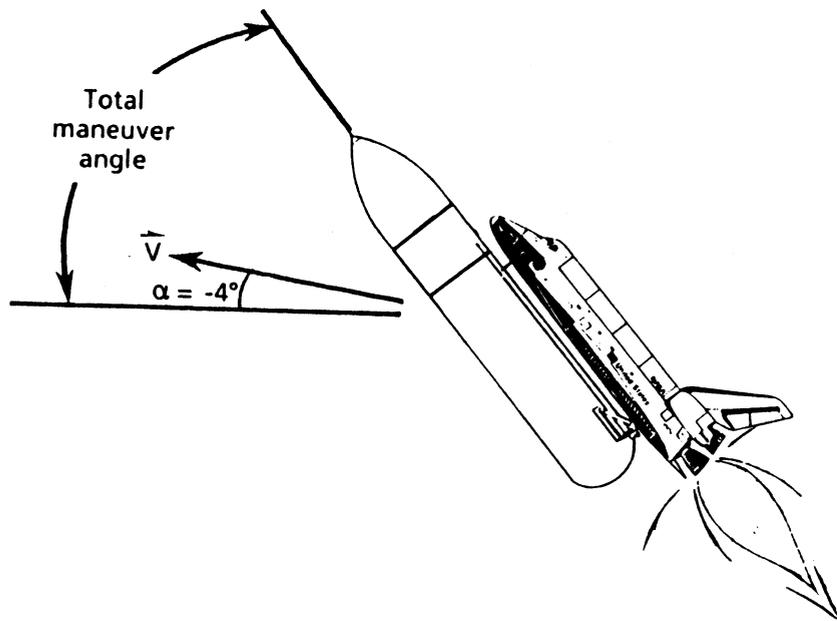


Figure 3-32.— Powered pitchdown maneuver.

To correct this situation, the PPD subtask will stop the PPD maneuver and calculate the length of time to delay. This delay is a function of the number of operating SSMEs and the time of engine failure after initiation of PPD. Once the delay time has expired, PPD will continue as before.

The main engines are commanded to the minimum power level in preparation for MECO by this subtask as well. The minimum power level for two or three active SSMEs is I-loaded to 65 percent. (Special case: For a three-engine RTLS, the minimum power level may be 91 percent if one SSME fails late in the trajectory). If only one main engine is operating, the minimum power level is 104 percent. If manual guidance and/or throttling is being performed, the throttle level is set to the value that existed at manual selection and is controlled by the SBTC.

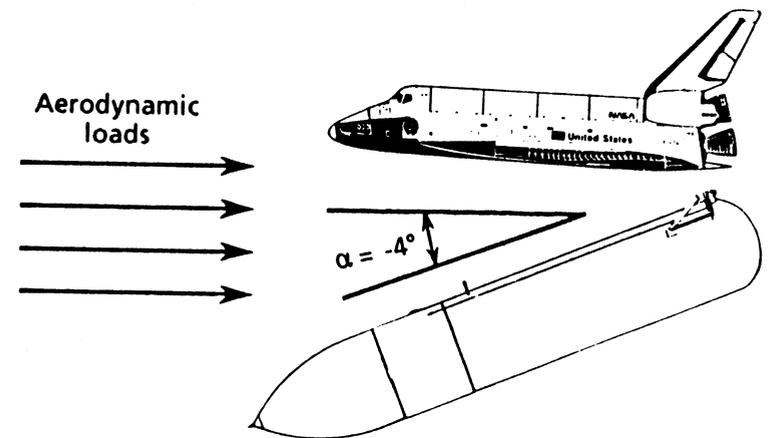


Figure 3-33.— RTLS external tank separation.

MPS Guidance Cutoff Subtask. The MPS guidance cutoff subtask has already been covered and is performing the same functions. The only difference is the calculation of parameters required to determine the point at which MECO should occur. These parameters are

- The position of the landing site
- The distance from the landing site
- The flightpath angle
- V_{GO} and T_{GO} to MECO

All these parameters affect the RTLS range/velocity MECO target, which is Earth-relative. This subtask is performed from the beginning of PPD until the MECO discrete is set. In the case of manual throttling, this subtask will be exited at main engine zero thrust.

RTLS Contingency Abort Subtask. This subtask determines if RTLS guidance should be scheduled in MM 602 and MM 603. Automatic glide RTLS (GRTLS) guidance is desired to provide the capability to complete an intact RTLS for a second SSME failure close to the RTLS MECO conditions (approximately 45 seconds before MECO). This subtask determines if the vehicle is heading back toward the launch site and checks to see that the current navigated velocity is greater than an I-loaded velocity ($V_{rel} \approx 2500$ ft/sec). If these conditions are met, the RTLS intact contingency flag is set true to schedule guidance. When this occurs, alpha recovery, load relief, and terminal area energy management (TAEM) guidance will be scheduled in MM 602 and MM 603. The RTLS contingency abort subtask is performed once when a second SSME fails and once when the MECO confirmed flag is set.

Contingency Abort Guidance

Beginning with software operational increment (OI) 21, guidance subtasks were added to perform automatic maneuvers in contingency situations where only one SSME remains in operation. These subtasks are covered in greater detail in the Contingency Abort 2102 Training Manual.

Questions

1. What type of guidance scheme is used for second-stage flight?
 - A. Open loop
 - B. Closed loop
 - C. Variable open/closed loop design
 - D. I-loaded MECO target
2. What are the parameters that define the MECO target?
 - A. Relative velocity, altitude, central angle, and orbit plane
 - B. Inertial velocity, radius, orbit plane, and flight-path angle
 - C. Central angle, time, horizontal velocity, and position
 - D. Turning rate vector, reference time, reference thrust vector, and V_{GO}
3. How is powered explicit guidance best described?
 - A. One-time steering parameter calculation utilizing prediction and correction process to steer the vehicle
 - B. Prediction process to maneuver midpoint then correction process to MECO target
 - C. Iterative correction process of one-time steering parameter prediction
 - D. Iterative steering parameter calculation utilizing prediction and correction process to steer the vehicle
4. What are the three steering parameters PEG calculates?
 - A. Thrust attitude angle, current thrust vector, and MECO thrust vector
 - B. R_{GO} , T_{GO} , and V_{GO}
 - C. Reference time, thrust turning rate vector, and reference thrust vector
 - D. All of the above
 - E. None of the above
5. What is linear tangent guidance?
 - A. A method whereby the tangent of the pitch attitude angle is varied for abort shaping
 - B. An open loop guidance scheme utilizing T-fail
 - C. A method using pseudo MECO targets to achieve a fuel optimal trajectory
 - D. A method by which a fuel optimum trajectory is achieved by varying the tangent of the thrust attitude as a function of maneuver time
6. Why are the subtasks within the PEG task necessary?
 - A. They compute dependent information required for the steering parameter calculations.
 - B. They compute independent information required for guidance to determine its current position.
 - C. They compute dependent information required to determine the current velocity vector.
 - D. They compute V_{GO} information to be used by flight control software.

7. What are external subtasks?
- They are subroutines which operate external to guidance software.
 - They are subroutines external to PEG which are used to support the PEG calculations.
 - They are subroutines which operate external to onboard vehicle software.
 - They are subroutines within PEG which are used to support external calculations.
8. Is the PEG algorithm used for abort guidance?
- Yes, it is used during ATO, TAL, and RTLS aborts.
 - No, the PEG algorithm was not designed for this purpose.
 - Yes, it is used for AOA aborts.
 - No, contingency aborts require manual guidance.
9. Compared to the nominal MECO target, which statement(s) is (are) true with regard to the TAL MECO target?
- Higher velocity
 - Higher radius
 - Shallower flightpath angle
 - Variable orbit plane that is a function of powered pitchdown time
 - All of the above
10. What is the purpose of ATO guidance?
- To gain pre-MECO AOA capability
 - To increase performance by wasting MPS propellant
 - To maximize vehicle performance by decreasing weight while thrusting in plane
 - None of the above
11. What is variable IY steering?
- Steering required to achieve an inclination of 28°
 - Steering performed to redirect thrust in-plane when an SSME fails during an ATO abort
 - Steering required to achieve an inclination greater than 28°
 - Steering used during TAL aborts to thrust out of plane toward the landing site
12. The three flight phases of powered RTLS guidance are
- T-fail, constant thrust, and constant acceleration
 - T-fail, G-limiting, and MPS guidance cutoff
 - Initialization, IY steering, and flyback
 - Fuel dissipation, flyback, and PPD
13. During RTLS, how does PEG determine when the flyback phase should be initiated?
- As a function of vehicle mass
 - As a function of mission elapsed time
 - At an I-loaded inertial velocity
 - At an I-loaded range from the launch site
14. During RTLS, what is the target that PEG aims for?
- Range/altitude ratio
 - Powered pitcharound
 - MECO
 - Powered pitchdown

15. What are maximum throttles?

- A. Any throttle setting higher than 104 percent up to a maximum of 109 percent
- B. 109 percent
- C. 104 percent
- D. Any throttle setting higher than 100 percent up to a maximum of 109 percent

16. What is abort shaping?

- A. Depressing the abort trajectory to prevent an AOA
- B. Lofting the abort trajectory to prevent an abort gap
- C. Lofting the nominal trajectory to prevent an abort gap
- D. Depressing the normal trajectory to prevent an AOA

17. What is T-fail?

- A. An abort shaping technique whereby PEG assumes an engine will fail at an I-loaded velocity
- B. An abort shaping technique whereby PEG assumes an engine will fail at an I-loaded mission elapsed time
- C. The latest mission elapsed time an RTLS abort can be declared
- D. An abort shaping technique whereby pseudo MECO targets are replaced with nominal MECO targets at an I-loaded mission elapsed time

18. What are pseudo MECO targets?

- A. An abort shaping technique whereby PEG assumes an engine will fail at an I-loaded mission elapsed time
- B. An abort shaping technique whereby "fake" MECO targets are replaced with actual MECO targets at an I-loaded velocity
- C. Alternate targets used during TAL aborts when the landing site crossrange exceeds vehicle capability
- D. An abort shaping technique whereby "fake" MECO targets are utilized until an I-loaded mission elapsed time

19. Which of the following subtasks calculates cutoff time based on desired cutoff velocity magnitude?

- A. MPS guidance cutoff subtask
- B. TGO subtask
- C. T-fail subtask
- D. MECO preparation subtask

20. Which of the following is NOT a guidance subtask?

- A. TVC retrim subtask
- B. TGO subtask
- C. T-fail subtask
- D. Desired position subtask

Answers

1. B. Guidance corrects its steering parameters each cycle to account for current conditions (page 3-1).
2. B. These parameters define the target for powered explicit guidance to steer to (page 3-3).
3. D. PEG cyclically computes steering parameters, predicts vehicle state at MECO, and corrects its prediction based on current conditions as a result of steering (page 3-6).
4. C. These parameters provide the foundation necessary to achieve the desired position and velocity at MECO (page 3-6).
5. D. Linear tangent guidance is the method employed to achieve a fuel optimum trajectory (page 3-6).
6. A. Each subtask constructs information necessary before PEG can calculate its steering parameters (page 3-8).
7. B. External subtasks are guidance software subroutines that are not directly responsible for calculating steering parameters; therefore, they operate external to PEG to perform guidance functions (page 3-22).
8. A. The PEG algorithm was designed to handle all phases of exoatmospheric flight (page 3 25).
9. C. The TAL MECO target has a shallower flightpath angle to prevent overstressing the vehicle on the ensuing entry. The velocity magnitude and orbit plane are dependent on landing site location with respect to current position. The TAL radius is identical to the normal MECO target (page 3-25).
10. C. ATO guidance maximizes vehicle performance after a main engine has failed (page 3-30).
11. B. Variable IY steering is necessitated on high inclination flights to maximize remaining performance by placing all thrust back in plane after an SSME has failed (page 3-31).
12. D. The three flight phases of powered RTLS guidance are fuel dissipation, flyback, and PPD (page 3-33).
13. A. This determination is made in the fuel dissipation subtask when the current vehicle mass is less than that which is required to ensure approximately 2-percent MPS propellant remains at MECO. When this occurs, powered pitcharound is begun and the flyback phase is entered (page 3-42).
14. D. During RTLS, the target that PEG aims for is powered pitchdown (page 3-36).

15. A. 109 percent is not always true. The max throttle logic depends on the situation. It is possible for PEG to command a power level higher than 104 percent but less than 109 percent (pages 3-44, 2-4, 3-25, and 3-43).
16. C. The nominal trajectory is lofted, which gains potential energy to prevent an abort gap should an SSME fail (page 3-9).
17. B. T-fail is an abort shaping technique whereby PEG assumes an engine will fail at an I-loaded mission elapsed time (page 3-9).
18. D. Pseudo MECO targets utilize an artificial MECO target, whereas T-fail assumes a main engine failure. Both are used to loft the nominal trajectory and are mutually exclusive (page 3-11).
19. A. The MPS guidance cutoff subtask calculates the cutoff time. The MECO preparation subtask merely sets a discrete based on T_{GO} (page 3-24).
20. C. T-fail is not a guidance subtask. It is a function within the T_{GO} subtask (page 3-9).

Section 4: Flight Control Hardware

Objectives

After reading this section, the student should be able to do the following:

- List the hardware that provides inputs to the DAP
- Explain the ways in which RGAs are used by the DAP

After guidance computes the required vehicle commands, these commands are sent to flight control. Ascent flight control is handled by both hardware and software (residing within the GPCs). The software is known as the digital autopilot (DAP). Before we get into the software, we will first discuss the hardware used for inputs to the DAP and the hardware the DAP commands to effect a vehicle maneuver for flight control. The following hardware sensors provide inputs to the DAP:

- IMUs
- Orbiter body mounted accelerometer assemblies (AAs)
- Orbiter rate gyro assemblies (RGAs)
- SRB RGAs

The following effectors utilize the DAP outputs for vehicle flight control:

- Main engines
- SRBs
- Aerosurfaces

Figure 4-1 shows the physical location of this hardware within the vehicle. A brief description of this hardware and how it interfaces with flight control follows. For more information on the hardware, each has an associated training manual/workbook.

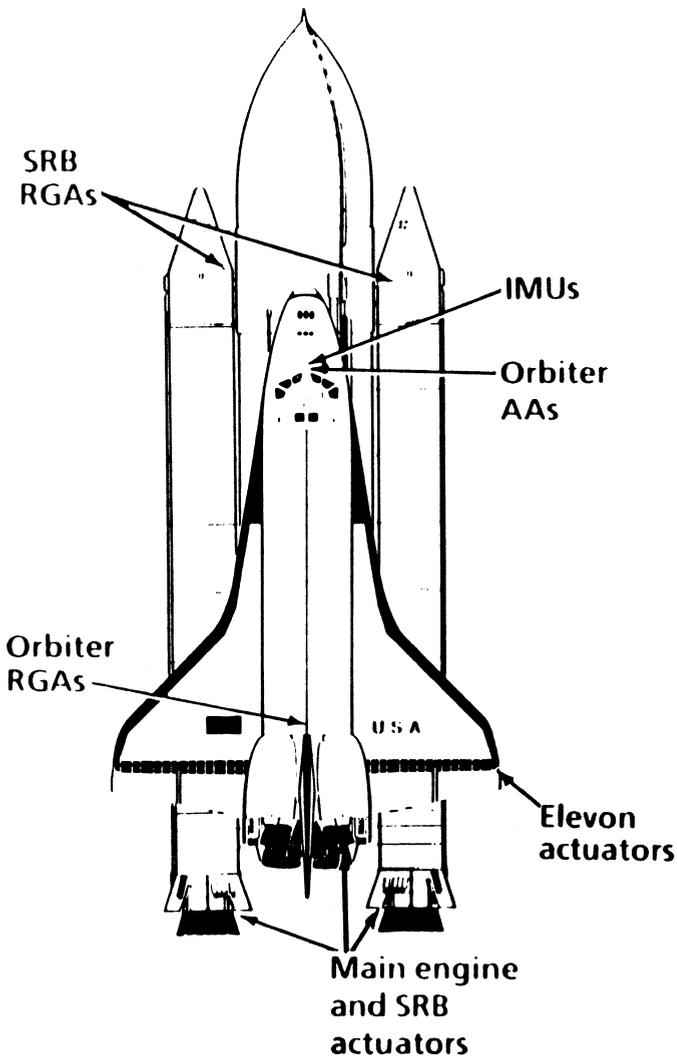


Figure 4-1. - Flight control hardware location.

IMU

Since the vehicle is inserted into an inertial orbit during ascent, IMUs are required to keep track of vehicle attitude, velocity, and position changes. Vehicle attitude information is provided via the gimbal angles within the IMU. Inertial acceleration is measured by the accelerometers within the IMU. An integrator multiplies this value by time to derive velocity in feet per second. By knowing the starting point (launch) and the velocity, the current vehicle position is determined by taking into account the elapsed time.

Orbiter AA

The AAs are mounted directly onto the orbiter body structure. They are oriented to measure normal acceleration (N_z), which is acceleration along the Z axis, and lateral acceleration (N_y), which is acceleration along the Y axis. The AAs are used only during first stage load relief. They are used for feedback to the DAP on orbiter body axis accelerations. As we discuss the DAP in further detail, its use will become more apparent.

Orbiter and SRB RGAs

The RGAs are used by the DAP for feedback on the pitch, roll, and yaw rates the vehicle is experiencing. This is necessary because the shuttle flies via a rate command system; therefore, rate feedback is required. The RGAs on the SRBs serve the same purpose; however, there is no roll gyro.

The Main Engines and SRBs

The main engines and SRB nozzles are gimbaled, providing thrust vector flight control to steer the vehicle.

Orbiter Aerosurfaces

Only the elevons are active during ascent. They are driven to relieve wing loading during the max \dot{q} regime. The rudder, speedbrake, and body flap are not driven and are placed in a position to reduce drag for ascent. Remember, the aerosurfaces are not used for flight control during ascent; only the main engines and SRBs are used for this purpose.

Section 5: Flight Control Software

Objectives

After reading this section, the student should be able to do the following:

- Explain the proper signal flow sequence through the flight control system
- Define the purpose of G/C steer
- Describe why rate gyro assemblies are necessary for flight control feedback

As stated earlier, the ascent DAP is the heart of the FCS. The ascent DAP is activated at SRB ignition and deactivated at SSME zero thrust, where the trans DAP takes over. The DAP converts steering commands from guidance and vehicle attitude, attitude rates, and acceleration inputs into control element (SSME and SRB TVC) deflections.

We will discuss the process which the DAP uses to convert this information to TVC deflections. The DAP can be operated in two modes, automatic and manual. The automatic system will be discussed first to explain the entire system. Then we will take a look at the differences associated with the manual mode, CSS.

The FCS can be broken out into three main software modules. These modules are as follows:

- The attitude processor
- G/C steer interface
- The DAP

Figure 5-1 illustrates the relationship between these modules.

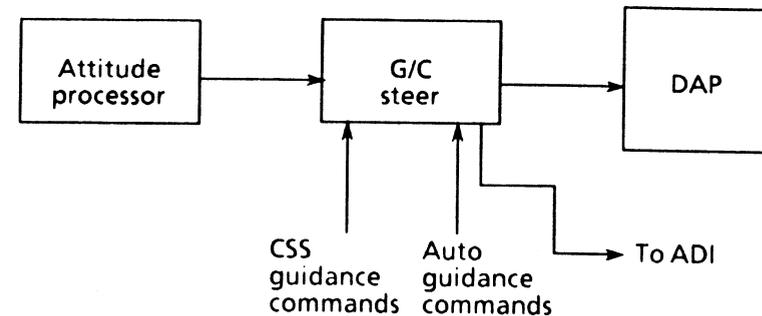


Figure 5-1.– Software interface.

The Attitude Processor

The function of the attitude processor is to provide current vehicle attitude-related information to G/C steer and the ADI. This attitude information is obtained from the gimbal angles within a selected IMU. These angles can be thought of as the actual vehicle attitude in the Aries Mean of 1950 (inertial) coordinate frame. Since flight control steers the vehicle in response to body axis errors, the inertial attitude is converted to the body frame. This is accomplished via a quaternion, which transforms information from one coordinate system to another.

IMU data is processed at a relatively slow rate; therefore, this module propagates the IMU angles with RGA data, which is supplied at a higher rate to prevent a loss of control (LOC). Flight control requires vehicle attitude information at this higher rate. Therefore, once every other cycle, the IMU angles are propagated with RGA data to derive the vehicle attitude (figure 5-2).

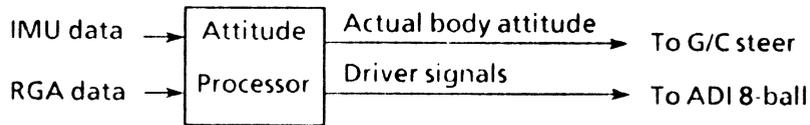


Figure 5-2.– Attitude processor.

Secondary functions within this module include a method by which redundancy management (RM) can select a good IMU. The attitude processor also provides body attitude information in the form of ADI driver signals which drive the 8-ball on the ADI. To recap, the attitude processor performs five functions.

- Provides attitude information
- Propagates IMU data
- Derives body axis attitude information
- Selects a good IMU
- Sends ADI driver signals

Current vehicle attitude information is required so that this attitude can be compared (in G/C steer) with the commanded attitude from guidance. This comparison would yield an attitude error for the DAP to fly out. An error is a delta between actual and required vehicle attitude. This delta is necessary for the DAP to "know" in what direction and how far the attitude must be corrected to get back on course.

G/C Interface

G/C steer utilizes the steering commands to develop an attitude error and attitude rate command that the DAP must steer out to get back on course. It also provides a smoothing function for interfacing between the low processing rate of guidance and the high processing rate of flight control. G/C steer performs four specific functions:

- Automatic steering - Provides the interface between guidance and flight control.
- Manual steering - Provides the interface between the pilot and flight control.
- Attitude errors - Generates the attitude error between the desired attitude and the actual vehicle attitude for the DAP to fly out. It also sends the attitude error signal to the ADI processor, which drives the attitude error needles on the ADI.
- Sends desired vehicle body axis rates to the DAP.

Figure 5-3 is a functional representation of G/C steer.

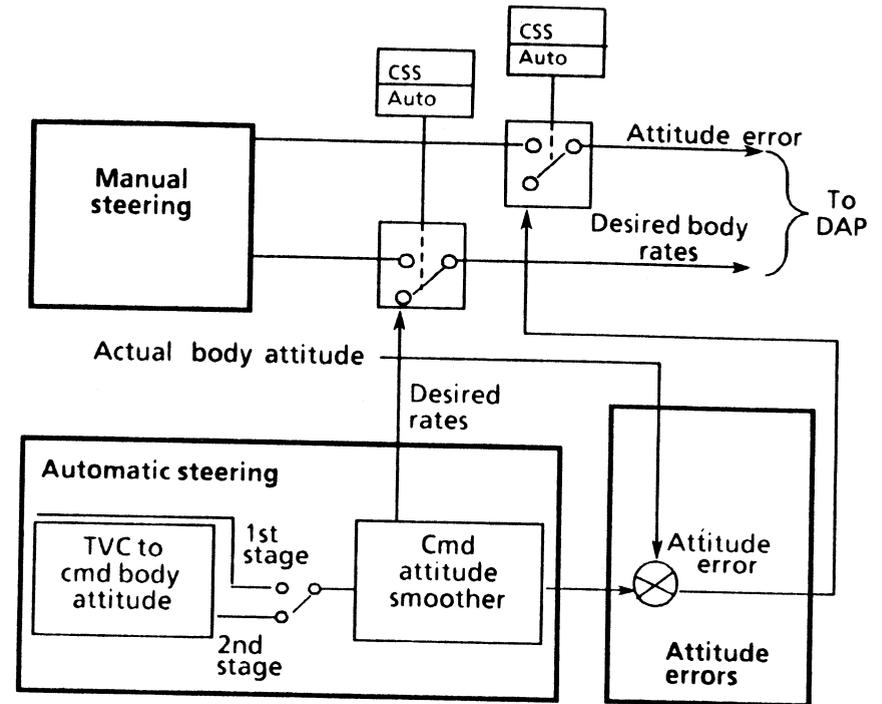
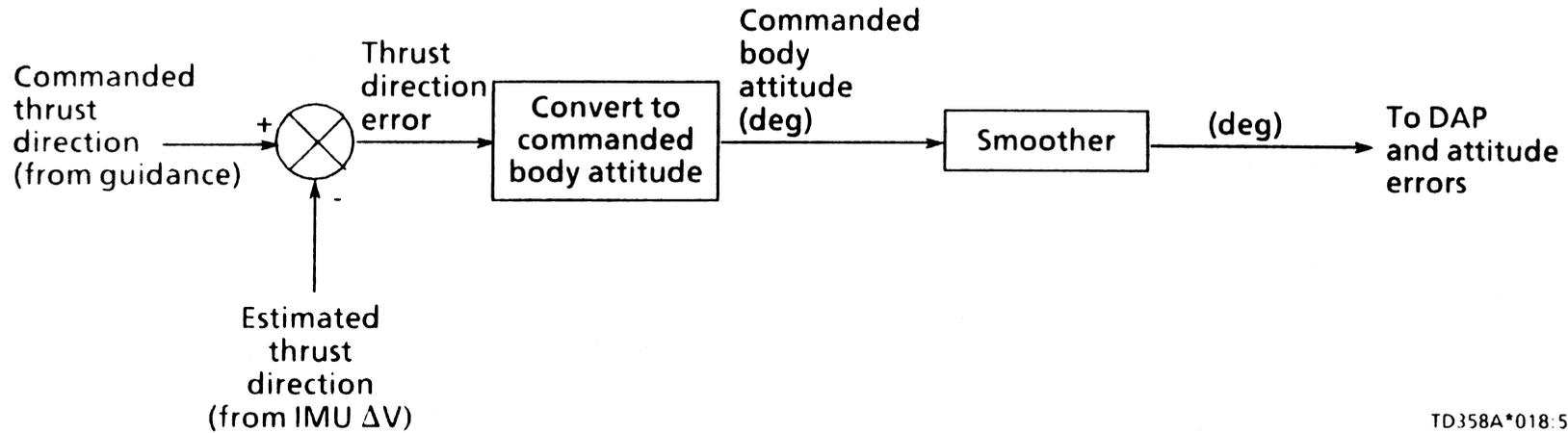


Figure 5-3.- G/C steer.

Automatic Steering

Basically stated, automatic steering differences the commanded thrust direction from guidance with that of the estimated thrust direction from IMU ΔV data to generate a thrust direction error. This error is then converted into a commanded body attitude in the form of

a quaternion. This is shown in figure 5-4. The thrust direction is converted to a body attitude so G/C steer can derive an attitude error. Note in figure 5-3 that there is a software switch that bypasses this function for first stage because the commanded body attitude comes directly from the I-loaded table of vehicle attitudes and no conversion is necessary.



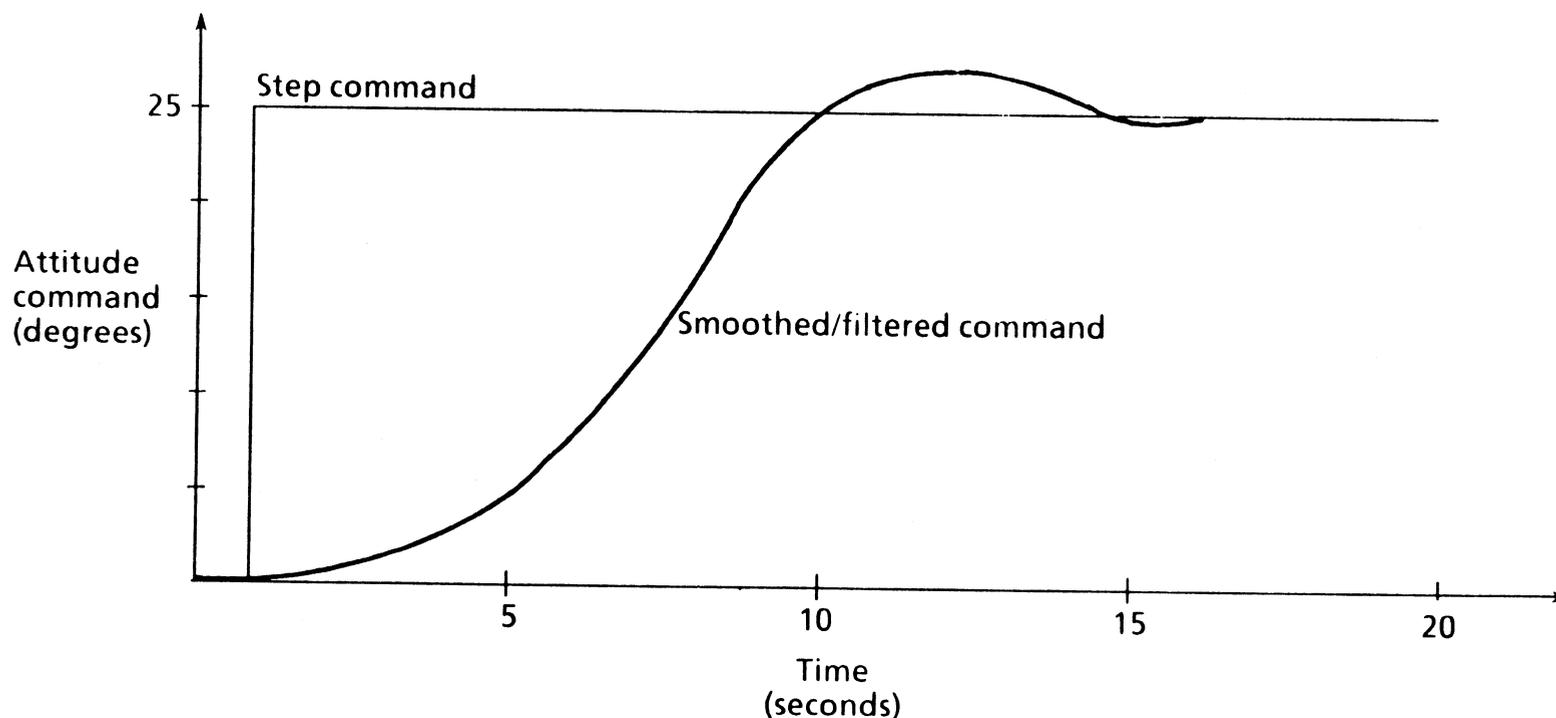
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Figure 5-4.– Derivation of commanded body attitude.

From here, the commanded body attitude passes through a smoother. With vehicle structural limits in mind, the commanded attitude must be smoothed or metered out so as not to exceed these limits during any maneuver. To understand what the smoother/filter does, think of a 25° attitude command going into the smoother and the curve shown in figure 5-5 being the output of the smoother. It actually took 15 seconds before the output command settled to 25°, thus smoothing the commanded attitude. This prevented overstressing the vehicle with one large instantaneous command.

Now look at what G/C steer does to the commanded body attitude in order to smooth it. Figure 5-6 shows the commanded attitude coming in from guidance, passing through the maneuver smoother, and coming out as a smoothed desired command.

The smoother takes the commanded body attitude angle and converts it into an angular acceleration via flight-phase-dependent steering gains. This is necessary because the limiter, which will limit the maximum maneuver rate, is using vehicle acceleration to control the rate.

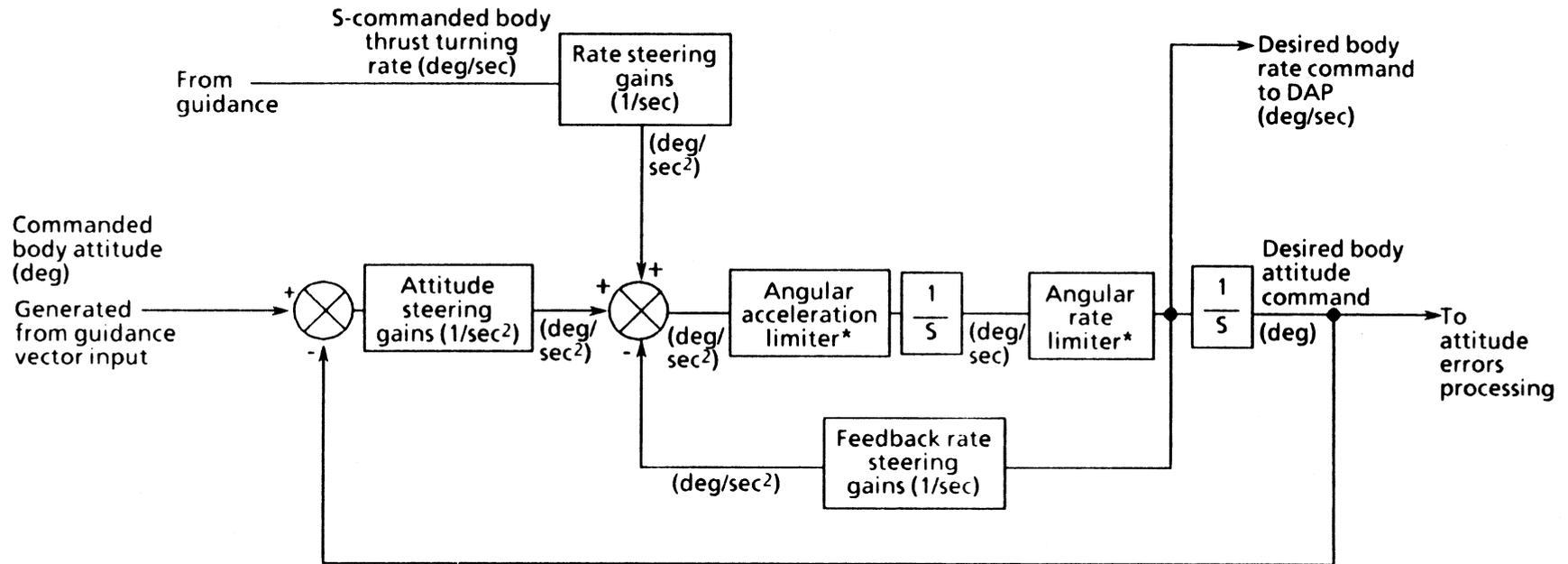


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Figure 5-5.— Step vs. smoothed command.

It is then converted back into a rate which is the attitude rate command sent to the DAP. Next, the rate is converted back to an attitude angle, which is the desired body

attitude. It is then sent to attitude errors and also fed back to generate the error between the commanded attitude and the smoothed attitude.



*Limits are function of flight phase per following table:

Flight phase	Acceleration limit (deg/sec ²)	Rate limit (deg/sec)
1 st stage	5.0	15
2 nd stage	1.25	5
RTLS PPA	2.5	10
RTLS PPD	2.39	12.25

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Figure 5-6.- Simplified maneuver smoother.

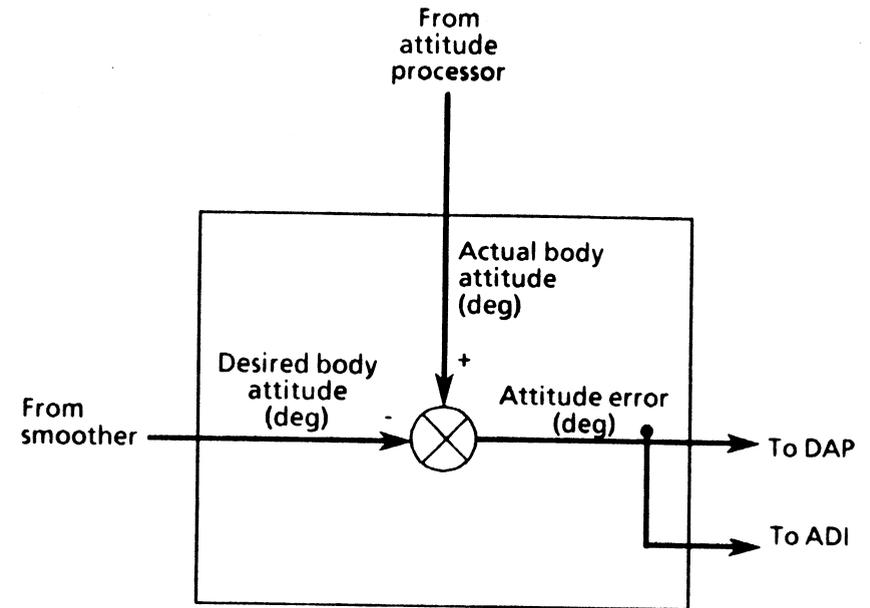
Attitude Errors Processing

From the smoother, the desired body attitude is differenced with the actual body attitude to generate the attitude error for the DAP. This is illustrated in figure 5-7.

Note that the attitude error is sent to the ascent ADI processor for crew display. During load relief the vehicle actually flies away from the commanded attitude. For this reason, the guidance error needles should not be used during this regime because they will be indicating errors that are required for load relief. There is logic to compensate for this. However, due to software lag time, this compensation is virtually useless.

We have now seen the two signals that G/C steer is responsible for generating, filtered attitude error and attitude rate commands for the roll, pitch, and yaw axes.

As can be seen in figure 5-3, G/C steer also contains logic for manual thrust vector control. Essentially, all manual steering does is bypass auto steering. More about this later.



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Figure 5-7.- Attitude error generation.

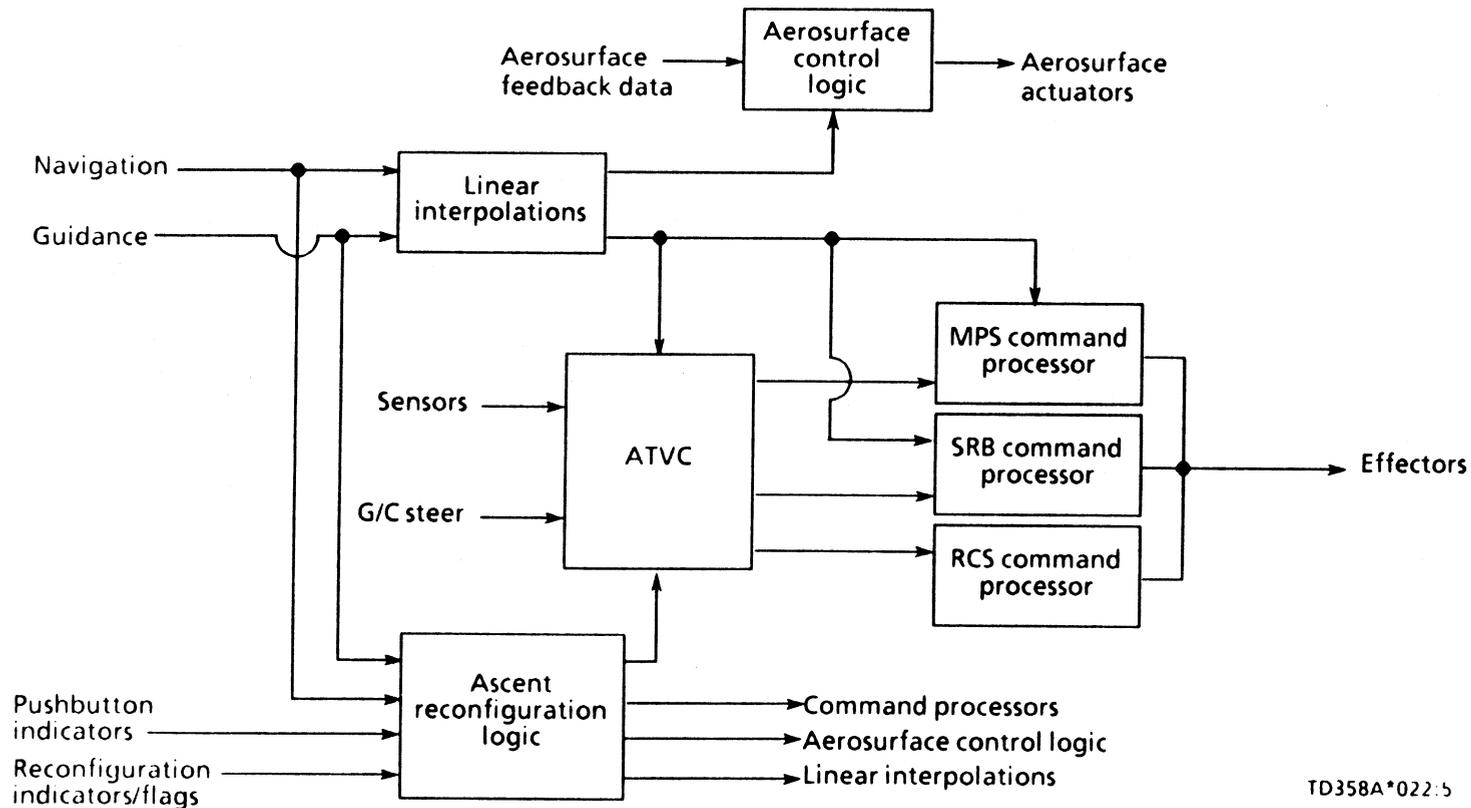
The Digital Autopilot

The ascent DAP is responsible for converting the vehicle attitude errors, attitude rates, and acceleration inputs into control element deflections. This is what is commonly called flight control. The DAP is divided into seven separate software modules.

- Reconfiguration logic
- Linear interpolations

- Ascent thrust vector control (ATVC)
- MPS command processor
- SRB command processor
- RCS command processor
- Aerosurface control logic

Figure 5-8 is a functional representation of the DAP. A discussion of each module follows.

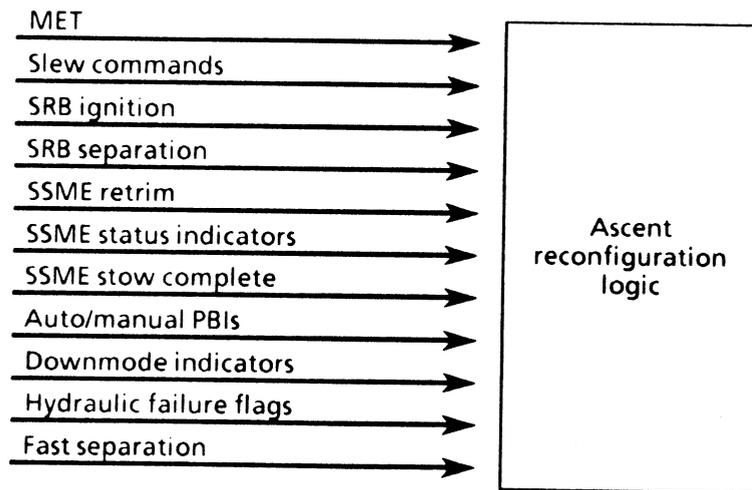


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Figure 5-8.- DAP software module interface.

Reconfiguration Logic

When vehicle control modes change, the DAP and G/C steer must be informed of these changes to prevent a loss of control. This module can be thought of as the book-keeper of flight control. Reconfiguration logic generates the indicators that are needed in order for the DAP to recognize a control mode change. Examples of control mode changes are SRB SEP, SSME failures, MECO, etc. Reconfiguration logic also receives indications from the cockpit pushbutton indicators (PBI's) to determine manual or auto flight control. Figure 5-9 illustrates the inputs to this module.



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Figure 5-9. – Reconfiguration logic inputs.

Linear Interpolations

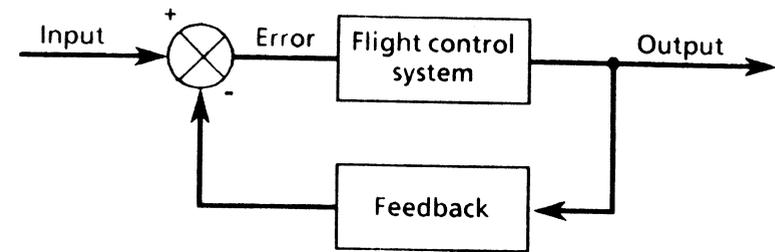
This logic is responsible for generating acceleration profiles, trims, elevon schedules, and scheduled gains as functions of Earth relative velocity magnitude, MET, and estimated vehicle mass. These trims, gains, and schedules are utilized by flight control either to enhance or to limit the output of a signal to obtain the flight control response necessary for a particular flight regime; for example, using a gain either to increase or to decrease an error signal. Remember, in all future diagrams if a trim, gain, or schedule is depicted, it comes from linear interpolations.

ATVC

Just as the DAP is the heart of the FCS, ATVC can be thought of as the heart of the DAP software. It is within this module that first- and second-stage control laws are scheduled, implemented, and switched. For this reason, first- and second-stage flight control and their differences will be discussed within this module.

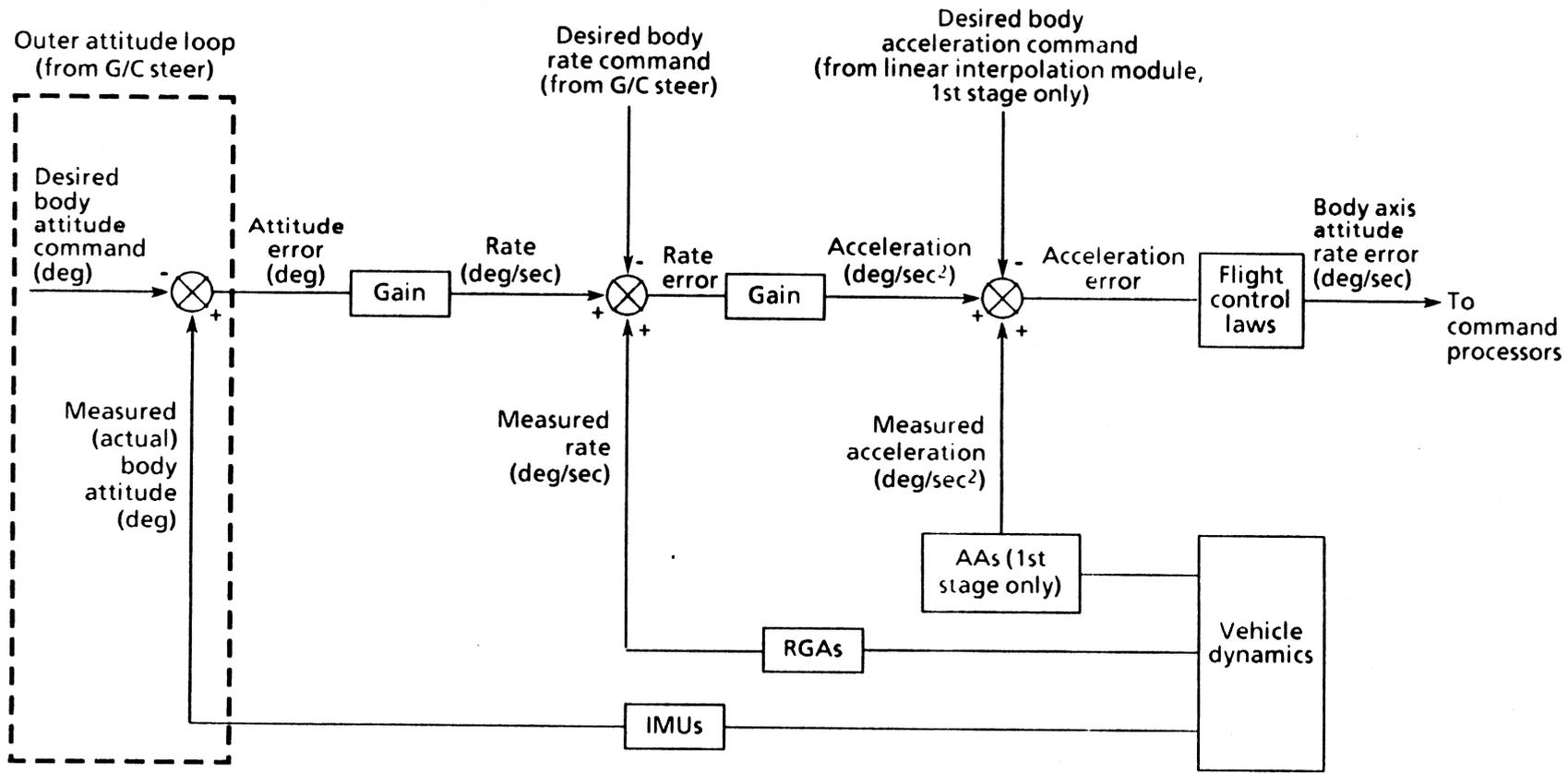
The function of TVC is to close the acceleration and rate loops within the outer attitude loop to generate body axis attitude rate errors. These errors are decreased by the SSMEs and SRBs. Before we go any further, a short review of a closed loop control system is required.

A closed loop control system is one in which the control action is dependent on the output. Figure 5-10 illustrates this point. The summing joint at left is used to generate an error which goes into the FCS. The FCS in turn utilizes the vehicle control laws and dynamics to generate an output. This output is fed back and summed with the input in order to reduce the errors going into the FCS. Body axis attitude errors and body axis rate commands from G/C steer are the commanded inputs to each axis. The primary feedbacks are rates via the RGAs and, in the case of first-stage flight, normal and lateral acceleration from AAs. IMU attitude information is also fed back. The outputs are body axis attitude rate errors for roll, pitch, and yaw. From here the output signals are sent to their respective MPS, SRB, or RCS command processors. Figure 5-11 is an overview of the ATVC control loop, which includes attitude feedback, rate feedback, and acceleration feedback.



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Figure 5-10.— *Simplified closed loop control system.*



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Figure 5-11.- ATVC control loop.

First-Stage Flight Control

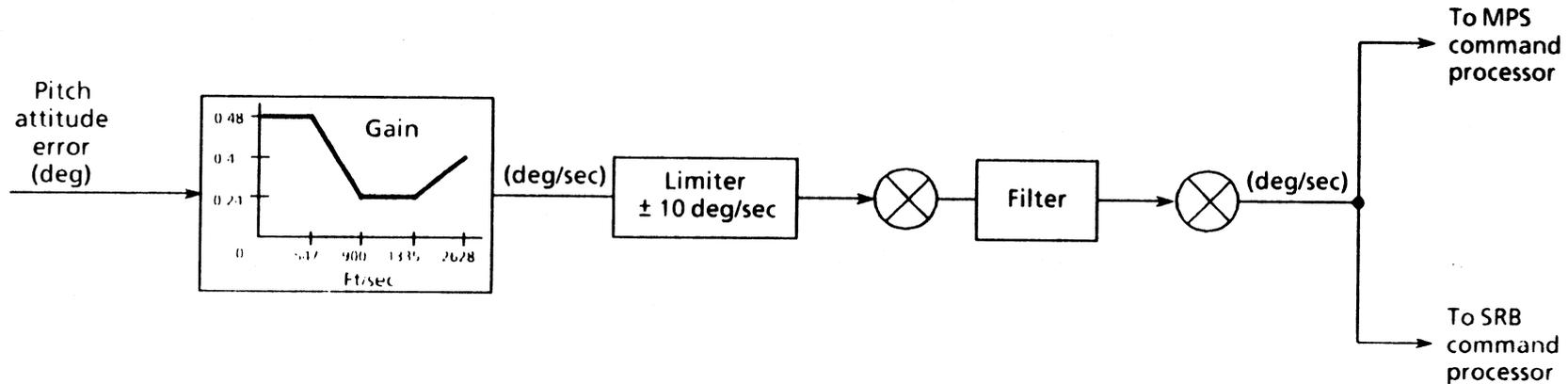
The inputs to TVC are attitude errors and rate commands. We will now take a look at what happens to these inputs. The approach taken will be to build the control loop in pieces to facilitate comprehension. We will start out with the pitch channel, then build the yaw channel, and finally the roll channel.

During first stage, the primary feedback inputs are body axis rates. Roll rate inputs come from the orbiter roll RGAs. Pitch and yaw inputs come from the SRB pitch and yaw RGAs. The SRBs do not have roll RGAs because they would be located so far from the c.g. that they would be indicating higher rates than actually exist. The reason the pitch and yaw rate inputs come from the SRBs is that the orbiter body flexes during first-stage flight. This flexing comes from atmospheric effects such as wind shears.

An SRB is a far more rigid structure; therefore, its rates are used to prevent feeding vehicle body flexing into flight control. Normal and lateral acceleration from the orbiter AAs are fed back in the pitch and yaw axes, respectively. Vehicle acceleration is a good indicator of loading; therefore, the accelerometers are used for load relief.

The Pitch Channel

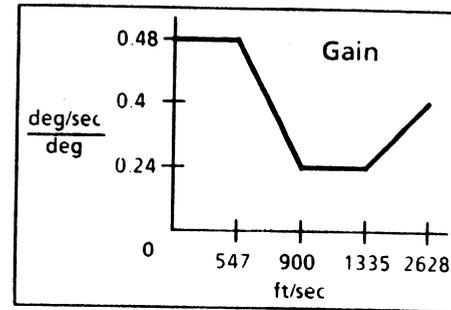
Recalling that one of the two inputs to TVC from G/C steer is attitude error, this attitude error exists for each axis: roll, pitch, and yaw. The attitude error provides a good starting point in that it can be thought of as the foundation of the flight control loops. Figure 5-12 shows the outer attitude loop, which is the attitude error. This is also known as the forward loop.



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Figure 5-12.- Forward loop.

After the attitude error reaches TVC, the first thing to occur is that it is multiplied by a gain which is variable with relative velocity. Remember that a signal that is "gained" is just multiplied by that gain value. Figure 5-13 shows an example of a gain. For example, if an attitude error of 5° comes into TVC at a $V_{rel} \leq 547$ ft/sec, this error is multiplied by a value of 0.48, which equals 2.4° . Once a V_{rel} of 900 ft/sec is achieved, the equation becomes $5^\circ \times 0.24 = 1.2^\circ$. This indicates that during load relief less attention is being paid to the attitude error. Because V_{rel} is a good indicator of dynamic pressure, one can easily see why these gains are functions of V_{rel} for load relief. The reason for the values of the gains will become more obvious as we complete the control loop. This gain also changes the units of the signal from an attitude error angle of degrees to an attitude rate error of degrees per second. This is necessary because the feedback loops utilize rate data from the RGAs. The signal is next limited to 10 deg/sec to prevent overstressing the vehicle by trying to command large rates.



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Figure 5-13.—Pitch attitude gain.

Now that we have an attitude rate error signal generated, we can take a look at the RGA feedback signals from the SRBs. The rate feedbacks from each SRB are averaged together to generate one rate feedback signal. This signal is compared with the commanded vehicle rate that was generated in G/C steer. This is accomplished by differencing the two signals to generate a rate error. See figure 5-14. Note that the polarity is actual vehicle rate minus the commanded rate. The rate error signal is then added to the attitude rate error that was previously computed.

Filters are used to prevent unwanted signals from getting out to the flight control effectors and causing unwanted vehicle perturbations.

- Lead/lag filter - Acts as a smoother to prevent a large step input to the FCS.
- High pass filter - Prevents unwanted frequencies from going out to the FCS. This filter filters out unwanted low frequency "noise."
- Low pass filter - Prevents unwanted high frequencies from going out to the FCS. These frequencies can develop as the result of vehicle vibration.

The next step for this error signal is through a filter. A detailed discussion of this filter is not necessary to comprehend the FCS. A brief explanation of the types of filters the FCS uses is presented.

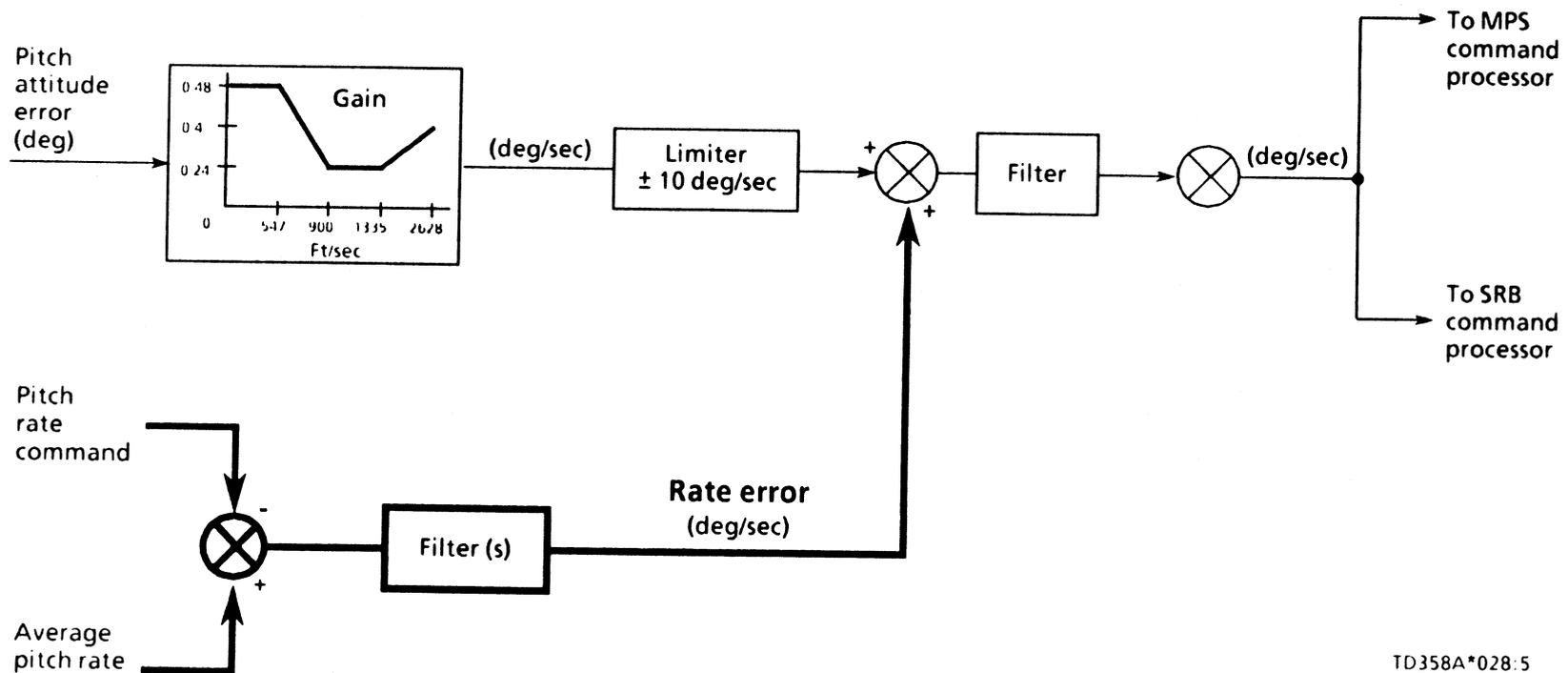


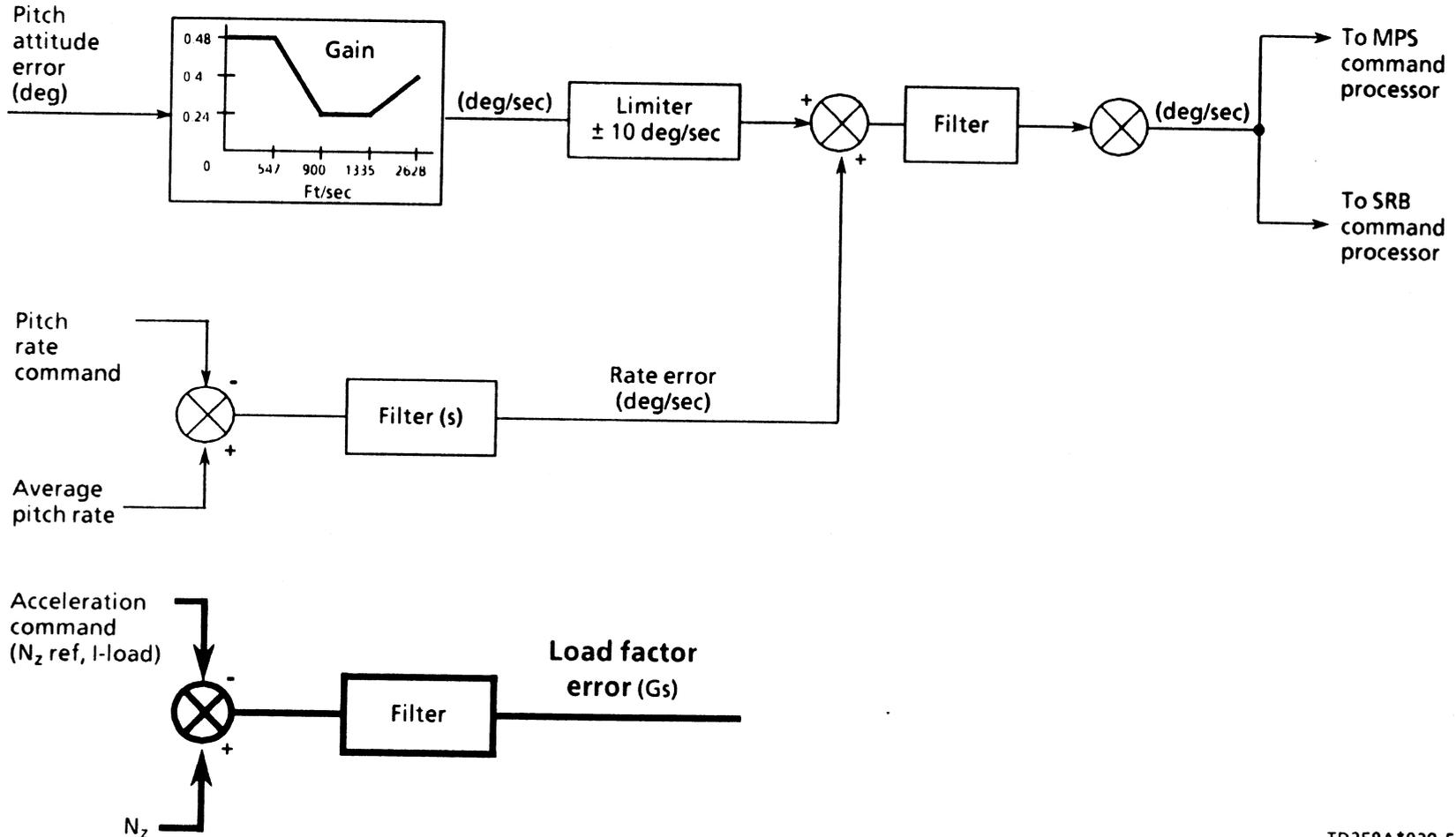
Figure 5-14.- Rate feedback loop.

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- Body-bending filter - Prevents extraneous rates from reaching the FCS as a result of vehicle vibration and flexing.

We will now discuss the final means of feedback, the accelerometer. The accelerometer we are concerned about in the pitch channel is mounted on the body Z-axis

and therefore measures N_z . This acceleration is differenced with a commanded acceleration which is I-loaded in the form of a table that is a function of V_{rel} . The resultant signal is called the load factor error. Note in figure 5-15 that the polarity is actual minus commanded. This error signal is then filtered.



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Figure 5-15.- Load factor error.

Ideally, the accelerometers should be placed at the vehicle c.g. This is impossible since the c.g. constantly moves throughout the entire flight due to the expulsion of propellant mass and separation of SRBs and ET. Basically, the c.g. lies below the orbiter at lift-off and gradually rises toward the orbiter as propellants are consumed. Therefore, the AAs must be compensated since they are not

located at the c.g. This is required because normal, not rotational, acceleration must be measured. Figure 5-16 illustrates how the AAs are compensated. Looking at this figure, notice that the pitch RGA signal is brought in and used to sum out the rotational acceleration.

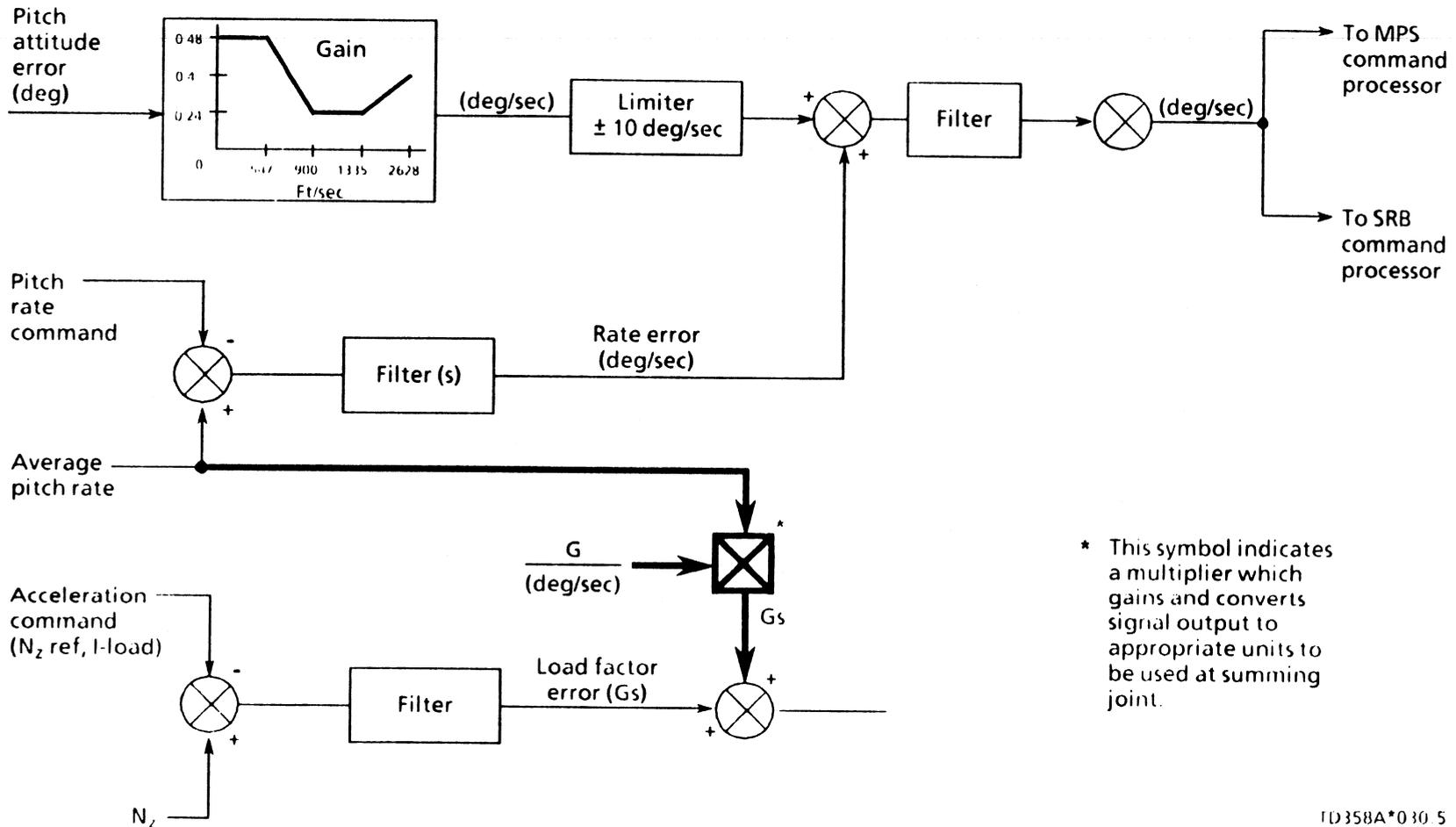
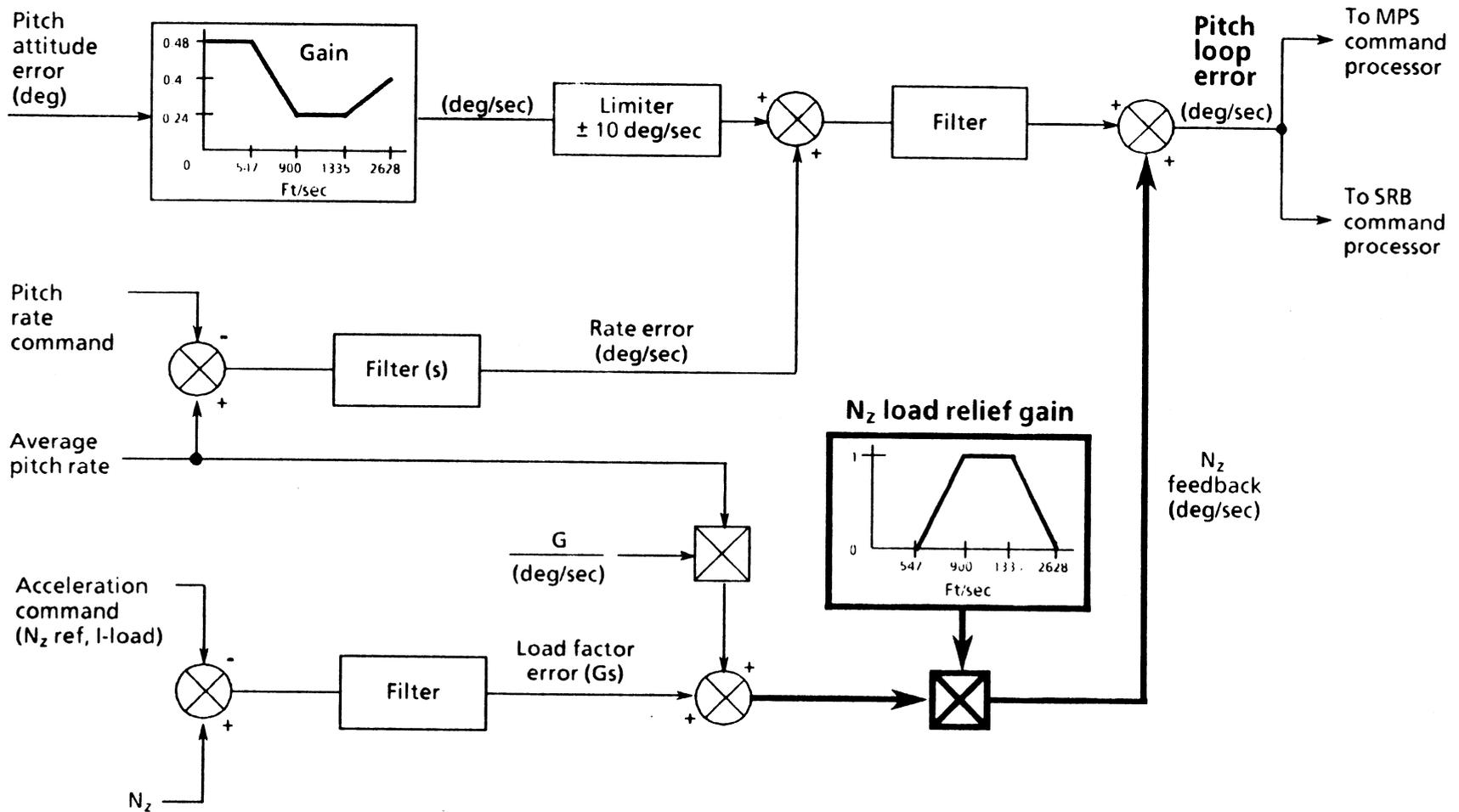


Figure 5-16.- Feedback to eliminate rotational acceleration.

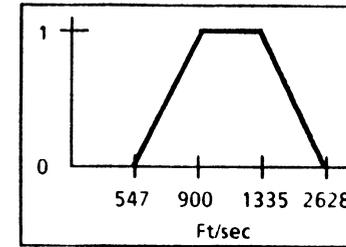
The resultant signal, a compensated acceleration error, is filtered and then gained by the N_z load relief gain. It is then added to the rate error signal that was generated in the outer attitude error loop as shown in figure 5-17.



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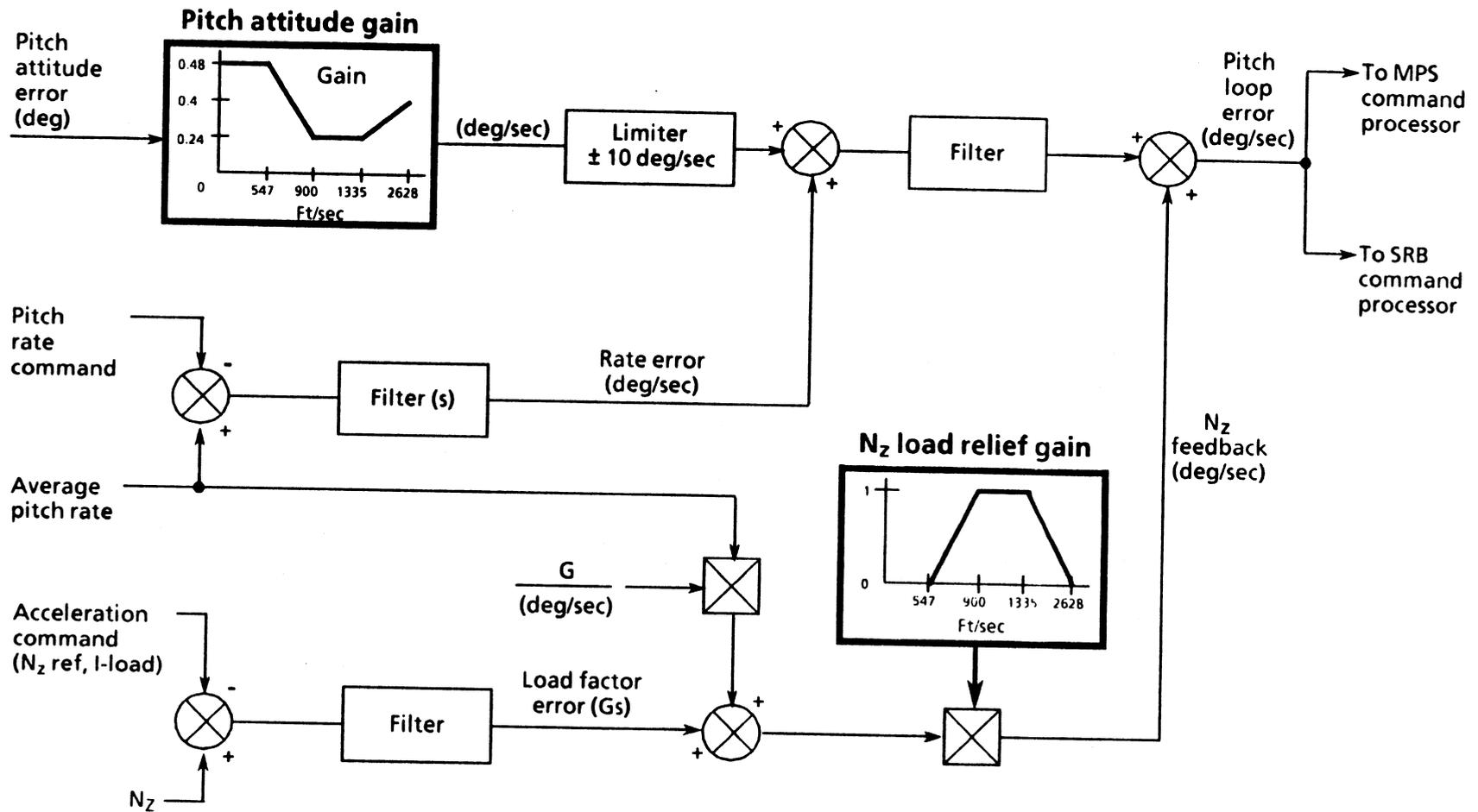
Figure 5-17.- Acceleration feedback loop.

Figure 5-18 shows the N_z load relief gain; note the direction of this gain. The N_z feedback gain is a sensitivity gain comparable to the gain which multiplies the pitch attitude error in the attitude error loop. These two gains play together; and, by looking at figure 5-19, one can see that, as the pitch attitude error gain decreases, beginning at V_{rel} of 547 ft/sec, the N_z load relief gain increases. Note that it actually increases from zero, which means the acceleration error was never heard until $V_{rel} = 547$ ft/sec (beginning of load relief). Load relief increases sensitivity to acceleration error and decreases sensitivity to attitude error. This means the vehicle flies out acceleration error as opposed to attitude errors to relieve loads. The net result is that the vehicle will actually steer off course in order to reduce the acceleration error to reduce loads in the high \dot{q} region. Winds and wind shear are the prime culprits that cause off-nominal loading.



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Figure 5-18.— N_z load relief gain.



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Figure 5-19.- Pitch control loop.

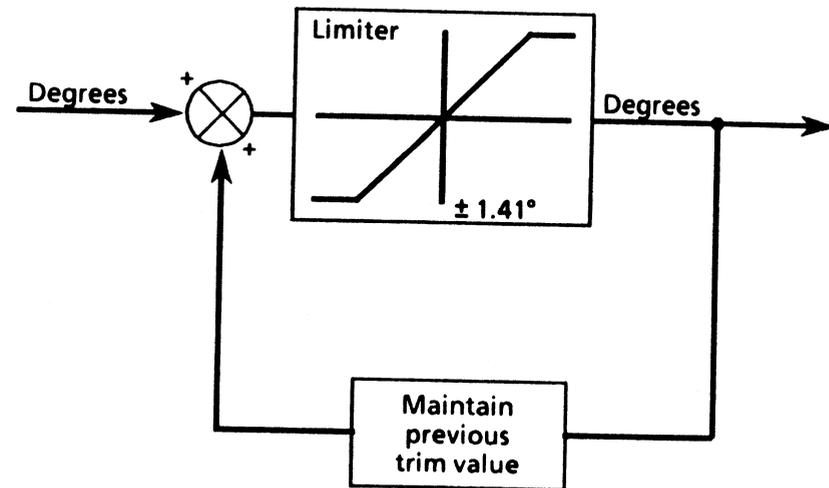
The Nz feedback next is added to the pitch attitude rate error to come up with the total pitch loop error. The Nz load relief gain converts the acceleration back into rate units of degrees per second. This is necessary so that "apples and oranges" will not be added together in the outer attitude error loop. From here the total pitch loop error is filtered and takes two paths, one to the MPS command processor and the other to the SRB command processor, after passing through the forward loop gain. This gain is a function of time; it is required to have an accurate way of tracking the greatly changing inertia of the vehicle.

Autotrim is necessary to trim out attitude errors that are not a result of main engine and/or SRB nozzle deflections. These attitude errors are caused by many factors such as a bent airframe, c.g. offset, thrust mismatch, etc. Effectively, autotrim adds a bias to correct for these types of problems.

Autotrim acts on the attitude error in the forward loop. Figure 5-20 illustrates autotrim in greater detail. Figure 5-21 illustrates where autotrim comes into the control loop. Autotrim is just a limited integrator which performs two functions.

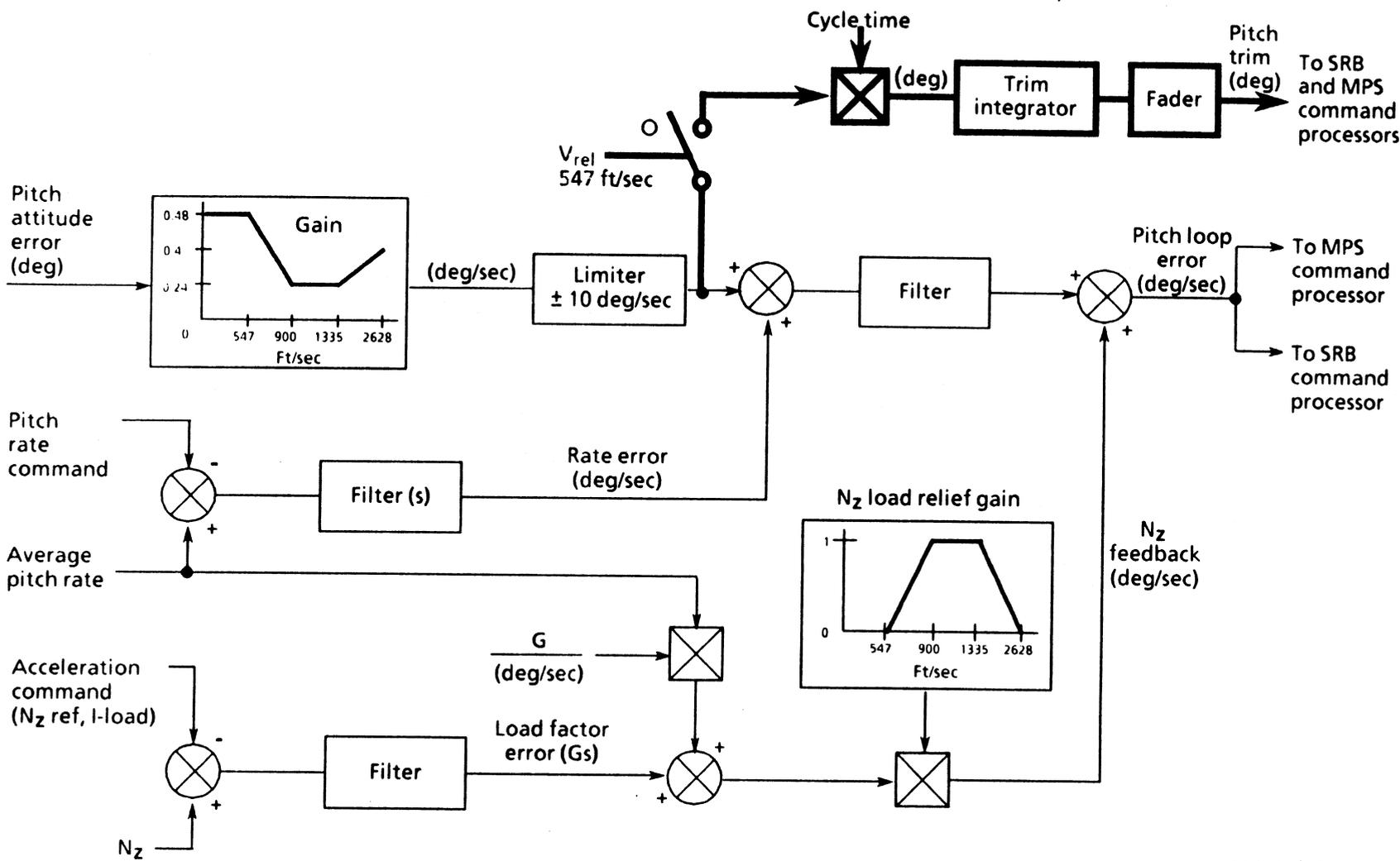
- Limits the maximum trim value to the command processors at $\pm 1.41^\circ$ for first stage and $\pm 7^\circ$ for second stage.
- Maintains the trim value to the command processors such that when an attitude error is nulled, the trim value will not go to zero. This concept is analogous to a pilot's "trimming out" a "heavy" stick to light control stick feel. Just because they feel no pressure does not mean they want to take the trim back out.

Looking at figure 5-20 again, an example would be in order. On the first pass through the integrator, let us say, a 0.5° input is present. Because this integrator is a ramp function, the output will be something less than 0.5° , say 0.4° . This gets fed back to be summed with the input. On the next pass through, the error will have decreased some due to the previous trim integrator output to say, 0.4° . This gets summed to 0.8° and sent to the integrator, and the output is 0.7° . It gets fed back and summed with 0.1° and will maintain a 0.8° trim even when the input is zero because $0 + 0.8^\circ = 0.8^\circ$. Thus, the trim value is maintained. In first stage, if an input is greater than 1.41° , the output will always be 1.41° ; hence, the name limited integrator.



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Figure 5-20.- Autotrim integrator.



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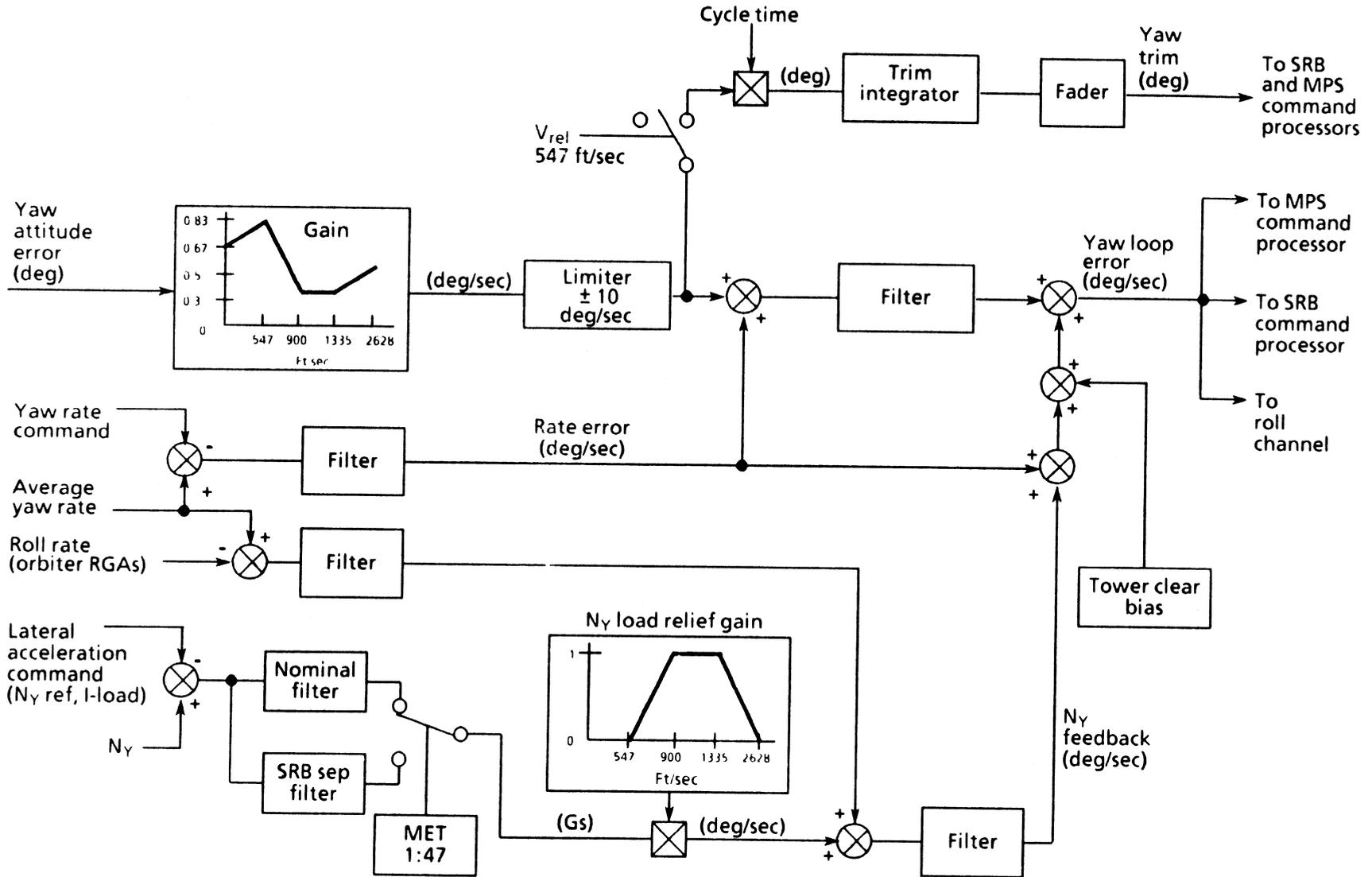
Figure 5-21.- Autotrim logic.

Note that the integrated error is summed downstream of the pitch attitude gain in order not to lose the maximum trim value in the gain multiplications. Note also, there is a software switch and a functional module called fader. These two elements play together. The switch terminates autotrim when $V_{rel} = 547$ ft/sec, which is the beginning of load relief. It is reenabled shortly after SRB separation, depending on engine failures. The output of the integrator is held constant through load relief to help reduce loads by not trimming, which would cause additional maneuvering and more loading. The fader then kicks in to ramp this frozen signal to zero prior to SRB separation.

This completes the pitch control loop for first stage TVC. Two pitch error signals have been generated: one gained for the MPS and the other gained for the SRBs. Each signal next goes to its respective command processor to be converted to main engine bell and SRB nozzle deflection commands.

The Yaw Channel

The yaw control loop is very similar in design to the pitch loop; therefore, there is no need to build it as meticulously as the pitch loop, and only the deltas will be discussed. Figure 5-22 presents the yaw loop in the same format as the pitch loop.



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Figure 5-22.- Yaw control loop.

The first major change is in the compensation for the AA offset from the c.g. Both roll and yaw rates are needed to correct for this. In a perfect situation, with a pure vehicle yaw maneuver, there would not be any roll crosscoupling. However, the shuttle is not a perfect situation, and when a yaw maneuver is commanded, the vehicle will roll as shown in figure 5-23 since the SSMEs are located above the c.g.

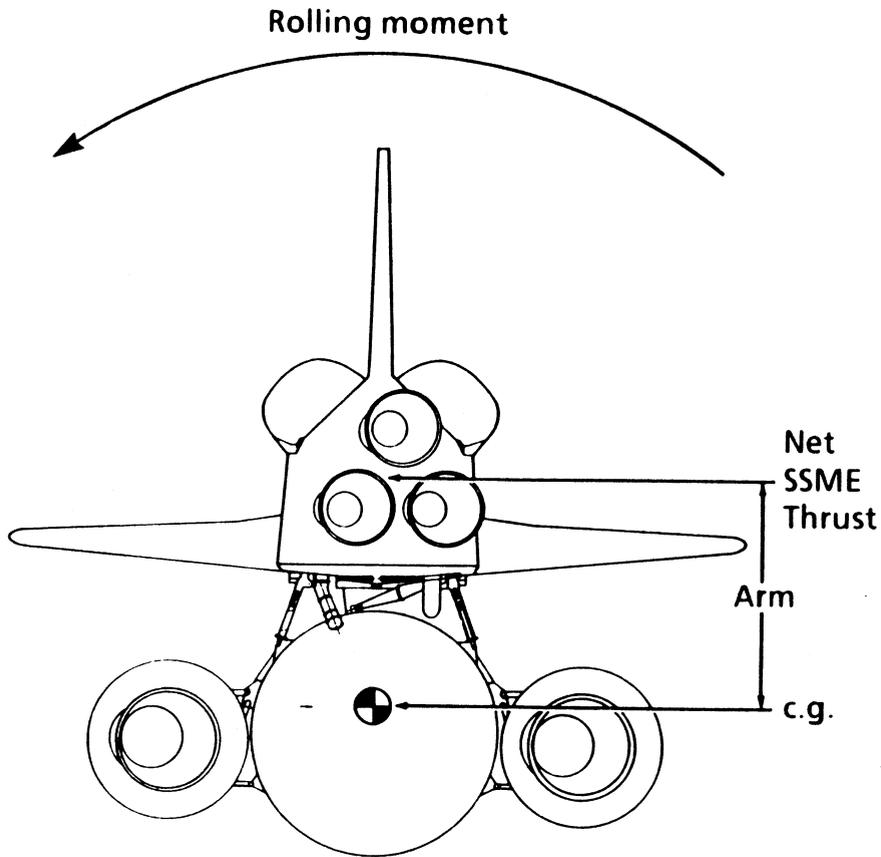


Figure 5-23.- Vehicle response to a right yaw command.

The second change is the addition of a software switch within the load factor error block. This switch controls two filters; its position will determine which filter is being used in the feedback loops to the command processors. In addition to load relief, the yaw accelerometers are used to aid in controlling the vehicle in yaw as it goes through SRB tailoff and prepares for separation. Beginning at an I-loaded time ($\approx 1:47$ MET), this switch will move from the nominal filter to the SRB separation filter. This is just a lag filter which increases the capability to control the yaw of the vehicle as the result of uneven SRB thrust tailoff. This signal, called Ny feedback, is then ramped toward zero at an I-loaded time ($\approx 2:04$ MET) as controlled by the Ny feedback gain or when SRB chamber pressure (P_c) becomes less than 110 psi.

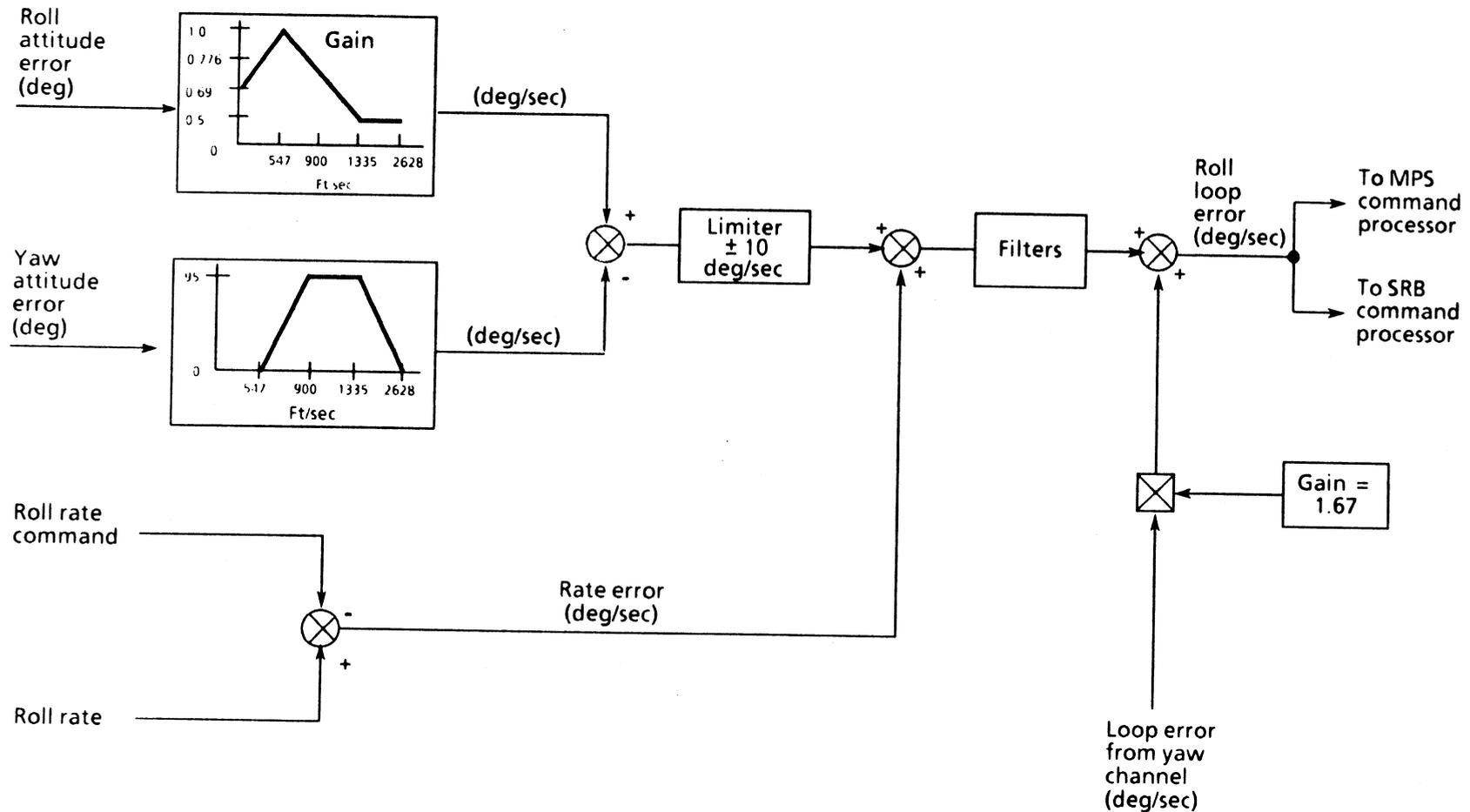
The last change is the addition of a yaw bias which is incorporated if SSME 2, the left engine, fails between lift-off and a MET of 2.5 seconds. This bias prevents vehicle recontact with the tower provided the GPCs know the SSME has failed.

As in the pitch loop, the yaw loop generates the total yaw loop error, which is then sent to the MPS and SRB command processors for conversion to bell and nozzle deflection commands.

The Roll Channel

Note in figure 5-24, the roll channel is different from the pitch and yaw channels in two respects. The first difference is that there is no autotrim for roll. The reason for this will become more apparent as we go on. The

second difference is that accelerations are not used for load relief. Roll has a unique way of performing load relief. The goal of load relief in the roll channel is to maintain the thrust vector in the desired trajectory plane while the yaw channel tries to reduce sideslip.

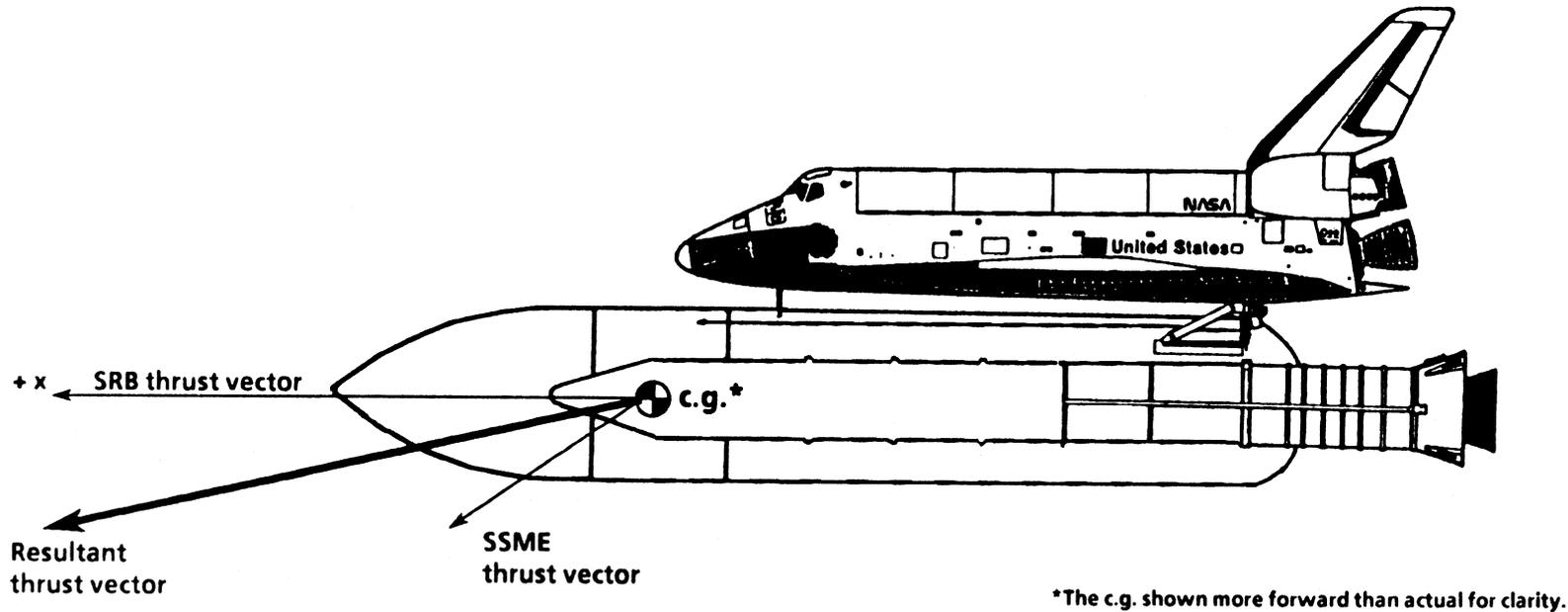


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Figure 5-24.- Roll control loop.

Figure 5-25 illustrates that the thrust vector is not parallel to the vehicle X-axis; it is actually canted down through the vehicle as the result of the SSME thrust vector. Remember, the yaw channel becomes more sensitive to the yaw acceleration error caused by sideslip from winds and less sensitive to yaw attitude errors during the load relief region. By flying out the yaw acceleration, the

sideslip is being reduced. So, by reducing sideslip, we are flying into the winds that are causing the loads problem. When the vehicle flies into the winds, it starts to fly out of the planned trajectory plane because of the SSME cant and the yaw into the winds. This costs the vehicle valuable performance capability.



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Figure 5-25.- Direction of resultant thrust vector.

Looking at figure 5-26 we can see that if the vehicle first yaws toward the wind, the sideslip is reduced, but the thrust vector has been rotated out of the desired plane. Therefore, the vehicle will roll to get the thrust vector back into the desired plane. This is the reason for the yaw attitude error crossfeed loop in the roll channel shown in figure 5-24. Note that in the load relief region (between V_{rel} 547 and 2600 ft/sec), the yaw attitude error is gained higher than the roll attitude error. This is necessary so that the vehicle will roll the thrust vector back into the desired plane proportional to how much it had yawed it out of plane.

Now back to autotrim. Since the vehicle rolls to help relieve loads, it would be senseless to try trimming to a load relief attitude that is bound to change continuously throughout first stage because of winds. This would only lead to excessive maneuvering and, consequently, vehicle loading.

One other minor change in the roll channel is the use of orbiter RGAs for roll rate feedback.

As is the case with the other channels, the roll channel generates a total roll loop error which gets sent to the appropriate command processors.

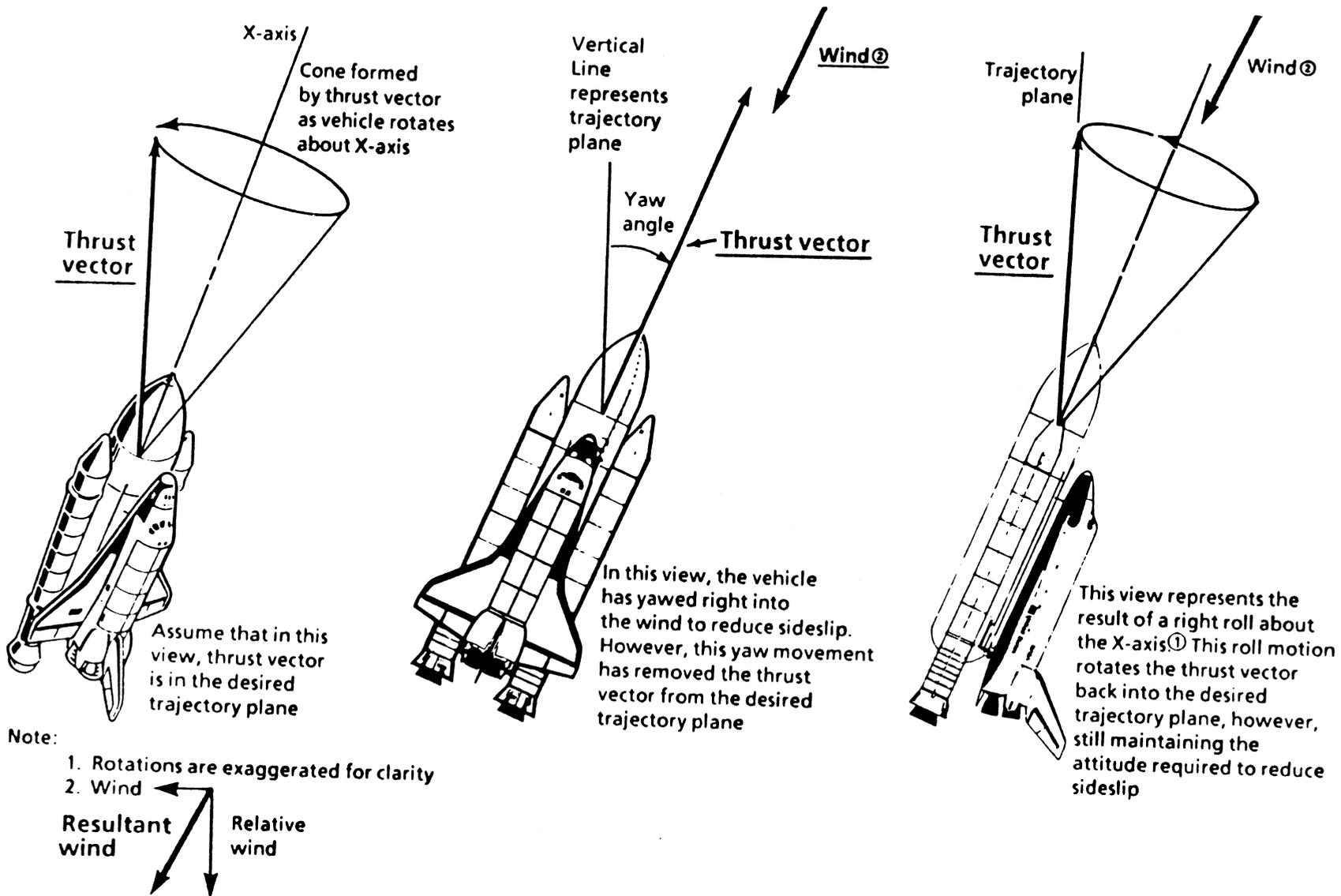


Figure 5-26.- Load relief operations.

Second-Stage Flight Control

Essentially, flight control is simplified in second stage. As shown on figures 5-27 and 5-28, there is a software switch in the Nz and Ny feedback loops labeled SRB null command. This switch opens just prior to SRB separation and takes the accelerometer feedback out of these control loops. Also at this time, the orbiter RGAs are used for rate feedback data.

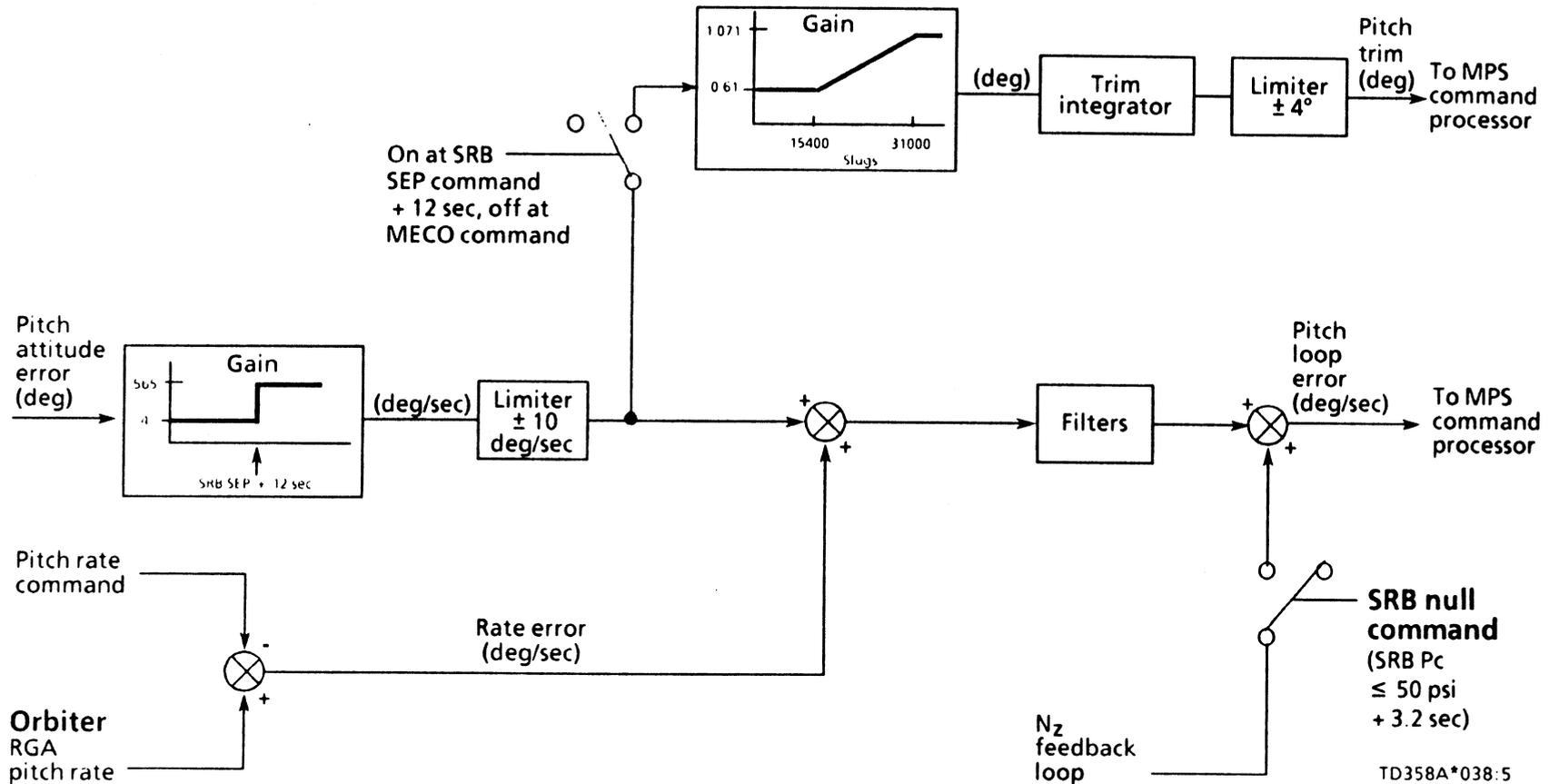
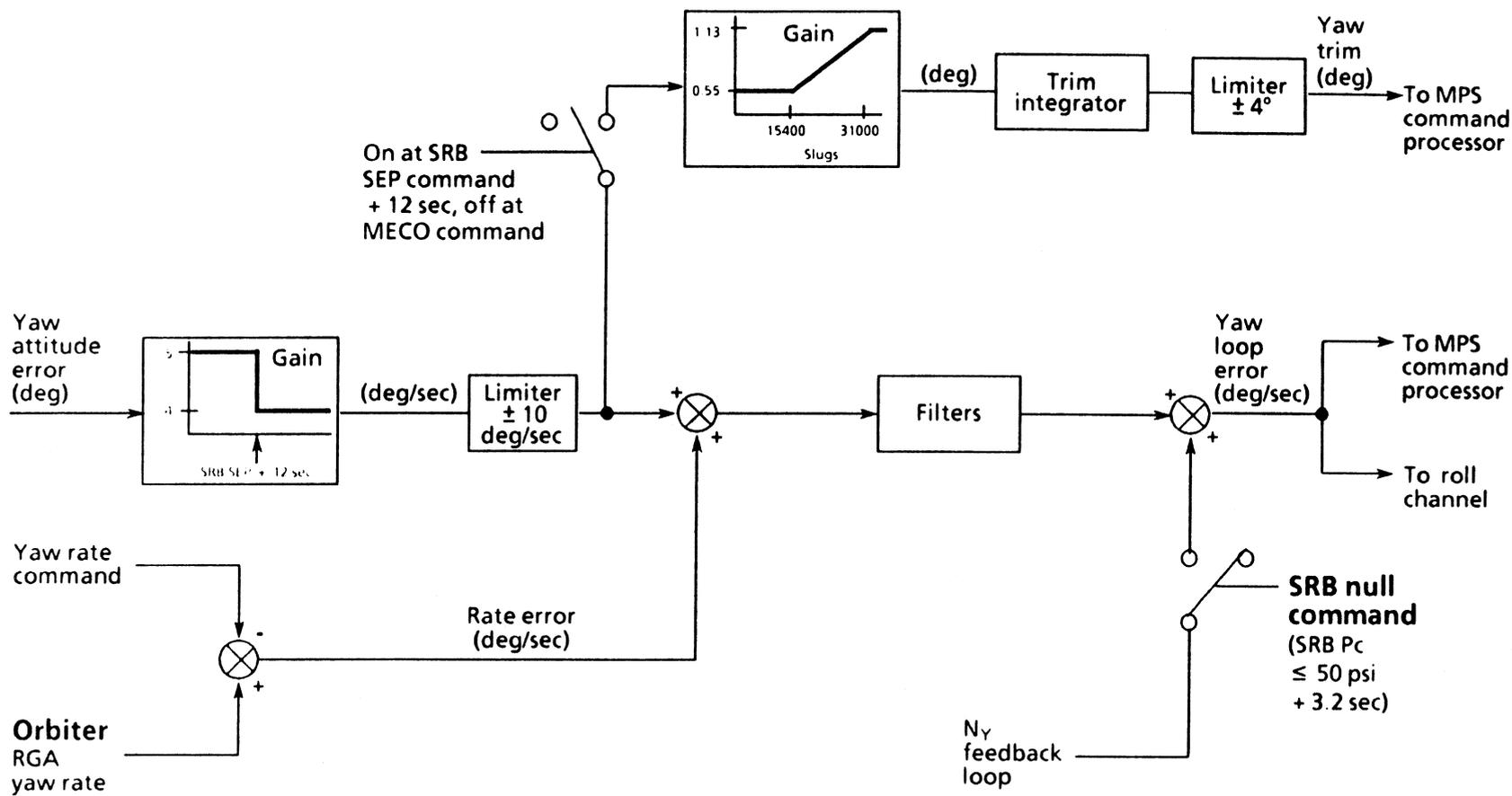


Figure 5-27. - Second-stage pitch control loop.



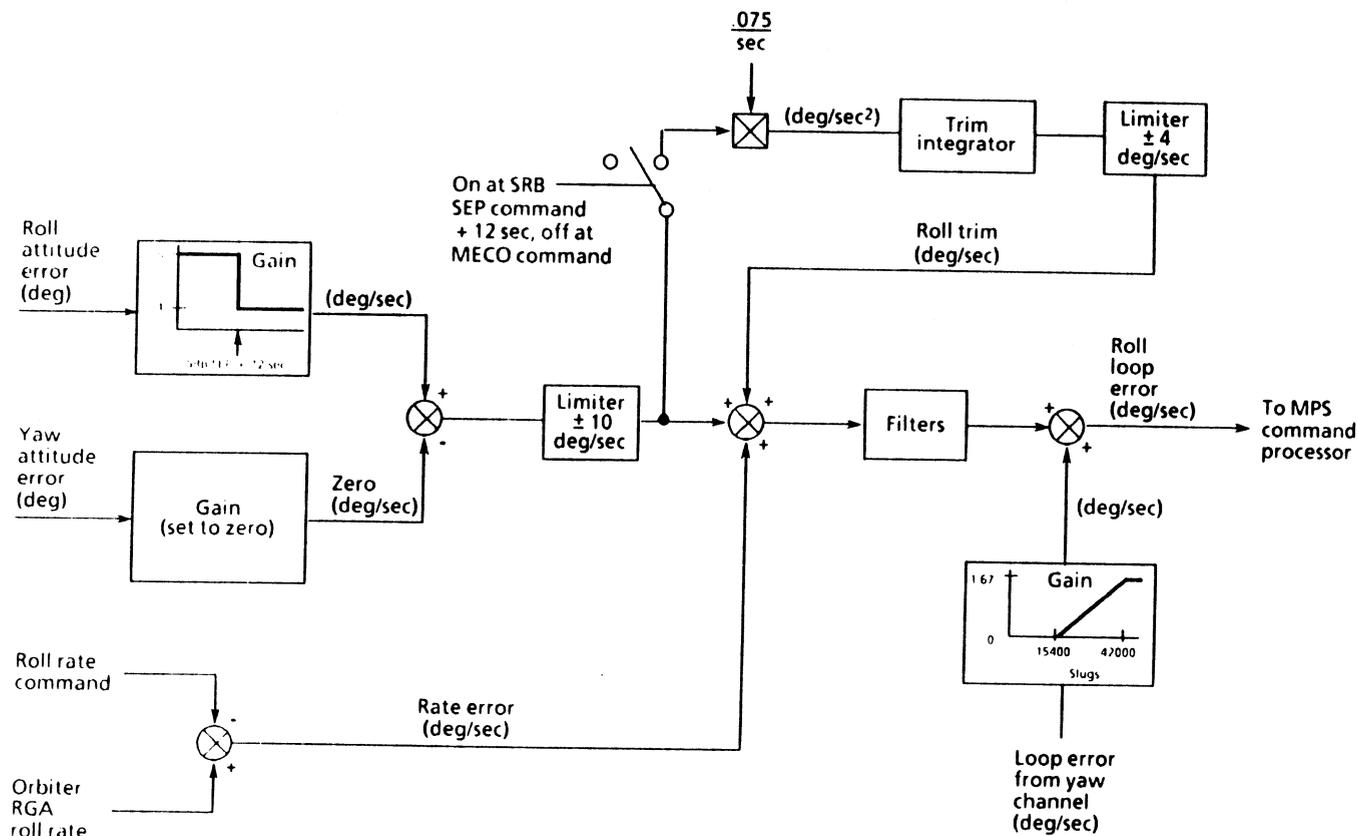
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Figure 5-28.- Second-stage yaw control loop.

Once second-stage flight is entered, there are very few changes to the flight control software. The ascent reconfiguration module has informed the TVC software that the SRBs have been separated, and now the vehicle is controllable only by means of the SSME bell deflections.

Figure 5-29 illustrates the roll channel for second-stage flight control, the only difference, since first stage is the addition of autotrim. Autotrim functions the same way it

did for first stage in the pitch and yaw channels. Autotrim is very useful in second-stage roll control because the left main engine is mounted on a weaker orbiter structure. Therefore, there is a greater possibility for thrust misalignment problems which would require autotrim. Note that there is still a yaw crossfeed. This is required for yaw and roll stability purposes because the vehicle c.g. is below the SSMEs.



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Figure 5-29. - Second-stage roll control loop.

There is one other minor difference in that the gains that were referenced to V_{rel} and MET in first stage are now referenced to mass. This is required in order to track the rapidly changing inertia. Also, the rate autotrim limit has been raised. As in first stage, the error signals in each axis go to the command processors.

MPS Command Processor

The major function of the MPS rate command processor is to convert the body axis attitude error (loop error) signals that are generated in TVC into pitch and yaw engine bell deflection commands. Figure 5-30 illustrates a top, rear, and side view of the main engines and the location of their actuators.

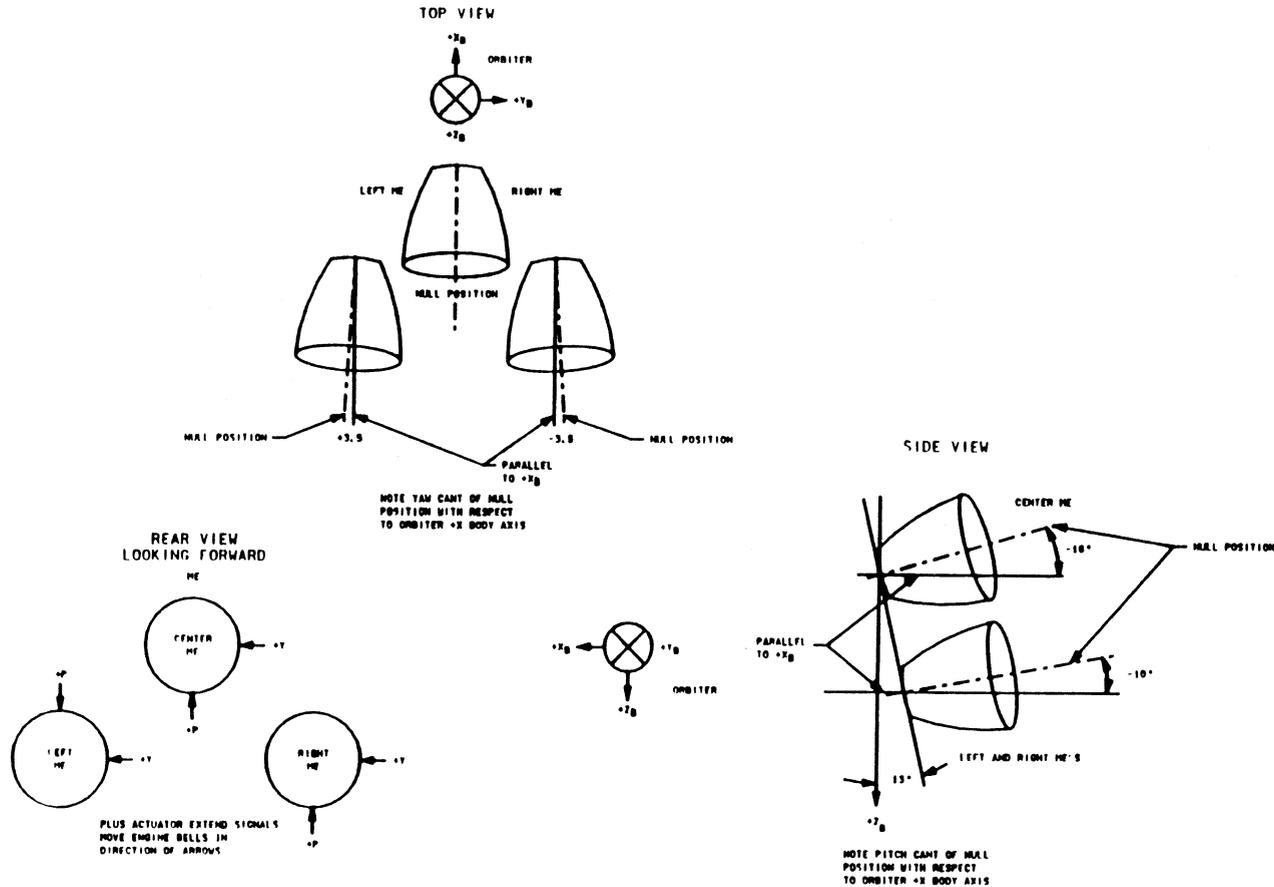


Figure 5-30.—SSME configuration.

Figure 5-31 shows a functional view of the MPS command processor. The TVC loop error signals, in degrees per second, come in from the ATVC module, where they pass through a rate gain that is a function of MET in first stage and mass in second-stage flight. As stated earlier, this is necessary in

order to track the greatly changing inertia of the vehicle. These gains also change the units of the signal to degrees. The pitch and yaw signals next go to their respective summing joints where they are added to the trim signals coming in from the pitch and yaw trim integrators. The roll trim was added within the TVC roll

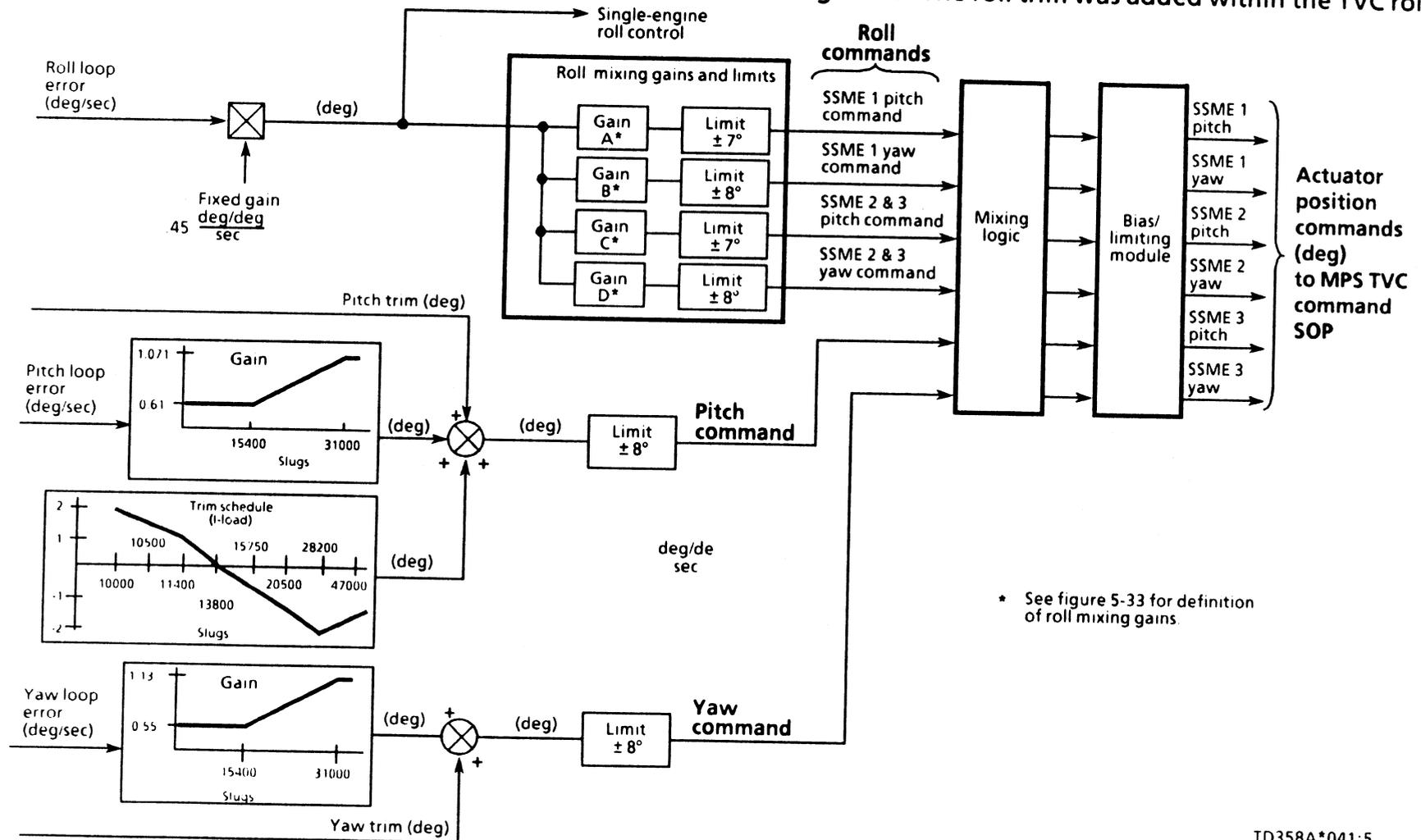


Figure 5-31.- MPS command processor.

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channel. There is no real reason why the roll channel was designed in this fashion.

From here, the body axis errors in roll, pitch, and yaw pass through position limiting. The position limiting is necessary to prevent engine bell saturation in one axis as a result of a maneuver in another axis. For example, if guidance requested a full right roll and a pitch up, the right engine bell will not be commanded all the way up for the roll. Instead, it would be limited to have some pitch control authority left over for the pitch up as well. The same limiting applies to the other axes.

Notice that the roll body axis error passes through roll mixing and gaining. The mixing is required since, to generate a vehicle roll, the engines must be deflected in some combination of engine bell pitch and yaw deflections as shown in figure 5-32. Therefore, this logic converts the body roll error into the appropriate combination of engine bell pitch and yaw errors. This combination is dependent on the number of SSMEs that are operating. When all three are operating, the roll body error is converted to a left and right main engine pitch error and a center engine yaw error. This is accomplished by multiplying various gains.

It is very important that the reconfiguration logic recognize the actual number of operating SSMEs and hydraulic systems. Otherwise, the mixing/gains will be inconsistent with the vehicle configuration, and a loss of control may result.

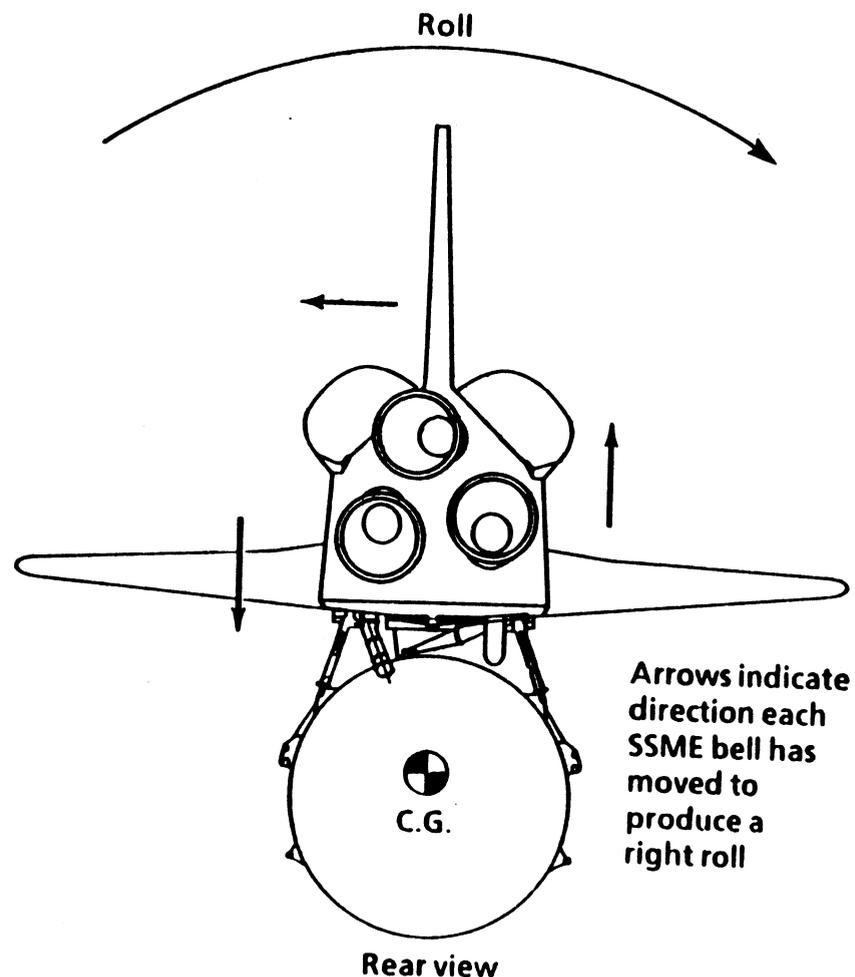


Figure 5-32. – Roll maneuver.

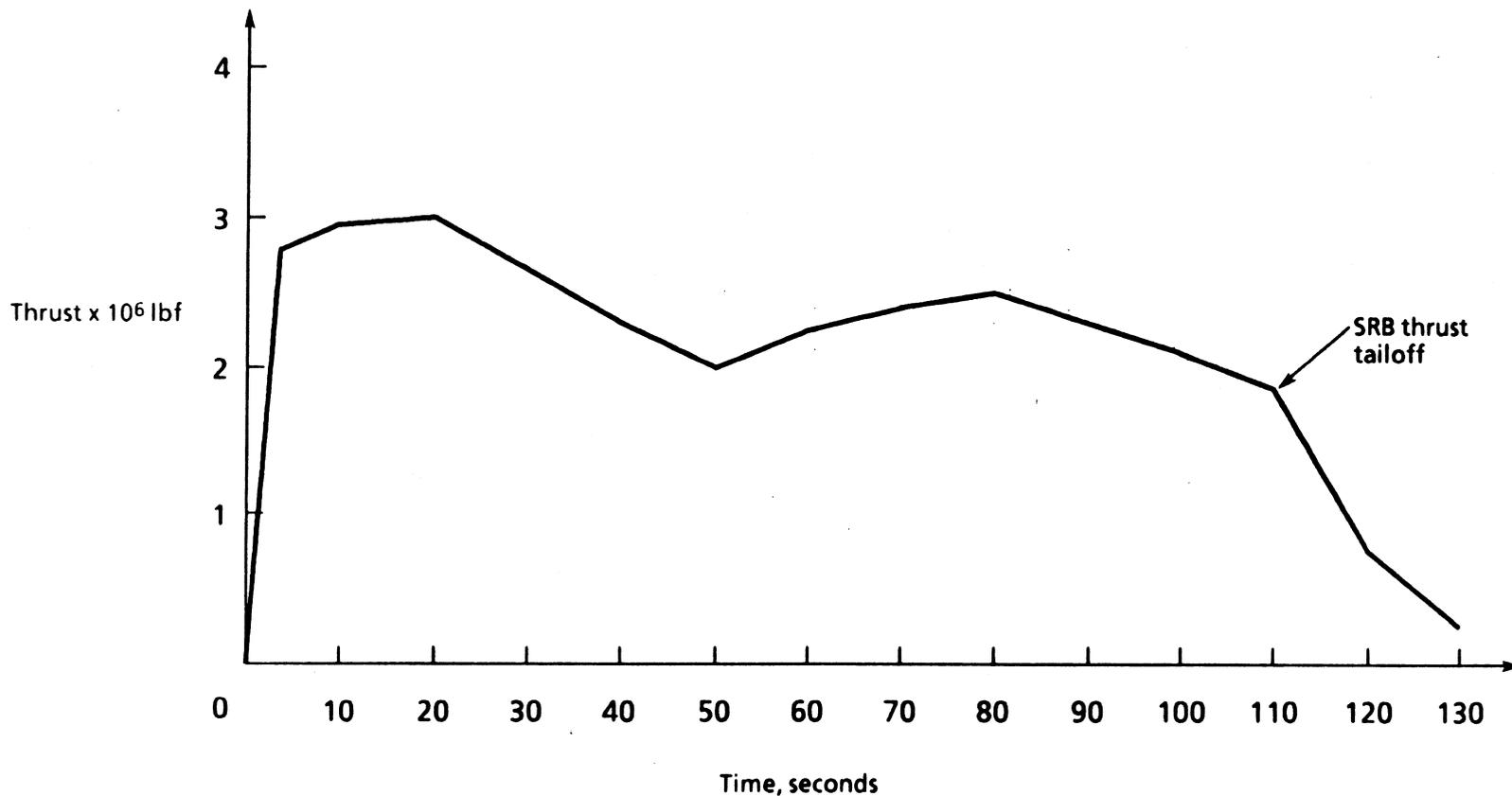
These gains are changed as the result of a mission phase change (such as first to second stage), an engine failure, or dual hydraulic system failures preventing SSME movement. As shown in table 5-1, the gains are lower in first stage than second stage since the SRBs have most of the control authority. In second stage, the gains are increased since the main engines are the only effectors providing vehicle control.

Table 5-1.- Roll-mixing gains.

Roll-mixing gains	Nominal first stage gains (L/O to 2 minutes MET)	Nominal second-stage gains (SRB PC 380 or 2 minutes MET to zero thrust)	Single SSME fail gains (SRB tailoff to zero thrust)			Dual SSME fail/SERC gains (SRB tailoff to zero thrust)
			Failed SSME/TVC*			
			Center	Left	Right	
A	0	0	0	2.5	2.5	0
B	0.75	0.75	0.75	2.0	2.0	0.35
C	1	2.5	2.5	2.5	2.5	0.35
D	0	0	0	2.0	2.0	-0.35

*Should one hydraulic system failure clear, these gains will switch back to the appropriate non-failure state.

If an SSME fails prior to SRB tailoff ($\approx 1:47$ MET), no special mixing is required for roll control because of SRB control authority due to thrust (≈ 2.5 million pounds each). Figure 5-33 shows an SRB thrust curve. If an SSME fails after tailoff, the gains are increased to provide the control authority necessary. If two SSMEs fail or the crew selects single-engine roll control ((SERC), via CRT display) after SRB tailoff, the gains are decreased in order to allow for roll control from the RCS jets.

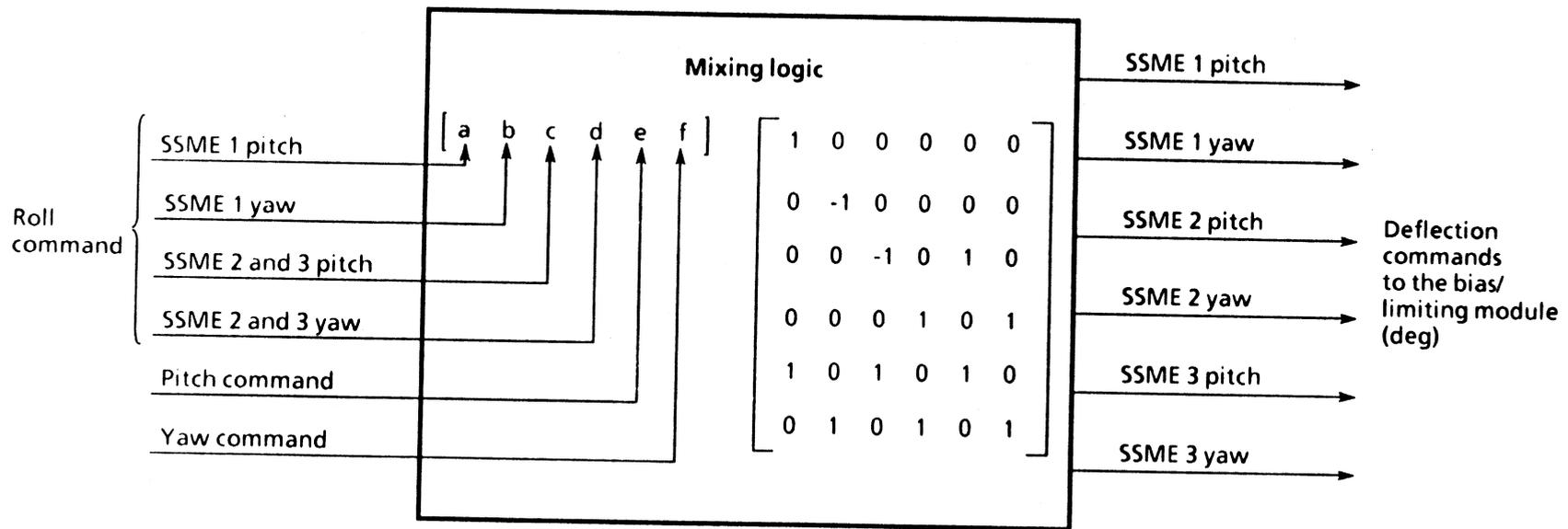


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Figure 5-33.- Typical SRB thrust curve.

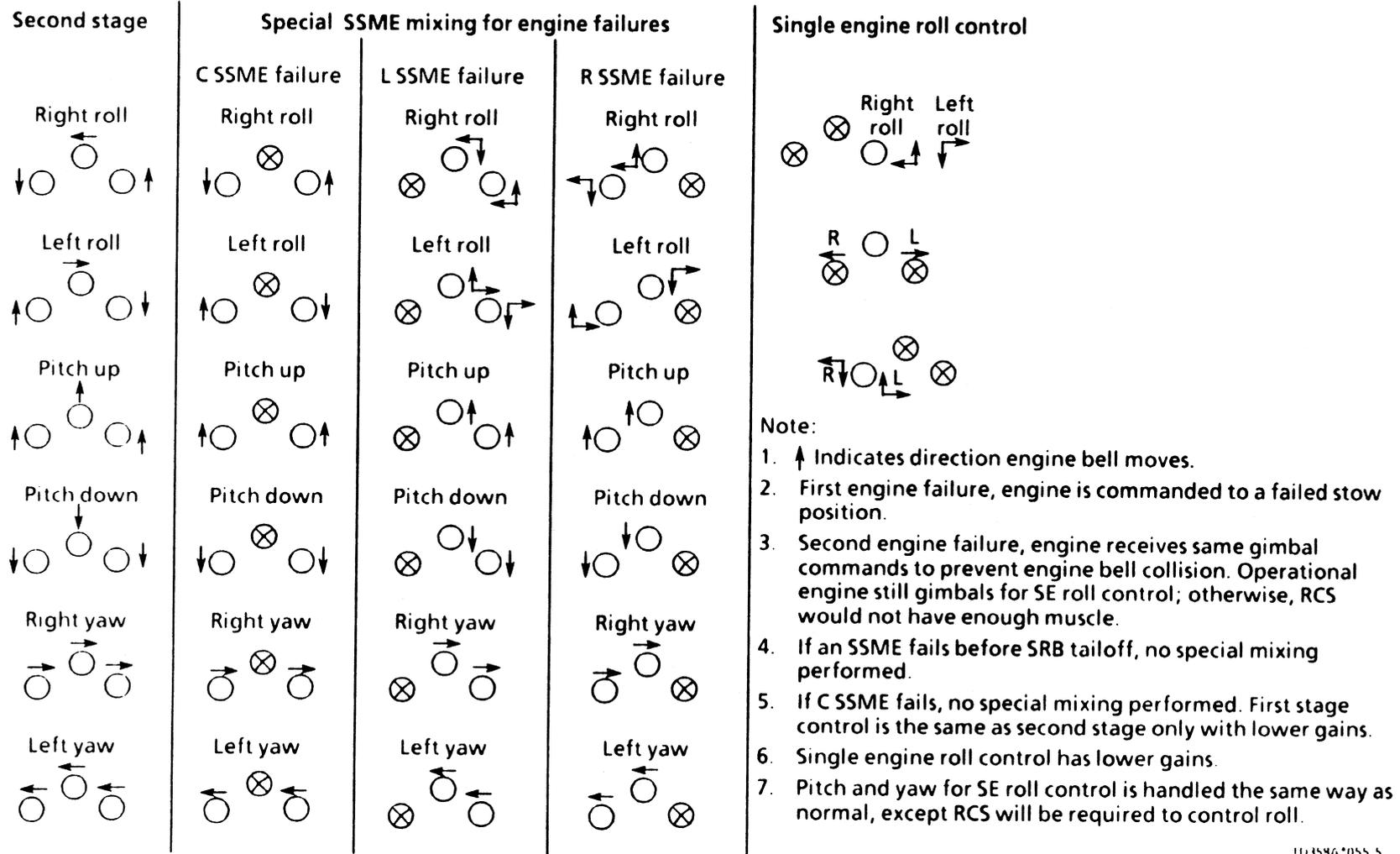
Mixing Logic

The mixing logic converts SSME bell pitch and yaw errors into commands for roll, pitch, and yaw maneuvers. Refer to figure 5-34; a matrix within this logic performs this task. The matrix relates the pitch and yaw errors to the main engine bell deflection commands necessary to accomplish a given maneuver. The matrix sign notations are established such that when the matrix is used, deflection commands are generated to achieve engine bell movements as shown in figure 5-35.



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Figure 5-34. - Mixing logic.



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Figure 5-35.- Engine bell dynamics resulting from mixing logic.

Before we leave this logic, a discussion of the MPS TVC command subsystem operating program (MPS TVC CMD SOP) is required. The MPS TVC CMD SOP is downstream of the MPS command processor. This is a software module that compensates the commands to the SSME pitch and yaw actuators. Compensation is necessary because of the physical orientation of the pitch actuators. Figure 5-36

illustrates this. Notice the left SSME pitch actuator is located 180° opposite the others, thus causing a sign convention problem. The MPS TVC CMD SOP multiplies the center and right (that's right, the center and right) SSME pitch actuator deflection commands by negative scale factors to change their sign convention to achieve the required maneuver.

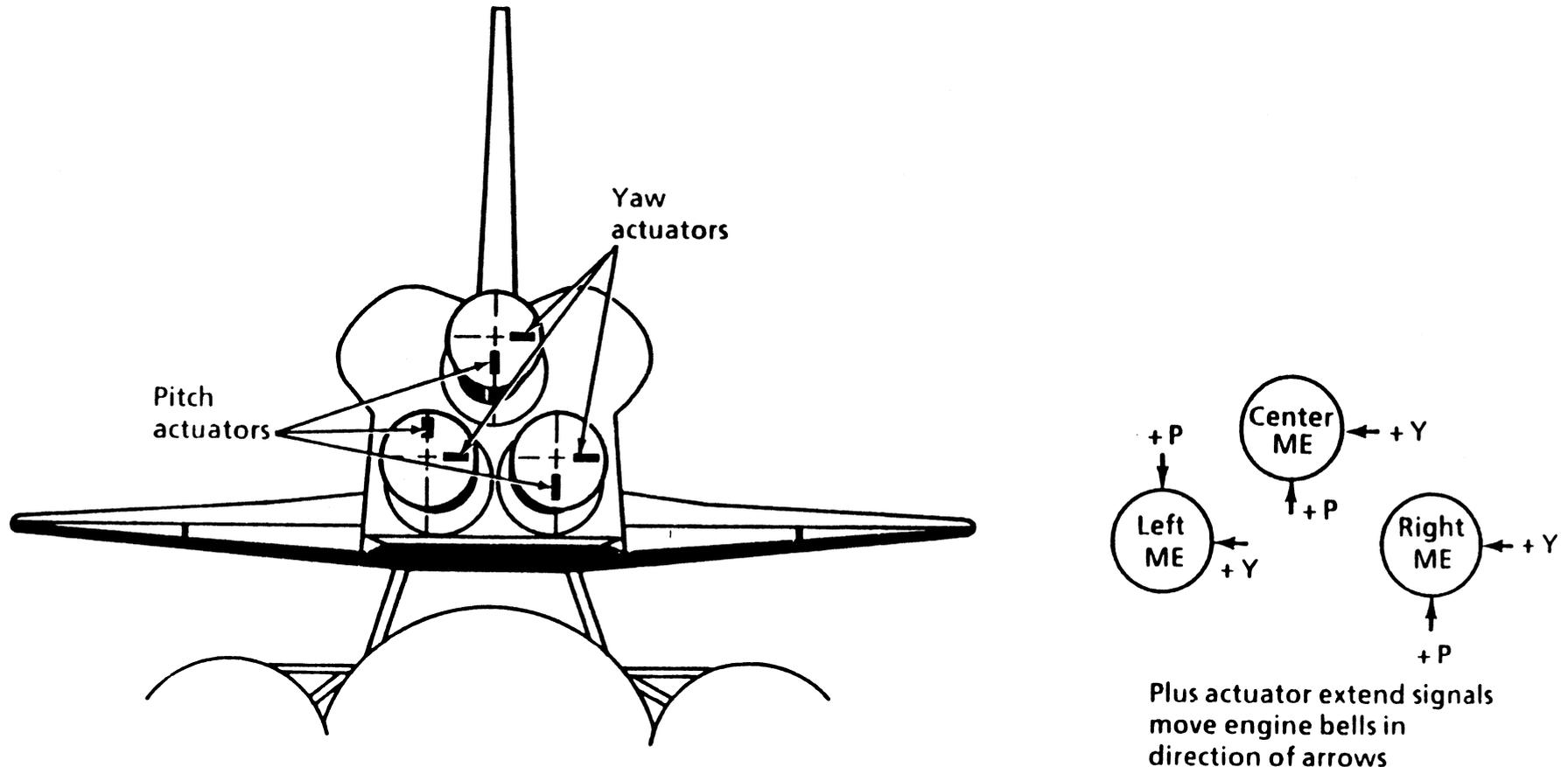
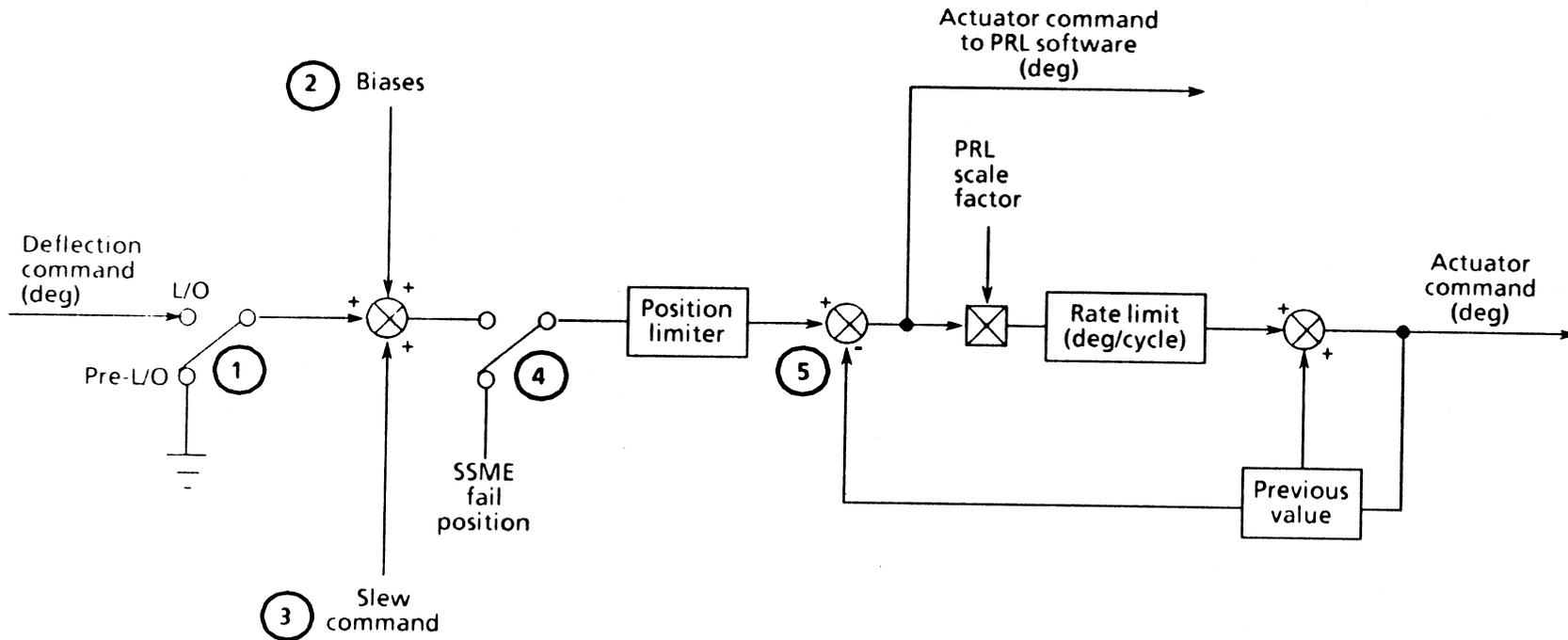


Figure 5-36.- SSME actuator orientation.

BiasLimiting Module

This module provides bias additions and rate limiting of each of the six main engine deflection commands. Figure 5-37 shows the pitch command channel for the center SSME; the other channels are handled in the same manner.



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Figure 5-37.- Command biasing and limiting.

Switch ① is a software switch, as are all the other switches. This switch is closed at lift-off, which allows the SSME bell deflection commands to get into this module. Pre-lift-off, this switch is left open to prevent "spurious" commands from reaching the engine bells before lift-off. Item ② is the biases that are added pre- and post-lift-off. The biases that are added pre-lift-off are as follows:

- Gravity position bias is used to "command" the SSMEs in pitch and yaw to the position the main engine bells hang to due to gravity before the auxiliary power unit (APUs) are brought on line and hydraulic system pressure is available. This prevents force fights within the ATVC servovalves when the APUs are brought on line.
- MM 101 initialization bias is used to carry over the bell position values from GNC OPS 9 for initialization in OPS 1.
- Start position bias adds a bias to drive the SSMEs to the pre-lift-off start position.

The post-lift-off biases are used in second stage to parallel or unparallel the null axes in pitch and yaw, provided there have been no engine failures. Note in figure 5-30 that the yaw null positions for the left and right SSMEs are offset from a line parallel to the body X-axis by $\pm 3.5^\circ$. To aid in thrust performance, a bias of 3.5° is added from the time the main engines are put in launch position until the MECO command. If, however, the left or right SSME fails or one APU fails, the bias is not added in second stage so that total control authority is available. This regime is really between SRB Pc less than 380 psi (I-load) and MECO. The bias is removed at the MECO command.

The pitch axes are not parallel to the body X-axis; they are canted up as shown in figure 5-30. Canting is necessary to fit the engines within the orbiter aft structure and provides a statically stable vehicle by pointing the thrust vector through the c.g. As a result, a small amount of thrust performance is lost due to this angle. For this reason, pitch paralleling is performed between SRB SEP + 12 seconds and an I-loaded vehicle mass close to MECO to regain some of this lost performance. Pitch paralleling adds a $+3^\circ$ bias for the center SSME and a -3° bias for the left and right SSMEs to parallel their net thrust. Once again, if an SSME or APU fails, pitch paralleling is not performed, in order for total control authority to be available. These biases are removed prior to MECO (when the TVC retrim guidance subtask sends the retrim discrete) to prevent engine bell contact near MECO when engine bell dynamics are greatest. For control authority reasons, if two SSMEs fail or the crew selects SERC, all biases will be zeroed. However, if SERC is selected, all biases will be latched at zero, regardless of subsequent deselection of SERC.

Item ③ is a slew command which is used prior to lift-off to test the ATVC actuators. This test is performed at T -3:25 for the SSME actuators.

Software switch ④ is "thrown" when an SSME fails. This sends a nozzle position command to the failed engine, preventing engine bell collision with the other operating engines. If a second engine fails, it is not locked into a nozzle fail position; instead, it will receive the same commands as the remaining engine. If two SSMEs fail simultaneously, the lowest numbered engine is placed in the failed position. Nozzle fail positions are $(-3, 0)$ for the center SSME, $(3, 3)$ for the left SSME and $(3, -3)$ for the right SSME.

From here, the main engine deflection commands pass through an engine bell software position limiter. These software limits really are not necessary since the

commands do not exceed the maximum size which can be used by the multiplexer/demultiplexers (MDMs). The software limits are $\pm 11^\circ$ in pitch and $\pm 9^\circ$ in yaw; the hardware limits are $\pm 10.5^\circ$ in pitch and $\pm 8.5^\circ$ in yaw.

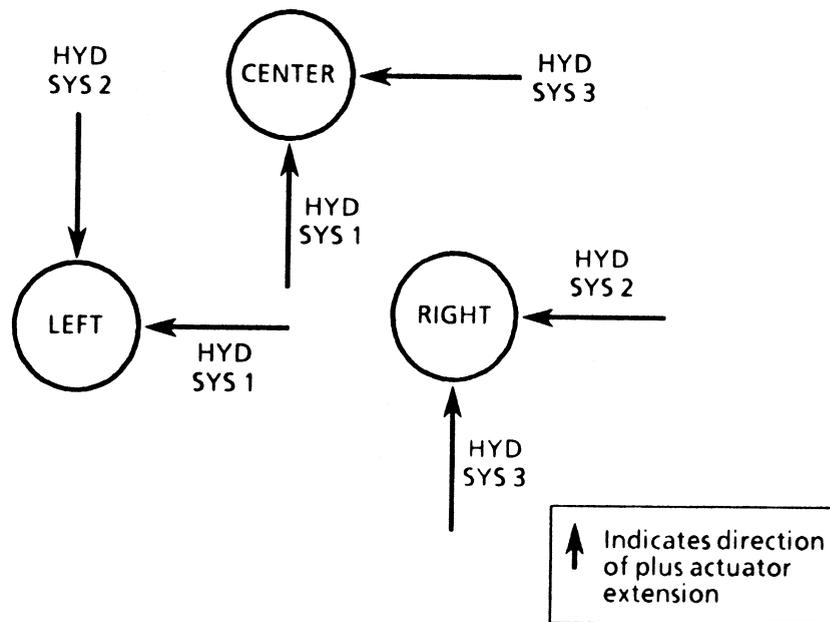
From here, the main engine deflection commands pass into item ③ logic. This logic limits the rate of change of the command. An example is necessary for the discussion of this logic. Assume the previous command was $+6^\circ$. A second later, the command is -4° (for simplicity, 1 Hz cycle time is assumed; however, the software is actually cycled at 25 Hz). As shown in figure 5-37, at the first summing joint we have $-4 - 6 = -10^\circ$. Next, it passes through a multiplier for a scale factor for priority rate limiting (PRL). There is no PRL in first stage; therefore, this scale factor is set to a value of 1. Then the rate limit is applied. The limits depend on the number of main engines and hydraulic systems failed. Nominally, the limit is 7 deg/sec. Since the -10° exceeds the rate limit, only a -7° is passed through the limiter. Finally, the signal is summed with the past value to generate the new command. The new signal is -1° deflection, not the total -4° command since it would exceed limits. The slack in the command would be picked up in the next second. As was mentioned earlier, the rate limits depend on failures. The limits for first stage are

- 7 deg/sec No failures.
- 5 deg/sec Single hydraulic system failure.
- 4 deg/sec Two hydraulic systems failed.
- 2 deg/sec If an SSME is failed, only on the failed engine. A second failed engine is not limited.

In second-stage flight, the rate change limit is fixed at 10 deg/sec and further limiting is accomplished by PRL.

PRL

As soon as second stage begins, at the SRB SEP command, PRL is available. PRL is rate limiting placed on engine bell deflection commands to scale them down proportionally to give priority to a larger command. This is necessary when an engine bell rate command exceeds a hydraulic system engine bell rate capability, which can happen anytime high rates are requested or any hydraulic system fails. There are three hydraulic systems. All the demands (commands) on a given hydraulic system are summed. If this sum exceeds an I-loaded rate capability, a scale factor is formed and multiplies all the demands on that hydraulic system to lower the total demand. Figure 5-37 shows where this scale factor comes in. Figure 5-38 illustrates which hydraulic systems interface with the SSME TVC actuators.



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Figure 5-38.—SSME actuator/hydraulic interface.

For example, the actuators that are driven by hydraulic system 1 when there are no hydraulic system failures are

Pitch actuator on ME1 (center) deflection given by P1

Yaw actuator on ME2 (left) deflection given by Y2

The demand of the actuator is a rate. Therefore, P1 is the center SSME pitch rate and Y2 is the left SSME yaw rate. Then |P1| and |Y2| are the demands on hydraulic system 1. If, for example, the DAP has commanded 14 deg/sec from the center SSME pitch actuator and 10 deg/sec from the left SSME yaw actuator, the total system demand is therefore

$$|P1| + |Y2| = \text{total demand}$$

or

$$14 + 10 = 24 \text{ deg/sec}$$

Under nominal conditions the I-load system capability is 18 deg/sec. Since 18 is less than 24, the commands must be scaled down.

If the engine bell deflection command is greater than the system capability, the actuator command is

$$\text{New command} = \frac{\text{System capability}}{\text{Total system demand}} \times \text{Actuator command}$$

or, in the example,

$$\text{new P1} = \frac{18}{24} \times 14 = 10.5 \text{ deg/sec}$$

$$\text{new Y2} = \frac{18}{24} \times 10 = 7.5 \text{ deg/sec}$$

$$\text{Total} = 18.0 \text{ deg/sec}$$

Now that the demands have been scaled appropriately, one can easily see that the priority has been placed on the center SSME pitch actuator since it is receiving a higher rate command.

When there are hydraulic system failures, the secondary hydraulic system will feed the affected actuators. This is accomplished by a switching valve. The MPS TVC 2102 training manual contains more information on this subject.

The I-loads used for PRL are provided in table 5-2. The rate capability decreases at the MECO command due to the demand on the hydraulic system for valve closures. Note that the rate capability is stepped up slightly when two hydraulic systems fail. This is necessary for adequate vehicle flight control since one engine is no longer gimbaling. If two hydraulic systems fail, the remaining APU is kicked up to 110 percent, which shortens APU lifetime but gets the job done. The center SSME pitch actuator is multiplied by 1.24. This pitch actuator has a larger rated load which is required to compensate for aeroloads on the center SSME since it has to drive the bell up into the airflow.

From the MPS command processor, the pitch and yaw engine bell deflection commands go to the MPS TVC command SOP where the commands are converted to voltages for use by the actuators.

Table 5-2. – I-loaded rate capacity.

Failures	Rate capability (deg/sec)	
	SRB SEP command to MECO command	MECO command
None	18.0	15.8
One hydraulic system failed	18.0	15.8
Two hydraulic systems failed*	18.6	16.5

*Remaining APU is run at 110 percent.

SRB Command Processor

The SRB command processor works in the same fashion as the MPS command processor. Figure 5-39 shows a functional block diagram of this processor. It processes the

body axis roll, pitch, and yaw loop errors from TVC, adds an I-loaded pitch trim schedule that is a function of V_{rel} plus SSME failures, and applies SRB mixing logic to generate SRB rock and tilt commands.

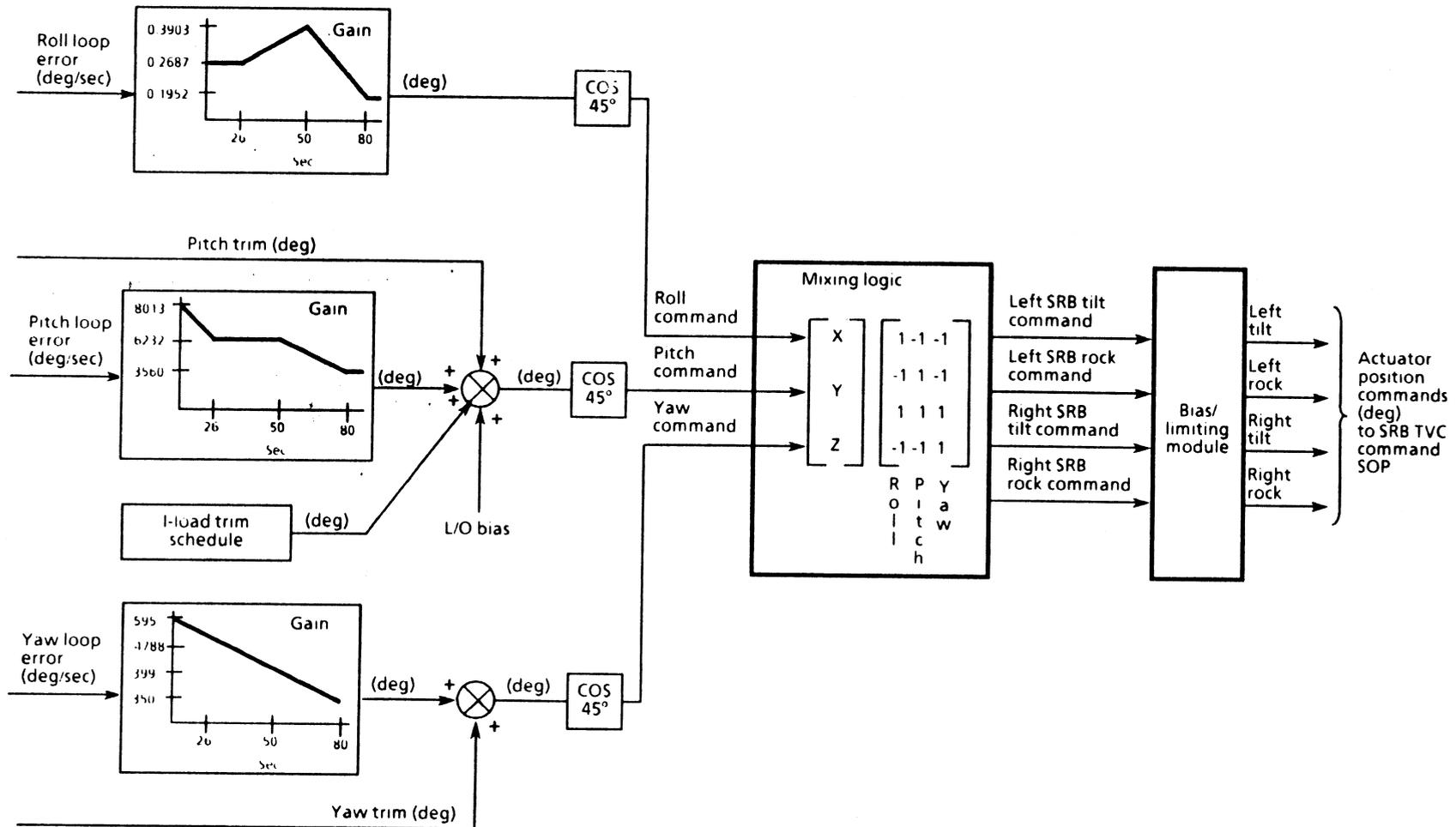


Figure 5-39.—SRB command processor.

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Reference figure 5-40 for the discussion of rock and tilt. This is a rear view, looking forward, of the SRB nozzles. Note that the SRB gimbal axes are at 45° angles to the orbiter body axes; therefore, the deflections are not pitch and yaw, but rock and tilt. The reason the SRB TVC actuators are mounted at 45° to the orbiter body axis is best explained by a right isosceles triangle as represented in figure 5-41.

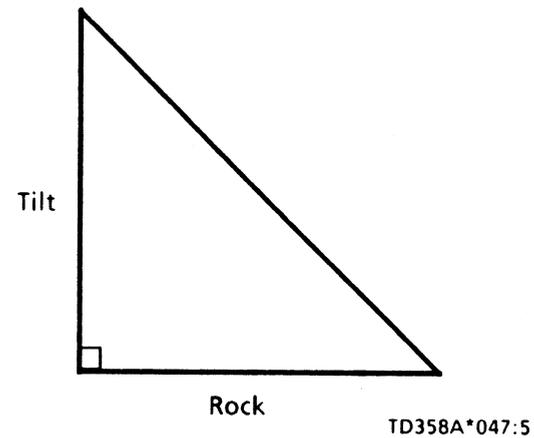


Figure 5-41.—Isosceles triangle.

If we let one leg be the tilt actuator and the other be the rock actuator, the hypotenuse is longer than either leg. That is the idea behind rock and tilt. By mounting the actuators at 45°, the SRB nozzles can be deflected further in the orbiter body pitch and yaw axes for better vehicle control. This also reduces the size of the actuators because bigger actuators are not required for the necessary nozzle deflection. Do not confuse rock and tilt with the vehicle motion dynamics. Rock and tilt are merely names given to the SRB actuators because of the 45° rotation. Therefore, to achieve vehicle roll, pitch, and yaw, both the rock and tilt actuators on each SRB must be extended and/or retracted.

As shown in figure 5-39, the SRB command processor is very similar to the MPS command processor; therefore, a detailed discussion will not be presented, but the differences will be pointed out.

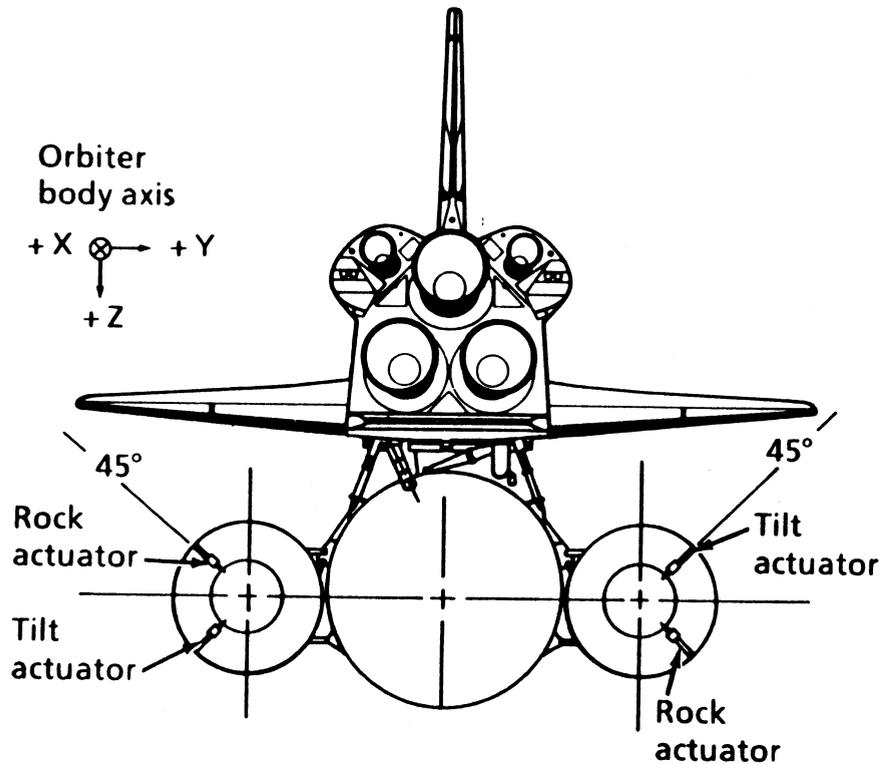


Figure 5-40.—SRB rock and tilt actuators.

As was the case in the MPS command processor, the body axis roll, pitch, and yaw TVC loop errors come into the processor and are multiplied by various rates and gains to develop SRB nozzle errors. Note that an I-loaded lift-off bias of 0.233° is added to the pitch channel. This is required for the vehicle to clear the holddown support during lift-off. This bias starts at lift-off and is ramped to zero in 2.12 seconds.

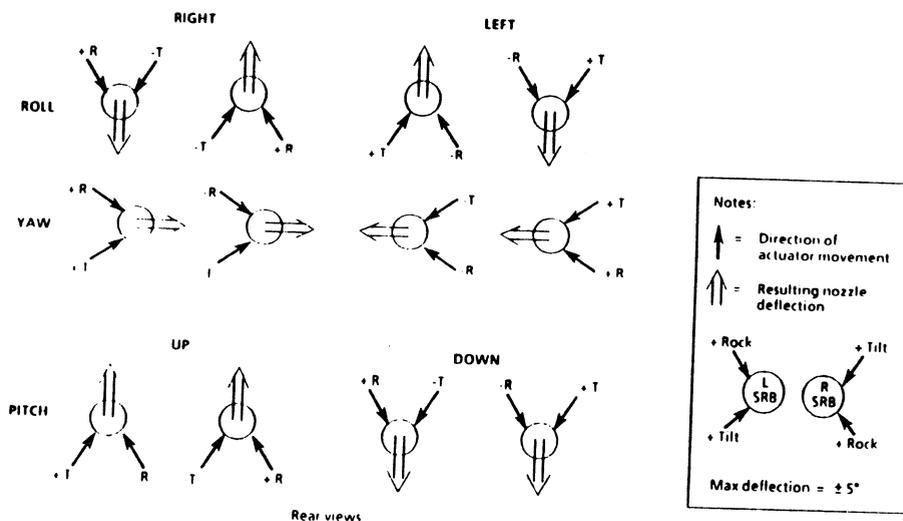
The next step is to factor in the cosine of 45° , which corrects for the gimbal axis rotation of 45° to come up with the SRB nozzle errors for roll, pitch, and yaw. From here the errors pass through the mixing logic, which is another matrix. This converts the roll, pitch, and yaw errors into rock and tilt commands to achieve a vehicle roll, pitch, or yaw maneuver. Figure 5-42 illustrates what the mixing logic does to achieve a given vehicle maneuver.

These deflections are used along with the SSME bell deflections to achieve the required maneuver. Once again, there is position biasing for pre-lift-off functions. The only bias is the slew command at T -21 seconds to check the actuators. Except for the holddown clearance bias, there are no post-lift-off biases as was the case in the MPS command processor. Next, the nozzle deflection commands go into position and rate limiting. This is performed exactly the same as in first stage for the SSMEs. The rate limit on the command signal is ± 7 deg/sec. The software deflection limits are as follows:

$\pm 6.364^\circ$ nominal

$\pm 2.83^\circ$ when $P_c < 110$ psi to prevent SRB recontact during the separation sequence

The hardware limits are $\pm 5^\circ$. Also, there is no PRL for the SRBs.

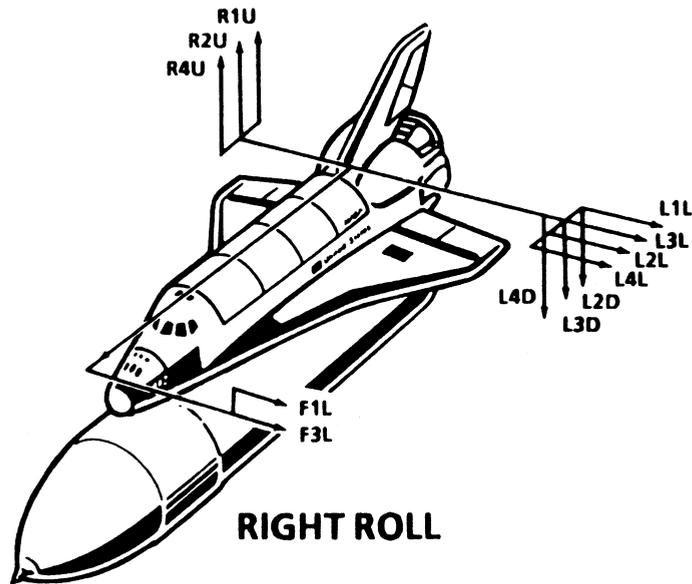


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Figure 5-42.- Rock and tilt mixing for roll, pitch, and yaw.

RCS Command Processor

The RCS is used to provide roll control during powered flight when the reconfiguration logic "knows" that two or three SSMEs have failed in first stage, two SSMEs have failed in second stage, or the crew has selected SERC. The RCS is required when only one main engine is operating because roll control cannot be achieved by differential pitch actuator deflections. The remaining engine will gimbal to aid the RCS jets in roll control. Without this gimbaling the jets would not have enough control authority.



The TVC roll error, in degrees, comes in from the MPS command processor to a software switch. This switch is grounded out until the RCS roll control enable discrete is set, which happens when the second SSME is confirmed failed or SERC has been selected by the crew on a CRT display. The signal next passes through a hysteresis loop with a deadband. If the error is greater than or equal to the deadband of 0.25° , the RCS jets are commanded on in order to null ($\text{error} \leq 0.2^\circ$) the roll error. Figure 5-43 illustrates the jets that fire for a right and left roll, respectively. The forward side-firing jets fire as well as the aft. Because the vehicle c.g. is below their thrust location, the forward jets cause a rolling moment.

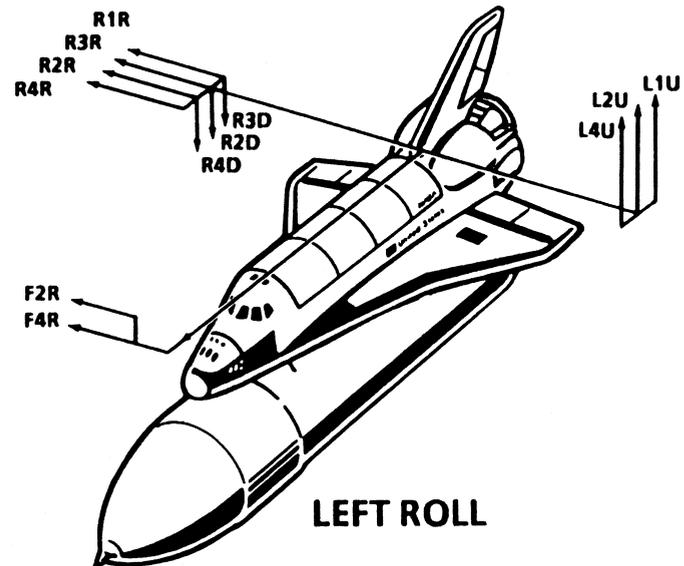


Figure 5-43.— RCS jets utilized for single-engine roll control.

If the vehicle is in the middle of an abort dump and a second SSME fails or SERC is selected, the RCS jet portion of the OMS propellant dump will be terminated (if the interconnect has been enabled) or never started. In this case, the jets will be commanded on as necessary for roll control. During single-engine roll control, the RCS jets are interconnected to the OMS propellant supply. The RCS jets are controlled by the ascent DAP until zero thrust from the main engines.

Aerosurface Control

Aerosurface control controls the commands to all aerosurfaces during ascent. This includes the body flap, rudder, speedbrake, and elevons. At T-3:55, this logic allows the aerosurfaces to be deflected as part of the aerosurface slew check to check their actuators. After the aerosurface slew is complete at T-2:50, the body flap and

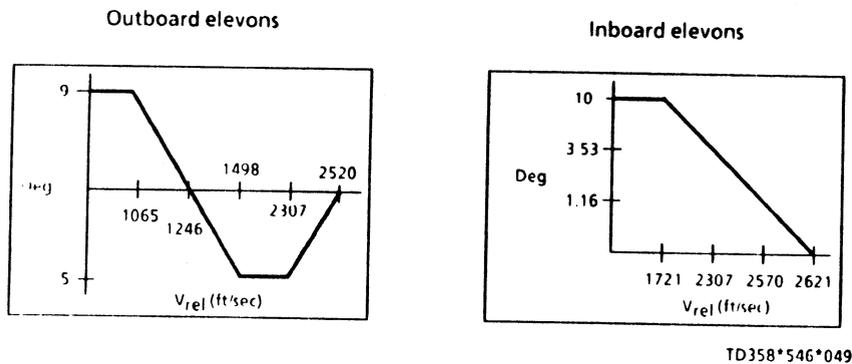


Figure 5-44.- Elevon schedules.

rudder are commanded to 0° and held in place during ascent; the speedbrake is commanded to a -9.9° to ensure that it is fully closed during ascent. The elevons are driven dynamically during the load relief region to reduce hinge moments (elevon loads). Since all surfaces but the elevons are held in place during ascent, only the elevons will be discussed.

Beginning at lift-off and continuing through the end of load relief, the elevons are driven by an I-loaded elevon schedule that is a function of V_{rel} . Figure 5-44 shows an example of an elevon schedule. This schedule is required in order to relieve elevon hinge moments during load relief. The load relief schedule is operative between lift-off plus 2.5 seconds and an I-loaded MET of approximately 100 seconds. After this MET has expired, a software switch is thrown which commands and holds the elevons in place at 0° deflection for the rest of the ascent. The elevons, when they are moved, are slaved together. This means the left and right inboards are driven together, and the left and right outboards are driven together. The inboards are driven independently of the outboards and vice versa, as illustrated in figure 5-45.

- The inboard panels deflect together.
- The outboard panels deflect together.

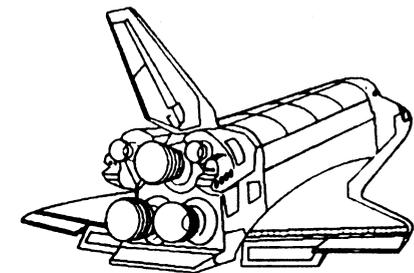
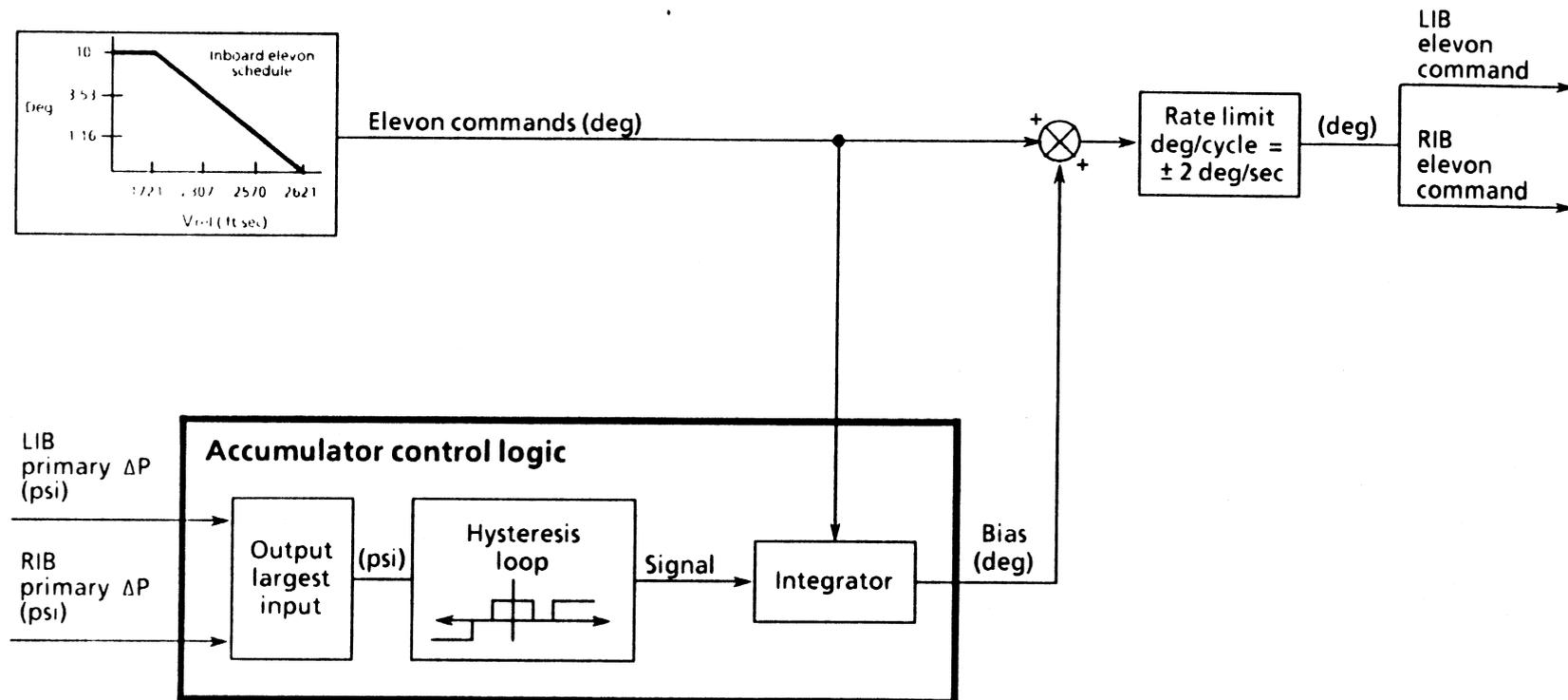


Figure 5-45.- Elevon load relief operations.

Figure 5-46 shows a functional view of the elevon logic. The same logic is performed for the inboards and outboards separately. Note that primary ΔP s are used as feedbacks to the elevon schedule commands.

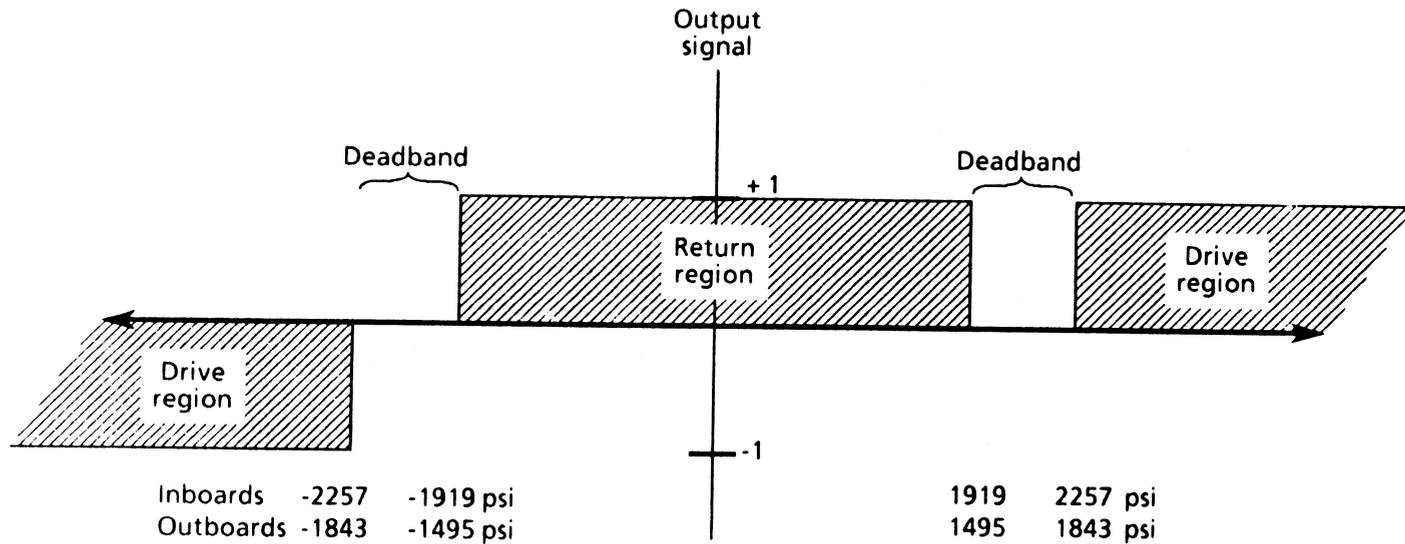


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Figure 5-46.- Elevon load relief logic.

Primary ΔP is a direct indicator of elevon loads. Therefore, the ΔP is used to reduce the hinge moments in the event the elevon schedules are not accurate. The largest value of the left and right inboard/outboard ΔP s goes into the accumulator control logic where it passes through a hysteresis loop with a deadband. The accumulator control logic is nothing more than an integrator. If the ΔP s grow too large, the scheduled elevon command is biased to relieve the hinge moment. Figure 5-47 shows an example of an elevon hysteresis loop.

For example, assume that the primary ΔP on the inboard elevon is large enough to be out in the drive region of ≥ 2257 psi. The drive and return regions are I-loads determined preflight. Therefore, a bias is added to the scheduled command to reduce the ΔP . This is indicated by a step increase to ± 1 . As long as the ΔP remains greater than or equal to 2257 psi, the bias is integrated over time. When the ΔP falls below 2257 psi, the integration stops. When the ΔP gets within the return region, the bias begins to be subtracted from the scheduled command. The same applies to the outboard elevon ΔP s.



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Figure 5-47.— Elevon hysteresis loop.

Manual Control Steering

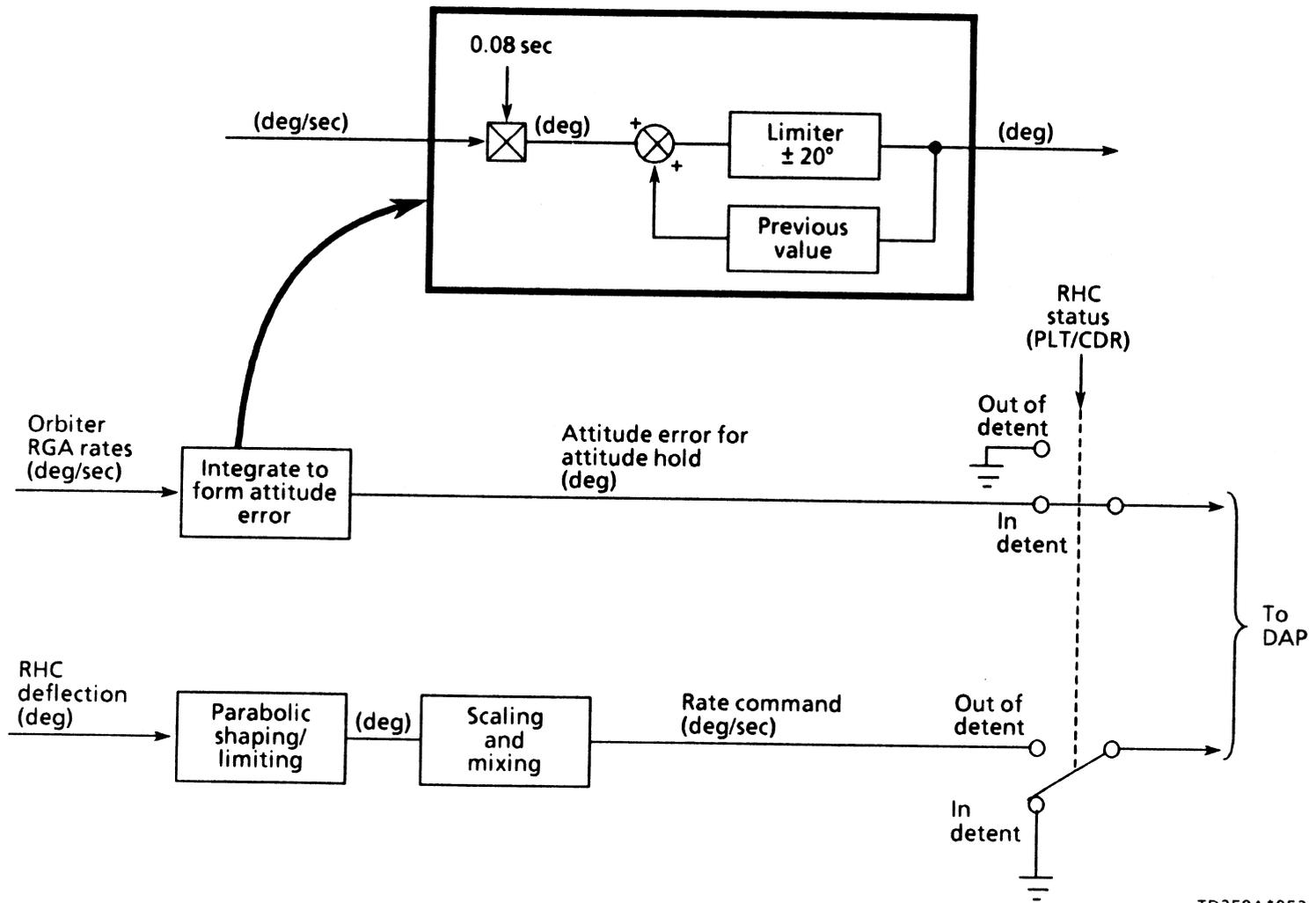
The capability exists to manually control the vehicle thrust vector during first- and second-stage flight (see figure 5-3). This is performed by substituting the crew's inputs from the RHC for the automatic commands from guidance. The switch shown in this figure accomplishes this function, which is controlled by the AUTO/CSS roll, pitch, and yaw PBIs. Everything downstream remains active, of course, to process the crew's inputs from the RHCs.

Manual thrust vector control (MTVC) is available beginning at lift-off. There is no "hot stick" capability. This means that any one of the CSS PBIs on panel F2 or F4 must be depressed before MTVC is available by way of the RHC. Once MTVC is activated, the vehicle is in a rate command/attitude hold mode in all affected axes. In order to give control back to the automatic system, any of the AUTO PBIs must be depressed.

Assuming MTVC has been selected, when the RHC is in detent, the vehicle is in attitude hold. This means that the DAP will hold the vehicle in the attitude it had when the RHC was put in detent. A rate command equal to zero replaces the rate command from automatic steering within G/C steer. When the RHC is in detent, the attitude error for the DAP is computed by integrating the orbiter RGA measured rates and limiting the results. The rate command, of course, is zeroed. There are maximum limits on vehicle rates that can exist when attitude hold is requested by placing the RHC in detent. Attitude hold will not be initiated until the rates are back within limits. The limits are as follows:

- 10 deg/sec roll rate
- 3 deg/sec pitch and yaw rate
- 20° roll, pitch, and yaw attitude error

As soon as the RHC is removed from detent, a rate command, proportional to the amount of deflection, replaces the rate command previously generated. The attitude error is zeroed as indicated by the software switch in figure 5-48.



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Figure 5-48.- Manual steering - rate command/attitude hold.

Figure 5-49 shows how the RHC inputs are handled. The RHC deflection (degrees) is input to a parabolic shaper. This is designed on crew preference as to how sensitive the stick is to control inputs. This aids in control stick "feel" with small inputs; small outputs result and get larger as control inputs increase.

The signal next goes through a rate command gain to change the RHC deflection command to a rate command. Notice once again, that the yaw is crossfed into roll for stable control of the roll axis. This is necessary to reduce pilot workload because the orbiter/stack c.g. is located below the crew.

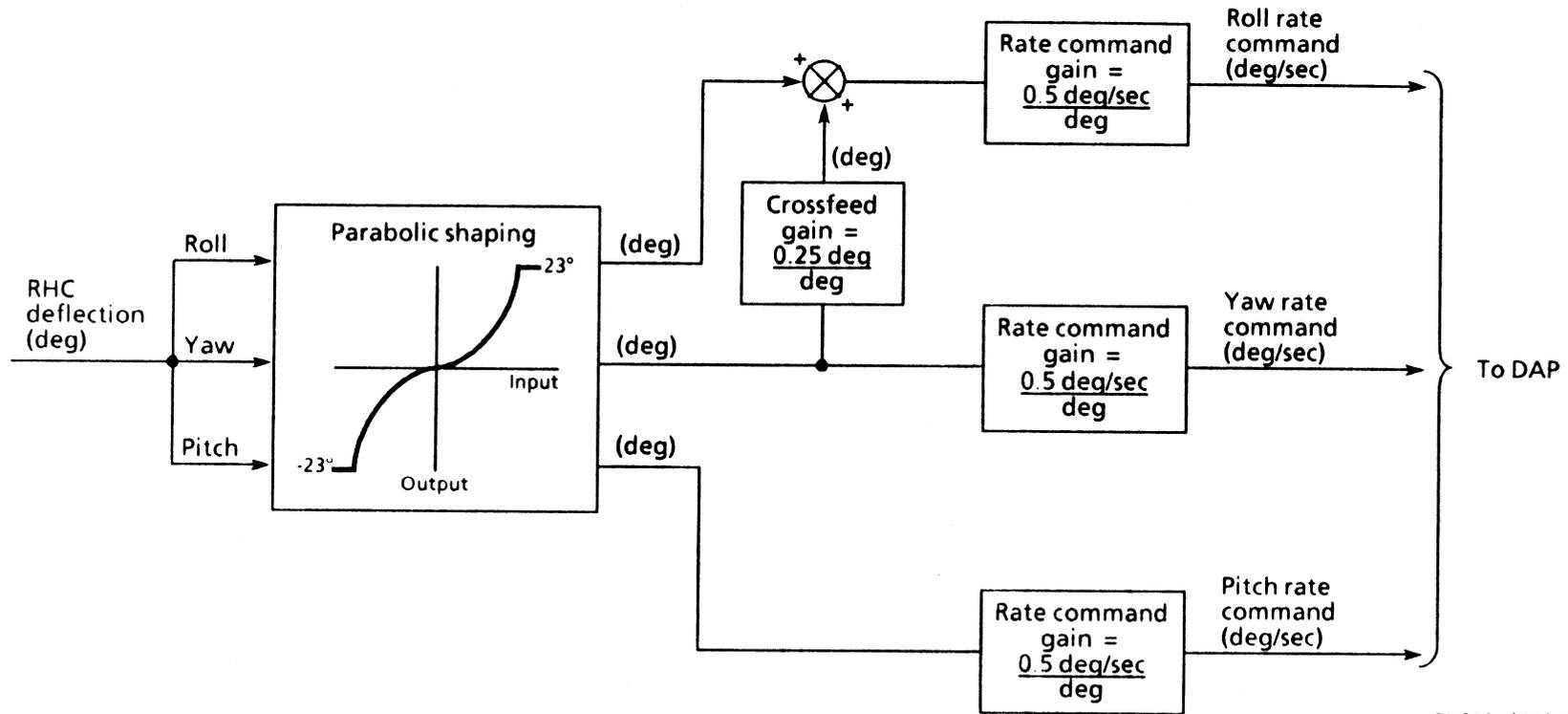


Figure 5-49.- RHC processing.

Questions

1. What is the proper signal flow sequence through the flight control system?
 - A. Command processor → attitude and rate errors → ATVC → effectors
 - B. ATVC → command processor → attitude and rate errors → effectors
 - C. Attitude information and guidance commands → G/C steer → DAP → effectors
 - D. Attitude information and guidance commands → DAP → G/C steer → effectors
2. What is the purpose of G/C steer?
 - A. To develop rate commands and attitude errors for the DAP to steer out
 - B. To develop actuator position commands for the DAP to steer out
 - C. To generate the DAP schedules, gains, and trims necessary for controllability
 - D. To generate an attitude command for the DAP to steer to
3. Which of the following is NOT an element of the DAP software module?
 - A. Reconfiguration logic
 - B. Attitude processor
 - C. Ascent thrust vector control
 - D. MPS command processor
4. What is the purpose of the ATVC software?
 - A. To develop rate commands and attitude errors for the command processors to steer out
 - B. To develop actuator position commands for the command processors to steer out
 - C. To generate the DAP schedules, gains, and trims necessary for controllability
 - D. To generate body axis attitude rate errors (loop errors) by utilizing the G/C steer outputs and sensor feedback
5. What does the MPS command processor do?
 - A. Commands the main engine throttle setting
 - B. Converts the loop error signal from ATVC into main engine actuator position commands
 - C. Generates FCS control mode indicators as a result of main engine failures
 - D. All of the above
6. Why are rate gyro assemblies necessary for flight control feedback?
 - A. Because acceleration feedback is required to relieve structural loading during maximum dynamic pressure
 - B. Because they accurately measure wind speed and direction to relieve structural loading during maximum dynamic pressure
 - C. Because navigation data is processed too slowly for flight control to use without augmentation
 - D. To propagate relatively slow AA data with faster RGA data

7. What does a gain do?
- Multiplies a signal to either increase or decrease its magnitude
 - Adds two signals together to convert their sign convention
 - Filters unwanted signals to prevent vehicle perturbations
 - Adds two signals together to increase their signal magnitude
8. Why are accelerometer assemblies necessary for flight control feedback?
- Because acceleration feedback is required to relieve structural loading during maximum dynamic pressure
 - Because rate feedback is required to relieve structural loading during maximum dynamic pressure
 - Because acceleration feedback is required late in second stage for 3g throttling
 - To propagate relatively slow RGA data with faster AA data
9. How does first-stage flight control relieve structural loading during maximum dynamic pressure?
- By varying gains which increase sensitivity to acceleration error and decrease sensitivity to attitude error
 - By using accelerometer feedback to reduce the load factor error
 - By using elevon primary delta pressure feedback to reduce hinge moments
 - All of the above
10. How does the roll channel relieve structural loading during maximum dynamic pressure?
- By adjusting the elevon schedule as a function of the roll attitude error
 - By rolling into the wind and yawing the thrust vector back into the desired trajectory plane
 - By rolling the thrust vector back in plane as a result of normal acceleration feedback
 - By rolling the thrust vector back in plane necessitated by yawing into the wind
11. What is the purpose of the auto trim integrator?
- To null attitude errors caused by SSME and SRB nozzle gimballing
 - To null attitude errors caused by thrust misalignment
 - To null attitude errors caused by wind shears
 - A and B above
 - B and C above
12. Which of the following is a function of the SRB command processor?
- Pitch and yaw paralleling
 - Priority rate limiting
 - Actuator slew check
 - Rock and tilt paralleling

13. How are the main engine pitch and yaw axes paralleled?
- The MPS command processor adds biases to the engine deflection commands.
 - The auto trim integrator adds biases to the engine deflection commands.
 - The SSME TVC retrim subtask multiplies scale factors to the engine deflection commands.
 - The mixing logic multiplies scale factors to achieve the appropriate combination of pitch and yaw actuator commands.
14. What is the purpose of the MPS command processor mixing logic?
- To mix SSME and SRB actuator deflection commands together to achieve first-stage vehicle rotational maneuvers
 - To generate a combination of pitch and/or yaw actuator commands to achieve vehicle rotational maneuvers
 - To provide the logic necessary to accommodate the pre-lift-off actuator slew checks
 - To provide the logic necessary to compensate for the physical orientation of the main engine pitch actuators
15. Which module commands a failed SSME to the engine fail stow position?
- SRB command processor
 - Reconfiguration logic
 - Ascent thrust vector control
 - MPS bias/limiting module
16. How are the pre-lift-off actuator slew checks performed?
- They are performed by the reconfiguration logic which incorporates a scale factor to the pre-lift-off biases at a specific time in the terminal countdown.
 - They are performed manually by ground support personnel at KSC through the launch processing system.
 - The bias/limiting modules within their respective command processors send slew commands at a specific time in the terminal countdown.
 - The flightcrew performs this function manually by deflecting the rotational hand controller in all three axes prior to lift-off.
17. What is the purpose of priority rate limiting?
- To add a bias to give priority to the SRB nozzle deflection commands until SRB thrust tailoff
 - To limit engine bell deflection commands proportionally to give priority to a larger command
 - To limit engine bell and SRB nozzle deflection commands to give priority to a larger command
 - To add an I-loaded rate limit to give priority to the center SSME actuator necessary to overcome aerodynamic loading
18. Why is the RCS command processor required?
- To allow for roll control when two main engines have failed
 - To maintain vehicle control after SRB separation
 - To allow for single-engine roll control when selected by the crew
 - A and C above

19. How do the elevons relieve structural loading during first stage?
- A. By varying gains which increase sensitivity to acceleration error and decrease sensitivity to attitude error
 - B. By using accelerometer feedback to reduce the load factor error
 - C. By adjusting an I-loaded position schedule with elevon primary delta pressure feedback
 - D. The elevons are not used for this purpose and are not moved during first-stage flight
20. How is manual flight control initiated?
- A. By depressing the CSS PBI to send a flag to G/C steer to replace automatic steering with manual steering from the RHC
 - B. By deflecting the RHC out of detent to send a flag to G/C steer to replace automatic steering with manual steering from the RHC
 - C. By parabolic shaping and limiting to generate a rate command for the DAP to use for thrust vector control
 - D. Through the use of an integrator to generate an attitude error for the DAP to use

Answers

1. C. G/C steer compares navigation and guidance inputs to develop an attitude error and rate command for the DAP. The DAP then develops a position command for the effectors (figures 5-1 and 5-8).
2. A. Rate commands and attitude errors are what the DAP uses to develop actuator position commands (page 5-3).
3. B. The attitude processor is a software module external to the digital autopilot software (figure 5-1).
4. D. The ATVC software closes the acceleration and rate loops within the outer attitude loop with sensor feedback (page 5-9 and figure 5-11).
5. B. The MPS command processor generates the signals necessary to move the main engine pitch and yaw actuators (figure 5-31).
6. C. IMU attitude data coming into the flight control loops is not being updated fast enough to maintain control; therefore, RGA data must be used to prevent an attitude "data gap" (page 5-2).
7. A. A gain is used to adjust FCS signals to achieve the desired flight control response. Gains are frequently used to change the signal units as well (page 5-13).
8. A. Acceleration error is an indirect indicator of external forces (winds) causing loading problems during maximum dynamic pressure; therefore, accelerometers are required to sense this (pages 5-12 and 5-18).
9. D. All these methods are employed to reduce their respective errors and relieve structural loading (pages 5-12, 5-18, and 5-51).
10. D. This question is more of a performance concern. The yaw channel flies out yaw acceleration error to reduce sideslip. When this occurs, the roll channel will roll the vehicle as necessary to place thrust vector back into the planned trajectory plane (page 5-27).
11. B. The autotrim integrator eliminates errors in the FCS caused by net thrust misalignment from vehicle c.g. (page 5-20).
12. C. Actuator paralleling and priority rate limiting are not performed with the SRB actuators. The actuator slew check is accomplished in the bias/limiting module of the SRB command processor (page 5-47).
13. A. Specifically, it is the bias/limiting module within the MPS command processor that performs this function (page 5-41).
14. B. This logic determines what combination of actuator movement is required for vehicle maneuvering (page 5-34).
15. D. The MPS bias/limiting module commands a failed SSME to the engine fail stow position (page 5-41).

16. C. The actuator slew checks occur automatically as a function of the countdown time (pages 5-41, 5-47, and 5-49):
- At T minus 3 minutes 55 seconds the aerosurface slew occurs.
 - At T minus 3 minutes 25 seconds the main engine slew occurs.
 - At T minus 21 seconds the SRB nozzle slew occurs.
17. B. Hydraulic system demand exceedances are prevented by giving priority to a larger main engine rate command (page 5-42).
18. D. The main engine actuators cannot provide adequate roll control when two engines have failed. The RCS command processor also allows the crew to select single-engine roll control after SRB thrust tailoff (pages 5-36, 5-48, and 5-49).
19. C. The elevon schedule is biased to relieve hinge moments whenever the primary delta pressure exceeds its hysteresis limits (page 5-51).
20. A. The CSS PBIs must be depressed before RHC commands will be used for flight control (page 5-52).

Section 6: Displays and Controls

Objectives

After reading this section, the student should be able to do the following:

- Explain how the crew can determine if single-engine roll control has been established
- Described how manual guidance is established
- Describe how manual throttles are established

Cockpit Panels

Overhead Panels 14, 15, and 16

Overhead panels (PO) 14, 15, and 16 are illustrated in figures 6-1, 6-2, and 6-3. All IMU and RGA power switches are contained in row A of these panels. As their names imply, these switches apply power to the orbiter RGAs and IMUs.

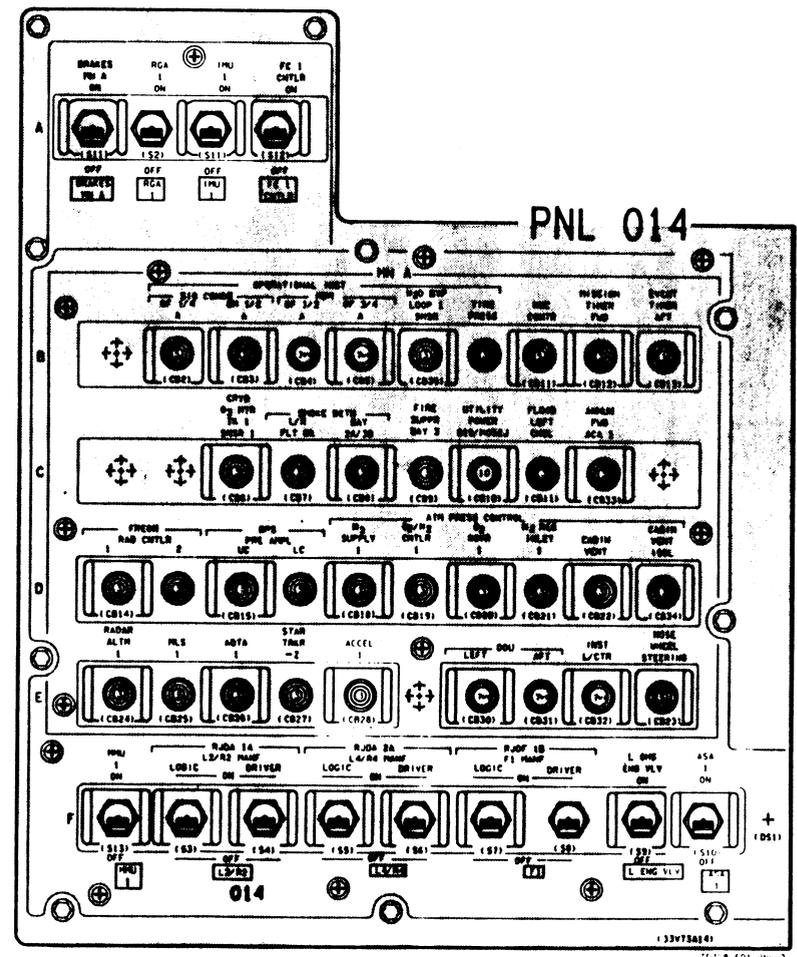
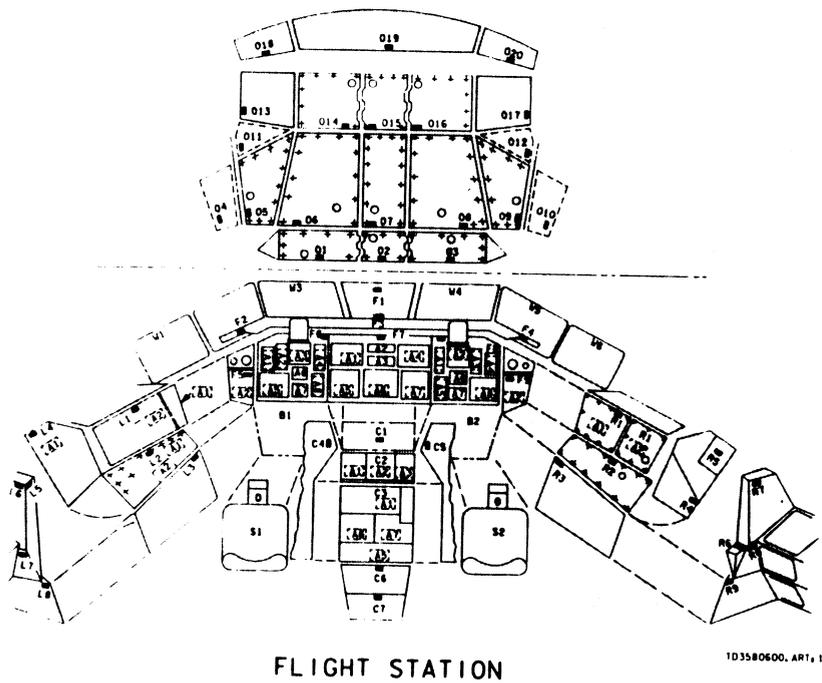


Figure 6-1.- Panel 014.

The orbiter RGAs are turned on prelaunch before the crew enters the vehicle. Once orbit (MM 201) is achieved, the RGAs are no longer required and are turned off to reduce power consumption and prevent unnecessary wear and tear.

The IMUs are turned on prelaunch and will remain on for the entire flight. Remember, the IMUs are used by Navigation to calculate state vector and attitude changes of the vehicle.

Since the SRBs are jettisoned, their RGAs are not required to be manually turned off; therefore, there are no crew-controlled switches for them.

On row E of PO 14 and PO 15 are the circuit breakers for AAs 1 and 2. These can be seen in figures 6-1 and 6-2. They act just like power switches and enable power to their respective AAs.

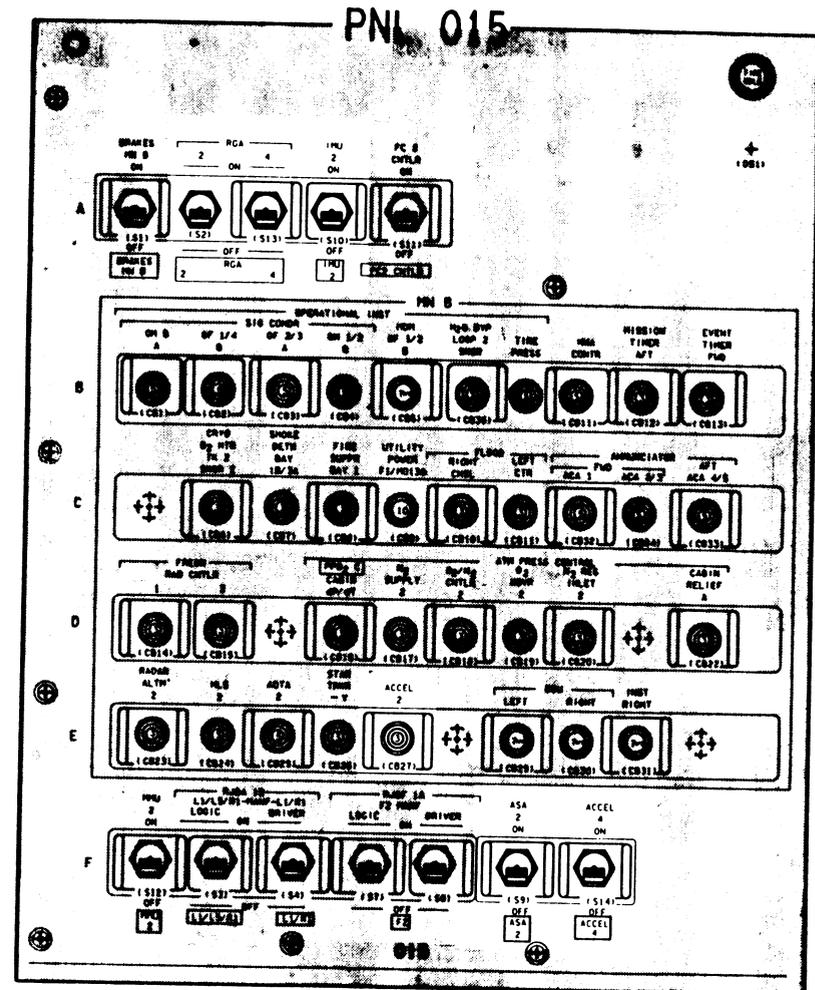


Figure 6-2.- Panel O15.

Row F of PO 15 and row F of PO 16 contain power switches for AAs 3 and 4. Prior to launch, all AA switches and circuit breakers are on to power the AAs. For more information on the AAs and RGAs, consult the Sensors 2102 Training Manual.

The aerosurface servoamplifier (ASA) power switches are also located on row F of POs 14, 15, and 16. These switches power up the ASA "black boxes" that enable the aerosurfaces to drive during load relief. The Effectors 2102 Training Manual discusses these switches in greater detail.

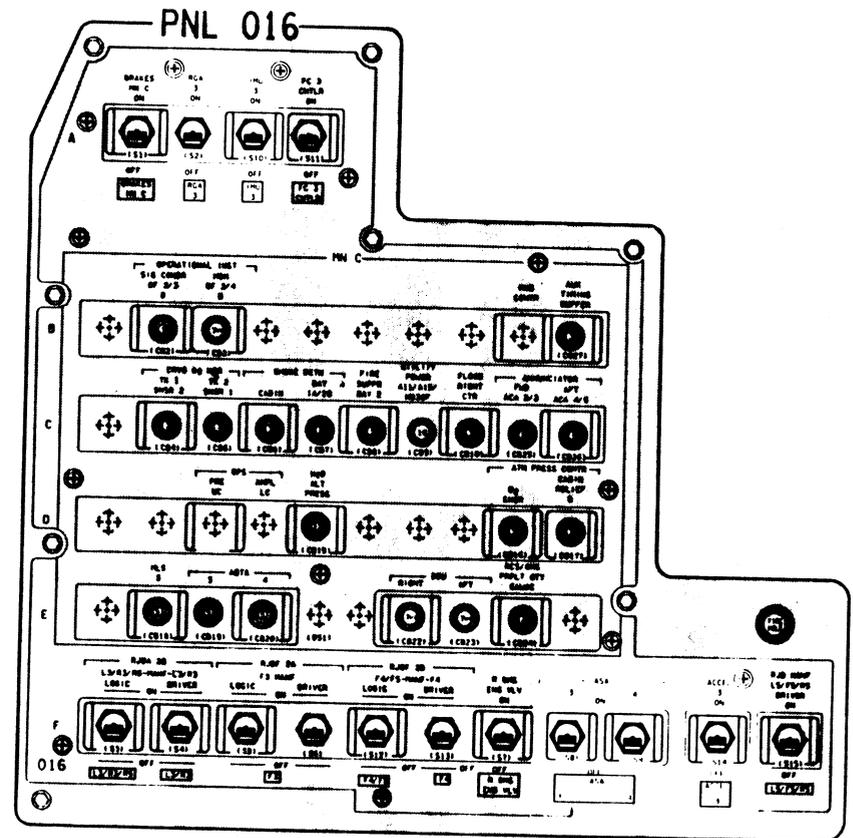


Figure 6-3.- Panel O16.

PO 17 is shown in figure 6-4. Row A on this panel contains the ATVC power switches. These switches control power to the SSME and SRB TVC system. Prelaunch, these switches are turned on to allow the SSME bells to gimbal and the SRB nozzles to be deflected for vehicle flight control.

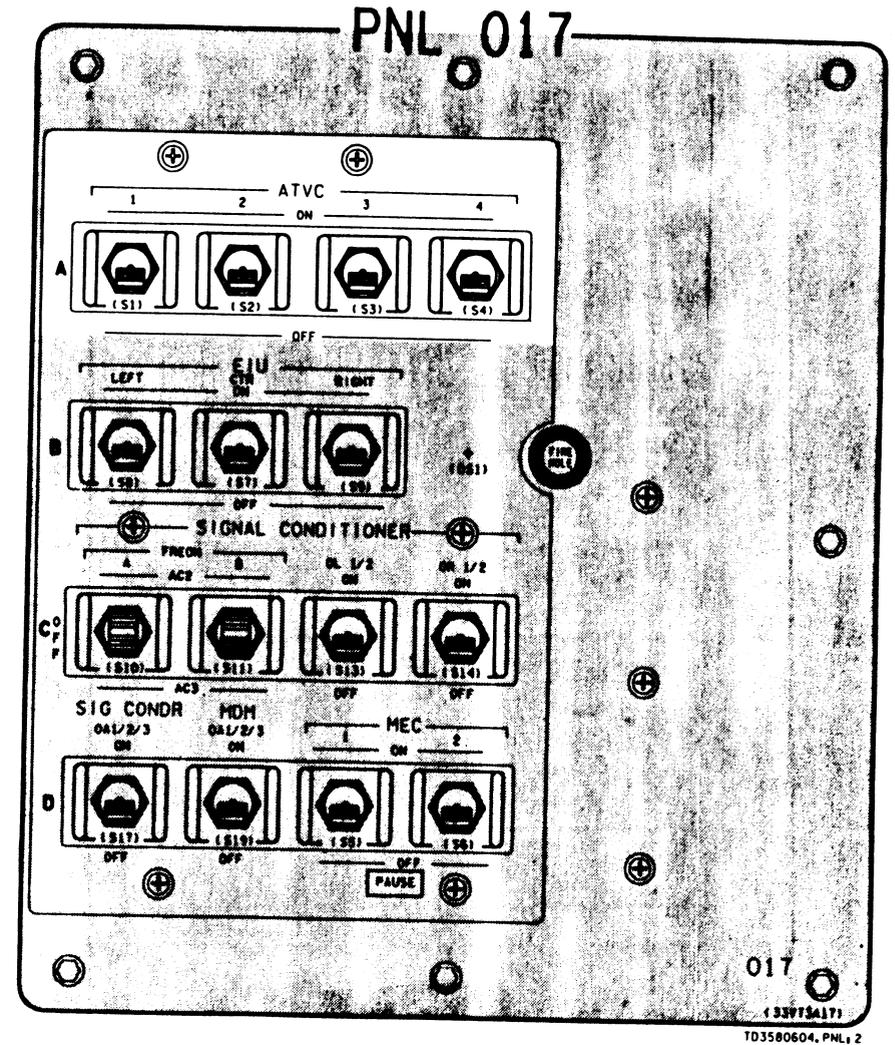
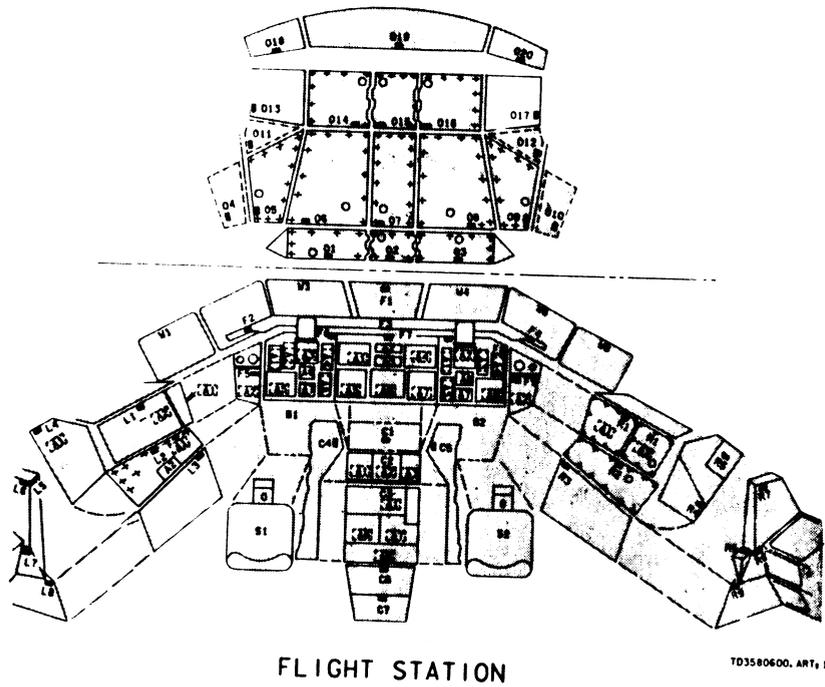


Figure 6-4.- Panel O17.

Overhead Panels 7 and 8

POs 7 and 8 are shown in figures 6-5 and 6-6. These panels contain the OMS and RCS valve switches and talkbacks. After a second SSME fails, the crew can look at the OMS and RCS crossfeed and RCS tank isolation valve talkback indications for an interconnected configuration. This interconnect, along with "SERC ON" being displayed on the ascent trajectory CRT, confirms to the crew that single-engine roll control has been established. The RCS jet activity lights on panel F6 cannot be used for this purpose because they will not illuminate during powered flight.

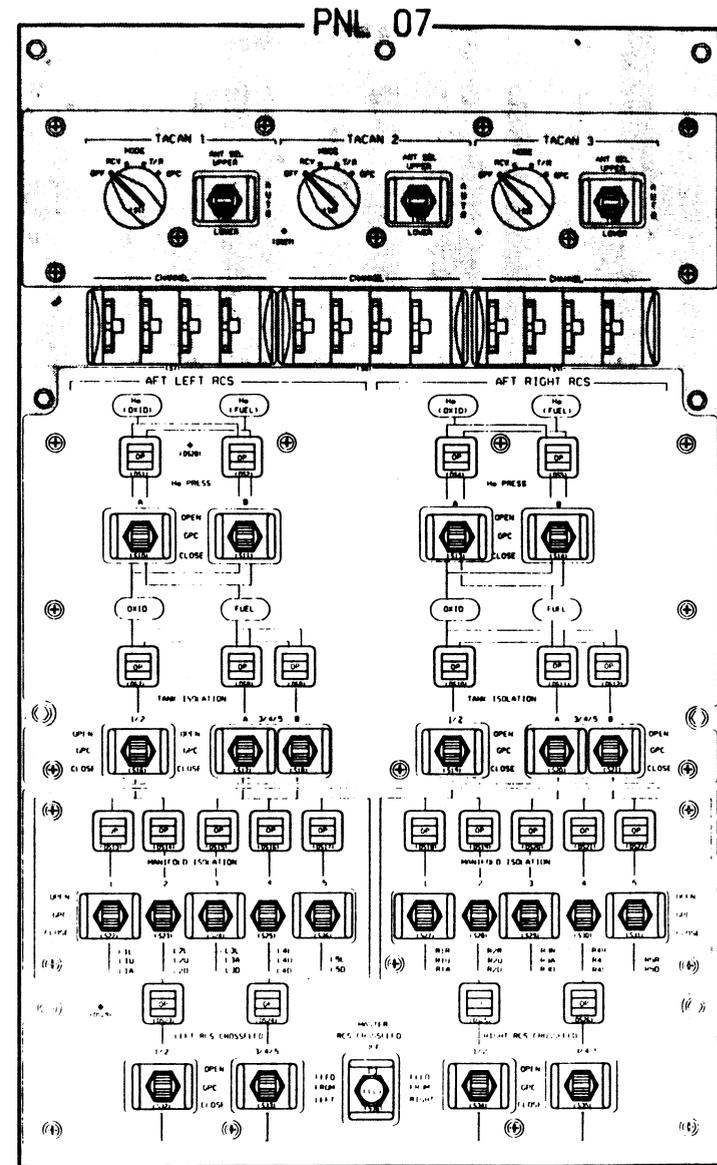
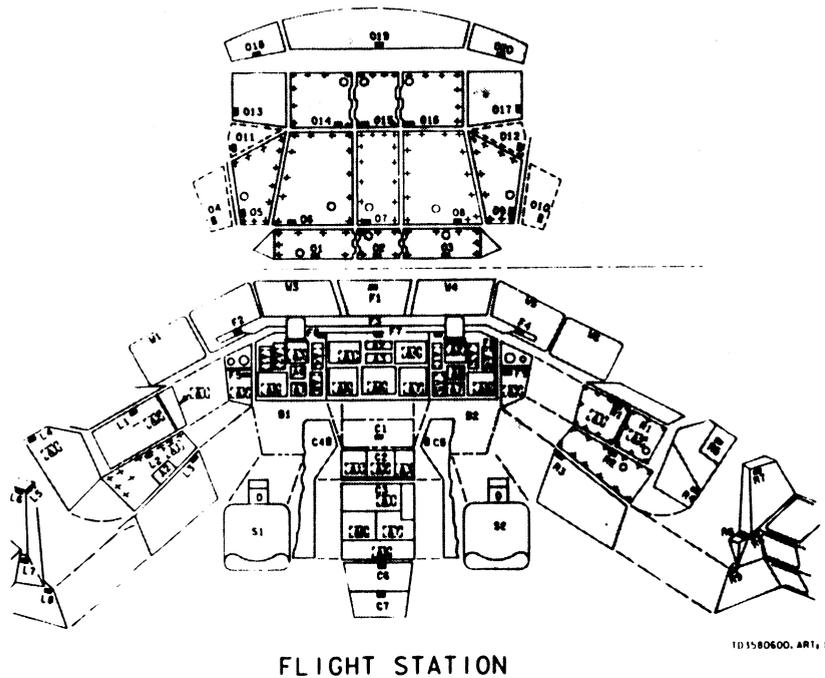


Figure 6-5.- Panel 07.

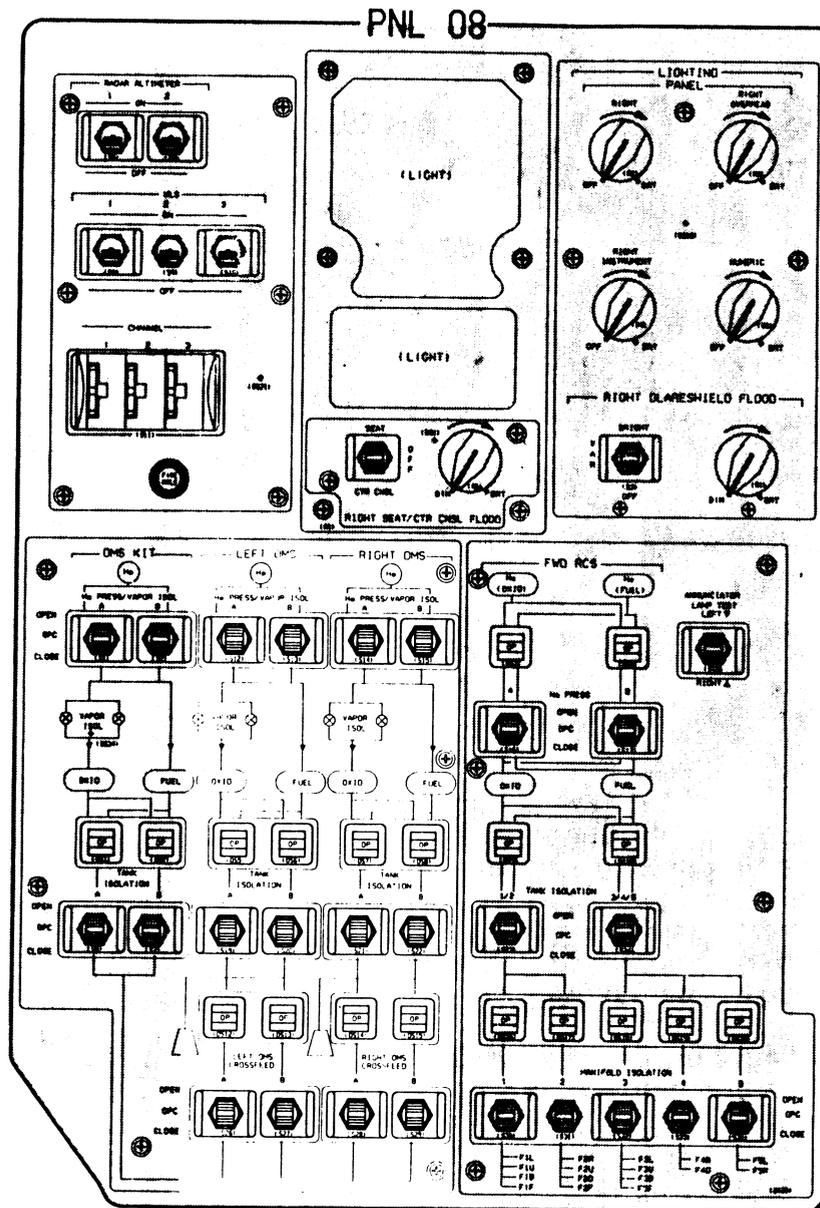
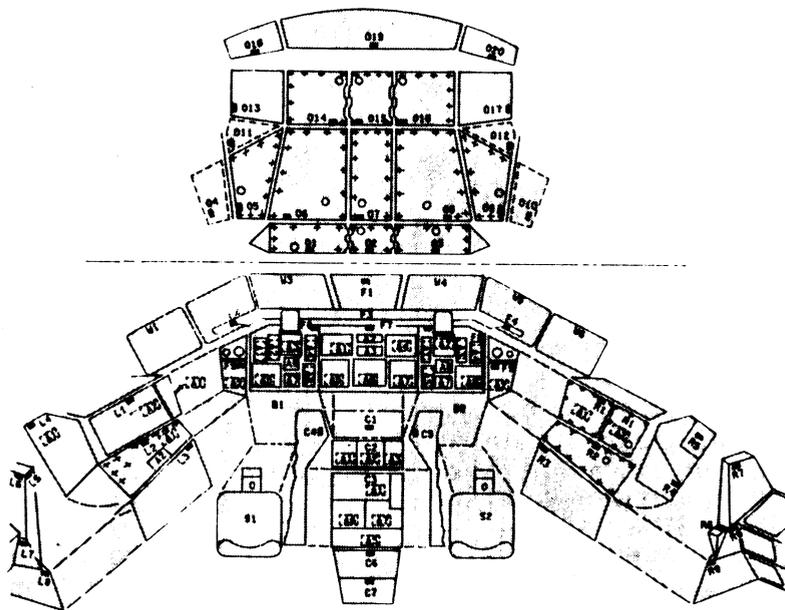


Figure 6-6.-Panel 08.

Forward Panels 2 and 4

Forward panels 2 and 4 (F2 and F4) have identical FCS mode PBIs; one set is available to the commander and the other to the pilot (figures 6-7 and 6-8). On these panels are located the AUTO/CSS PBIs for the pitch and roll/yaw axes. Automatic (AUTO) and CSS are mutually exclusive of each other. These PBIs are used to select either automatic or manual steering/guidance. These PBIs are initialized pre-lift-off in AUTO and are downmoded from automatic to manual steering in all axes by pushing any CSS PBI. In OPS 1, the RHC is not a hot stick and will not downmode to manual steering by deflecting the RHC only. Control is returned back to automatic steering by pushing an AUTO PBI. Lights will illuminate within the PBI to indicate the control mode the vehicle is in.

The speedbrake/throttle (SPDBRK/THROT) PBIs are also located on panels F2 and F4. This PBI is used to select AUTO throttling. Manual throttle is selected by the pilot pushing the takeover pushbutton (PB) on the SBTC and moving the SBTC until the throttle command matches the automatic command within 4 percent. The takeover PB is then released, and the crew has manual throttle control with the SBTC. The crew must perform MECO if manual throttling is selected because guidance cannot issue the MECO command. To return to automatic control, the SPDBRK/THROT PBI is depressed. When the pilot has manual control of the throttle, his MAN light will be illuminated and the commander's AUTO light will extinguish. Remember that the commander's SBTC is not enabled in powered flight.



FLIGHT STATION

TD3580600, ART. 1

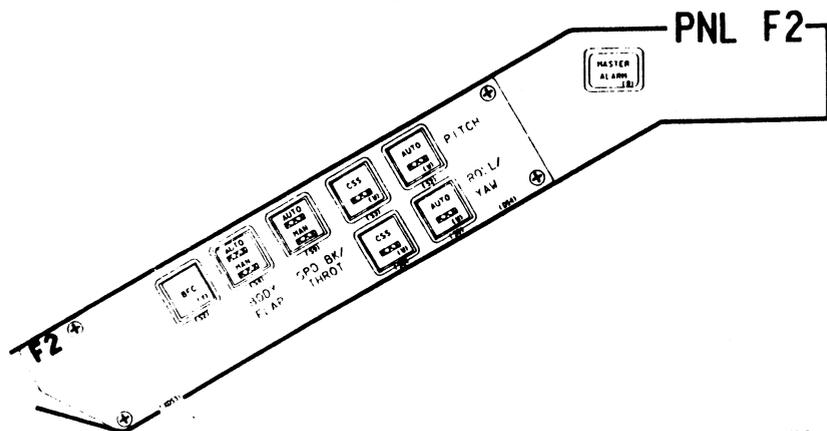


Figure 6-7.- Panel F2.

TD3580407, PM. 2

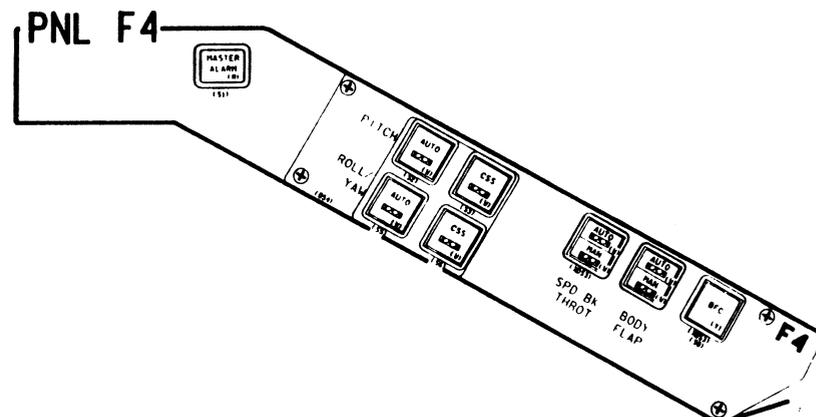


Figure 6-8.- Panel F4.

TD3580608, PM. 2

Forward Panels F6 and F8

F6 and F8 are similar to one another. F6 (figure 6-9) contains the commander's flight instruments, and F8 (figure 6-10) contains the pilot's flight instruments. Note that each panel has an ADI, a horizontal situation indicator (HSI), an alpha/Mach indicator (AMI), and an altitude/vertical velocity indicator (AVVI). All these indicators are scanned by the commander and pilot during ascent to monitor the vehicle performance from guidance, navigation, and flight control. These indicators will now be discussed in further detail.

Attitude Direction Indicator

The ADI is shown in figure 6-11. The ball within the ADI indicates the vehicle attitude as a result of software-generated attitude commands from guidance. Remember, though, that the IMUs provide gimballed angles in order to display attitude information. The attitude displayed on the ball can be one of three reference frames, depending on the ADI attitude switch position. This switch is shown in figure 6-10. The selectable reference frames are

- INRTL - Inertial attitude
- LVLH - Local vertical/local horizontal attitude
- REF - Reference attitude

The crew uses the guidance-commanded ADI attitude to compare it against the CRT displays and onboard FDF. This verifies that the software is flying the vehicle in the correct direction.

Reference figure 6-11 for the guidance error needles. These needles identify how far the current vehicle attitude is from the guidance-commanded attitude. The roll, pitch, and yaw errors are displayed by the top, side, and bottom needles, respectively. These needles are fly-to indicators. At the base of each needle is a scale with graduation marks to allow the crew to deduce the magnitude of the attitude error. Full-scale deflection is defined by the position of the ADI error switch. This switch has three positions: HIGH, MED, and LOW. The full-scale deflection values for all three axes are

- HIGH = 10°
- MED = 5°
- LOW = 1°

Located around and to the outside of the ADI ball are the vehicle body rate pointers. These pointers are shown in figure 6-11. They provide a continuous indication of the vehicle rates in degrees per second. Roll, pitch, and yaw body rates are indicated by the top, right, and bottom pointers, respectively. Graduation marks on either side of the null (center) position allow the viewer to determine the rate magnitude based on full-scale deflection. Full-scale deflection is defined by the ADI rate switch located next to the ADI error switch. Like the error switch, the ADI rate switch has three positions: HIGH, MED, and LOW. Full-scale deflection values for all three axes are

- HIGH = 10 deg/sec
- MED = 5 deg/sec
- LOW = 1 deg/sec

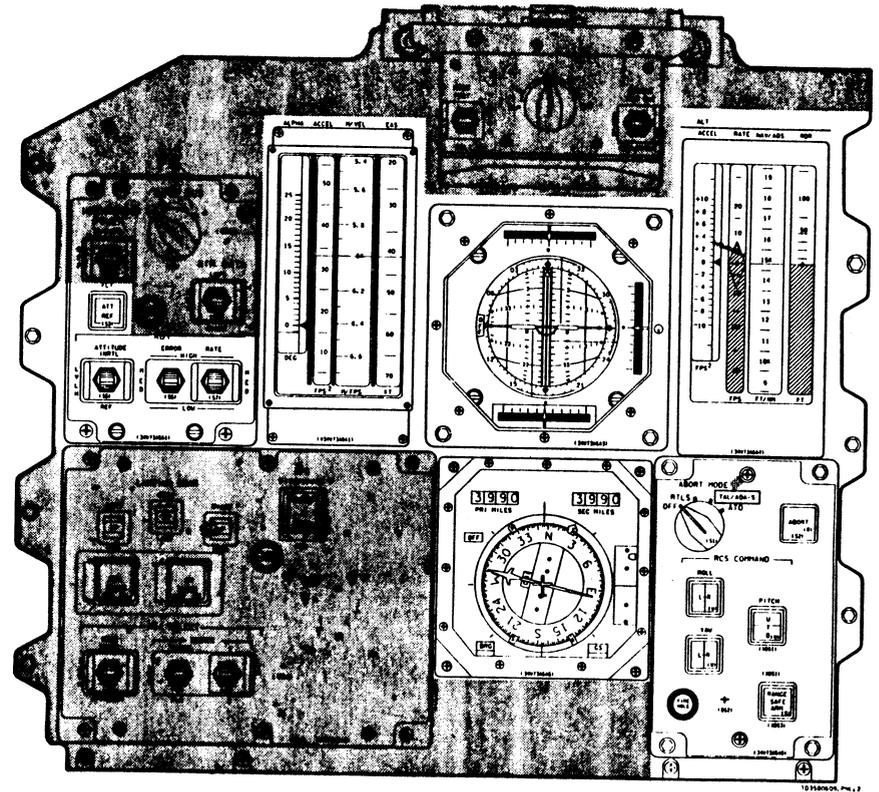
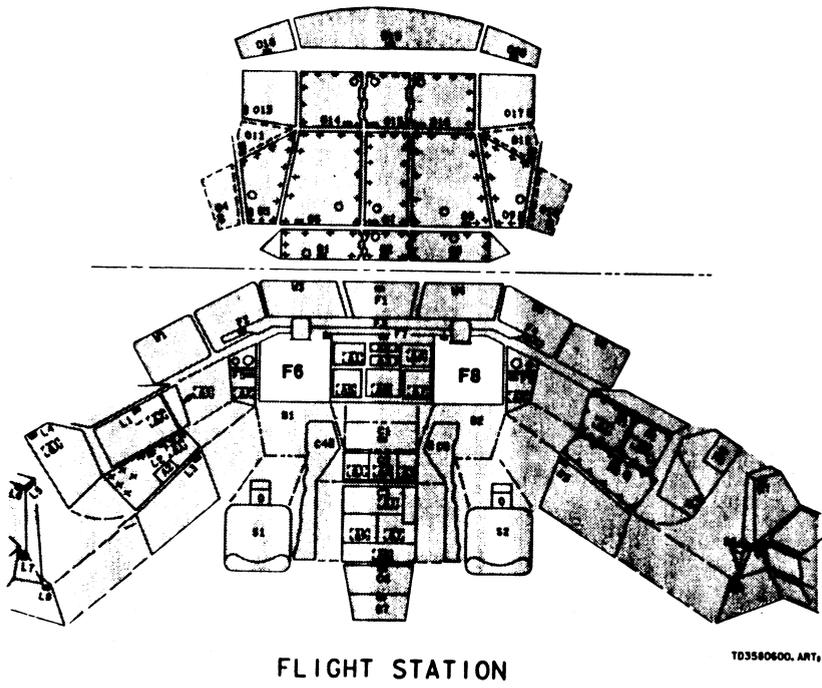


Figure 6-9.- Panel F6.

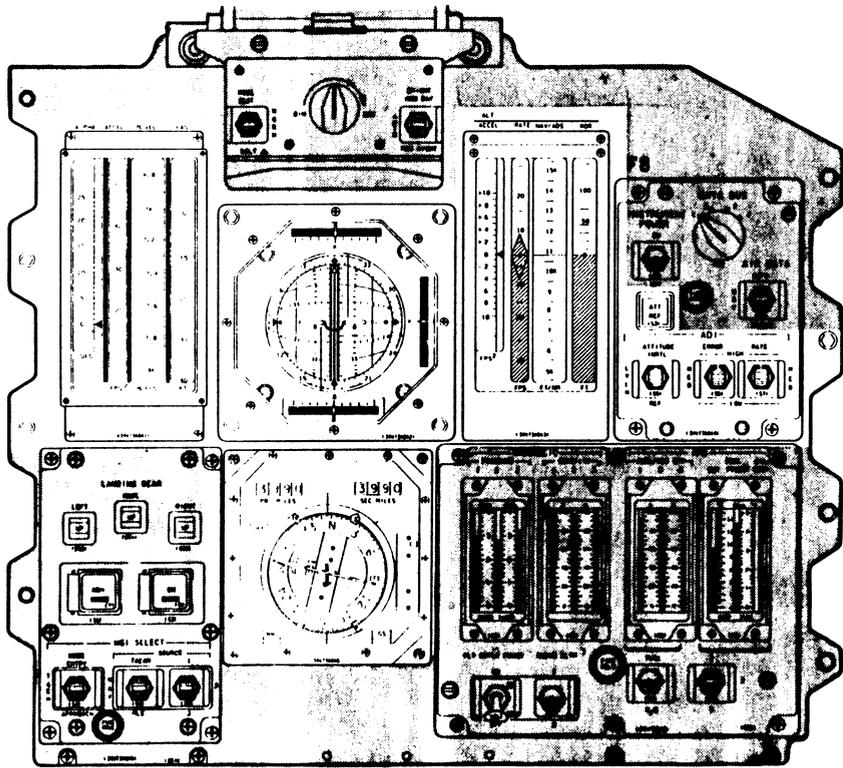


Figure 6-10.- Panel F8.

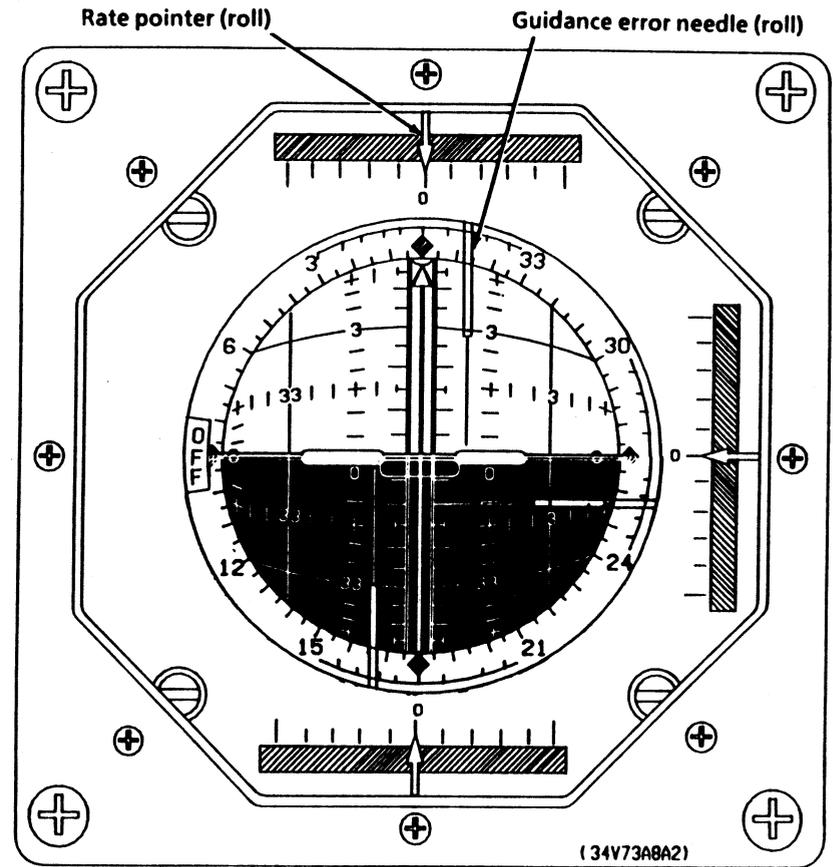


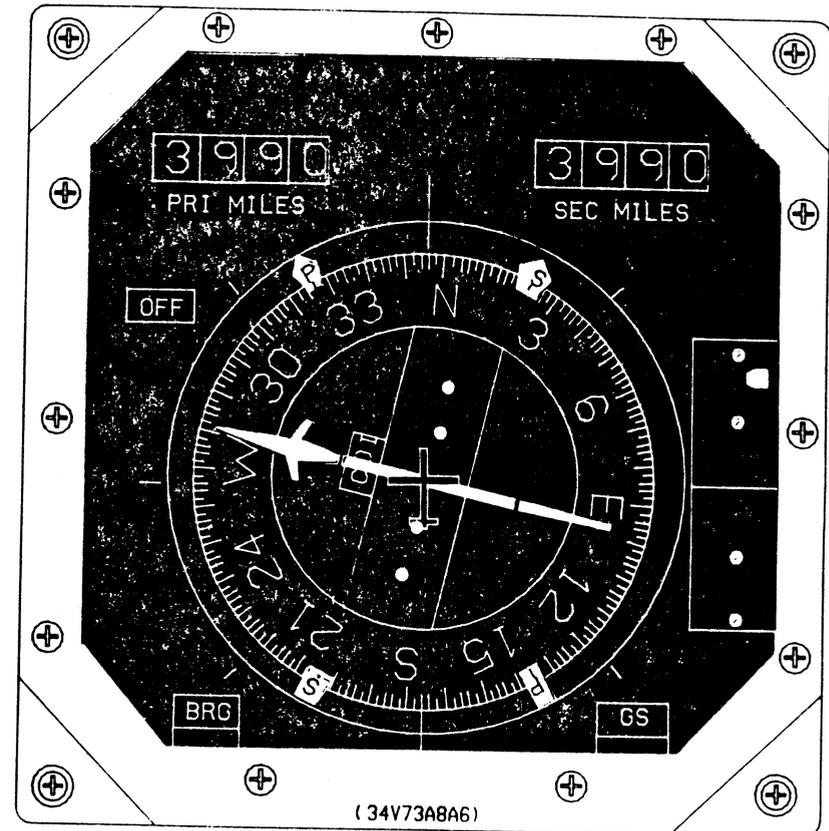
Figure 6-11.- ADI.

Horizontal Situation Indicator

An HSI is shown in figure 6-12. The HSIs are located immediately below the ADIs on panels F6 and F8. Like the ADIs, the HSIs serve two main functions during ascent.

- Provide an independent source (from nav) to compare with auto guidance
- Provide information needed by the crew to fly an ascent manually

During MMs 102, 103, and 601, the HSI provides information relative to the target insertion plane.



TD3580612. PNL, 1

Figure 6-12.- HSI.

The heading represents yaw, which is the heading of the body + X axis with respect to the target insertion plane as shown in figure 6-13. North on the compass card points along the target insertion plane; therefore, yaw angle is read as the angle between north and the current vehicle heading.

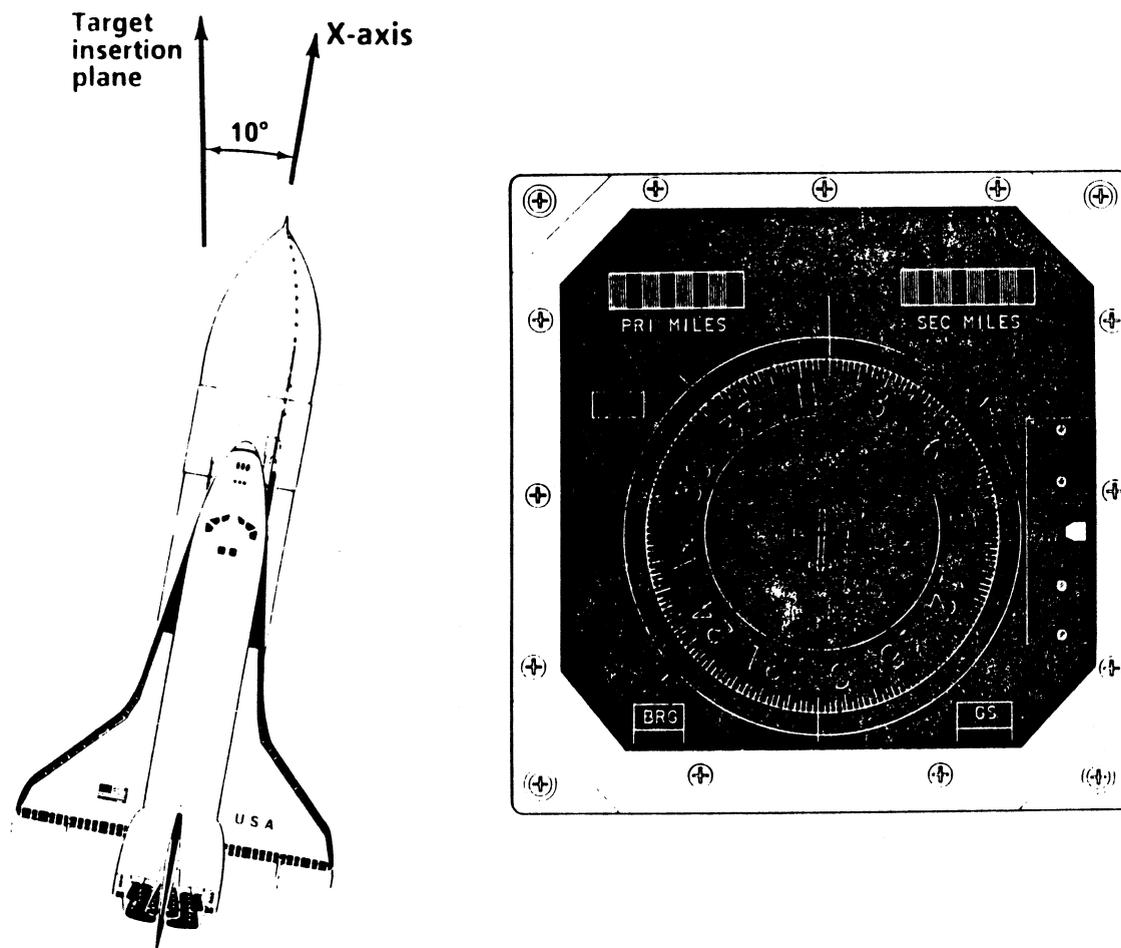


Figure 6-13.- HSI indication of yaw.

The course pointer provides the heading of V_{rel} with respect to the target insertion plane as illustrated in figure 6-14.

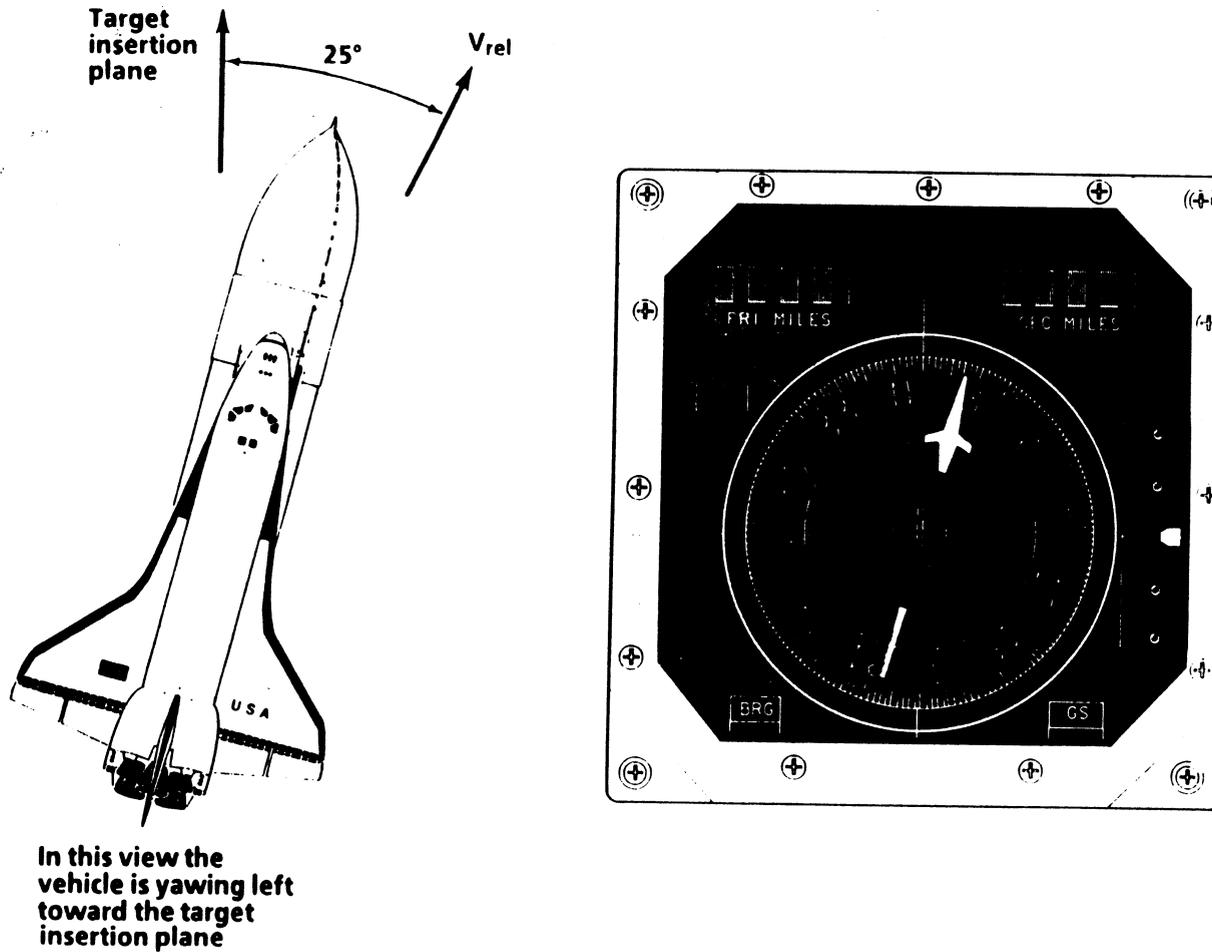


Figure 6-14.— Course pointer indicates heading of V_{rel} .

The course deviation indicator deflection indicates the estimated sideslip angle, beta (β). As shown in figure 6-15, this is the angle between the body X-axis and V_{rel} .

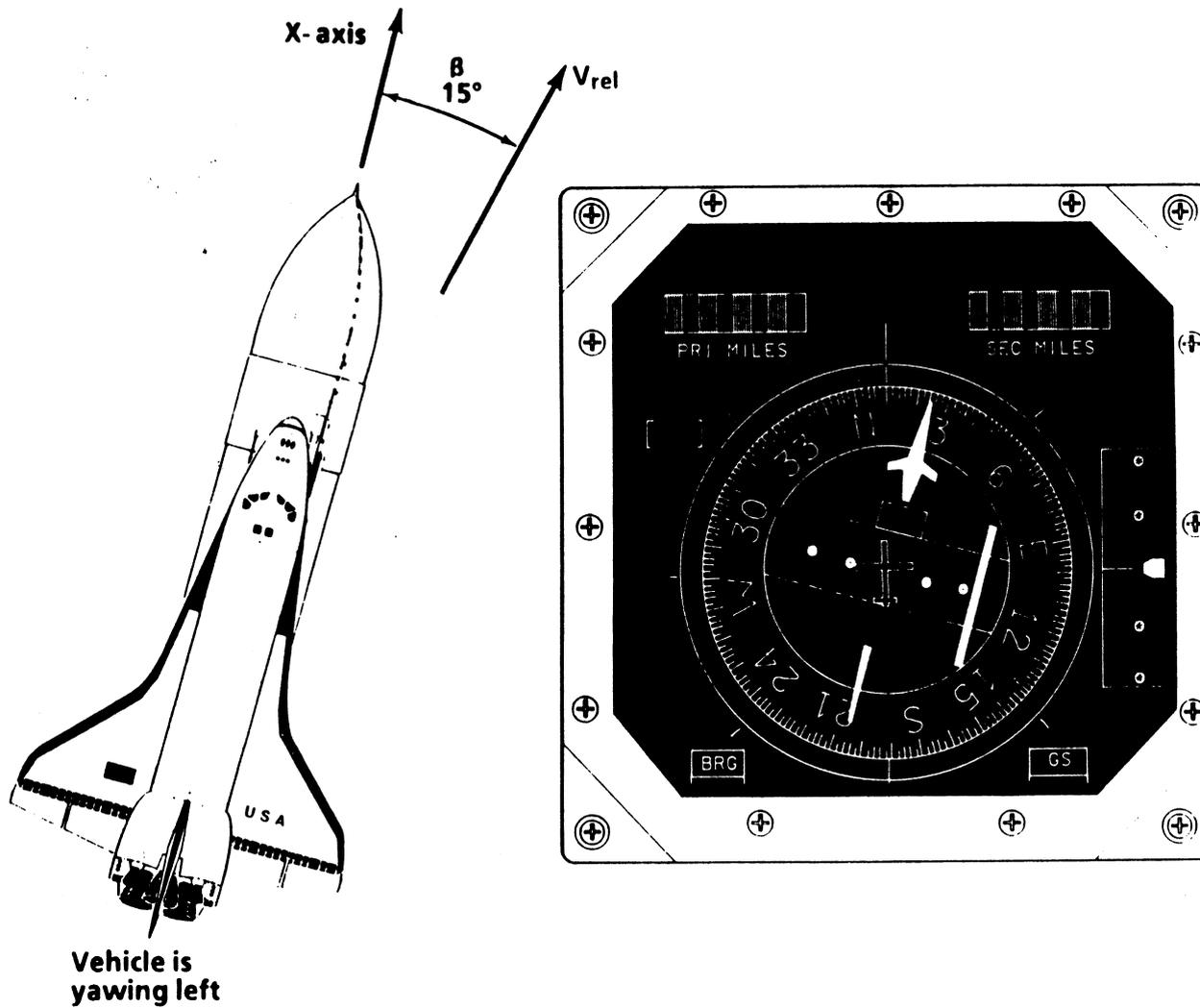


Figure 6-15.—HSI indication of sideslip.

The secondary bearing pointer in figure 6-16 provides the heading of the inertial velocity vector (V_i) with respect to the target insertion plane.

The other indications, primary range, secondary range, and glide slope deviation, are not driven during powered flight.

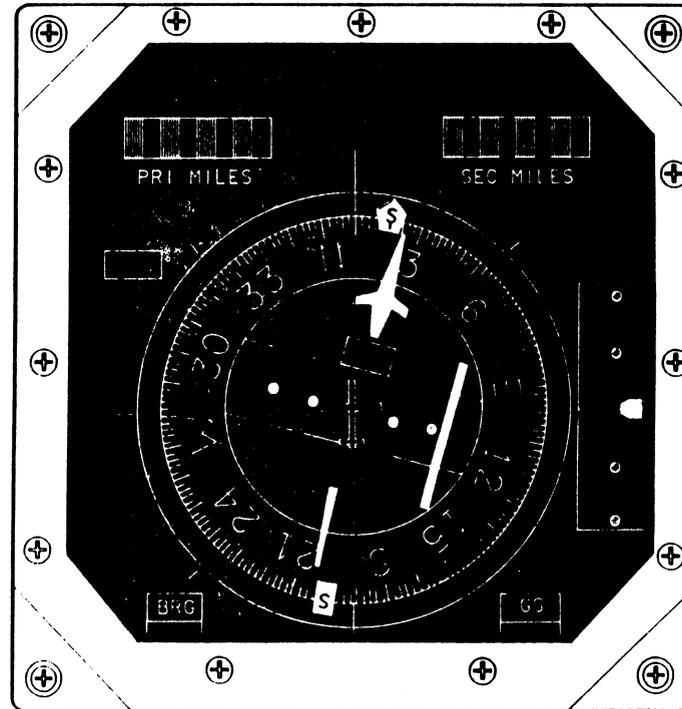
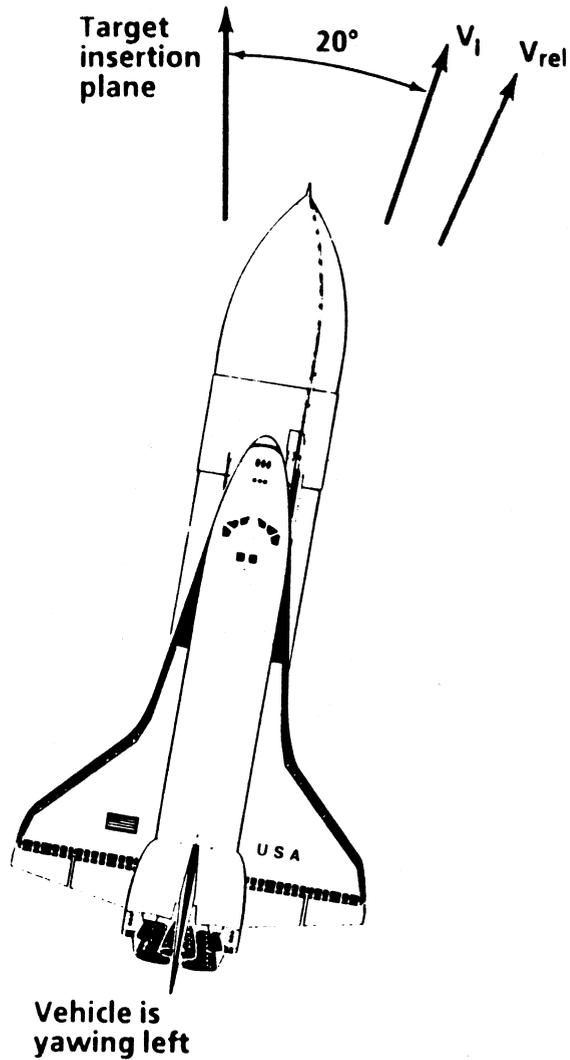


Figure 6-16.- HSI indication of inertial velocity vector.

The primary bearing pointer is driven only in MM 601 when it displays the heading to waypoint 1 (figure 6-17).

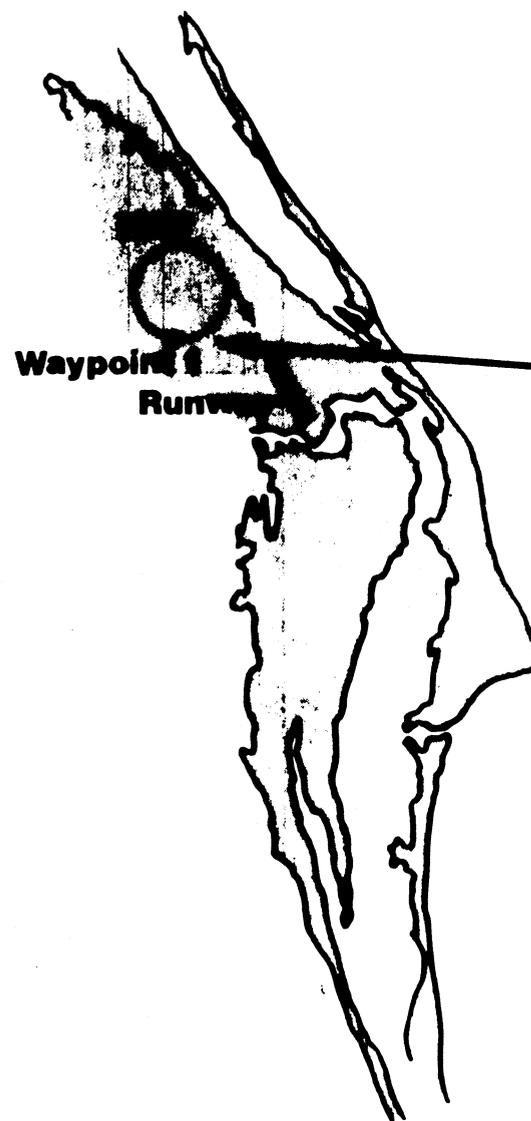
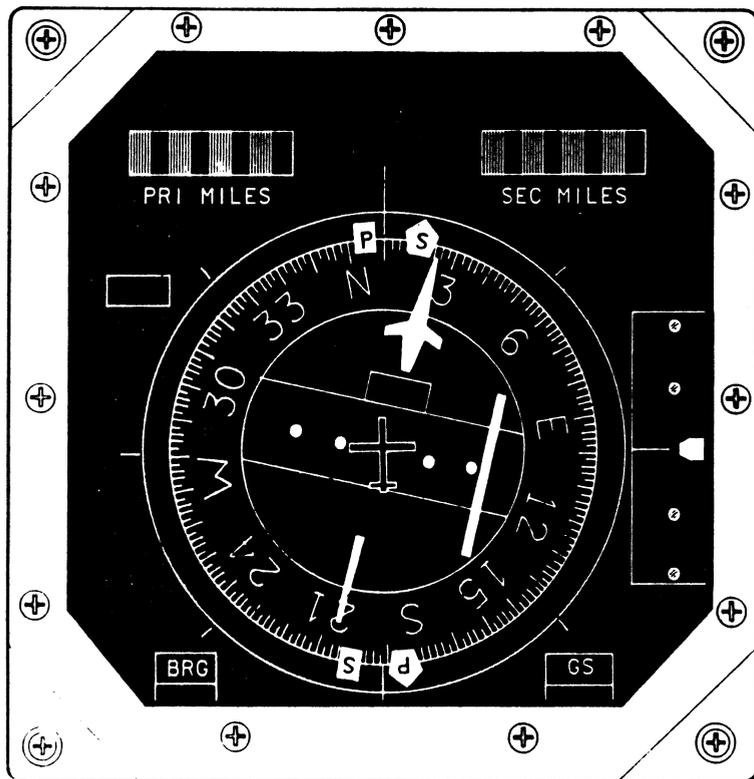


Figure 6-17.- Primary bearing pointer operation.

Alpha/Mach Indicator

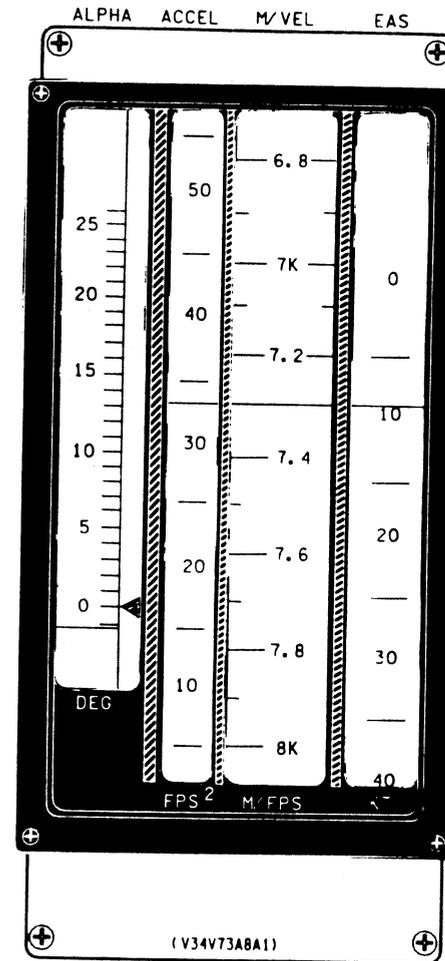
The AMI shown in figure 6-18 provides the crew with nav-derived velocity and aerodynamic information during powered flight.

The ALPHA tape displays the angle of attack in MMs 103 and 601. The scale ranges from -18° to $+60^{\circ}$. During MM 102, the ALPHA flag is displayed to indicate to the crew that this information is not available.

The ACCEL tape displays total vehicle acceleration in MMs 103 and 601. The scale ranges from -50 to $+100$ ft/s^2 . During MM 102, the ACCEL flag is displayed to indicate that this information is not available.

The M/VEL tape displays the magnitude of the inertial velocity vector during MMs 103 and 601. During MM 102 this tape displays the magnitude of the Earth relative velocity vector. The scale ranges from Mach 0 to Mach 4 and from this point in feet per second from 4000 to 27,000.

The EAS tape displays equivalent airspeed (EAS) in MMs 102, 103, and 601. The scale ranges from 0 to 500 knots. EAS is primarily used in first-stage flight as an indication of the dynamic pressure.



TD3580618. PNL 2

Figure 6-18.- AMI.

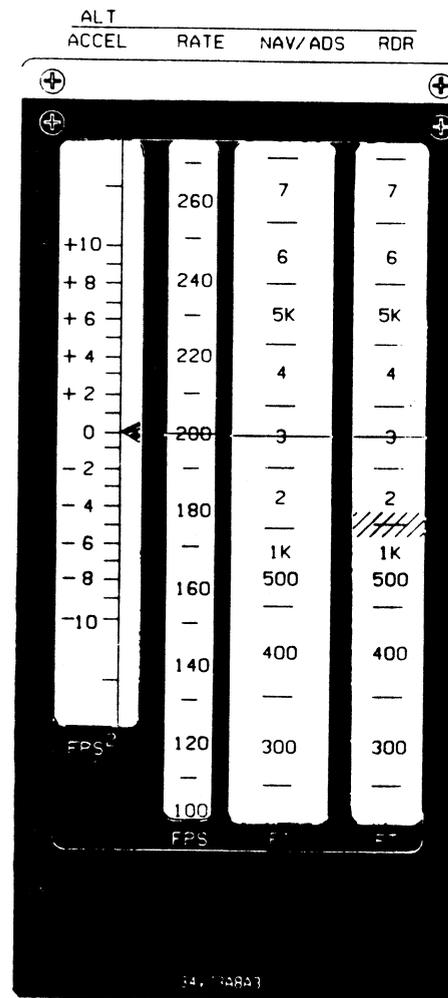
Altitude/Vertical Velocity Indicator

As the name implies, the AVVI in figure 6-19 provides nav-derived altitude and vertical velocity information during powered flight.

The NAV/ADS tape displays vehicle altitude with respect to the runway in MMs 102, 103, and 601. The scale ranges from 1100 to 400,000 feet and from 40 to 165 nautical miles. The feet and nautical mile scales overlap between 40 and 60 nautical miles.

The ALT RATE tape displays \dot{H} in MMs 102, 103, and 601. The scale ranges from -2940 to +2940 ft/s. Essentially, \dot{H} can be thought of as vertical velocity rate.

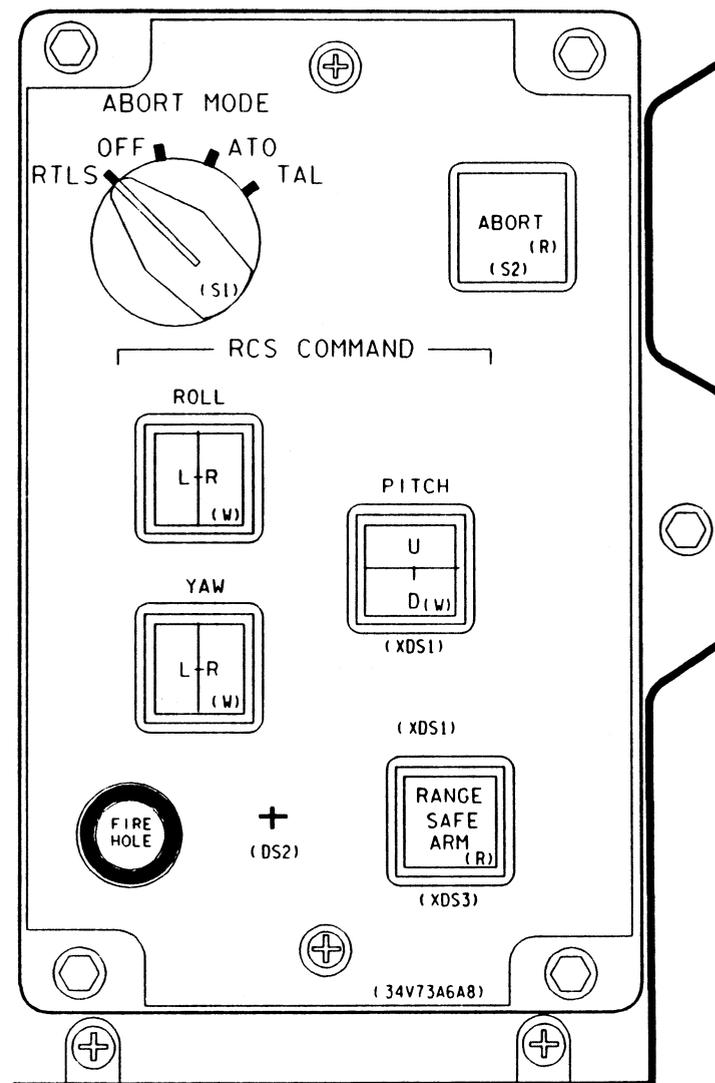
The other tapes, altitude acceleration (\ddot{H}), and radar altitude, are not driven during ascent.



TD3580619, PNL 2

Figure 6-19.- AVVI.

The abort rotary switch in figure 6-20 is available only to the commander on panel F6. This switch is used in conjunction with the abort PBI adjacent to it. These switches, when used, schedule abort guidance and flight control for RTLS, TAL, or ATO. They are used together in an arm/fire fashion. In order to mode guidance into an abort mode, the rotary switch is moved to the desired abort and the abort PBI is depressed. Of course, as soon as an abort is selected, a different set of MECO targets is selected for guidance.

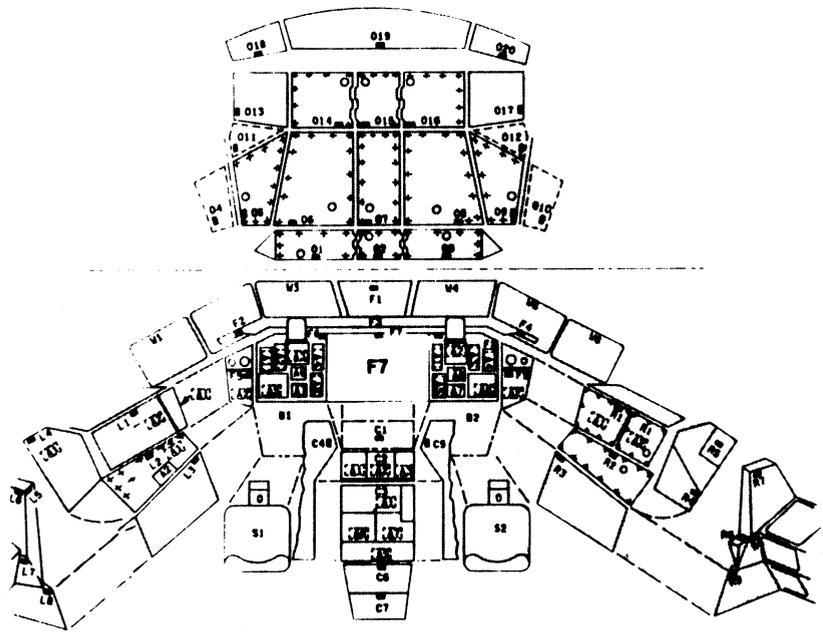


TD3580620. PNL 3

Figure 6-20.- Abort rotary switch.

Panel F7

The main engine status lights are located on F7 (figure 6-21). The main engine status lights will illuminate red whenever redlines are exceeded, indicating an SSME failure. These lights will illuminate yellow when off-nominal conditions exist on an SSME. There is one light for each main engine. More information can be obtained on these lights in the MPS 2102 Training Manual.



TD3580600, ART. 1

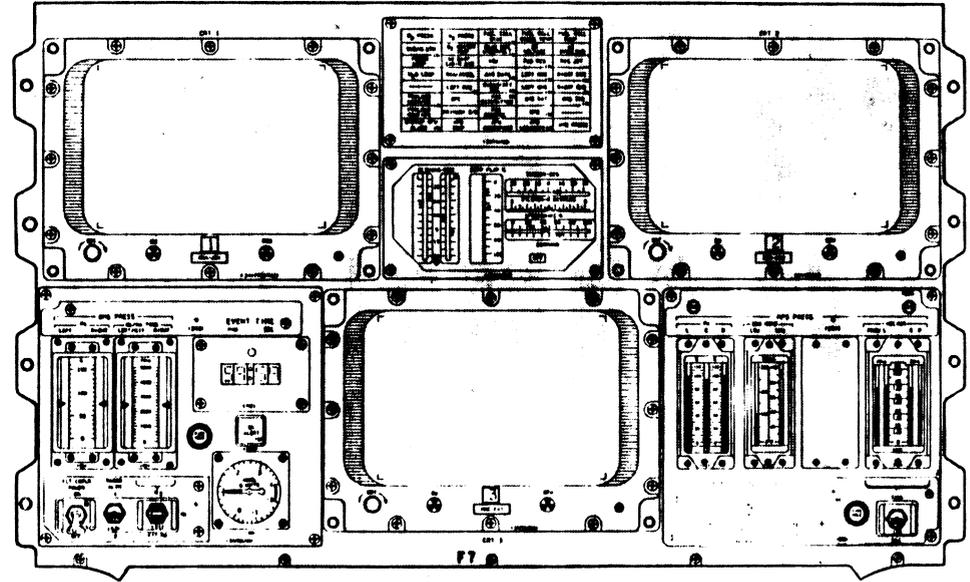


Figure 6-21.- Panel F7.

Center Panel 3

Several guidance and flight control switches are located on C3, figure 6-22. These switches are the FCS channel switches, the SSME limit shutdown switch, and the main engine shutdown PBs.

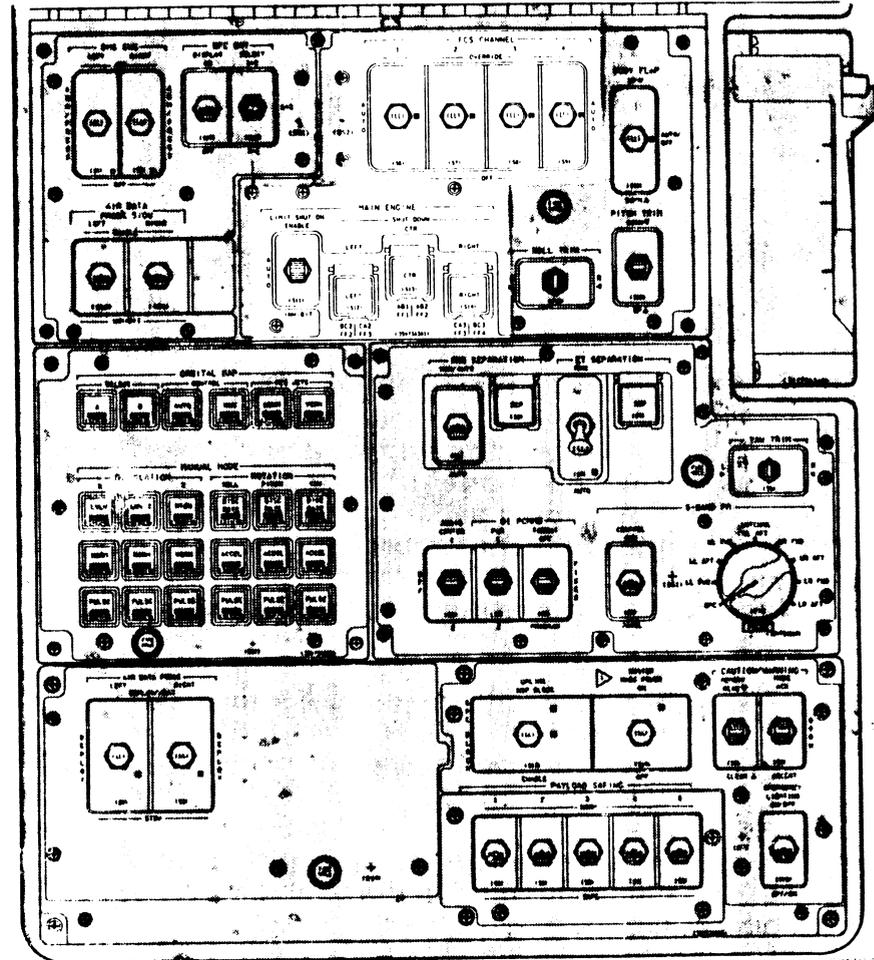
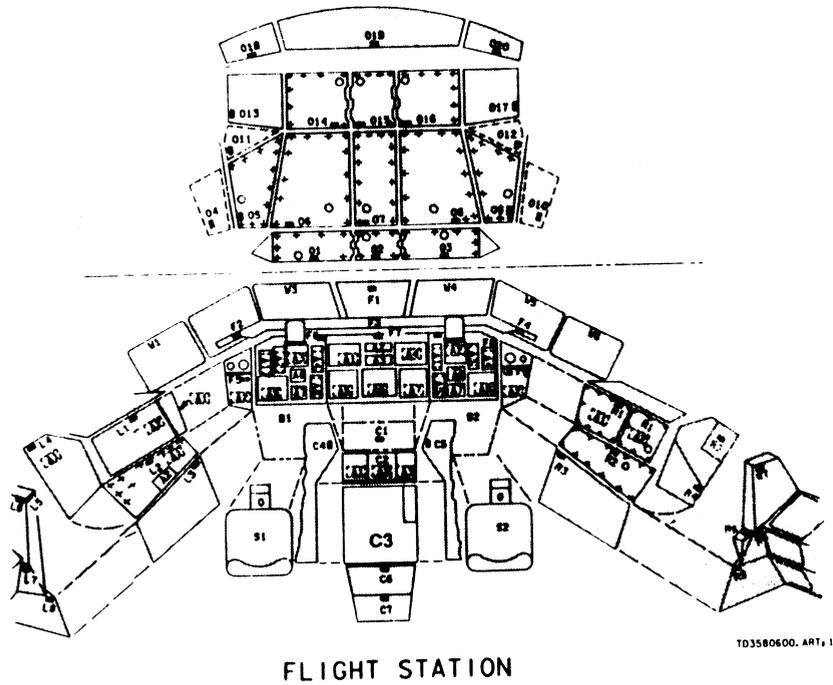


Figure 6-22.- Panel C3.

The FCS channel switches are three-position lever lock switches. The three positions are AUTO, OVERRIDE, and OFF. There is one switch for each FCS channel. These switches are used to control and protect the servovalves on channels 1 through 4 for the aerosurface, main engine, and SRB actuators. Under nominal conditions, these switches will remain in AUTO. If two FCS channels fail on any actuator, the remaining channels are taken to OVERRIDE to protect their servovalves. The FCS/EFF 2102 Training Manual contains more information on these switches.

The SSME limit shutdown switch is also located on C3 between the commander and pilot. This lever lock switch has three positions:

- ENABLE - Enables limit shutdown on all SSMEs
- AUTO - Enables only first SSME limit shutdown
- INHIBIT - Inhibits limit shutdown on all SSMEs

Depending on switch position, this switch either enables or inhibits an SSME to shut down if engine redlines are exceeded. The AUTO position allows the first engine to automatically shut down when these limits are exceeded. Any subsequent engine limit violations will be ignored by the GPCs. This switch has a major impact on vehicle guidance and flight control. If an SSME suffers an engine failure while limits are inhibited, guidance and flight control will not mode to two-engine logic. If two SSMEs fail while inhibited, single-engine roll control will not be enabled. Therefore, it is very important to know the position status of this switch for guidance and flight control moding.

Located next to the limit shutdown switch are the guarded main engine shutdown PBs. These PBs are used for two purposes.

- To manually shut down a main engine
- To inform guidance and flight control that an SSME has failed

The first purpose is obvious; however, the second is not. There are two times when guidance/flight control will not know of an SSME failure and will not mode to the appropriate logic:

- When an SSME fails while limits are inhibited
- When an SSME that had previously suffered a data path failure (GPCs cannot see engine data) fails

In these cases, guidance/flight control is moded by depressing the appropriate shutdown PB. As soon as this is accomplished, guidance and flight control will "see" the affected engine as failed and mode into the necessary logic. Under all other situations, guidance and flight control will mode automatically when a change in engine status/configuration has occurred.

CRT Displays

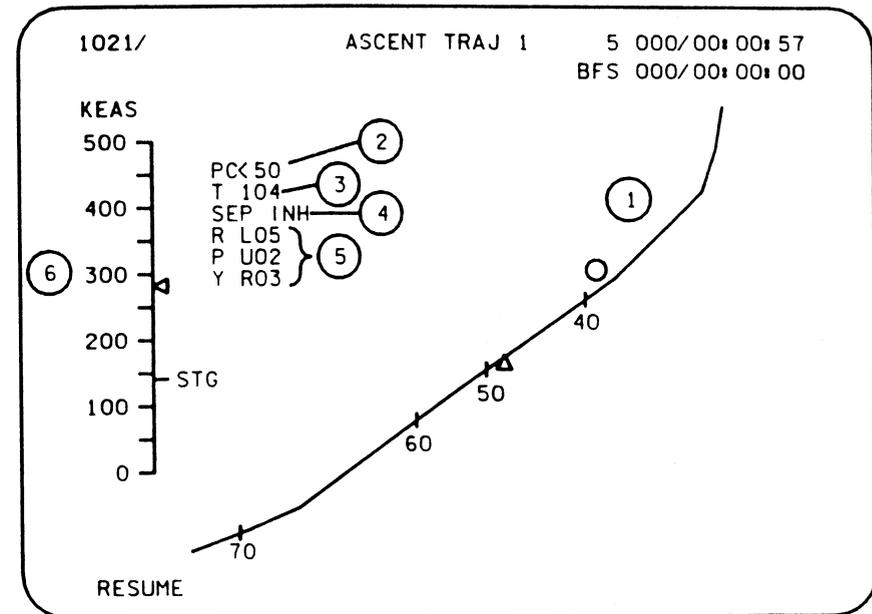
The ascent CRT displays contain ascent trajectory information for the crew. These CRT displays are mission specific and are designed for each flight based on expected performance. The central plots are all altitude vs. velocity profiles that provide the trajectory information. In this manner, the crew is able to determine how well guidance is performing to keep the vehicle on course. These displays are also used to manually fly ascents and aborts. There are four displays used to monitor trajectory data for all ascents. These are

- ASCENT TRAJ 1
- ASCENT TRAJ 2
- ASCENT TRAJ
- RTLS TRAJ 2

We will discuss each of these in further detail.

ASCENT TRAJ 1

The ASCENT TRAJ 1 display shown in figure 6-23 automatically comes up when the SSME start command is issued. It is used to monitor first-stage performance. At the end of first stage (MM 103 transition), this display will automatically be dropped and replaced with the ASCENT TRAJ 2 display. The ASCENT TRAJ 1 display is available in the backup flight system (BFS) (pre- and postengage) for MM 102.



TD3580623. CRT, 2

Figure 6-23.- ASCENT TRAJ 1 display.

① The central plot is an altitude vs. V_{rel} profile relative to the launch site. Though no scale is present on the display, the horizontal scale ranges from 0 to 5000 ft/sec. The vertical scale ranges from 0 to 170k feet. Tick marks on the curved line correspond to LVLH ADI ball angles of 70°, 60°, 50°, and 40°. These angles are used in conjunction with the ADI and the ascent FDF to determine guidance and vehicle performance. The current vehicle state from navigation is shown by a moving triangle. The predicted state for 20 seconds in the future is indicated by a moving circle. If an SSME fails during first stage, lofting will occur; therefore, the triangle and circle will track slightly above the curved line.

② $P_c < 50$ is displayed in this field, double overbright and flashing, when both SRB P_c sensors indicate < 50 psi. This flag indicates that the SRB separation sequence has been started.

③ T is a digital readout of the throttle command from guidance in percent. The range is from 50 to 104 percent. If manual throttling is in progress, this will indicate the manual command.

④ SEP INH is displayed in this field, double overbright and flashing, when the dynamic pressure and/or vehicle body axis rates exceed I-loaded values. The I-loaded values are

- $\bar{q} = 63$ pounds per square foot (psf)
- Roll rate = 5 deg/sec
- Pitch rate = 2 deg/sec
- Yaw rate = 2 deg/sec

These are the maximum limits for safe SRB separation. If these limits are exceeded, SRB separation will be inhibited (SEP INH) until they are satisfied. If this does not occur, SRB separation will automatically be commanded based on a backup time of 131 seconds MET (I-load).

⑤ In these fields, the BFS-computed roll (R), pitch (P), and yaw (Y) ADI errors are displayed for comparison with the primary avionics software system (PASS)-driven ADI error needles. The errors are used as fly-to indicators. For example, the following errors are indicated:

- R LO5
- P U02
- Y R03.

In this example, the crew would manually roll left 5°, pitch up 2°, and yaw right 3° to null the errors. The ADI errors are displayed preengage only. Postengage, these parameters are not displayed.

⑥ The scale on the left indicates equivalent airspeed in knots (KEAS) with a moving triangle. The STG tick mark indicates safe staging EAS (≈ 138.9), which is a good indicator of dynamic pressure.

ASCENT TRAJ 2

The ASCENT TRAJ 2 display is shown in figure 6-24. This display is available in the BFS (pre- and postengage) for MM 103. It will automatically be displayed upon the MM 103 transition and will remain up until the completion of the post-MECO -Z maneuver for ET separation. This display is primarily used to monitor second-stage performance.

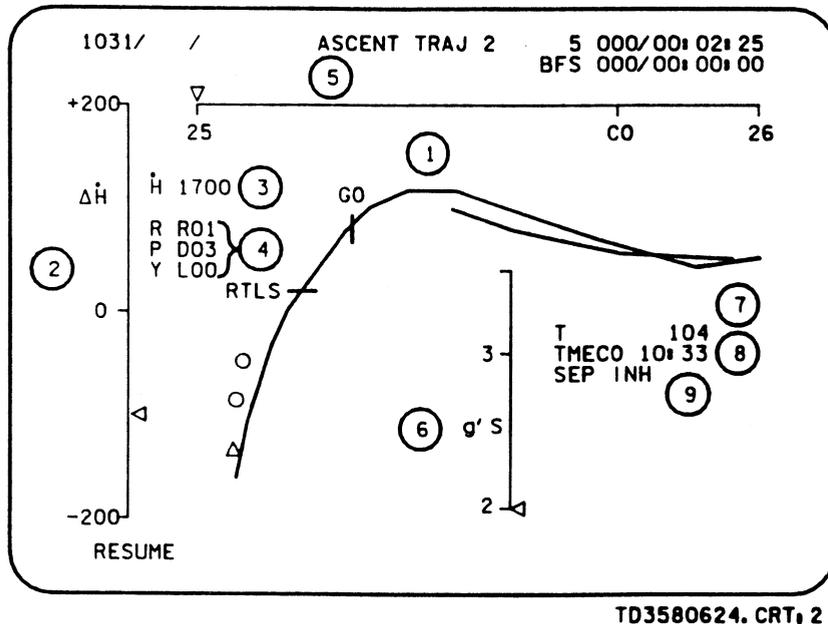


Figure 6-24.—ASCENT TRAJ 2 display.

① The central plot is an altitude vs. V_i profile for a nominal ascent and TAL. Although no scale is present on the display, the horizontal scale ranges from 5000 to 26,000 ft/sec. The vertical scale ranges from 140k to 525k feet. Two tick marks, RTLS and GO, are displayed on the nominal trajectory. Both of these marks indicate abort mode boundaries. The RTLS mark indicates negative return, and the GO mark indicates press to MECO. Just below the nominal trajectory is the TAL trajectory. The current vehicle state from navigation is shown by a moving right-pointing triangle. Predicted states for 30 and 60 seconds in the future are indicated by moving circles.

- ② On the far left side of the display is a ΔH scale which indicates the actual radius rate (essentially, vertical velocity) from nav, minus the expected radius rate for the present velocity. The present ΔH is indicated to the crew by a moving left-pointing triangle which flashes if off scale.
- ③ The \dot{H} is a digital readout of the present vertical velocity in feet per second from navigation.
- ④ This area indicates the ADI errors in R, P, and Y computed by the BFS software. These errors will be displayed only preengage. Postengage, these parameters are not displayed. The crew uses these parameters to compare with the PASS-driven ADI error needles. Next to R, P, and Y will be a number that indicates the magnitude of the error and one of the following letters: U, D, L, or R. The letters indicate that a fly-to correction is necessary to null the errors. For example, in this figure a pitch down of 3° and a right roll of 1° are required to null these errors.

- ⑥ The scale across the top is an inertial velocity scale x 1000 fps. Since most MECO targets are between 25,000 and 26,000 ft/sec, the scale is normally displayed with these limits; however, this is a flight-specific number, and the limits could be specified differently. The CO tick mark indicates where the main engines are commanded to shut down to attain the nominal MECO V_i . If a TAL abort is selected, this scale represents a ΔV to go biased between two flight-specific limits (remember, a TAL MECO is based on an R/V line). In this case, the CO tick mark represents a zero ΔV to go and is the TAL MECO indicator. Present velocity from navigation and TAL ΔV to go are indicated by a moving downward pointing triangle. Both scales are used to indicate when MECO will occur.
- ⑥ A vertical scale is located under the central plot to indicate the total load factor in g's. The current load factor is indicated by a moving left-pointing triangle, which will flash when 3g are exceeded. This scale is used to monitor how well the G-limiting subtask is performing to maintain the average acceleration below 3g. It is also used during manual throttling to indicate when to throttle back to maintain 3g.
- ⑦ This is a digital readout of the current throttle command from guidance or the speedbrake thrust controller. The readout ranges from 65 to 109 percent.
- ⑧ A digital readout of the predicted MECO time, TMECO, is displayed in this field. The time is displayed in minutes and seconds from lift-off. The stability of TMECO indicates when guidance has converged.
- ⑨ SEP INH will be displayed, double overbright and flashing, when safe automatic ET separation conditions do not exist. Safe ET separation conditions are: rates less than or equal to 0.7 deg/sec in each axis and closed feedline disconnect valves.

Ascent Trajectory Display

The ascent trajectory display is shown in figure 6-25. This display provides more information when an RTLS abort is selected than a nominal ascent or any other abort mode. The title of the display is dynamic. It is ASCENT TRAJ unless an abort is initiated, at which time it indicates the appropriate abort mode. This display automatically comes up at SSME start command and will remain up until MM 104 is entered (completion of -Z translation), or MM 602 if RTLS. It is available in the PASS for MMs 102, 103, and 601.

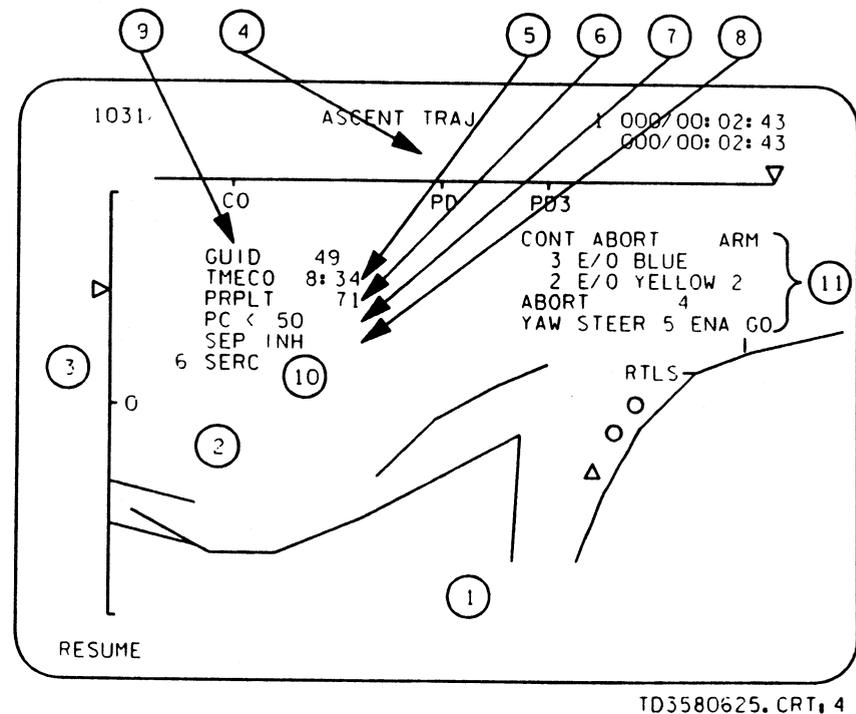


Figure 6-25.—ASCENT TRAJ display.

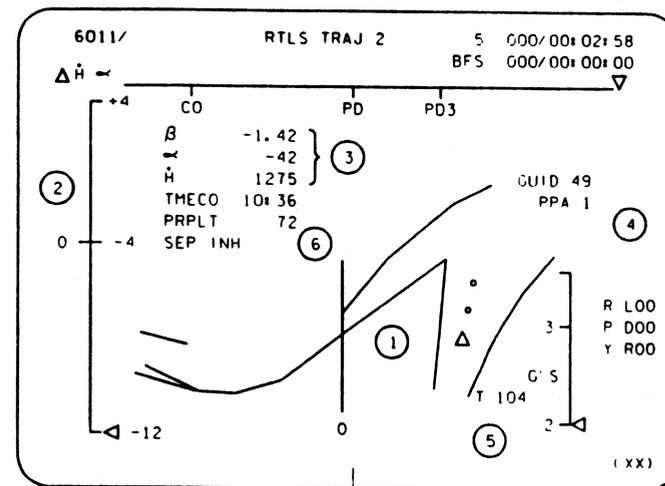
- ① There are three central plots on this display. The far right plot is the nominal ascent trajectory and is an altitude vs. V_{rel} profile. There are two tick marks on this trajectory, RTLS and GO. RTLS indicates the negative return boundary. The GO mark indicates press to MECO. The two plots pointing to the left are RTLS trajectories. Both plots are altitude vs. horizontal component of downrange velocity. The reason downrange velocity is used for RTLS is that the MECO target is Earth relative. The uppermost RTLS plot represents the highest lofting, which usually occurs when an SSME fails close to the SRB SEP timeframe. The lower RTLS plot represents the resulting trajectory when an SSME fails at lift-off. If an SSME fails at any other time, the resulting trajectory generally falls within this envelope. The current vehicle state from navigation is shown by a moving upward-pointing triangle. Predicted states 30 and 60 seconds in the future are represented by two moving circles. Once again, no scale is provided for the central plots. The horizontal scale ranges from -7000 to 10,000 ft/sec. The vertical scale ranges from 140k to 525k feet.
- ② Associated with the central plot are two lines which represent RTLS ET separation dynamic pressure constraints. The top line represents the altitude vs. velocity at which \bar{q} equals 2 psf. If the vehicle state is below or on this line, \bar{q} is greater than or equal to 2 psf. The lower line represents the altitude vs. velocity at which \bar{q} equals 10 psf. If the vehicle state is below this line, \bar{q} is greater than 10 psf. A \bar{q} of 2 is the minimum dynamic pressure necessary to aid RTLS ET separation (figure 3-33). A \bar{q} of greater than 10 psf is the maximum dynamic pressure that can exist for safe separation. Since these represent tank constraints, they are not flight specific and remain the same for every mission.
- ③ The vertical scale on the left is ΔH and is used for RTLS only. It is defined as H from navigation minus a reference H that is a function of delta range (ΔR). Present ΔH is indicated by a moving right-pointing triangle that flashes if off scale. The scale ranges from +200 to -200 ft/sec. This scale is a fly-to indicator.
- ④ The horizontal scale across the top is ΔR . The ΔR is driven only during RTLS. The scale ranges from -143 nautical miles on the left to +17 nautical miles on the right. ΔR is the glide range potential based on the current energy state. This state is a function of horizontal component of V_{rel} and flightpath angle. Present ΔR is indicated by a moving downward-pointing triangle which will not flash if off scale. The PD tick mark indicates where PPD will occur with two active SSMEs. The PD3 tick mark indicates where PPD will occur with three SSMEs operating. Both marks are at MECO -20 seconds. The CO tick mark indicates when MECO will occur. At approximately MECO -45 seconds, the triangle will move from right to left toward the CO tick mark. This scale indicates how well the PPD and guidance cutoff subtasks are performing. It will also inform the crew when to manually pitch down and shut down the main engines. One more point before we leave this scale: a flashing F for fine countdown will be displayed on the right side of the scale when the fine count discrete is set.
- ⑤ TMECO is the predicted time of MECO, from guidance in minutes and seconds from lift-off.
- ⑥ PRPLT is the percent MPS propellant remaining in the ET. This is a calculated quantity from guidance based on vehicle acceleration, mass, and number of operating SSMEs.

- ⑦ This field is where $P_c < 50$ is displayed, double overbright and flashing, when both SRB chamber pressures are less than 50 psi. $P_c < 50$ informs the crew that the SRB separation sequence has started. At second-stage initiation, this field is blanked.
- ⑧ SEP INH is displayed in this field when automatic SRB and/or ET separation is inhibited.
- ⑨ When an RTLS abort is selected, guidance status prior to PPA will be displayed. The following mutually exclusive information can be displayed:
 - GUID INIT - Indicates that the guidance solution is being initialized.
 - GUID xxx - Typically a two-digit (although three digits are possible) integer indicating percent deviation of the predicted final mass from the RTLS PPA mass target. As PPA is approached, the numeric value will decrement to zero.
 - GUID INHB - Indicates either that PPA is inhibited because guidance is unconverged or that PPA is being delayed because of an attitude violation.
 - GUID PPA - Indicates that PPA has been initiated.
- ⑩ Item 6 can be toggled to enable or disable single engine roll control. An asterisk will be displayed when SERC has been enabled. "ON" will be displayed double overbright to the right of "SERC" when single engine roll control has been established by the DAP.

- ⑪ These fields provide information and item entry controls to support the automated contingency abort guidance and flight control. SSME failure configuration and contingency abort region determine the algorithm to be scheduled. These fields are covered in greater detail in the Contingency Abort 2102 Training Manual.

RTLS TRAJ 2

One can see in figure 6-26 that this display is very similar to the ASCENT TRAJ display. Therefore, only the differences will be discussed. This display is automatically called up at RTLS abort in second stage and will remain up until MM 602 is entered. It is available in the BFS for MM 601 only.



TD3580626. CRT. 3

Figure 6-26.- RTLS TRAJ 2 display.

① The central plot is an altitude vs. horizontal component of V_{rel} profile. The horizontal scale ranges from -7000 to 10,000 ft/sec. The vertical scale ranges from 140k to 525k feet. The only difference to the central plot is a vertical line across the RTLS flyback profiles. Note that this line has a zero at the bottom. This line represents the point at which the horizontal V_{rel} equals zero. The vehicle is not moving forward or backward with respect to the Earth; therefore, the flightpath angle is 90° at this point, with the vehicle falling straight down.

② The scale on the far left now includes angle of attack with a moving left-pointing triangle. The alpha triangle will not flash if off scale. The limits of this scale range from $+4^\circ$ to -12° .

③ These fields display the following from navigation:

- β = sideslip angle in degrees
- α = angle of attack in degrees
- H = radius rate in feet per second

④ Item 1, PPA, issues a powered pitcharound command to the BFS to force it into PPA during an RTLS. This is necessary for a BFS-engaged RTLS because there is no manual guidance and flight control. If BFS auto guidance does not pitch the vehicle around, item 1 is executed. An asterisk will be displayed as soon as BFS powered flight guidance sets the RTLS flyback discrete to true.

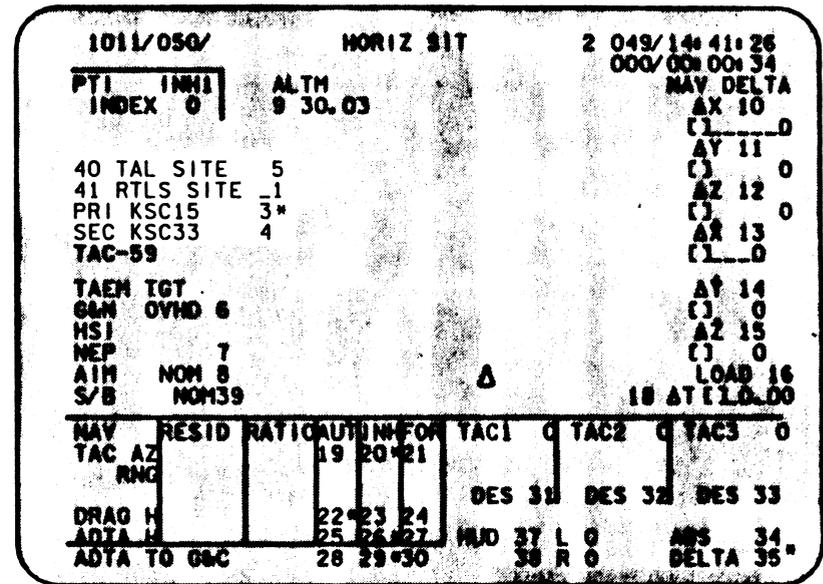
⑤ T is a digital readout of the throttle command in percent, from guidance. This field does not reflect commands from the SBTC.

⑥ SEP INH is displayed in this field only when automatic ET separation is inhibited.

The remainder of the display is identical to the ASCENT TRAJ display.

Horizontal Situation Display

SPEC 50, shown in figure 6-27, is the Horizontal Situation display. It is used to select alternate TAL and RTLS landing sites if the primary (I-loaded) sites are not desired. Once the primary site changes, the MECO target that guidance flies to will change. This is the purpose of item entries 40 and 41; they change the MECO target. SPEC 50 is available in the PASS and BFS for OPS 1 and 6.



TD3580627. CRT, 3

Figure 6-27. – Horizontal Situation display.

Item 40 allows selection of an alternate TAL landing site before TAL abort select. The display is initialized with the primary TAL site and thereafter reflects crew inputs. An index of 1 to 5 will specify the TAL/RTLS site. If a TAL abort is selected, item 40 will blank and the TAL site index will appear next to item 41. As soon as this occurs, the primary and secondary runways for a given TAL site will be displayed next to item entries 3 and 4. An alternate TAL site can be selected via item 41 after TAL abort select but before MECO -10 seconds. After this time, no change in the TAL MECO targets will occur.

Item 41 is initialized with RTLS. The primary and secondary runways for the selected RTLS site are displayed unless a TAL abort has been selected, in which case TAL is displayed. An alternate RTLS site can be selected anytime before PPD. After this time, no change in the RTLS MECO targets will occur. Items 3 and 4 allow selection of either the primary or secondary runway of a selected landing site. The runway IDs are displayed next to items 3 and 4, with an asterisk indicating which has been selected.

Should a contingency abort be declared, "CONT" will be displayed next to item 41. This will enable the site index to expand to 25 in order to select more landing sites for the contingency abort at hand. In this situation, the MECO targets associated with this item entry will be ignored by guidance.

OVERRIDE Display

SPEC 51 is the OVERRIDE display and is shown in figure 6-28. It contains item entries to select a TAL or ATO abort and enable maximum throttles.

100/001/		OVERRIDE		XX X DDD/HH:MM:SS		DDD/HH:MM:SS	
ABORT MODE		ELEVON	FILTER	ENTRY FCS		ATMOSPHERE	
TAL/AOA	1	NOM 17*	NOM 20*			NOM 22*	
ATO	2	ALT 18	ALT 21			N POLE 23	
ABORT	3					S POLE 24	
MAX THROT	4						
PROPLT DUMP				TRU STAT	ATT DES	PRL	
				1	25	SYS AUT DES	
				2	26	1	20* 31
OHS DUMP TT8	47			3	27	2	29* 32
INH ICNCT						3	30* 33
CONT DUMP				ADTA			
				L	99616	0.0	0.10 34
ARM	6			R	99616	0.0	0.10 35
START	7*				99616	0.0	0.10 36
STOP	8				99616	0.0	0.10 37
AFT RCS 13	INH			ET SEP		HOLL MODE	AUTO
14 TIME	- 0			AUTO	38	AUTO SEL	42
				SEP	39		
FWD RCS 15	INH			ET LMB DR		VENT DOOR CNTRL	
16 TIME	- 1			CLOSE	40	OPEN	43
NAV EDIT	AC			RCS RM HANG		CLOSE	44*
				CL OVRD	41		
					12		
						00:13	

TD3580628. CRT, 2

Figure 6-28.- OVERRIDE display.

Abort Mode

If the abort rotary switch on F6 were to fail, TAL/ATO aborts could still be selected by use of this display. Item 1 or 2 allows the crew to select either a TAL (there is no pre-MECO AOA selection) or ATO abort, respectively. These item entries are mutually exclusive, with the selected mode indicated by an asterisk. The abort sequence is initiated by the execution of item 3 and is indicated by an asterisk. When this is performed, the vehicle will start steering toward the abort MECO target. Incidentally, an RTLS abort is selected by OPS 601 PRO as a backup for the abort rotary switch.

Max Throttles

Item 4 enables the maximum SSME throttle level and rescales the SBTC to the new level. Execution of item 4 enables auto or manual guidance to command the SSME throttles to a setting higher than the nominal 104 percent up to a maximum of 109 percent. Once executed, an asterisk is displayed next to item 4. If this is performed in MM 102 or MM 601 (pre-PPA), max throttles will not occur until an SSME fails. Once maximum throttles are enabled, there is no turning back to nominal throttles without using the SBTC, because the command is latched.

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Questions

1. Which of the following sensors does NOT have a dedicated power supply switch in the cockpit?
 - A. Orbiter RGAs
 - B. IMUs
 - C. AAs
 - D. SRB RGAs
2. How can the crew determine if single-engine roll control has been established?
 - A. When the SE ROLL light illuminates on panel F7
 - B. By monitoring the OMS/RCS valve talkbacks and the SERC status on a CRT display
 - C. By monitoring the RCS jet activity lights on panel F6
 - D. By depressing the appropriate main engine shutdown pushbuttons when two out of three engines fail
3. How is manual guidance established?
 - A. When the commander or pilot depresses the CSS PBIs on panel C3
 - B. Via an item entry on SPEC 51, the OVERRIDE CRT display
 - C. When the commander or pilot depresses the CSS PBIs on panel F2 or F4, respectively
 - D. Simply by the commander or pilot deflecting the respective RHC out of detent
4. How are manual throttles established?
 - A. When the pilot depresses the takeover pushbutton on panel F4 and moves the SBTC to match the auto throttle command within 4 percent
 - B. Via an item entry on SPEC 51, the override CRT display
 - C. When the pilot depresses the takeover pushbutton on his SBTC and moves it to match the auto throttle command within 4 percent
 - D. When the commander depresses the speed-brake/throttle PBI on panel F2 and moves his SBTC to match the auto throttle command within 4 percent
5. Which of the following flight instruments provides navigation-derived velocity and aerodynamic information?
 - A. HSI
 - B. AMI
 - C. ADI
 - D. AVVI
6. What information does the ADI provide to the crew?
 - A. Commanded attitude from guidance
 - B. Attitude errors
 - C. Commanded attitude from navigation
 - D. A and B above
 - E. B and C above

7. What information does the HSI provide to the crew?
- A. How well guidance is performing to achieve the targeted trajectory plane
 - B. How well the vehicle is performing to achieve the desired attitude
 - C. Vehicle rate error
 - D. How well the vehicle is performing to achieve the desired trajectory
8. How is abort guidance established?
- A. Abort rotary switch
 - B. Abort PBI
 - C. Via CRT inputs
 - D. Either A and B above or C above
9. Why are the main engine shutdown pushbuttons on panel C3 important to guidance and flight control?
- A. They are used to inform the software of an SSME failure.
 - B. They are used to manually shut down an SSME.
 - C. They are used to inhibit redline limit shutdown on all SSMEs.
 - D. They are used to enable redline limit shutdown on all SSMEs.
10. If automatic guidance is unconverged, how would the crew fly an ascent or abort manually?
- A. By substituting guidance software with flight control software
 - B. By selecting manual steering and using the flight instruments, trajectory CRT displays, and flight data file to obtain the information necessary to fly the proper trajectory
 - C. By engaging the backup flight system
 - D. None of the above

Answers

1. D. There is not a requirement for these cockpit switches (page 6-3).
2. B. Both indications should be used, as an interconnect may already be established for other reasons, such as an abort dump (page 6-6).
3. C. The CSS PBI must be depressed before automatic steering is replaced with manual steering from the RHC. The correct panel for this PBI is F2 or F4, not C3 (page 6-8).
4. C. The takeover pushbutton is located on the SBTC, not panel F4. The commander's SBTC is not enabled in powered flight (page 6-8).
5. B. The AMI provides navigation-derived velocity and aerodynamics information (page 6-19).
6. D. Commanded attitude and attitude error from guidance are displayed by the ADI. Vehicle rates are indicated as well (page 6-10).
7. A. The HSI indicates the heading of the vehicle and velocity vectors with reference to the target trajectory plane. The difference between the target trajectory plane and the current trajectory plane provides an indication of guidance health (assuming navigation software is good) (page 6-13).
8. D. Abort guidance is scheduled in one of two ways (pages 6-21 and 6-33).
 - By selecting the appropriate abort with the abort rotary switch and depressing the abort PBI
 - Via CRT displays
9. A. There are situations when guidance and flight control cannot recognize that an SSME has failed. In these situations the main engine shutdown pushbuttons are used to inform the software of the engine failure(s) (page 6-24).
10. B. This topic was not specifically addressed. Good job if you answered this question correctly (pages 6-10, 6-13, and 6-25).

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Section 7: BFS Differences

Objectives

After reading this section, the student should be able to do the following:

- Explain why, prior to engagement, BFS guidance and flight control software must perform and duplicate all computations and commands independently of the PASS
- List the functions that are not capable of support during postengagement

Since the BFS is constrained by the same vehicle performance, controllability, and structural limitations as the PASS, very few functional differences exist between the two systems. This section will discuss the differences that do exist. The differences are summarized below.

Prior to engagement, BFS guidance and flight control software must perform and duplicate all computations and commands independent of the PASS. This enables the BFS sequencers and events to follow the PASS to allow BFS engagement at any point in the ascent trajectory. Without this independence, bad guidance and/or flight control commands would be transferred to the BFS at engagement, possibly rendering it useless.

Postengagement, there is no capability to support the following functions:

- Pre-MECO manual steering
- Manual throttling

- Single-engine roll control (this function is planned to be added sometime in the future)

Without these functions, contingency abort capability is lost during powered flight.

Unless a launch hold is desired, it would be absolutely stupid to engage the BFS prior to lift-off. For this reason, BFS flight control software does not have the capability to perform prelaunch actuator slew checks. In fact, the BFS software does not have the capability to command any of the actuators prior to SRB ignition.

If, for whatever reason, fast separation is selected in MM 102, BFS flight control will freeze the current vehicle attitude from the time the MECO command is issued until entry into MM 602. This is necessary because manual steering capability does not exist until MM 602.

Subsequent to BFS engagement, the pilot's flight instruments are slaved to indicate what the commander has selected. Also, only two CRTs are immediately available for trajectory monitoring (a third or fourth CRT can be assigned if desired).

The BFS software code organization and, in some circumstances, its calculation of certain parameters is different from the PASS. For example, BFS guidance calculates orbiter plus ET mass using the middle value of the measured SSME chamber pressures, whereas the PASS performs this calculation using the commanded SSME throttle setting. These differences should not lead to any significant functional/operational differences. For the most part, the BFS calculates parameters and develops

commands almost exactly as the PASS does. However, it is a good idea to be aware that the BFS does execute its software differently on occasion.

Once the BFS is engaged subsequent to SRB ignition, ascent flight control will implement and process the control laws (discussed in section 5) that convert navigation inputs, guidance commands, vehicle attitude, attitude rate, and acceleration into effector commands in much the same way as the PASS.

Appendix A: Suggested Reference Documents

1. The following training manuals referenced in the main text cover related space shuttle systems, guidance, navigation, and flight control subjects:
 - ATO/AOA/TAL 2102
 - Contingency Abort 2102
 - FCS/Effectors 2102
 - Flight Control System Supplemental 2102
 - Inertial Measurement Unit 2102
 - Main Propulsion System 2102
 - MPS Thrust Vector Control 2102
 - Navigation 2102
 - Quaternions Supplemental 2102
2. Flight procedures handbooks provide detailed explanations of crew procedures for all phases of flight. The Ascent/Aborts Flight Procedures Handbook contains relevant procedures to the information presented in this manual.
3. The STS Operational Flight Rules document outlines preplanned decisions for failure situations.
4. The space shuttle software drawings contain detailed information on PASS and BFS software code execution logic.
5. The Shuttle Operational Data Book (SODB) is a compilation of data on all shuttle systems.
6. The Guidance and Control System Brief provides hardware and software operational information.
7. The Functional Subsystem Software Requirements (FSSR) document provides detailed descriptions of the primary flight software. It is divided into separate books on various segments of the software. The following are relevant to ascent guidance and control:
 - GNC - Guidance Ascent/RTLS
 - GNC - Flight Control Ascent, volume I
 - Effectors
 - Sensor/Controller
 - Sequencing
8. The BFS Program Requirements Document (PRD) serves the same purpose as the FSSR for the BFS software. Three volumes are of relevance to ascent guidance and control:
 - Guidance and Navigation
 - Flight Control
 - GNC Hardware Interface Programs
9. Volume II of the NSTS Flight Software Initialization Load Document specifies the flight-specific I-load requirements for the PASS and BFS software.

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M	mass	R	roll
M	mass flow rate	R/V	range/velocity
M50	Mean of 1950	RCS	reaction control system
MM	major mode	REF	reference
MAN	manual	RGA	rate gyro assembly
MDM	multiplexer/demultiplexer	RGO	range to go
ME	main engine	RGO	range to go position vector
MECO	main engine cutoff	RGO _x	radial component of range to go
MET	mission elapsed time	RGO _{xy}	radial and crossrange range to go
MPS	main propulsion system	RGO _y	crossrange component of range to go
MTVC	manual thrust vector control	RGO _z	downrange component of range to go
		RHC	rotational hand controller
n. mi.	nautical mile	RIB	right inboard
NSTS	National Space Transportation System	RM	redundancy management
N _y	lateral acceleration	RPL	rated power level
N _z	normal acceleration	RTL	return to launch site
OI	operational increment	SBTC	speedbrake/thrust controller
OMS	orbital maneuvering system	SE	single engine
OPS	operational sequence	SERC	single-engine roll control
		SODB	Shuttle Operational Data Book
P	pitch	SOP	subsystem operating program
PASS	primary avionics software system	SPD BRK	speedbrake
PB	pushbutton	SPI	surface position indicator
PBI	pushbutton indicator	SRB	solid rocket booster
P _c	chamber pressure	SSME	space shuttle main engine
PEG	powered explicit guidance		
PFG	powered flight guidance	\dot{T}	thrust turning rate vector
PLT	pilot	\dot{T}_C	current thrust vector
PO	overhead panel	TGO	time to go
PPA	powered pitcharound	\dot{T}_M	MECO thrust vector
PPD	powered pitchdown	\dot{T}_R	reference thrust vector
PRD	Program Requirements Document	TAEM	terminal area energy management
PRL	priority rate limiting	TAL	transoceanic abort landing
PRTL	powered return to launch site	THROT	throttle
psf	pounds per square foot	TMECO	mission elapsed time of main engine cutoff
		TVC	thrust vector control
\bar{q}	dynamic pressure		
\bar{q} alpha	dynamic pressure in pitch plane		
\bar{q} beta	dynamic pressure in yaw plane		

Appendix C: Acronyms, Abbreviations, and Symbols

α	angle of attack	F	forward
AA	accelerometer assembly	FCS	flight control system
ADI	attitude direction indicator	FDF	flight data file
AMI	alpha/Mach indicator	fps	feet per second
AOA	abort once around	FSSR	Functional Subsystem Software Requirements
APU	auxiliary power unit		
ASA	aerosurface servoamplifier	g	gravity, acceleration
ATO	abort to orbit	G/C	guidance/control
ATVC	ascent thrust vector control	GN&C	guidance, navigation, and control
AUTO	automatic	GPC	general purpose computer
AVVI	altitude/vertical velocity indicator	GRTL	glide return to launch site
β	sideslip angle	H	altitude
BFS	backup flight system	H	altitude rate
		H	altitude acceleration
c	center	HA	height of apogee
c.g.	center of gravity	HAC	heading alignment circle
CA	constant acceleration	HP	height of perigee
CDR	commander	HSI	horizontal situation indicator
cmd	command		
CRT	cathode-ray tube	I-load	software initialization load
CSS	control stick steering	IMU	inertial measurement unit
		INRTL	inertial
DAP	digital autopilot	IY	orbital plane
ΔP	pressure difference		
ΔR	position difference	k	1000
Δt	time difference	KEAS	knots equivalent airspeed
ΔV	velocity difference	KSC	John F. Kennedy Space Center
EAS	equivalent airspeed	L/O	lift-off
ET	external tank	LIB	left inboard
		LOC	loss of control
		LPS	launch processing system
		LVLH	local vertical/local horizontal

V velocity vector
VEX exhaust velocity
VGO velocity to go
V_i inertial velocity
V_{rel} Earth relative velocity
Y yaw

