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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE

JANUARY 24, 1972

APOLLO MISSION TECHNIQUES

MISSION J-2 AND MISSION J-3
UPDATE

TECHNIQUES DESCRIPTION

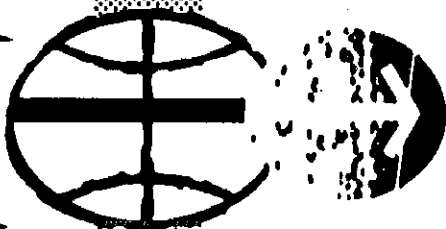
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MSC INTERNAL NOTE

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MISSION J-2 AND MISSION J-3 UPDATE
TECHNIQUES DESCRIPTION

January 24, 1972

SUPERSEDES
APOLLO MISSION TECHNIQUES
MISSION J-1 UPDATE
MSC-04278
May 26, 1971

APOLLO SPACECRAFT PROGRAM OFFICE
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Approved: _____

Philip C. Shaffer
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Apollo Data Priority Coordination

FOREWORD

This Apollo Mission Techniques Update Document and the reference set of techniques documents listed below describe the J-2 and J-3 (Apollo 16 and 17) Mission Techniques.

Mission H-2 and Subsequent

- Launch Phase Aborts
- Earth Parking Orbit and Translunar Injection
- Translunar Midcourse Corrections and Lunar Orbit Insertion
- Lunar Descent
- Abort from Lunar Powered Descent and Subsequent Rendezvous
- Lunar Surface Phase
- Lunar Powered Ascent
- Transearth Injection, Midcourse Corrections, and Entry
- Contingency Procedures

Mission H-3 and Subsequent

- Manual Ascent
- Tracking Data Selection Controllers Procedures
- Attitude Reference Initialization

Mission J-1

- Lunar Orbit Activities

These documents contain the officially approved guidance and control sequence of events, data flow, and real-time decision logic for the J-2 and J-3 missions. The purpose of these documents is to ensure compatibility of all related MSC and supporting contractor activities.

A data priority working group has been established under the direction of Chief, Apollo Data Priority Coordination, ASPO. This group, which is comprised of representatives of MSC and support contractors, meets to coordinate the various associated activities and develop agreed-upon mission techniques. TRW assists in the development of the techniques and documents them for ASPO. Changes in Mission Techniques after publication of this update will be documented in Data Priority Coordination memos authored by PA/Chief, Apollo Data Priority Coordination.

This document supersedes the Apollo Mission Technique Mission J-1 Update, MSC Internal Note MSC-04278, May 26, 1971.

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1. INTRODUCTION

1.1 SUMMARY OF UPDATED MISSION TECHNIQUES

The purpose of this report is to present the J-2 and J-3 missions updates to the basic set of mission techniques description documents (References 1 through 13). Each section of this report presents the updated techniques for the corresponding mission phase document in the referenced set. The majority of the updates involve mission-dependent parameters or corrections and improvements to existing techniques with two notable exceptions:

- Pre-PDI Timeline (Section 5.1) which changed Descent Aborts (Section 7.1)
- NO SLA SEP Procedures (Section 13.1)

The updated techniques are accompanied by supporting graphs, tables, and parameter values; however, since the purpose of this document is to present the techniques for both Apollo 16 and 17, only typical parameter values are shown (usually for a March 17, 1972 Apollo 16 launch). The mission-dependent parameters are identified by mission and launch date to alert the reader to their applicability should the launch date slip.

Although each section presents a review of the major events within the particular mission phase prior to discussing the updates, it is not the intent of this document to replace the baseline set of mission techniques documents. To aid the reader in locating the baseline information, the first sentence of each section indicates the reference (baseline) document and each subsection title is accompanied by the page number of the reference document in which the subject is discussed.

1.2 SPECIAL ITEMS

This section discusses specific techniques or mission activities that overlap mission phases.

1.2.1 Real-Time GET Adjustment*

The time between lift-off and LOI changes from day to day because the geometry changes between the launch site and the TLI point and because

* Reference 14

the earth-moon distance changes. The translunar trip time can vary several hours from month to month, and the T - 24 hour launch opportunities result in about 1 day between LOI and DOI being spent waiting for the sun to rise at the landing site. The amount of detailed trajectory information required premission is greatly reduced if parameters such as GET of MSFN AOS/LOS, sunrise/sunset, and time of arrival at selenographic targets do not change. It is also desirable to maintain the same lunar ground tracks for a given mission, independent of launch day or month.

In real time, the trajectory is sensitive to all premission sources of event time changes plus trajectory dispersions. The real-time GET adjustment to alleviate most of the premission and real-time updating of these data is accomplished by changing GMTLO (GMT of lift-off) by an amount equal to the change in GET. This change will occur in the RTCC, LGC, and CMC, but not in the Remote Site Data Processor (RSDP). The RTCC, LGC, and CMC time must be kept synchronized for the transmission of valid state vector information between these computers. The RSDP time base will not be changed so that the CSM and LM systems data are chronologically continuous.

The GET update criteria is 1 minute for the translunar coast (TLC) phase and 2 minutes for the lunar orbit phase. Three GETLO updates are scheduled: the first, at about 53 hours:30 minutes GET; the second, prior to LM activation during revolution 4 (about 83 hours GET); and the third, during revolution 60 of lunar stay (about 193 hours GET).

- a) At about 53 hours:30 minutes, the GET of arrival at lunar longitude of 180° in revolution 2 of lunar orbit will be predicted and compared to the flight plan time. If the change is greater than 1 minute, GMTLO will be changed so that the GET of longitude arrival and all events in that time interval will be equal to that of the flight plan.
- b) Prior to LM activation during revolution 4 (about 83 hours GET), the GET of arrival at longitude of 180° will be predicted for revolution 26. If this GET is more than 2 minutes different from the flight plan, GMTLO will be updated to correct the GET of longitude arrival and all events in that time interval.

- c) During revolution 60 of the lunar stay (about 193 hours GET) the GET of arrival of longitude of 180° in revolution 66 will be predicted. If this GET is more than 2 minutes different from the flight plan, GMTLO will be updated to correct the GET of longitude arrival and all events in that interval.

Potentially, there are other opportunities to correct event times, but in practice, these will probably not be exercised in order to minimize the number of time base changes. (A single GET adjustment was required during Apollo 14, and no adjustment was required during Apollo 15.)

1.2.2 REFSMMAT Selection

Table 1-1 presents the REFSMMATS to be used for certain mission activities.

Table 1-1. REFSMMAT Selection

<u>Mission Activity</u>	<u>REFSMMAT</u> *	<u>Remarks</u>
MCC-4	PTC	PTC used unless MGA > 60 deg, then MCC-4 and SIM-bay door jettison with MCC-4 PREFERRED (0, 0, 0)
SIM-bay door jettison	PTC	PTC used unless MGA > 60 deg or PREFERRED used for MCC-4, then SIM-bay door jettison with MCC-4 PREFERRED (0, 0, 0)
LOI**	PREFERRED (0, 0, 0)	This REFSMMAT is computed at about 60 hours GET when the LOI abort chart is updated. (Uplinked to CSM after SIM-bay door jettison)
DOI	LANDING SITE	Loaded on the pass after LOI and used for the alignment on the first night pass. Maintain until plane change unless a special REFSMMAT is required for SIM-bay operations
TEI	PREFERRED (180, 0, 0)	

* All REFSMMATS are computed by MCC-H.

** Prior to Apollo 15, MCC-4 and LOI used the LANDING SITE REFSMMAT.

2. LAUNCH PHASE ABORTS

This section discusses the updates to the Launch Phase Abort Techniques (Reference 1). Following a brief discussion of the launch opportunities and objectives, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

The countdown will allow a launch on time using azimuth limits of 72 to 96 degrees which causes only a slight launch profile change from Apollo 15 (80-degree azimuth). Mission planning allows launch attempts during each of 3 consecutive months. During the first and second months, T - 0 and T + 24 hours launch opportunities are available. The planning for the third month includes launch opportunities for T - 24, T - 0, and T + 24 hours.

2.1 SATURN V LAUNCH ABORT AND CONTINGENCY MANEUVER DESCRIPTION REVIEW (Page 2-1)

A complete description of the launch abort modes are found in the baseline document; however, Figure 2-1 identifies these modes and presents the updated launch timeline and contingency capabilities. There are two contingency orbit insertion (COI) updates that should be noted in conjunction with Figure 2-1. First, the Mode IV COI maneuver consists of only one SPS burn manually steered or at fixed attitudes (SCS). Prior to the H-3 mission, the COI Mode IV maneuver could have been one or more fixed-attitude (SCS) burns. The second item, which affects all COI maneuvers (AK, MTO, and ESS) is the lowering of the safe perigee altitude from 75 nautical miles (pre-Apollo 15 limit) to 70 nautical miles (Reference 15). Figures 2-2 and 2-3 show typical suborbital abort and COI capabilities. MTO is not shown in these figures since it is available at anytime during the launch phase if a guidance reference failure occurs.

2.2 ABORT AND CONTINGENCY MANEUVER CUES

2.2.1 Onboard Abort and Contingency Maneuver Cues (Page 3-1)

A manual abort or contingency maneuver decision is based on two onboard cues. The crew abort and contingency maneuver limits are specified in the baseline document.

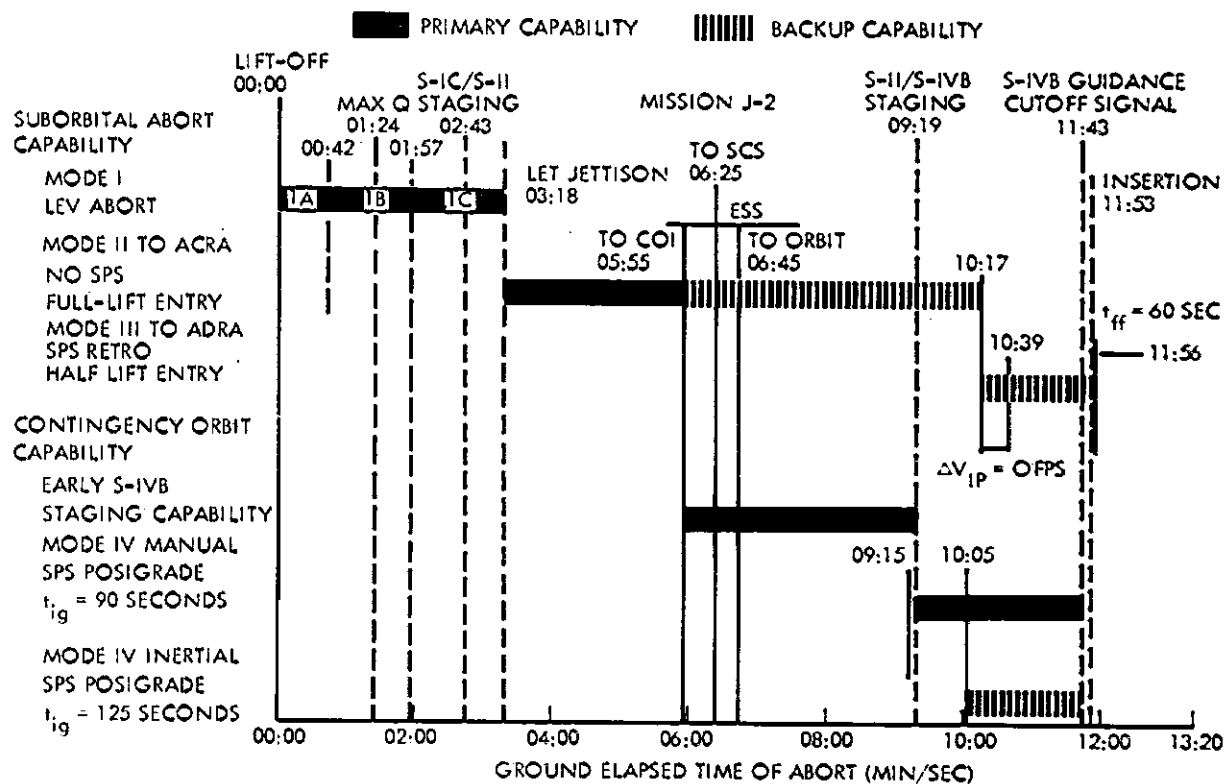


Figure 2-1. Typical Launch Timeline and Contingency Capability (Reference 16)

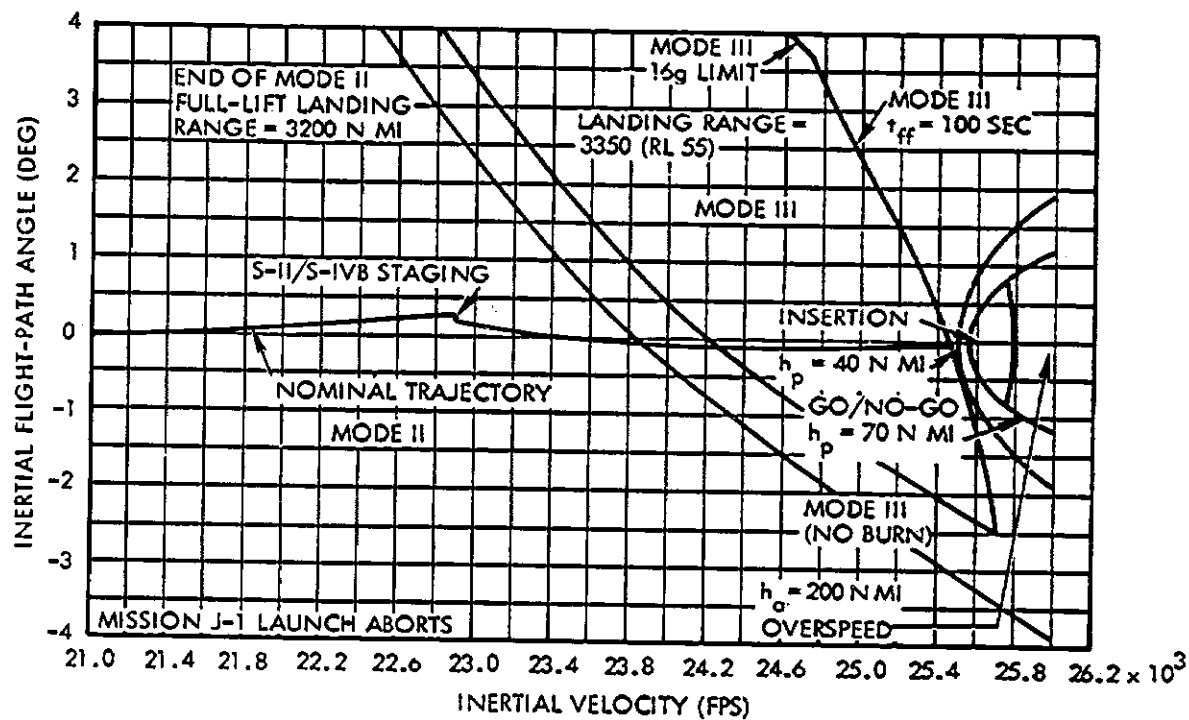


Figure 2-2. Typical Suborbital Aborts (Reference 16)

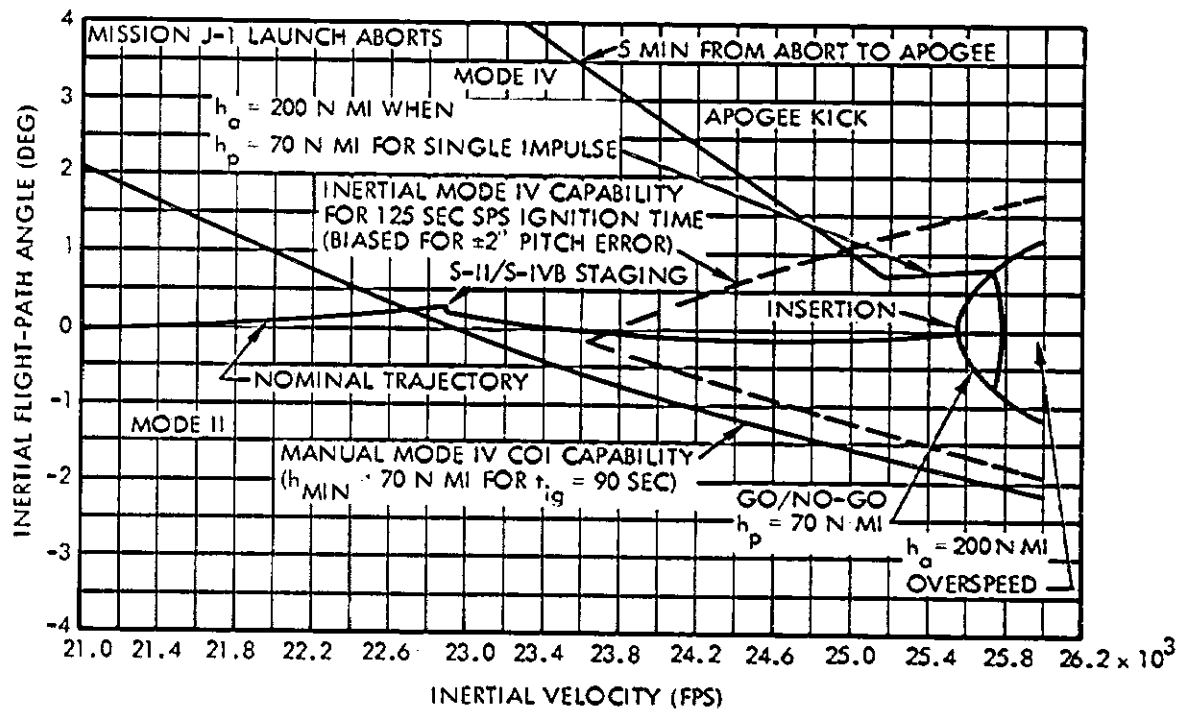


Figure 2-3. Typical Contingency Orbit Insertion (Reference 16)

A flow chart of the various combinations of two onboard cues which might result in exceeding an abort limit is shown in Figure 2-4. The figure contains several updates to the reference flows:

- a) If the LV GUID LIGHT comes on, the crew no longer waits for a secondary cue; rather, they execute MTO.
- b) Following confirmation of two cues with MCC-H, the reference flow indicates decisions to be made concerning Mode I and Mode II aborts. The updated version of the flows have this information in the Abort and COI Mode Decision Flow Chart (Section 2, 3).
- c) The logic for disabling the EDS, if one engine is out or at 2 minutes GET, has been added to the final page of Figure 2-4.
- d) The Max Q region abort limit no longer requires the GUID LIGHT ON or S-IC control engine out after 0 + 50 GET cues. Now, if the GET is less than 75 seconds, the attitude limit (5 degrees) and the angle of attack (AOA) limit (100 percent) are valid Mode I abort cues; however, a crew abort action is not necessary because over rates (4 degrees/second) will occur and the EDS will initiate an automatic abort. After 75 seconds, if a 5-degree error in any axis occurs and an AOA greater than 100 percent occurs, a Mode I abort is executed by the crew. Although overrates may occur, which would result in an automatic EDS abort, launch-escape-vehicle constraints are violated; therefore, the Mode I abort is initiated as soon as the two cues are present.
- e) Due to electrical reconfiguration of the S-IVB cutoff discretes, the S-IVB upstage switch cutoff signal is now issued simultaneously with the guidance cutoff command (Reference 17). Therefore, it is no longer meaningful to manually back up the guidance cutoff with the upstage switch. Cutoff will now be backed up manually with the THC, which was previously a second level backup. As usual, the THC must be returned to detent in less than 3 seconds to prevent CSM SEP.

2. 2. 2 MCC-H Abort and Contingency Maneuver Request (Page 3-9)

In addition to the onboard abort cues, MCC-H can command abort lights A and B (two redundant lights) ON, which are regarded by the crew as a single abort cue. An abort light ON plus a MCC-H voice request to abort results in an immediate abort by the crew. The KSC Launch Operations Manager can send an abort request from the time the LET is

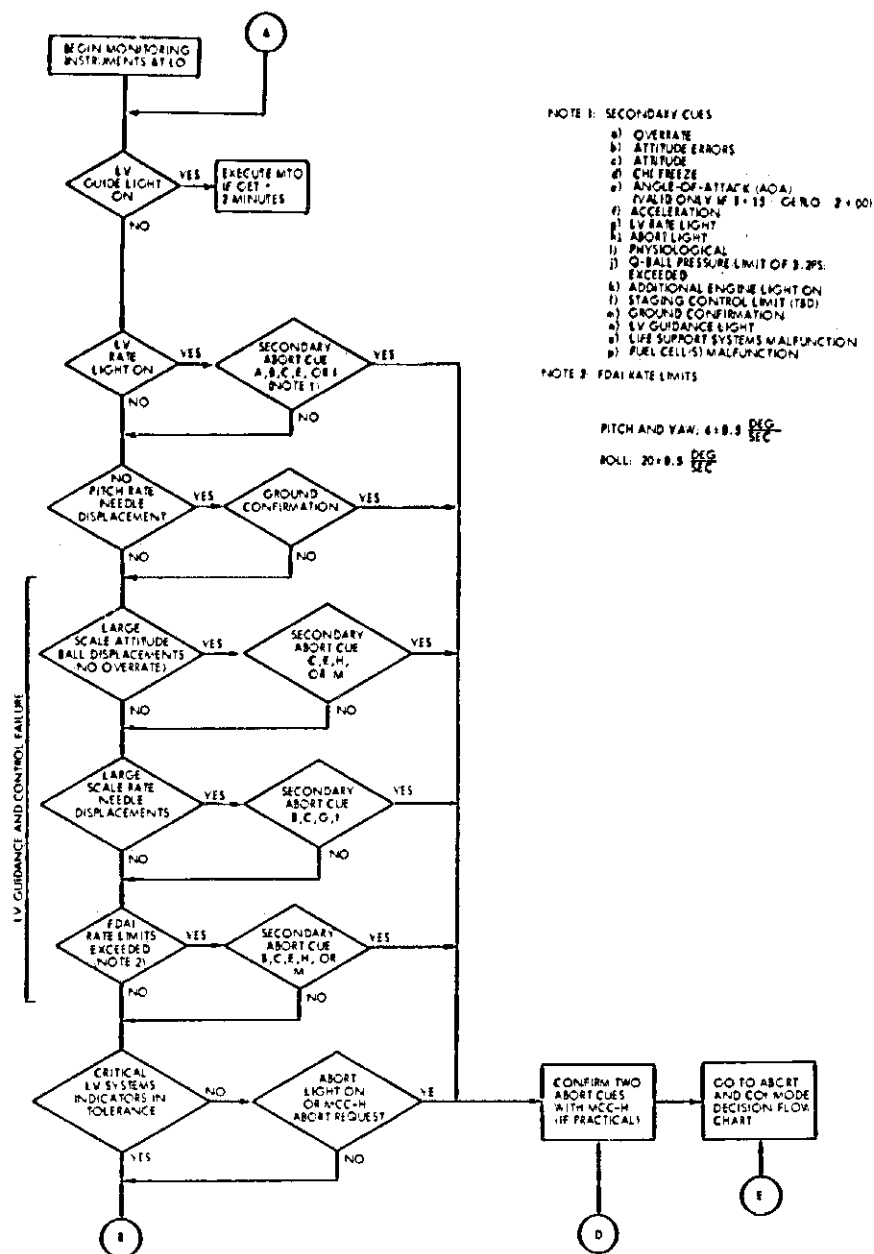


Figure 2-4. * Onboard Launch Abort and Contingency Maneuver Cue Flow Chart

* This figure replaces Figure 3-1 in Reference 1.

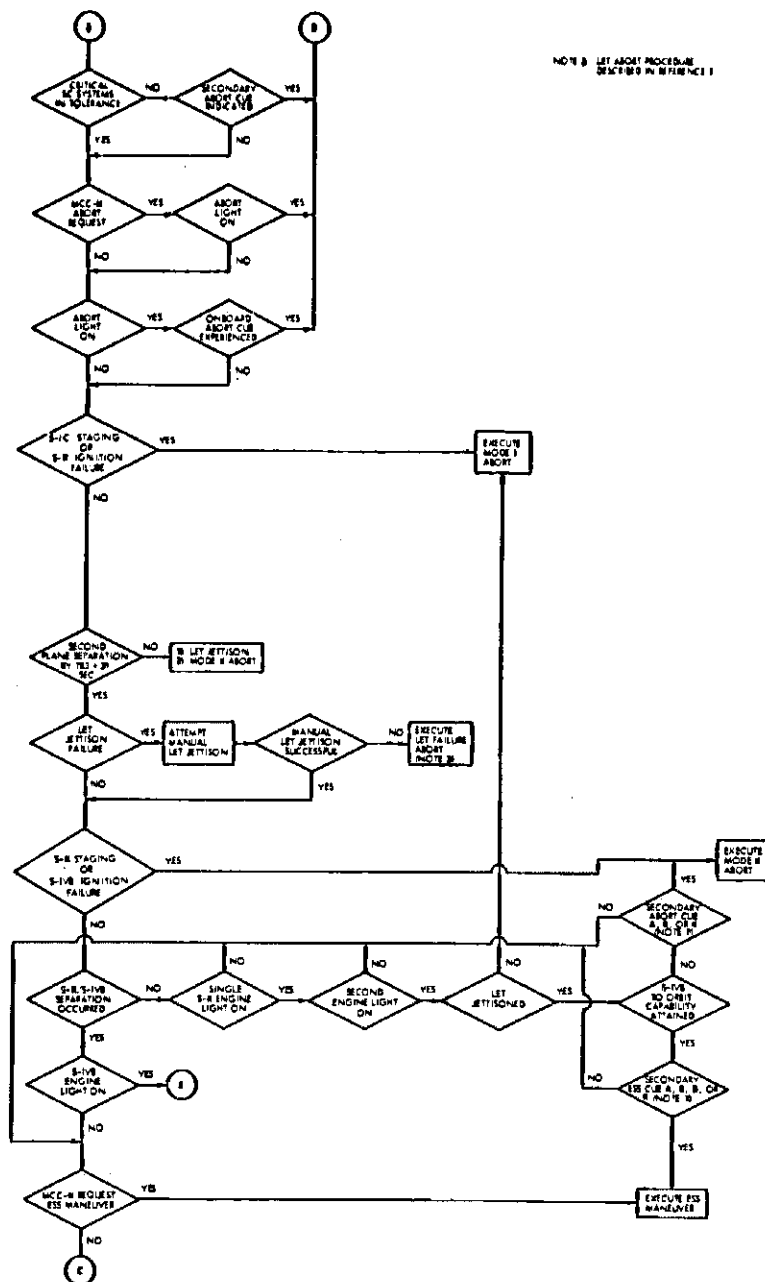


Figure 2-4. * Onboard Launch Abort and Contingency Maneuver Cue Flow Chart (Continued)

* This figure replaces Figure 3-1 in Reference 1.

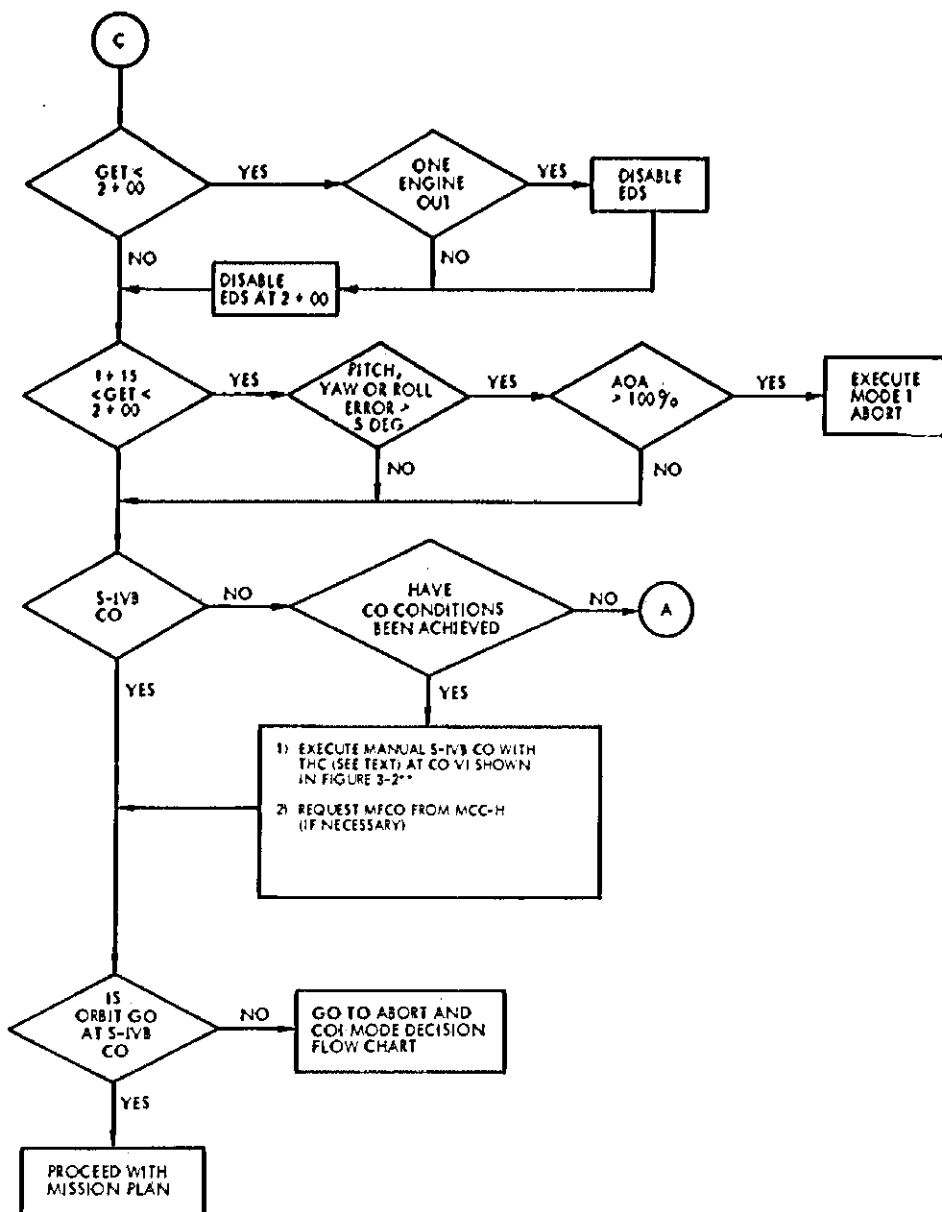


Figure 2-4. * Onboard Launch Abort and Contingency Maneuver Cue Flow Chart (Continued)

* This figure replaces Figure 3-1 in Reference 1.

** Reference 1

armed until the vehicle clears the top of the umbilical tower (about 0 + 10 GET). The Range Safety Officer (RSO) can also shut down the vehicle by transmitting the main fuel cutoff (MFCO) command which also lights the abort request light in the spacecraft. Figure 2-5 shows the updated logic flow for MCC-H initiated abort/contingency request. The major change is as follows:

At one time, the aborts light could indicate either an abort or COI and would subsequently be turned off for a COI. The lights now will be turned on only for an abort and will not be turned off later.

2.3 ABORT AND COI MODE DECISION FLOW CHARTS (Page 4-1)

The updated Abort and COI Mode Decision Flow Chart (Figure 2-6) is based on the assumption that two valid abort or COI cues have been ascertained by the crew. It shows the crew and ground procedures plus the data needed by the crew to execute the proper maneuver.

2.3.1 Onboard Procedures (Page 4-1)

The Mode I aborts are relatively straight forward and essentially automatic with no data required by the crew from the ground.

Abort Modes II, III, and Mode IV and AK COI are considered in the flows (Figure 2-6) for the following cases:

Case A: Voice communication between crew and MCC-H exists.

Case B: Voice communication between crew and MCC-H is lost, but the onboard Q&N is operational (i.e., Q&N GO).

Case C: Voice communication between crew and MCC-H is lost, and the onboard Q&N is nonoperational.

Cases B and C have not been changed; however, Case A has. Previously in a Case A situation, the abort or COI mode and the SCS maneuver data were determined by the ground and then voiced to the crew. Now, the SCS maneuver data are voiced to the crew only if the Q&N is NO-GO.

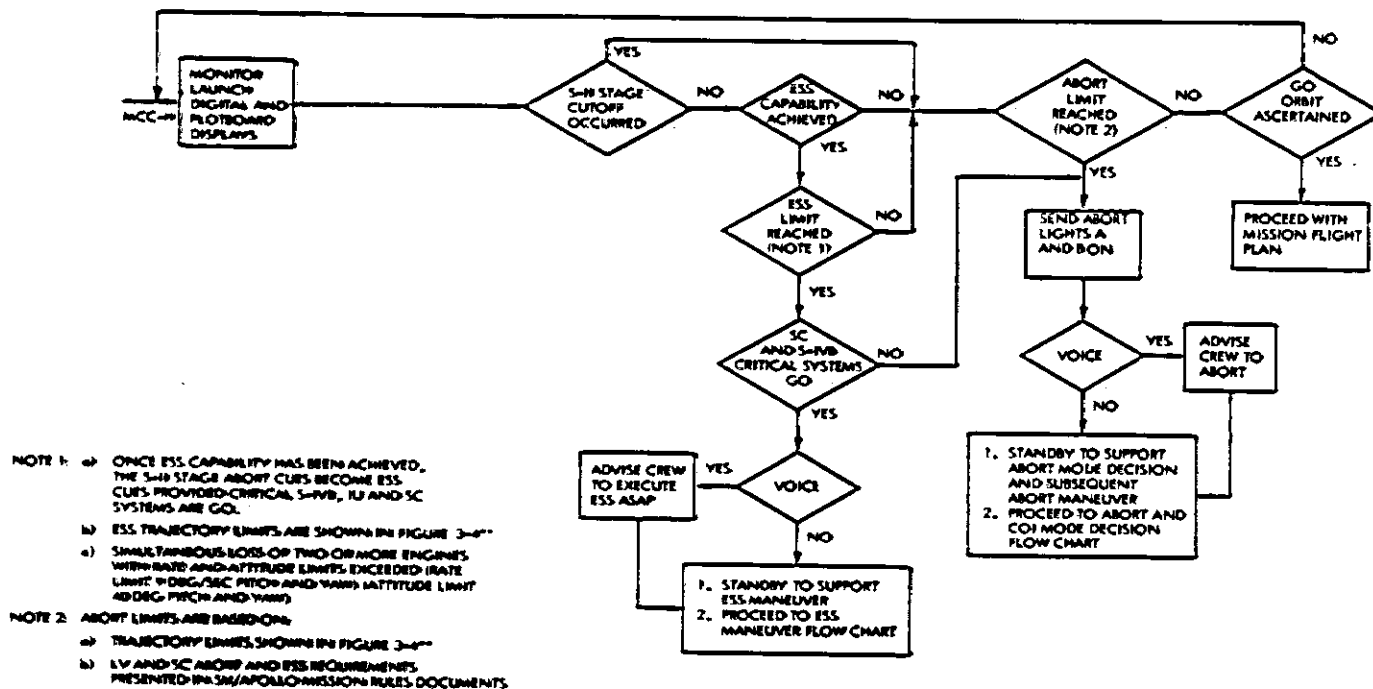
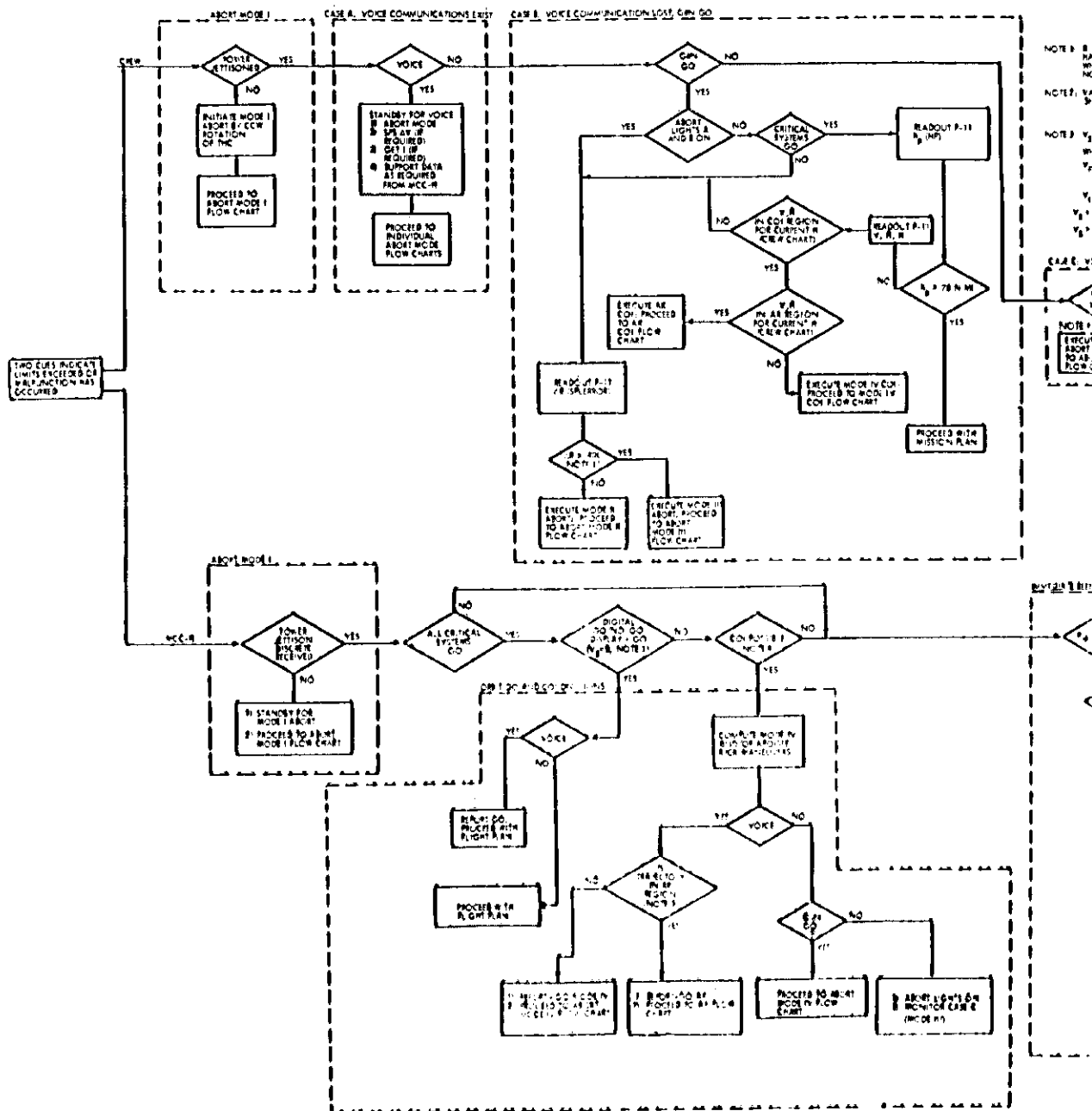
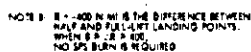


Figure 2-5. * MCC-H Abort and Contingency Maneuver Request Flow Chart

* This figure replaces Figure 3-3 in Reference 1.

** Reference 1 (Page 3-11): Inertial velocity versus inertial flight path angle with limits for heating, structural breakup, maximum entry load, and time of free fall.





NOTE: VARIABLE 24 CORRESPONDS TO SP5 IN
SECTION 2. IN PICTURE 2-4

NOTE 9: $\nabla_1 = \nabla_2 = \nabla_3$

●

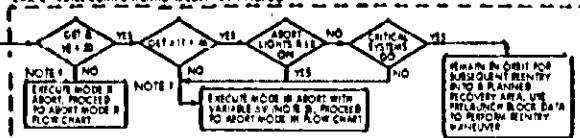
V_p = VELOCITY REQUIRED AT THE PRESENT ALTITUDE h_1 AND FLIGHT PATH ANGLE ψ_1 TO ACHIEVE AN ORBIT HAVING A PERIGEE OF 70 NM

 v_{∞} = PRESENT INERTIAL VELOCITY

75-0-10-701400 CO

70 NMI NO-60

CASE 8: VOICE COMMUNICATIONS LOST CAN NO. GO



NOTE: NO
EXECUTE MODE II
ABORT, PROCEED
TO ABORT MODE II
FLOW CHART

NOTE: NO EXECUTION TO A

REMAIN IN ORBIT FOR
SUBSEQUENT REENTRY
INTO A PLANNED
RECOVERY AREA, USE
PRELIMINARY BLOCK DATA
TO PERFORM REENTRY
MANEUVER

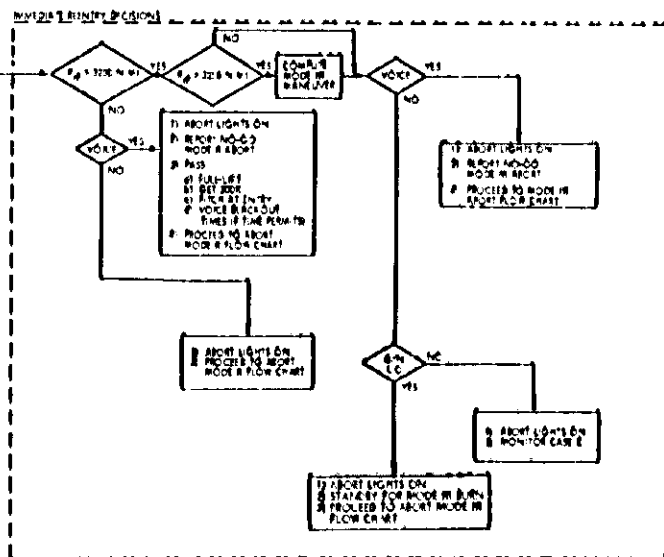


Figure 2-6. Abort and COI Mode Decision Flow Chart (Two Abort Cues Ascertained)

* This figure replaces Figure 4-1 in Reference 1.

Figure 2-6 shows other changes when compared to the baseline flow chart (Figure 4-1 of Reference 1).

- a) All references to "G&N in P00" by MCC-H when the G&N is NO-GO have been deleted.
- b) The GET of 300K' is actually used rather than 400K'.

(Note: GET 300K' should replace GET 400K' throughout the Launch Abort Mission Techniques).

2.3.2 MCC-H Procedures (Page 4-10)

The ground determines the required abort mode from a variety of launch digital and plotboard displays monitored throughout the launch phase. There are two categories of abort mode decisions.

a) Orbit GO and COI Decisions

The orbital GO decision is based on a RTCC computed value of the velocity (V_{ps}) required at the present altitude and γ to achieve a perigee altitude (70 nautical miles). When the inertial velocity becomes equal to, or greater than, V_{ps} , a safe or GO orbit has been achieved.

If a GO orbit has not been achieved and critical systems are GO, MCC-H determines if COI capability exists (COI capability is outlined in Section 2.1).

The discussion of the orbit GO and COI decisions in Reference 1 are still valid with the exception of the Mode IV voice callout data. If the Mode IV maneuver is used, the ground will stand by to back it up with a single impulse burn at about 2 + 05 (if possible) in case the maneuver cannot be initiated prior to 1 + 30. Previously, the Mode IV voice callout data consisted of (1) report GO Mode IV and (2) the SCS maneuver parameters. Now, the maneuver parameters will be passed only in the event the G&N is NO-GO. The voice data are

- 1) Report GO Mode IV
- 2) If G&N is NO-GO, pass:
 - GETI
 - ΔV_{INS}
 - t_B
 - Pitch at Ignition

b) Immediate Reentry Decision

If a COI maneuver is not possible, then an immediate reentry maneuver (Mode II or III) is determined. If the RTCC computed full-lift impact point (R_{IP}) is no greater than 3,200 nautical miles, Mode II will be indicated on the launch digital display. If communications exist, MCC-H will voice the following to the crew:

- 1) Report NO-GO, Mode II abort
- 2) Pass: Full-lift entry profile
GET 300K'*
Pitch at 0.05g*
GET drogue
Recovery time*
Blackout times (if any)

The three items listed above which have an asterisk represent changes or additions in Reference 1 and are reflected in the updated Abort Mode II Flow Chart (Figure 2-7). These changes also apply to the Mode III and IV flow charts. Since these were the only changes for the Mode III chart, that chart is not reproduced. The Mode IV chart has been updated and is presented in the following section.

2.4 ABORT MODE AND CONTINGENCY MANEUVER FLOW CHARTS
(Page 5-1)

The different abort mode flow charts are presented in detail in Reference 1 and the intent of this section is to update only those flows which required substantial changes.

2.4.1 Abort Mode I Flow Chart (Page 5-1)

The reference flow chart does not require any changes; however, it has been pointed out in References 18 and 19 that a water landing is required if the CM RCS is not dumped for early aborts (Mode I). Retention of the CM RCS is planned as a result of the Apollo 15 parachute failure.

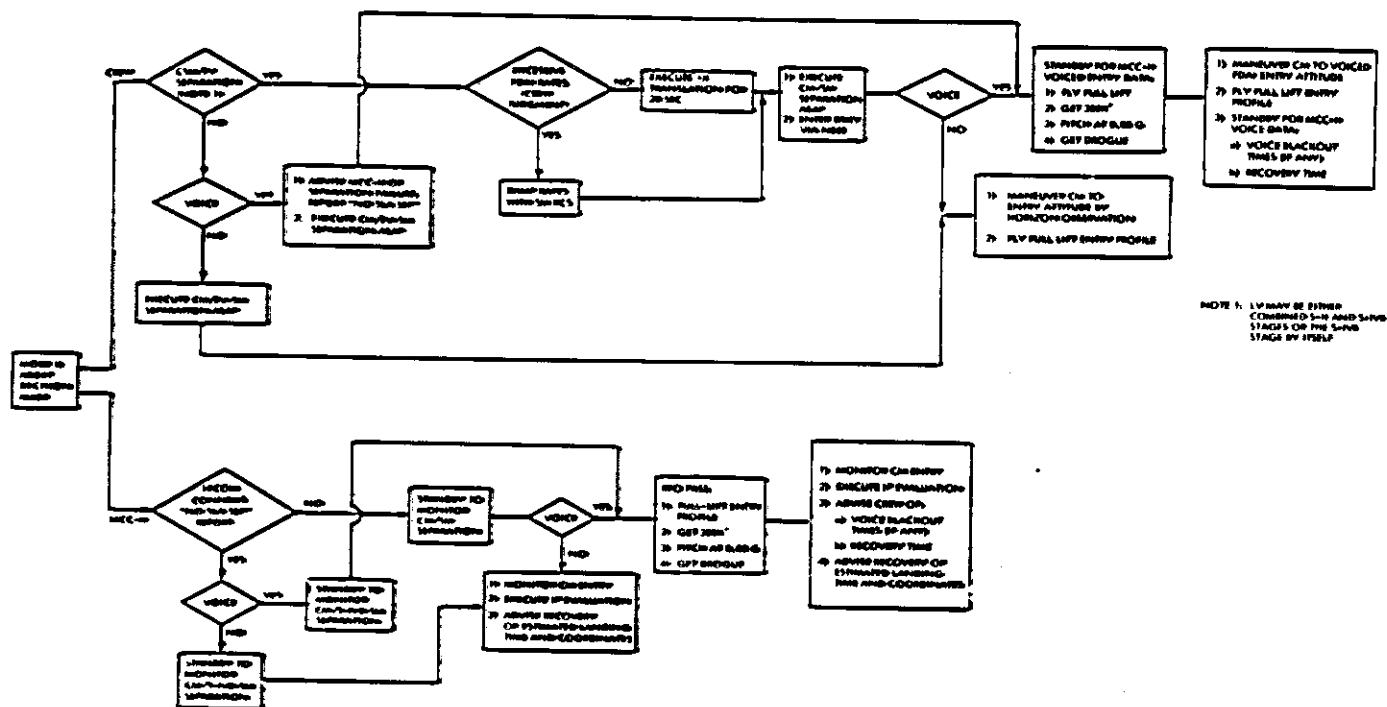


Figure 2-7. * Abort Mode II Flow Chart

* This figure replaces Figure 5-3 in Reference 1.

The Mode I abort procedures given in Table 2-1 will be used for the voice and no voice cases, with Table 2-2 presenting the basis for selecting 75 seconds as the RCS burn/no burn crossover time for the no-voice Mode I aborts. It should be emphasized that these procedures will not cover all abort situations (i. e., launch vehicle dispersions with a tracking or communication failure); however, these procedures cover the most probable error cases.

2.4.2 Abort Mode II Flow Chart (Page 5-5)

The Mode II abort flow (Figure 2-7) was updated to include the updated voice data passed to the crew. This was discussed in Section 2.3.2.

2.4.3 Abort Mode III Flow Chart (Page 5-6)

Although the Mode III Abort Flow Chart is not included in this update, there are two items that should be noted. The first update is that of the data voiced to the crew as was discussed above. The second item concerns changing the time of free fall (T_{FF}) during the burn from 1 + 40 to 1 + 00 seconds. $T_{FF} = 1 + 40$ is based on allowing sufficient time to get to the burn attitude which is already accomplished. The baseline document describes the procedures for CSM/S-IVB separation and SPS failures. Possible SPS burn or entry attitude updates which result from estimated T_{FF} becoming less than 60 seconds are also taken into account. The crew can obtain a T_{FF} estimate from either the DSKY display or by voice from the ground. (MCC-II predicts the T_{FF} estimate at SPS termination.)

2.4.4 Mode IV COI Flow Chart (Page 5-11)

The Mode IV COI Flow Chart and the associated rationale given in the baseline document required major revisions for the J series missions; therefore, rather than listing all the changes, the complete section is replaced by the following paragraphs and Figure 2-8.

Table 2-1. Mode I Abort Procedures (Reference 19)

<u>Time of Abort</u>	<u>Procedure</u>	
	<u>Voice</u>	<u>No-Voice</u>
0-61 (Mode Ia)	Auto dump ¹	Auto dump
61-75	Ground call ²	Burn RCS
76 through Mode I	Ground call	No burn
Mode II	No burn	No burn

- Notes:
- 1 Auto dump means that the RCS is automatically dumped through the aft heat shield
 - 2 Real-time procedure used whereby the ground will advise the crew when the predicted impact point is satisfactory (water landing). This call is based upon the SC's predicted impact point following an abort accounting for both predicted LES impulse and predicted parachute drift (from wind data measured at T - 120 minutes). These data plus tracking uncertainties are used to construct a limit line extending seaward from an acceptable water depth of 10 feet. The ground call assumes the presence of tracking and communication with the spacecraft. S-band tracking during a Mode I abort is not guaranteed and may require switching of the omni antennas during the abort sequence.

Table 2-2. Basis for Selection of 75 Seconds as Burn/No-Burn Crossover for No-Voice Abort (Reference 19)

Analysis Inputs and Assumptions

- Nominal LV state vector at abort (AS-511 preliminary operational trajectory)
- 100-degree launch azimuth (worst case)
- EDS pitch and pitch-rate limits - UP range
- -3 sigma tower misalignment applied to Apollo 15 LES
- Nominal Apollo 15 abort weights and mass properties
- Upper limit on baroswitch (maximum time on mains)
- Wind profile - 99 percentile annual from 65 degrees (worst azimuth)

Analysis Results

- Aborts through 70 seconds - land landing
- Aborts beyond 70 seconds - water landing

Recommend Biasing Crossover Time by 5 Seconds

- Accounts for possible differences between
 - AS-511 preliminary and final OT LV state vectors
 - Apollo 15 and Apollo 16 LM thrust vector alignment
 - Apollo 15 and Apollo 16 weights and mass properties
 - Allows flight crew to train with constant crossover time
-

The nominal Mode IV COI sequence of events after CSM/S-IVB separation consists of the following:

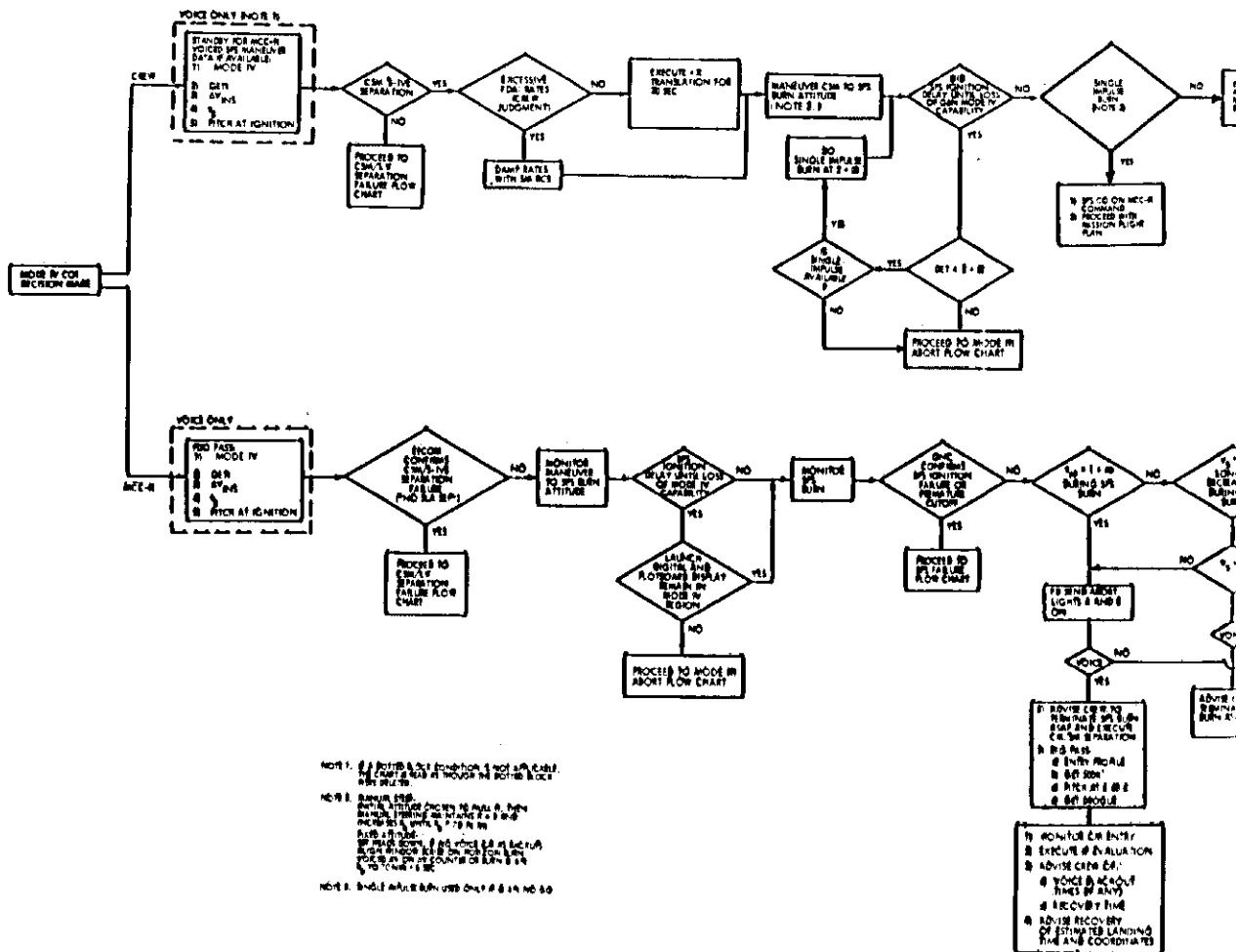
Digital Event Timer	Event
0 + 04	Start ullage +X translation (maintain for 20 seconds)*
0 + 06	If necessary, damp rates with SM RCS
0 + 24	Terminate +X translation. Maneuver CSM to FDAI insertion attitude
ASAP (1 + 30 max)	Ignite SPS thrust (posigrade burn). Burn until required SPS ΔV is achieved.

Like the Abort Mode III Flow Chart, the Mode IV COI Flow Chart (Figure 2-8) considers the possibilities of either a CSM/S-IVB separation failure or an SPS failure. The procedures for these failures are referred to in the CSM/S-IVB separation failure and SPS failure flow charts, respectively. Either failure results in an immediate reentry.

The Mode IV burn should begin as soon after the Mode IV maneuver decision as possible. The crew will pitch the vehicle as required to establish \dot{h} at zero until the appropriate cutoff V_1 (onboard chart) is attained ($h_p \geq 70$ nautical miles). The initial pitch attitude should be biased high to preclude h_p becoming less than 70 nautical miles. This Mode IV COI Q&N technique is the prime method. It assumes the presence of an acceptable Q&N and the capability to perform a SCS burn. If the Q&N technique is not possible, the ground will pass a SCS fixed attitude single impulse burn at 2 + 05 GETI if possible. Figure 2-8 shows the logic following a Mode IV COI burn decision.

The COI attempt will be abandoned for an immediate reentry if T_{FF} reaches the 1 + 40 threshold and is decreasing. Once the specified SPS burn is completed, an orbit GO/NO-GO decision is made identically as for an S-IVB cutoff. The procedure followed is the same as that presented in the Abort and COI Mode Decision Flow Chart (Figure 2-6).

*For Mode IV Aborts, a 7-second four-jet ullage is sufficient to guarantee booster separation.





2.4.5 Apogee Kick COI Flow Chart (Page 5-15)

The Apogee Kick COI Flow Chart is the same as the Mode IV COI Flow Chart (Figure 2-8) with the following exceptions:

- a) The AK SPS burn GETI occurs when the CSM reaches the apogee plus 2 minutes* position in orbit as opposed to 1 + 30 after the J-IVB cutoff of the Mode IV.
- b) TFF monitoring is not meaningful because the TFF from the apogee position in orbit is always much greater than the threshold value of 1 + 40.

2.4.6 Other Flow Charts (Pages 5-15 through 5-27)

The pre-TLI CSM/S-IVB separation failure techniques are correct as given in the baseline document. The NO SLA SEP procedures for post-TLI are given in Section 13.1. The techniques in the other flow charts in Reference 1 remain virtually unchanged. These include the SPS Failure Flow Chart, Manual Take-over Flow Chart, and ESS COI Maneuver Flow Chart.

2.5 NO TLI DEORBIT (Page 17-18 of Reference 20)

As discussed in Section 3.1, the circular earth parking orbit altitude was lowered from 100 to 90 nautical miles to increase the launch vehicle payload capability. Lowering the earth parking orbit altitude caused the freefall time (RETEI) to 400,000 feet following a contingency deorbit to be reduced, which impacted the NO TLI SPS deorbit procedures. In a retrofire for a nominal 2-1 (west Atlantic) landing, the value of RETEI was increased by changing the horizon position in the window from 31.7 to 15 degrees (Reference 21). This maneuver attitude reduces the negative radial component of the retrofire maneuver which increases RETEI as shown in the table on the following page.

*Apogee plus 2 minutes is the update rather than apogee (pre-Apollo 15).

<u>Circular Orbit</u> <u>(n mi)</u>	<u>(Heads-up)</u> <u>Window/Horizon</u> <u>(deg)</u>	<u>Burn</u> <u>(ft/sec)</u>	<u>RETEL</u> <u>(min:sec)</u>	<u>H_p</u> <u>(n mi)</u>
100 (Apollo 14)	31.7 (old)	325	10:41	-14
90 (Apollo 15)	31.7 (old)	325	8:42	-22
90 (Apollo 15)	15 (new)	250	9:46	-19

It should be noted that the 31.7-degree line is still used for Mode III and Mode IV launch aborts. Also, Reference 17 states that the 2-1 landing is not available for launch azimuths greater than 86 degrees because there are islands in the footprint.

Specifically, the procedural changes which reflect the 15-degree window mark change are

- a) The TIG - 5 minute horizon check is now at TIG - 3 minutes and the 3-degree window mark is used instead of the previous 12-degree mark.
- b) The TIG - 2 minute backup GDC align angle is now at the 7-degree rather than 24-degree window mark.
- c) The TIG attitude is now at the 15-degree window mark rather than the 31.7-degree window line.
- d) The attitude for CSM/S-IVB separation for pre-TLI deorbits is also changed from the 31.7-degree window line on the horizon to the 15-degree window mark.

3. EARTH PARKING ORBIT AND TRANSLUNAR INJECTION

This section discusses the updates to the Earth Parking Orbit and Translunar Injection Techniques (Reference 2). Following a brief review of the major mission events, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

The launch vehicle will place the spacecraft into a 90-nautical mile circular earth parking orbit. The S-IVB will be reignited during the second revolution to place the vehicle on a translunar trajectory targeted so that a transearth return to an acceptable entry corridor can be achieved with the RCS for at least 5 hours after translunar injection (TLI) cutoff (i. e., up to the time of CSM docking with the LM) and so that an acceptable earth return can be achieved using the DPS until at least pericynthion plus 2 hours if the lunar orbit insertion (LOI) burn is not performed.

3.1 NINETY-NAUTICAL MILE PARKING ORBIT (Entire Document)

The altitude of the circular earth parking orbit has been lowered from 100 to 90 nautical miles to gain payload capability. Lowering this value decreases the Mode IV launch abort capability (Section 2.4.4); however, it increases the launch vehicle payload capability by approximately 700 pounds (Reference 22). The 90-nautical mile orbit also significantly reduces the free-fall time to 400,000 feet altitude following a contingency deorbit, thus requiring some changes in the NO TLI deorbit procedures discussed in Section 2.5.

3.2 TLI GO/NO-GO PHILOSOPHY (Pages 2-1 through 2-3)

The discussion of the TLI GO/NO-GO decisions in the baseline document is correct with the exception of the minimum acceptable apogee criteria for a minimum lunar landing mission. Reference 2 states that "if an accelerometer fails prior to the TLI burn, HOSC will determine the probability that the IU can still guide the booster into a 65,000-nautical mile apogee orbit. If that probability is one percent or more, TLI is GO (Mission Rule)." For Apollo 16, TLI is GO if a reasonable probability (Flight Director's discretion) exists to achieve an apogee of 29,000 nautical miles.

In the presence of TLI underspeeds, the capability to fly a lunar orbit mission is increased on Apollo 16 by decreasing the TLI apogee requirement from 43,000 nautical miles (Apollo 15) to 29,000 nautical miles for Apollo 16. This is equivalent to a 35-second TLI underburn rather than the 23 seconds for Apollo 15. The Apollo 15 value was based on the first mid-course correction (MCC-1) at TLI + 3 hours. Reference 18 shows that lunar orbit capability (correcting for underspeeds) is gained as the time of execution of MCC-1 after TLI is reduced. The allowable underburn (35 seconds) is the result of performing MCC-1 at TLI + 1 hour. This requires transposition, docking, and LM extraction (TD&E) to be completed by TLI + 50 minutes.

3.3 OPTICS GO/NO-GO STATUS CHECK (Page 4-4)

On missions prior to Apollo 15, TLI was NO-GO if an acceptable IMU was not available after a P52. For Apollo 15 and subsequent missions, the SXT and SCT may fail and TLI still be GO. In this case, the CSM COAS will be calibrated and P54 (Backup IMU Realign) will be used to establish an acceptable IMU.

3.4 TLI PREPARATIONS (Page 4-9)

The TLI sequence of events is essentially unchanged except for the event times between insertion and TLI. The TLI preparation consists of testing the EMS ΔV counter and checking that the TLI burn monitoring and G&N switch settings are properly positioned. (The reference document states that the EMS is not required to monitor the TLI burn, which is no longer valid, as discussed in Section 3.6.) The TB6 light will come on soon after MSFN verifies TLI GO and Pyro Arm. This light signals the beginning of the S-IVB restart preparations and remains lighted for 38 seconds. If the TB6 initiate light does not come on, the crew will continue their normal procedures. If the sequence does not actually start, nothing is harmed, and if it starts, there is no reason to stop for a light failure. The digital event timer (DET) is started from about 51:00 minutes, when the light goes out. TLI ignition should occur when the DET reads 00:00 minutes.

Prior to ignition, P15 (TLI Cutoff) is selected, and the DSKY displays the AV components. The CMC enters the average-G routine and the PIPA's are checked. TLI is NO-GO if a PIPA has failed. After this, the crew selects the display of VI, \dot{H} , and H.

3.5 TLI BURN ATTITUDE CHECK (Pages 4-12 and 6-9)

The attitude check (COAS is set to the horizon) performed about 25 minutes after earth orbit insertion (EOI) on missions prior to Apollo 15 has been deleted because it is redundant with MCC-H and crew monitoring throughout the earth parking orbit phase.

3.6 TLI CUTOFF MONITORING (Page 5-1)

The S-IVB TLI burn is monitored by observing spacecraft attitude, attitude rates, and velocity. The burn is terminated if any one of the following limits are exceeded:

- a) Pitch or yaw attitude rate exceeds 10 degrees/second.
- b) Pitch or yaw attitude deviates more than 45 degrees from the nominal.
- c) Auto shutdown has not occurred by 2 seconds after CMC indicates the nominal cutoff velocity.

The first two limits are identical to those stated in the baseline document; however, the third one has been changed from the following:

CMC indicated velocity is greater than its nominal limit at injection (VI PAD data) and if the ET indicates a value greater than the nominal burn time (BT PAD data) plus 6 seconds.

The reason for the update is to reduce the magnitude of a possible overburn which may require subsequent excessive midcourse corrections. The previous technique could result in a cutoff velocity of VI plus the uncertainty of burn time (approximately ± 10 seconds) plus 6 seconds. Terminating the burn at PAD VI plus 2 seconds will insure the capability of a lunar landing in the case of an overburn. Should this cutoff be earlier than desired, the effect is not catastrophic, and a midcourse correction can be performed to obtain the lunar landing capability. For this same reason, the TLI burn is not trimmed.

On Apollo 15, the technique was to burn to depletion in the event of a G&N failure. The rationale for this decision involved the inability of the entry monitoring system (EMS) to adequately support the desired cutoff accuracy and the fact that there was only a few seconds of overburn capability. However, the Apollo 16 vehicle is capable of overburns up to 15 seconds (Reference 17) at TLI; therefore, the EMS will be used for thrust termination if the CMC fails. Dispersion analysis indicates the ground-predicted injection velocity error can be up to 65 feet per second, and the EMS has a 1.3 percent (3 σ) error; therefore, TLI will be terminated when the EMS reads -200 feet per second, and the CMC is failed. The lunar orbit or free return trajectory will then be re-established with TLC midcourse corrections.

For Apollo 15 and subsequent missions, the inertial velocity target for P15 (TLI Cutoff) and the EMS ΔV targets passed to the crew by MCC-II for TLI will not contain the J-2 tailoff ΔV . This means that the CMC V_{GO} and EMS ΔV residuals are nominally equal to the tailoff ΔV after TLI cutoff.

3.7 MINIMUM LUNAR LANDING PRIORITY (New Section)

The minimum lunar landing mission profile is based on the progressive alterations to the nominal profile. These alterations to the nominal profile, given by priority are

- 1) Relax return inclination
- 2) Earliest possible midcourse correction (\sim TLI + 3 hours)
- 3) Give up LOPC-2
- 4) One extra day in transearth coast
- 5) Reoptimize landing-site approach azimuth
- 6) Earliest possible midcourse correction (\sim TLI + 1 hour)
- 7) Slowest possible translunar coast - higher SEA (DPS abort and MSFN coverage limited)

- 8) No post-ascent science (earliest possible TEI)
- 9) Minimum TEI ΔV (If it saves ΔV and stays within 300 hours total mission time)
- 10) Shortest possible lunar surface stay time - 1 EVA (minimizes LOPC-1)
- 11) LM yaw steering on lunar ascent could save about 50 feet per second of SPS fuel

4. TRANSLUNAR MIDCOURSE CORRECTIONS AND LUNAR ORBIT INSERTION

This section discusses the updates to the Translunar Midcourse Correction and Lunar Orbit Insertion Techniques (Reference 3). Following a brief review of the major mission events, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

Shortly after injection, the CSM will be separated from the S-IVB/SLA LM using SM RCS. The CSM will turn around and dock to the LM. Separation of the combined LM/CSM from the S-IVB nominally will be achieved within 2-1/2 hours after injection. The S-IVB will then perform an evasive maneuver to alter its circumlunar coast trajectory clear of the SC trajectory. At approximately 5-3/4 hours after launch, the S-IVB will perform a midcourse correction to achieve a trajectory that results in an S-IVB lunar impact. A second midcourse correction will be performed by the S-IVB at approximately 9-1/2 hours GET which reduces the 3 σ error uncertainty of ± 335 kilometers.

During TLC, SC midcourse corrections will be made, if required, using MSFN for navigation. The SIM-bay door will be jettisoned about 4.5 hours prior to LOI. The SPS will be used for LOI which will place the SC in a 60- by 170-nautical mile lunar orbit.

4.1 SEQUENCE OF EVENTS (Page 1-2)

There are several updates to the reference timeline as shown in Figure 4-1. The major updates are

- LM intravehicular transfer (IVT)
- Suit donning practice
- Star/horizon sighting
- SIM bay door jettison

The first two items are briefly discussed in the following paragraphs, and the last two are discussed in Sections 4.4 and 4.5, respectively.

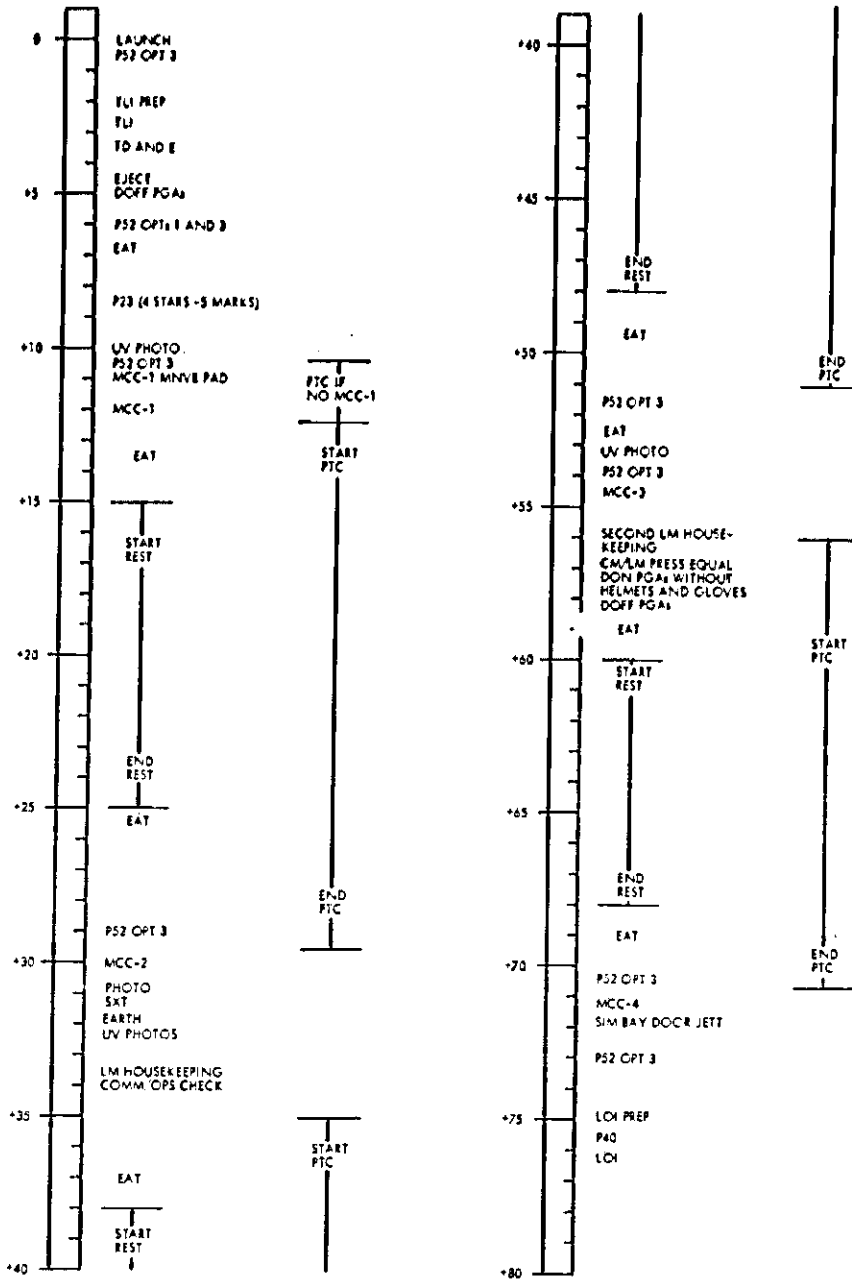


Figure 4-1. Typical Mission Sequence from Launch to LOI

The LM IVT has been moved about a day earlier than shown in the reference document. Moving IVT to about 34 hours GET allows more time for problem solving, should any malfunctions occur. The second IVT is scheduled at about 56 hours GET. The PGA DON/DOFF exercise was moved from 72 hours GET (on Apollo 15) to the second IVT (LM house-keeping period) for subsequent missions. For Apollo 15, the crew was required to be suited up for the jettison of the SIM-bay door. This requirement was dropped for Apollo 16 and 17. The PGA DON/DOFF exercise is still desirable training and moving it back to the MCC-3 period relaxes the timeline just prior to SIM-bay door jettison. Figure 4-2 shows a more detailed timeline for the pre-LOI period.

4.2 OPTICS GO/NO-GO (Pages 2-1 and 5-8)

The first LOI preburn activity consists of performing an IMU alignment (P52) and drift test (Figure 4-2). The alignment is checked using auto optics to a third star. In the event of a CSM optics (SXT, SCT) failure, the LM IMU will be powered up and aligned using P51 and P52. The CSM IMU will then be aligned to the LM IMU (nominal docked alignment). The LM IMU is required because an uncalibrated CSM COAS cannot necessarily provide the desired alignment accuracy and may be blocked by the LM structure. The 3σ alignment accuracy (per axis) for the COAS given in Reference 13 is

Calibrated 0.7 deg

Uncalibrated 2.1 deg

The same reference shows the docked alignment accuracy is 0.81 degree, which is adequate for the subsequent MCC's and LOI. Therefore, LOI is GO in the event of a CSM optical (SXT, SCT) failure. (The docked alignment accuracy can be improved to 0.18 degree using the eigenvector alignment technique discussed in Reference 13, with nominal PTC and MCC attitude maneuvers during TLC.)

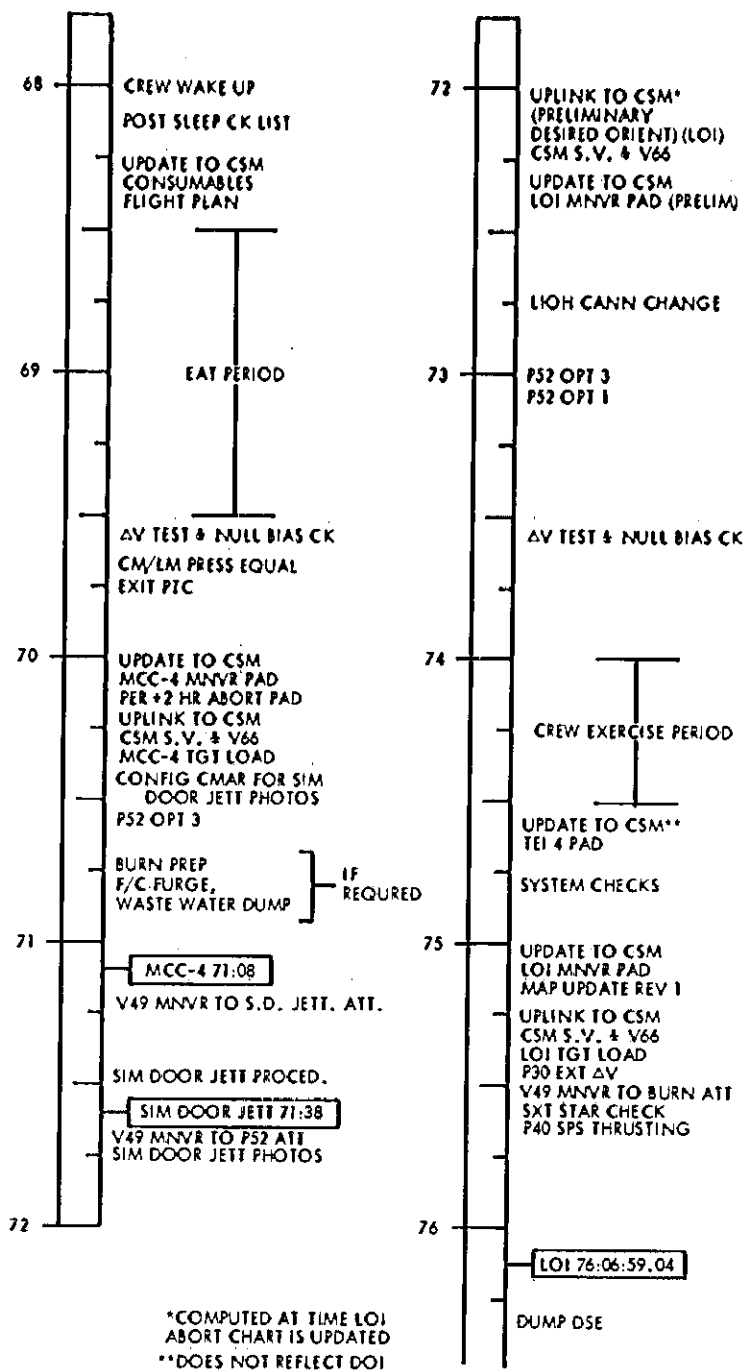


Figure 4-2. Pre-LOI Timeline

4.3 MCC TRIM RULE (Page 3-6)

The MCC trim rules given in the baseline document are correct with the exception of the MCC-4 value. The baseline document states that MCC-4 VGX is trimmed to within 1 foot per second. The value has been lowered to 0.2 foot per second which is consistent with the other MCC values. It should also be pointed out that an SPS MCC maneuver will be terminated if the pitch or yaw attitude rates exceed 10 degrees/second or if the attitude deviates more than ± 10 degrees with shutdown being performed at burn time plus 1 second if required.

4.4 P23 STAR/HORIZON SIGHTINGS (Pages 4-1 and 4-2)

One batch of P23 star/horizon sightings will be made on subsequent Apollo missions. This batch of sightings at 8 hours 30 minutes GET, will consist of five marks on each of four stars; two with the near horizon and two with the far horizon. These are evaluated by MCC-II to determine the earth horizon bias (ΔH) above the Fisher ellipsoid earth model in the CMC. The horizon serves as a reference for the crew in the event of a no-communication transearth and entry (Section 13.5). The TLC P23 sightings correspond to the range of transearth P23 sightings at about EI - 5 hours. Moving this batch earlier than on previous missions allows the sightings to be performed prior to PTC initiation, thus deleting a maneuver. On missions prior to Apollo 15, two batches of sighting were scheduled; however, flight-to-flight variances in the value of ΔH 's indicated that multiple batches are not necessary. Should difficulties develop in the first batch of sightings, another batch would be performed at about 28 hours GET. The horizon bias (ΔH), will be updated only if the calculated ΔH differs from the CMC E-memory value by more than 5 kilometers. The uplink, if required, will occur before TLI + 55 hours.

The ground will provide a P23 sighting attitude which avoids reflected light from the LM. Also, the star selection process will insure that the line of sight is far enough from the sun to prohibit scattered light interference, and the sun elevation angle of the horizon substellar point is greater than 15 degrees.

4.5 SIM-BAY DOOR JETTISON (New Section)*

The SIM will be flown on Apollo 15 and subsequent Apollo missions. This module is mounted within one of the bays of the SM and is protected during launch, earth orbit, and translunar coast by the surface structure of the SM. This surface covering to the SIM bay (commonly referred to as the SIM-bay door) is pyrotechnically severed and jettisoned from the CSM at LOI - 4.5 hours with the CSM plus X-axis pitched down 148 degrees from the local horizontal posigrade, yawed 58 degrees out of plane to the north and rolled right 59 degrees (lunar-referenced). This attitude avoids sunlight in the SIM bay, permits S-band steerable antenna communication with the earth, and results in a middle gimbal angle of 0 degree (PTC REFSMMAT), which precludes the possibility of gimbal lock (Reference 24). Sunlight must not be allowed in the SIM bay if the panoramic camera is unstowed. If the camera is stowed, the allowable time for continuous sunlight in the SIM bay is 13 minutes. Reference 25 presents the results of a thermal analysis of the panoramic camera lens barrel and closeout panel. The critical temperature envelope is contained within the values of

$$\theta = 60 \text{ to } 145 \text{ deg}$$

$$\phi = -10 \text{ to } 110 \text{ deg}$$

where θ and ϕ are the two angles defining the sun line with respect to the spacecraft coordinate system. The attitude is also constrained by a requirement for high-gain antenna (S-band communication) coverage to insure the ground is receiving high-bit-rate data should problems occur as a result of the jettison.

No recontact hazards are anticipated. The relative motion for the normal jettison velocity is 13.7 feet per second which places the SIM-bay door 26 nautical miles above, 5.6 nautical miles behind, and 8.4 nautical miles north of the spacecraft at LOI. Maximum jettison dispersions in direction (± 10 degrees) and velocity (± 2.7 feet per second) do not adversely affect this relative position, which is safe for either a nominal, partial, or

*Reference 23.

zero LOI burns. Reference 26 indicates that SIM-bay door jettison may occur from LOI - 5.5 hours to LOI - 1.5 hours (nominally performed at LOI - 4.5 hours) and poses no recontact hazards. Also, no recontact hazard exists for LOI abort maneuvers.

4.6 LOI IGNITION DELAY (New Section)

The baseline document does not address the problem of a late LOI burn. A late LOI is not anticipated; however, a limit has been established (References 27 and 28). The perilune altitude decreases linearly from about 60 nautical miles (on-time LOI) to impact (132 seconds late LOI). Since 5 nautical miles is the minimum acceptable perilune altitude, the maximum value for a late LOI is 120 seconds. Should this be violated, LOI will not be attempted.

It is possible to recover from a delayed LOI situation with either a one- or two-impulse DOI maneuver sequence (Reference 29). The single-impulse DOI would occur approximately two revolutions after LOI; whereas, the two-impulse sequence would require the first burn about 1-1/2 revolutions after LOI and the second about 1/2 revolution after that. The single burn is preferable in order to maintain the timeline and allow landmark tracking for descent targeting. (The single-burn ΔV cost is about 1-1/2 times greater than the 260 feet per second for the two-impulse with a 120-second LOI delay.) Therefore, the single impulse burn is performed if the SPS reserves allow. In the event the limited tracking data between LOI and first apolune (AOS + 35 minutes) indicate an unsafe perilune, a small burn to raise h_p will also be performed. It should also be noted that a late LOI will probably require a GET update (Section 1.3.1), and a delay in LOI introduces errors into the LOI abort charts (Section 13.3).

5. LUNAR ORBIT ACTIVITIES

This section discusses the updates to the Lunar Orbit Activities Mission Techniques (Reference 4). Following a brief review of the major events in this mission phase, the changes and additions are discussed in the order in which the subjects appear in the baseline document.

Once in lunar orbit, the CSM will be used to transport the LM into an orbit of approximately 8 by 60 nautical miles. Should this maneuver (DOI) result in an unsafe orbit, the CSM will perform a DOI bailout maneuver to return the vehicle to a safe orbit. Further adjustment to the orbit will be made with the DOI TRIM in revolution 9 if required to insure appropriate PDI conditions during revolution 13. Two astronauts will subsequently enter the LM and perform LM checkout. The CSM will be separated from the LM using the SM RCS.

The CSM orbit will be circularized (CIRC) at approximately 60 nautical miles shortly after CSM/LM separation (1/2 revolution before PDI) and commence landing site (SXT) observation. Just prior to PDI, the final LM descent targeting and landing site updating will be accomplished.

During lunar orbit, the SIM equipment will be placed in operation. The Particles and Fields Satellite (P&FS) will be ejected from the SIM bay prior to TEI. The SIM cameras and booms will be retracted into the SIM bay prior to any SPS maneuver. If any boom fails to retract, it will be jettisoned prior to SPS thrusting.

About 10 revolutions prior to the LM ascent during revolution 50, the CSM will perform a plane change to pass over the landing site approximately at lift-off. About 38 minutes after LM insertion, an APS TPI maneuver will be performed. Other LM MCC and TPI maneuvers will be made using the LM RCS. After docking, both LM crewmen will transfer to the CSM with the lunar surface samples and the exposed film. The LM will be separated from the CSM using the SM RCS during revolution 54.

During revolution 73 (two revolutions prior to TEI), the CSM will perform an SPS shaping maneuver to place the SC in the appropriate orbit ($h_a \approx 85$ nautical miles, $h_p \approx 55$ nautical miles) for the ejection of the P&FS. The ejection occurs approximately 30 minutes after the maneuver.

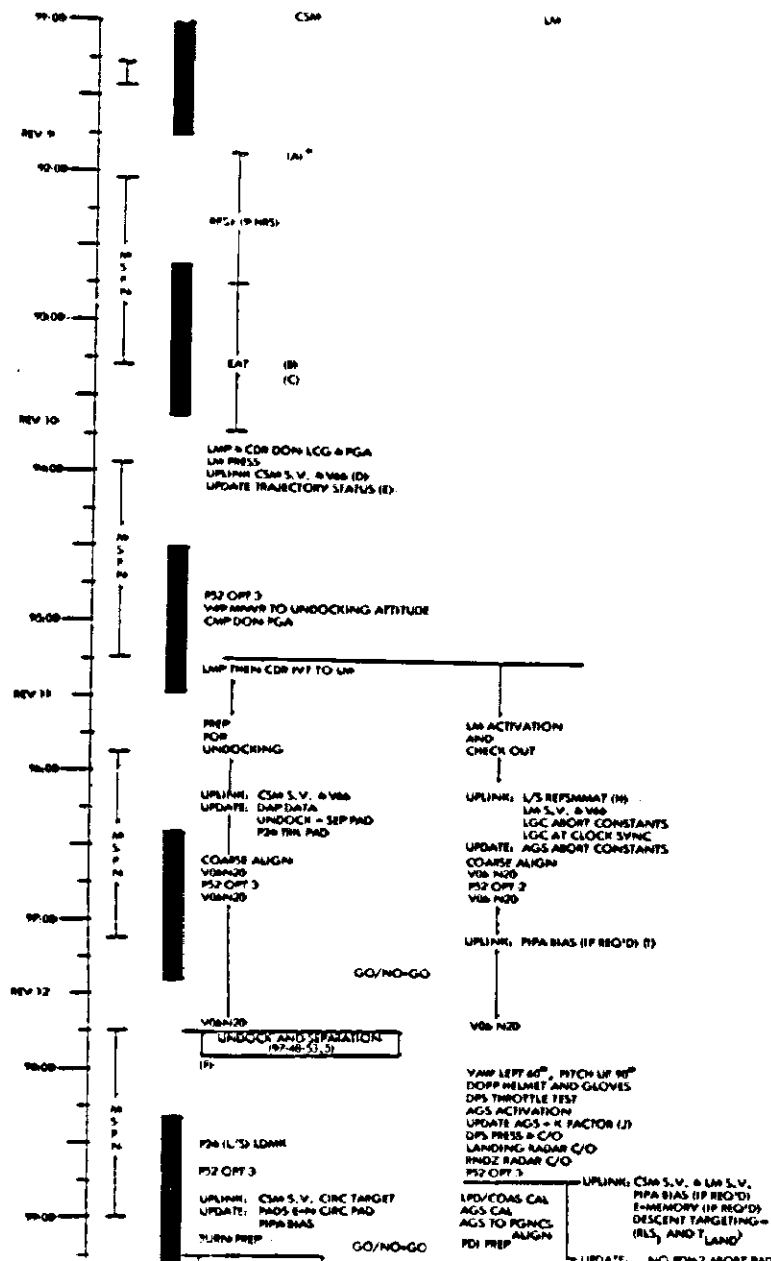
5.1 LUNAR ORBIT TIMELINE (Page 2-2)

The pre-PDI LM activation timeline has been shortened (Figure 5-1) to increase post-touchdown EVA time on the lunar surface. This shortened sequence results in a lunar landing on revolution 13 rather than 14. The deleted revolution was the one which contained the high orbit landmark tracking and results in PDI being just one-half revolution after CIRC. This phasing change causes major revisions in the descent abort profiles (Section 7.1). Landmark tracking is not required for landing because the landing site errors (for Apollo 16) are not large enough to present a crew safety problem. This could change for Apollo 17, depending on the final site selection. Additional relaxation of the pre-PDI timeline was obtained by deleting the television coverage, certain updates, and the DOI TRIM. (If a DOI TRIM is required, the crew will wake up 2 hours earlier to accomplish the maneuver.) Also, several updates and uplinks are moved or combined when compared with the baseline timeline.

5.2 DESCENT ORBIT INSERTION MONITORING (Page 4-1)

The descent orbit insertion (DOI) maneuver is performed two revolutions after LOI to place the LM in the proper condition for PDI (60° by 8-nautical mile orbit with the LM at pericynthion for PDI). The physiological cues and the attitude and attitude rate limits (± 10 degrees and ± 10 degrees/second) for monitoring DOI have not changed; however, the trimming rules have.

The DOI burn (for Apollo 16) will place the docked vehicle on a lunar impact trajectory with overburn of 1 second (10 feet per second) or greater. Thus, a manual SPS shutdown will be executed by the crew at the ground predicted burn time. If a DOI overburn is indicated, the crew will immediately trim (no limit) the valid system to within 1 foot per second to insure maintenance of a safe orbit.



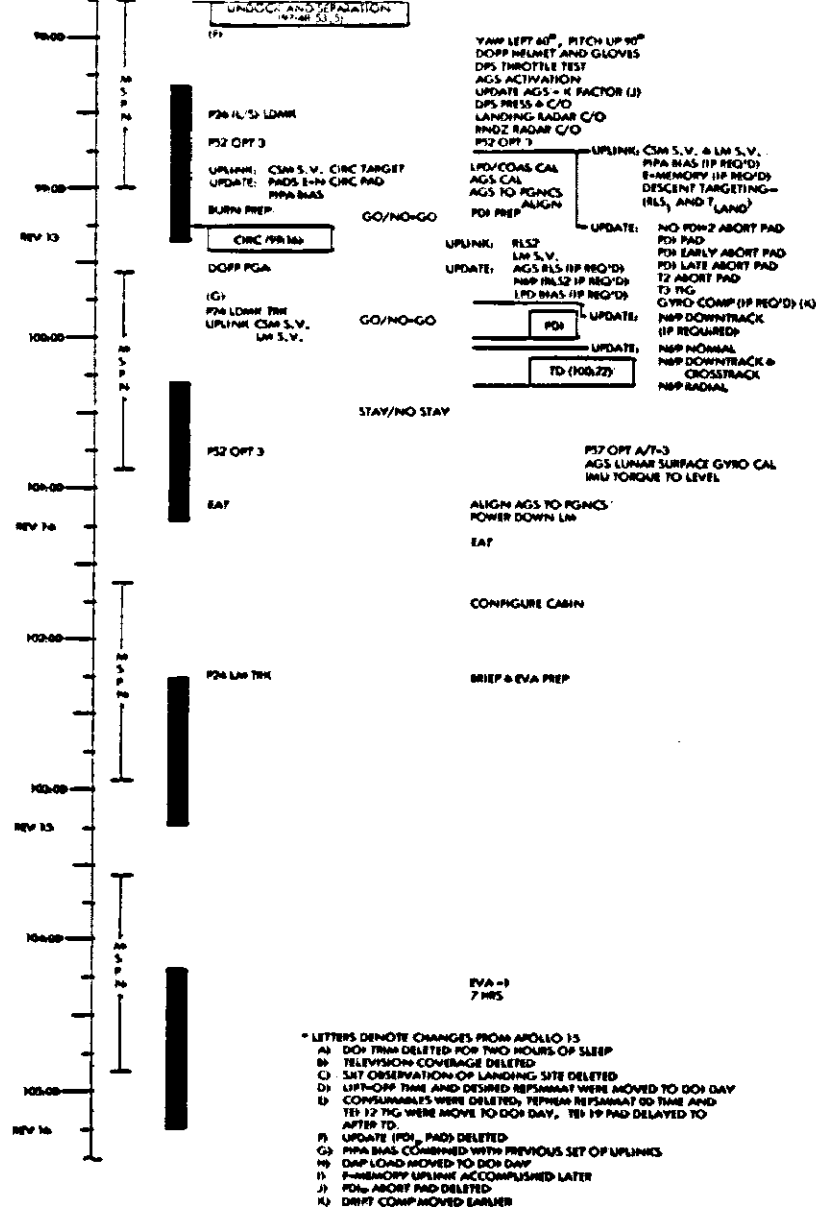


Figure 5-1. LM Activation to PDI Timeline

Several items must be considered in selecting the method of trimming RCS thrusters or SPS. The 3 σ error in predicted burn time is 0.5 second (5 feet per second); therefore, if the G&N shows a residual greater than +5 feet per second, the EMS must be believed so long as it is within 5 feet per second of the nominal ΔV . The monitoring technique will permit an overburn of 2 to 4 feet per second (crew reaction time of 0.2 to 0.4 second), and this must be considered in evaluating pericyynthion conditions. If a positive residual remains in the G&N, the monitoring technique has caused shutdown, and any overburn residual in the EMS must be believed. Overburn residuals in both the G&N and EMS imply G&N problems external to navigation and require trimming of the conservative (larger) residual; whereas, an underburn will not be trimmed. The -X RCS thruster has a 2.2-foot per second maximum on-time impingement constraint, and the SPS has a 0.5-second minimum on-time constraint; therefore, the DOI overburn trimming rules are as follows:

- Underburns (positive residuals) will not be trimmed.
- Overburns will be trimmed to within 1 foot per second.
 - For overburns less than 2.2 feet per second, trim to within 1 foot per second with the -X SM RCS thrusters.
 - For overburns greater than 2.2 but less than 10 feet per second, pitch 180 degrees and trim to 1 foot per second using +X SM RCS thrusters.
 - Residuals greater than 10 feet per second will be trimmed using SPS.
- If the GNCS has obviously malfunctioned, the negative residual indicated by the EMS will be trimmed.

It should be emphasized that DOI will be executed in the single bank mode and that there will be no change in the PU valve during the last 30 to 60 seconds of the LOI burn in order to have a constant thrust for subsequent burntime estimates. (Burntime predictions using the dual bank burns are less accurate than single bank Δt_{ij} 's because of the uncertainty about the time when the second bank starts.) DOI is the only burn greater than 6 seconds to be performed in the single-bank mode.

5.3 DOI BAILOUT MANEUVER (Page 4-2)

In the unlikely event that the MSFN DOI monitoring techniques indicate an unsafe orbit, a canned DOI bailout maneuver will be accomplished at AOS + 10 minutes. Reference 26 indicates the bailout maneuver can be slipped to as late as AOS + 12 minutes and still yield a 4-nautical mile h_p altitude (1.5-second overburn). The h_p for a STAY decision is defined as that which will allow the minimum acceptable clearance over the most constraining terrain peak in revolution 3. The minimum acceptable clearance is determined by adding a pad of 1 nautical mile to the RSS of the 3 σ uncertainties of the MSFN short arc and radius of the most constraining peak.

It should be noted that SPS gimbal motors will be turned ON if the preliminary MSFN decision is NO STAY. They will not be turned ON until a NO STAY decision, if the preliminary vote was STAY.

Table 5-1 defines the DOI bailout maneuver. This maneuver has an apolune attitude which allows a single maneuver back to the PDI orbit and high-gain antenna coverage (Reference 28).

Table 5-1. DOI Bailout Maneuver^a

<u>Parameter</u>	<u>Pitch up 83 deg to LVLH at AOS + 10 min</u>
Resultant orbit if maneuver is performed at AOS + 10 min	61.1 x 5.3 n mi
Resultant orbit if maneuver is performed at AOS + 12 min	54.8 x 3.3 n mi
Attitude requirements (gimbals)	IOA = 118.6 deg MOA = 0.0 deg OGA = 0.0 deg
Attitude requirements (LVLH)	Pitch = 83.0 deg Yaw = 0.0 deg Roll = 0.0 deg

^aAssuming 1.5-second overburn at DOI resulting in a 58.2 σ by 0.5-nautical mile orbit.

5.4 DOI TRIM MANEUVER (Page 4-3)

The DOI TRIM maneuver (not to be confused with trimming DOI) is not scheduled for Apollo 16 due to its low probability. Should a DOI TRIM (orbit adjustment for PDI) be required, the crew will wake up 2 hours earlier than planned on PDI day and perform the maneuver. It should be noted that if a DOI TRIM maneuver is performed and for some reason PDI is slipped a revolution, the crew will sleep first and then perform EVA-1.

The DOI TRIM will be performed if the estimated h_p at PDI is less than 30,000 feet or greater than 70,000 feet. The burn will be targeted for a nominal pericynthion* of 53,000 feet and the inplane horizontal component will be nulled to 1 foot per second. This makes descent nominal, aids the P23 sightings, and restores the manual abort angles. The DOI TRIM will be executed if the wedge angle component (relative to the landing site) is greater than 0.5 degree for the same reasons. The DOI TRIM maneuver will be monitored the same as the DOI maneuver using the same bailout maneuver execution criteria.

5.5 STATE VECTORS, REFSMMATS, ALIGNMENTS, AND DRIFT CHECKS (Page 4-4)

The REFSMMAT's computed on the DOI day will be used for landing in order to conserve time. It is understood that yaw will be in error by the amount of plane error at that time and that pitch will be in error by the predicted landing time error times the moon's rotation rate.

The docked P52 and the P52 just after undocking (Figure 5-1) should be accomplished on time to provide adequate time between the P52's to get a drift check; MCC-II has to compute and update the drift compensation (if required) before LOS in revolution 12. This means that DPS, RR, and LR checkout can be slipped until after the second P52, if required. An unacceptable drift check could result from a bad first P52, (e.g., larger torque angles than expected). To avoid this contingency, three V06N20's are scheduled. (This verb/noun combination yields a decimal display of present ICDU angles.) The sequence is

- Coarse (docked) alignment
- V06N20 (CSM and LM) immediately following coarse alignment

* Pericynthion is 15° east of the landing site (PDI).

- P52 (first one) - four marks on each star
- V06N20 (immediately following P52)
- V06N20 (preceding undocking)
- Undocking

The crew will record and relay the V06N20 data to the ground if it is done out of coverage; however, P52 stars are selected to allow steerable antenna coverage.

5.6 UNDOCKING/SEPARATION (Page 4-6)

Soft undocking is used to separate the CSM and LM (1/4 revolution prior to PDI) in order to minimize the downtrack errors at PDI resulting from orbital perturbations. The undock/separation maneuver can be performed as much as 45 minutes earlier at the nominal inertial (IMU) attitude, and up to 40 minutes late at the nominal local vertical (ORDEAL) attitude. The inertial attitude is good for slips up to 8 minutes. Slipping undock/separation past AOS (about 8 minutes late) is not desirable since this would compromise the landmark tracking and subsequent activities.

Should an abort be required after separation and before CIRC, the terminal phase (TPI, TPM) programs will be used to accomplish rendezvous. The TIG for TPI is at separation plus 75 minutes which yields the proper lighting conditions for docking and completes the nominal relative motion resulting from the separation maneuver.

5.7 LPD BIAS UPDATE LIMIT (Page 4-8)

After undocking and before LOS (revolution 12), the LM's IMU is aligned to REFMMAT (P52). A star check using auto optics is used to verify the alignment and calibrate the landing point designator (LPD). The LPD's position is keyed into the DSKY as a COAS sighting (0 degree shaft, ± 40 degrees trunnion). If the preflight calibration values, as determined by MCC-II from the crew observation, do not agree within ± 2 degrees in elevation and azimuth, MCC-II will calculate the LPD calibration values for a DSKY entry. (The limit originally was ± 1 degree; however, the crew has stated that they cannot read the LPD to that accuracy.) The crew also has charts to accomplish this update.

5.8 CSM CIRC MANEUVER TRIM RULES (Page 4-10)

After separation, when the low orbit landing site observations have been completed, MCC-H will voice PAD data for the CIRC maneuver to the CMP, and the targeting data for the maneuver will be uplinked to the CMC (P27). The CIRC maneuver is a SPS burn of approximately 70 feet per second (3.9 seconds) which places the CSM in an orbit which is targeted to be 60 nautical miles circular at rendezvous.

The baseline trim rules have been updated to cover the gross execution error case. The trim rules are

- Trim all residuals to within 0.2 foot per second, if all are less than 5 feet per second. The nominal CIRC maneuver will be loaded into the LGC via P76.
- Do not trim if any residual is greater than 5 feet per second; P76 (ΔV target) values will be adjusted for the errors and loaded into the LGC.
- For underburn greater than 20 feet per second, an SCS restart will be attempted. Banks will not be switched.

If the CIRC maneuver is trimmed with the CSM RCS to within reasonable operational limit (0.2 foot per second), the update of the CSM state vector by the nominal maneuver via P76 yields an acceptable state vector. The 0.2-foot per second limit preserves the orbital period and insures that the LM is in the plane determined by revolution 13 landmark tracking. If any residual is greater than 5 feet per second, the desirability of keeping the state vector nominal is dropped in favor of conserving RCS (previous mission residuals are typically less than 1 foot per second). This is acceptable since the LGC will be updated with the correct CSM vector. The ground will uplink a vector to the LM reflecting the confirmed CIRC, if required.

5.9 P24 LANDMARK TRACKING REQUIREMENTS (Page 4-10)

P24 sightings which are used to determine the relative position of the landing site with respect to the orbit are scheduled (Table 5-2) for Apollo 16; however, they are not required. (On previous missions, where

Table 5-2. Pre- and Postlanding P24 Tracking Schedule

<u>Tracking on Revolution</u>	<u>Tracking Source</u>	<u>Orbit</u>	<u>Purpose</u>	<u>Remarks</u>
3	SXT LNMK	low	OJT	
12	SXT LS LNMK	low	R_{LS1}	PDI targeting
13	SXT LS LNMK	high	R_{LS2}	Final PDI targeting
14	SXT on LM	high	CM position	
14	SXT LNMK	high		

the map uncertainties were larger, P24 sightings were required and may be required for Apollo 17 depending on map knowledge of the selected landing site.) The location of the Apollo 16 landing site is well known and the 3 σ landing ellipse (without the tracking) does not jeopardize the lunar traverse objectives; thus, a mandatory requirement for P24 tracking on Apollo 16 does not exist. Nonetheless, P24 tracking is highly desirable to retain nominal traverses and for postflight evaluation. Reference 30 indicates that the 3 σ a priori Apollo 16 landing site position (map) uncertainties are 3,800 feet downtrack, 7,900 feet crosstrack, and 4,350 feet radially.

To aid in the acquisition of P24 landmark tracking, the downtrack error will be estimated based on MSFN range data at AOS of revolution 12 (Reference 31). This is used to update the pitch initiation time for P24 landmark acquisition which requires an accuracy of approximately 0.5 second.

5.10 PRE-PDI DOWNTRACK N69 (New Section)

The baseline document describes the techniques for updating the landing site position (R_{LS}) to improve the relative accuracy of the orbit with respect to the landing site. Figure 5-1 shows the sequence of R_{LS} uplinks and N69 updates before and after PDI. On Apollo 15, the LMA experienced an approximate 5,000-foot error in downtrack at PDI. This was caused by water dumps, undocking, and uncoupled attitude maneuvers. Therefore, for subsequent missions, a N69 downtrack value will be estimated based on MSFN tracking shortly after entering the descent phase of the RTCC. This is loaded after the R_{LS2} uplink if the estimated uprange error is greater than 10,000 feet. This is all accomplished before P63 is called to allow TIG to reflect the update (Reference 31). The allowable uprange error for a nominal descent engine is 38,000 feet.

Reference 31 also indicates that the CSM water dump which accounted for a small portion of the anomalies noted on Apollo 15 has been moved earlier to allow a longer unperturbed tracking interval. Also, all minimum impulse maneuvers are coupled via jet selection in the DAP, and LLS observations are performed in fixed attitude rather than with a rate to reduce jet firings.

5.11 SIM-BAY ACTIVITIES (New Section)

On Apollo 16, the dead-band and weight constants will be varied during revolutions 14 through 17 to determine if the average pointing error (which is caused by gravity gradient torquing) during wide dead-band (WDB) operation and the RCS consumption during narrow dead-band (NDB) operation can be reduced. The nominal WDB for SIM-bay activity is 3.0 degrees; however, for test purposes the WDB will be set to 2.0 degrees for a one-revolution period during revolutions 14 and 15 (after TD and before the sleep period). The NDB (1/2-degree setting) test involves one revolution with

80 percent of the nominal CSM weight loaded in the CMC during revolutions 16 and 17 to force all firings to be minimum impulse*. One revolution with a 5-degree dead-band setting will be scheduled for calibration of the Apollo 16 data with that of Apollo 15.

Also, SIM-bay activity (uncoupled attitude control) is not scheduled for the two revolutions prior to ascent to allow perturbation-free MSFN tracking.

5.12 DIRECT RENDEZVOUS (Page 4-22)

The purpose of the direct rendezvous is to reduce the ascent/rendezvous work day. The direct rendezvous is accomplished in slightly over 90 minutes, while the coelliptic rendezvous requires approximately 197 minutes.

The baseline document indicates several assumptions used to govern the rules for commitment to a direct rendezvous; one of which is the maintenance of two independent means of rendezvous navigation through the tweak decision (INS + 1 minute). The philosophy has now changed somewhat in that one fully operational prime system (LGC/RR with valid angles/IMU, AEA/RR via tapemeter/LM COAS/CSM LITE/ASA (Gyros) or CMC/Usable SXT/LM LITE/IMU) is better than two of the degraded (nonprime) systems. These nonprime systems are LGC/RR without angles, AEA/RR with certain ASA failures, AEA/VHF/CSM LITE/LM COAS, CMC/VHF without SXT marks prior to TPI, and CMC/VHF/OPTICS with failed IMU. Their solutions are usually adequate, but the inherent procedural difficulties and larger errors cause them to be less desirable than a prime system. Therefore, unless one prime system (Table 5-3) is fully operational through the tweak decision point, a commitment for direct rendezvous will not be made.

* A similar exercise performed on Apollo 15 indicated reduced firings; however, the test was not of sufficient duration to be conclusive.

Table 5-3. Required G&N Systems Needed for Direct Rendezvous

Primary Rendezvous Navigation Technique	System Requirements				
	Computer	Sensor/Optics	Parameters Supplied by Sensor/Computer Interface	Tracker Light	Platform
LGC/RR	LGC	Rendezvous radar	Range, range rate, shaft and trunnion angles		IMU
AGS/RR	AEA	Rendezvous radar LM COAS	Tapemeter: range and range rate	CSM	ASA
CMC/Sextant	CMC	Useable sextant	Sextant shaft and trunnion angles	LM	IMU

Notes:

1. Direct rendezvous is GO as long as any one of these techniques remains available.
2. Loss of all three of these techniques will result in execution of the coelliptic sequence rendezvous.

Table 5-4 presents the requirements for direct rendezvous. If rendezvous navigation is initiated with a single G&N system and this system fails prior to computing TPI, either the ground TPI or an alternate BOOST/HAM sequence will be executed. The BOOST/HAM sequence may be executed by either vehicle; however, it is preferable to utilize the LM and execute a 50-foot per second BOOST at insertion plus 50 minutes. The ground TPI will be used in those cases where good confidence is held in the MSFN insertion vector and another source is available for general orbit confirmation (PGNCS, AGS, or MSFN short arc solution).

5.13 POST-ASCENT TWEAK AND BAILOUT MANEUVERS (Page 4-26)

The timeline and attitude for the post-ascent tweak and bailout maneuvers, if required, are

<u>Time</u>	<u>Event</u>	<u>FDAI Angles (deg)</u>
0	Insertion	P = 258 Y = 0 R = 0
1	MCC-H pass tweak maneuver- P47 ΔV 's and GETI (FDAI angles, if required for tweak execution with RCS thruster or system problems)	
3	Execute tweak (if required). Burn X, Z, and then Y.	P = 258 Y = 0 R = 0
	MCC-H pass direct rendez- vous bailout maneuver (ΔV and and GETI)	} Insertion attitude
5	Execute bailout if required	P = 242 Y = 0 R = 0
		} or { 10 deg in overhead window (burn along local horizontal
12	CSM bailout if required	P = 35 Y = 0 R = 180

Table 5-4. Direct Rendezvous Requirements

Lift-off

- a) One operational navigation system for ascent
- b) One primary navigation technique available (rendezvous)
- c) Predicted wedge angle at insertion must be zero*

Insertion

- a) One primary rendezvous navigation technique available
- b) Tweak delta V less than 60 fps (RSS)*
- c) Post-tweak pericynthion greater than 5 n mi*

Special Case Agreements

- a) Direct rendezvous is GO for loss of either LM buss or ascent battery if any onboard navigation technique exists.
- b) The Y-component of the tweak may be scrubbed for thruster problems or RCS fuel considerations.
- c) There are no RCS thruster or delta V problems that make the direct rendezvous NO-GO on the surface.
- d) Will knowingly commit to a CSM-active TPI/TPF due to LM RCS problems. This sequence is more desirable than the longer CSM bailout/rescue for these same problems.
- e) A "failed" sextant must be optically failed.
 - 1) The sextant is GO for these failures:
 - Mark button
 - Single or multiple axes electronic drive failures
 - "Frozen" sextant - FOV usable
 - Loss of reticle illumination
 - Failures in the sextant/CMC interface such that the CMC shaft or trunnion angles are static or dynamic/erroneous.
 - 2) The sextant is NO-GO for these failures:
 - Loss of visibility through sextant
 - Loss of eyepiece
 - "Frozen" sextant - FOV unusable

*Limits are discussed in Reference 4 (baseline document)

On Apollo 15, the tweak maneuver was performed to insure terminal approach within -95 to 210 degrees (operational values, third quadrant approach) provided the maneuver did not exceed 60 feet per second (RSS), and the out-of-plane component was less than 30 feet per second. However, widely dispersed approach angles and high or low closing rates are not desirable from a crew standpoint, particularly if a subsequent RR failure occurs; therefore, the terminal approach trajectory will be maintained near the nominal for Apollo 16 and 17. MCC will pass ignition time and body velocities to be gained (VG's) for the P47 execution. The tweak will be performed if either inplane component is greater than 5 feet per second (tweak MSFN 10 = 3 feet per second) or if the out-of-plane component is greater than 10 feet per second. Insertion residuals are trimmed with the active system to within 10 feet per second if communication exists and 2 feet per second if there is no communication. If communication exists during ascent, the ground will recommend the trim system at $T_{go} = 60$ seconds. If communication does not exist, the RR will be used to determine the correct system if the AGS and PGNCs differ by more than 10 feet per second in X- or Z-axes after insertion.

It should be pointed out that the CSM vector time tag will be adjusted for downtrack errors, since failure to do so results in incorrect tweak (and TPI) solutions on the ground and incorrect initial solutions onboard. (The V_Z sensitivity is approximately 1 foot per second per 1,000 feet of down-track error.) Also, it should be noted that the AGS residual must be frozen (not recomputed) for trimming at insertion.

The LM and CSM bailout maneuvers are discussed in the baseline document (Reference 4). Briefly, the bailout maneuver (Reference 32) adjusts the LM (or CSM) targeting for a standard CSI/CDH coelliptic sequence.

5.14 PRE-TPI ACTIVITIES (Page 4-27)

The normal procedure following the tweak maneuver for Apollo 16 and 17 will be for the CSM (in auto optics) to track the LM with the SXT and VHF for the 5 minutes (from INS + 6 minutes to INS + 11 minutes) it normally takes to service the LM and generate the postinsertion LM vector

Table 5-4. Direct Rendezvous Requirements

Lift-off

- a) One operational navigation system for ascent
- b) One primary navigation technique available (rendezvous)
- c) Predicted wedge angle at insertion must be zero*

Insertion

- a) One primary rendezvous navigation technique available
- b) Tweak delta V less than 60 fps (RSS)*
- c) Post-tweak pericynthion greater than 5 n mi*

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- a) Direct rendezvous is GO for loss of either LM buss or ascent battery if any onboard navigation technique exists.
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It should be pointed out that the CSM vector time tag will be adjusted for downtrack errors, since failure to do so results in incorrect tweak (and TPI) solutions on the ground and incorrect initial solutions onboard. (The V_Z sensitivity is approximately 1 foot per second per 1,000 feet of down-track error.) Also, it should be noted that the AGS residual must be frozen (not recomputed) for trimming at insertion.

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5.14 PRE-TPI ACTIVITIES (Page 4-27)

The normal procedure following the tweak maneuver for Apollo 16 and 17 will be for the CSM (in auto optics) to track the LM with the SXT and VHF for the 5 minutes (from INS + 6 minutes to INS + 11 minutes) it normally takes to service the LM and generate the postinsertion LM vector

for the CMC (Reference 33). If after the 5 minutes of tracking, the last SXT and VHF marks yield a N49 display, MCC-H will uplink the LM vector to the CMC. If a N49 is not displayed, the LM vector is not uplinked. A V93 (W-matrix reinitialization) is needed after the 5-minute tracking period to reduce the V_z errors at TPI (2 feet per second normal and 20 feet per second (3σ) if the LM light is out).

The pre-TPI tracking orientation is heads-up which is required to provide MSFN steerable antenna coverage.

The other Apollo 16 and 17 pre-TPI techniques are identical to those discussed in the baseline document with the exception of the use of the MSFN relay. Previously, the MSFN relay has been turned off so that the MSFN data could be used for voting should the RR and VHF not agree. The MSFN relay can result in a data transmission delay of about 3 seconds between the LM and CSM which can invalidate this comparison. However, the Apollo 16 crew has requested that the MSFN relay be turned ON before ascent to avoid the no-VHF communication contingency; therefore; the MSFN relay will be turned ON prior to ascent and a VHF communication check made at lift-off minus 5 minutes.

5.15 TPI RENDEZVOUS MANEUVER (Page 4-28)

At insertion + 10 minutes, MCC-H will pass up the MSFN TPI solution. This will indicate if sufficient APS propellant is available to accomplish TPI; however, to conserve RCS, TPI will be executed with APS propellant unless the APS tanks are known to be empty. Prior to Apollo 16, if an APS depletion was likely, TPI was performed using the RCS.

The only other update to the TPI procedures discussed in the baseline document involves the post-TPI burn report. For Apollo 16 and 17, the normal burn reports for the executed solution (burn and residual) are all that will be passed to the crew at AOS after TPI. All other solutions will be compared during postflight analysis.

5.16 RECOVERY PROCEDURES FOR AGS INCORPORATION OF INVALID RR MARK (Page 4-29)

In the event incorrect RR data are incorporated into the AGS rendezvous solution following steady state RR performance, the following N49 reject procedures are used (Reference 18):

- a) Perform 411 + 0 to inhibit auto RR transfer.
- b) Determine the cause of the N49; angle, range, or range rate (R3 code)
- c) Did error affect AGS?
- 606R -XXXXXX used range angles in last update (-sign clue)
+00000 used range rate in last update (+sign clue)
- No Verify with 317,440
Read and compare to RR data
Proceed with manual updating schedule
- Yes Proceed with step (d)
- d) Are there at least 10 minutes of tracking left?
- No Perform V47 to protect against subsequent PGNCS failure and perform V47 again near the maneuver time (TIG - 5 minutes) to get rid of errors in the PGNCS at the previous V47
- Yes Proceed to step (e)
- e) Check on error magnitude
- Angle Error: Angle less than 5 degrees (no 525 or 525 angle < 5 degrees): reinitialize filter and continue manual updating
- Angle greater than 5 degrees: do V47, reinitialize filter, and continue manual updating
- Range Error: 317 R and RR range within 5 nautical miles: reinitialize filter and continue manual updating
- 317 R and RR range differ by > 5 nautical miles: do V47, reinitialize filter, and continue manual tracking
- Range Rate Error: 440 R and RR range rate within 15 feet per second: reinitialize filter and continue manual updating
- 440 R and RR range rate differ by > 15 feet per second: do V47, reinitialize filter, and continue manual marks

5.17 TPI₂ PROCEDURES (New Section)

To avoid large braking ΔV 's, a TPI₂ burn may be executed following MCC-2. Prior to TPI, the TPI and TPF ΔV 's are calculated. If both of the LM RCS systems are operational and the ΔV_{TPF} is greater than 100 feet per second, the LM will execute TPI₂. If only one LM RCS system is operational, the LM will execute TPI₂ if ΔV_{TPF} is greater than 55 feet per second. If the LM cannot perform MCC-2 or it is otherwise known that the LM cannot support braking, then the CSM will perform TPI₂ if ΔV_{TPF} is greater than 55 feet per second. If ΔV_{TPF} is greater than 55 feet per second, the CSM will anticipate a failure at MCC-2 and be configured to perform TPI₂, if required. It should be noted that the tweak maneuver should keep the ΔV_{TPF} to less than 40 feet per second for all reasonable cases.

5.18 LM JETTISON AND DEORBIT (Page 4-30)

The LM jettison timeline (Figure 5-2) and procedures are essentially the same as to those given in the baseline document with exception of the separation (SEP) maneuver and the time of the uplink of the deorbit PAD's. The CSM SEP maneuver (P30) is now 2 feet per second postgrade (Apollo 15 used 1 foot per second retrograde) and is executed 5 minutes after LM jettison. This allows SEP to be slipped and protects against a larger LM ΔV resulting from higher than nominal tunnel pressure. The uplink of the deorbit PAD's are more to the end of the equipment transfer period (revolution 53), i.e., closer to the deorbit maneuver to improve targeting.

5.19 SHAPING MANEUVER AND SUBSATELLITE EJECTION (Page 4-31)

The Apollo 16 mission (not Apollo 17) includes the ejection of a one-year lifetime (Reference 34) Particles and Fields Satellite (P&FS). To satisfy the one-year requirement, the CSM will perform a shaping maneuver to place the SC into orbit with an h_a of 85.4 and h_p of 54.8 nautical miles (relative to the landing site). The shaping maneuver is executed at 218:26:05 and requires a 1.4-second SPS burn ($\Delta V = 39$ feet per second).

The P&FS is ejected at the ascending node (the night side of the orbit) two revolutions prior to TEI and approximately 30 minutes after the shaping maneuver. The P&FS is ejected (spring release ΔV of 4 feet per second) north, normal to the ecliptic. Ejection to the south at the descending

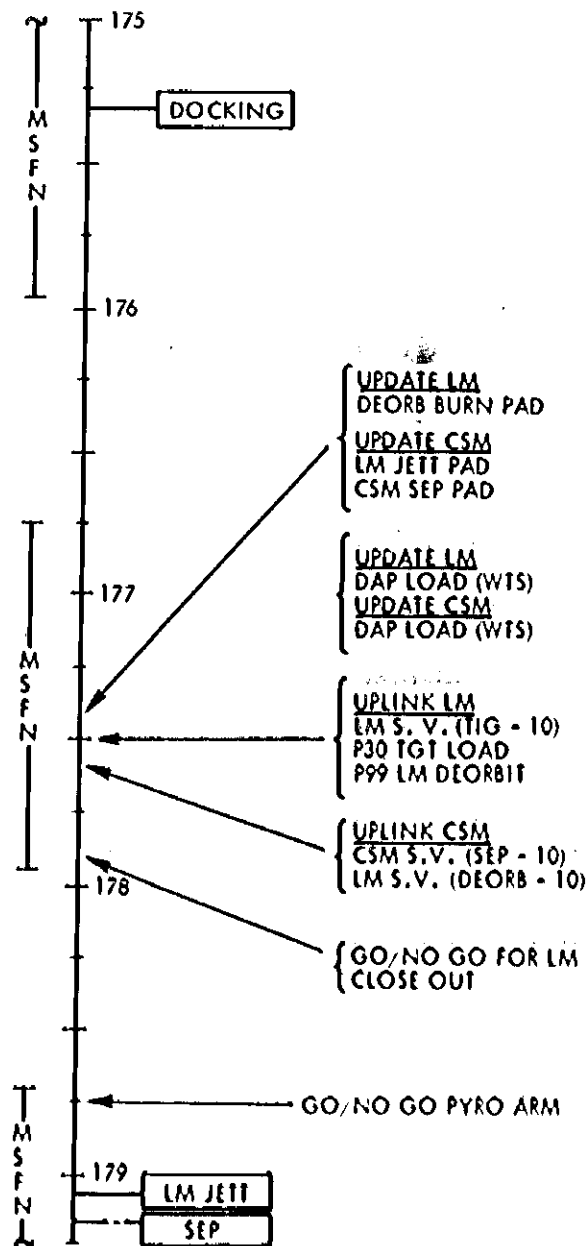


Figure 5-2. LM Jettison Timeline

node permits photography; however, unacceptable coning (spin axis wobble) of the satellite will result. Ejection at the descending node presents a potential recontact hazard during TEI. The ejection to the north at the ascending node imparts a posigrade inplane velocity component (approximately 1.8 feet per second) causing the satellite to fall behind, but not lap the CSM.

6. LUNAR DESCENT

This section discusses the updates to the Lunar Descent Mission Techniques (Reference 5). Following a brief review of the major mission phase events, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

The Apollo 16 descent profile is essentially the same as that of Apollo 15 except that PDI occurs at an h_p altitude of 53,000 feet as opposed to 50,000 feet for Apollo 15. The higher PDI altitude for Apollo 16 improves the success probability of the P24 landmark sightings. (The landing ellipse (35) is ± 1 kilometer downrange and ± 1.3 kilometers cross-range with landmark tracking and 4.5 and 8 kilometers, respectively, without it.) The significant events are

<u>Event</u>	<u>Approximate Time from PDI</u>
Maximum DPS thrust	26 seconds
LR altitude updates	4 minutes
LR velocity updates	5 minutes
Throttle recovery	7 minutes
High gate	9 minutes
Touchdown	12 minutes

6.1 STEERABLE ANTENNA COVERAGE (Page 4-2)

On the lunar landing revolution, the LM will maneuver to a yaw attitude of 20 degrees left prior to AOS and retain the attitude until PDI + 3 minutes. This attitude is necessary to insure continuous steerable antenna coverage. This yaw is removed at the rate of 5 degrees per second beginning at PDI + 3 minutes.

The AOS will be loaded (514, 515, 516) to yaw 0 degree left for steerable coverage in the event of an abort. Previous missions required a yaw maneuver for steerable coverage.

6.2 DPS IGNITION MONITORING (Pages 4-8 and 5-6)

The DPS ullage and ignition monitoring techniques have been updated so that if either auto ullage or auto ignition occurs, PDI is still GO. (Previously, if ullage did not occur by TIG - 7.5 seconds, PDI was NO-GO, and PDI was delayed one revolution, or a PDI + 12-minute abort was performed.) The rationale for the update is centered around the assumption that if either auto ullage or auto ignition occurs, the LGC is issuing the correct commands, and the failure is down stream and is correctable using manual techniques. If neither ullage nor ignition is performed automatically, the LGC is assumed to be malfunctioning and PDI is NO-GO.

6.3 NOUN 69 TARGET UPDATE (Page 4-9)

The LGC R_{LS} is scheduled to be updated before and after PDI. A pre-PDI backup R_{LS2} N69 technique is described in Reference 4, and another pre-PDI downtrack N69 correction prior to P63 entry is discussed in Section 5.10. The post-PDI N69 LGC target updates are given below:

- a) N69 NOMINAL (Z-component only) - This downrange propagation correction has been used since Apollo 11 and occurs at PDI + 2 minutes. It corrects state vector errors incurred prior to PDI and is based on either the PFP or MSFN doppler residuals prior to PDI.
- b) N69 DOWNTRACK NAV (Z-component only) - This is a correction to compensate for a downtrack position error incurred by a downtrack navigation error. By measuring a confirmed PGNCS navigation error at PDI + 2 minutes, the downrange position error (at the time of LR velocity convergence) can be estimated. This assumes a linear error growth and considers the effective error during LR velocity correction. If the estimated error at touchdown is greater than 3,000 feet, the magnitude with its sign reversed is volced to the crew for input to the PGNCS via N69 at PDI + 5 minutes.
- c) N69 CROSSTRACK NAV (Y-component only) - This is a correction to compensate for a estimated crosstrack navigation position error at touchdown greater than 3,000 feet. It is computed in the same manner as N69 DOWNTRACK and is input by the crew at the same time as the downtrack correction.

- d) N69 RADIAL NAV (X-component only) - This correction compensates for altitude errors resulting from either a navigation error during powered flight or an H error at PDI. The computation of this parameter is essentially the same as for the two preceding parameters, except this position error is a function of two velocity errors:

- Downrange propagation error
- Vertical navigation error

This correction will be voiced to the crew only if MCC-H is certain that LR data will not be received prior to pitchover. (The correction is not required if LR data are available.) This update permits continuing powered descent to a PGNCs-indicated altitude of 6,000 feet without the LR data, provided the H residual after 10,000 feet does not exceed 20 feet per second. The upper limit for this update is the LR data acceptance value; the lower limit is 500 feet (operational value). This correction is voiced to the crew for input by PDI + 8 minutes.

6.4 GUIDANCE THRUST COMMAND LIMIT (Page 4-15)

The baseline document discusses the monitoring of the DPS guidance thrust command (GTC) during descent. These techniques indicate that for extreme low-thrust cases, the PGNCs commanded thrust will increase and possibly become cyclic, resulting in large LM attitude excursions. This incompatibility between PGNCs guidance and a low thrust DPS is detected on a display of GTC versus horizontal velocity. Also, the baseline document discusses a limit based on GTC which predicts if throttledown will not occur by P64. Throttledown must occur by P64 plus 15 seconds.

Analysis has shown (References 35 and 36) that low thrusting DPS performance at full throttle is reliable and, given sufficient time to reduce horizontal velocity to a value suitable for throttle recovery, can be used to achieve a safe lunar landing. (The PGNCs can handle DPS thrust levels down to about 8,000 pounds.) In order to accomplish this, it is necessary to shift the targeted landing site downrange. The analysis also has shown that a direct relationship exists between DPS full-throttle thrust and the required shift in R_{LS} . The maximum permissible shift is 130,000 feet due

to unacceptable landing terrain beyond that point. This corresponds to a DPS thrust of about 8,900 pounds. A ridge (15,000 feet wide and 1,500 feet maximum height) which crosses the groundtrack between 66,000 and 81,000 feet downrange is also unacceptable for landing. Although, highly unlikely, the following techniques, using N69 downtrack inputs and +X RCS thrusting will be used to cope with a low-thrusting DPS problem.

Once the low-thrusting DPS is detected, a N69 input will be made provided the downrange correction is between 15,000 and 130,000 feet. (A N69 input of 81,000 feet will be used if the indicated correction falls between 66,000 and 81,000 feet.) A chart showing the N69 magnitude as a function of DPS thrust will be used to determine the correct N69 value. For values between 100,000 and 130,000 feet, it will be necessary to perform the input in at least two steps since the maximum possible N69 computer inputs limit is 99,999 feet. N69 inputs will probably be performed in 20,000-foot increments until GTC convergence.

In order to achieve the nominal EVA traverse objectives, it is necessary to land within 15,000 feet of the LLS. One-minute performance of +X RCS, beginning at PD1 + 2 minutes, is equivalent to a 15,000-foot downrange N69. Therefore, for low thrusting DPS, N69's between 0 and 30,000 feet, the one-minute +X RCS will be performed.

Figure 6-1 summarizes all of these limits; however, it should be noted that having to perform these technique is very improbable since the nominal throttle-down point protects against 33 low DPS, 33 negative N69, and a failed DPS shutoff valve.

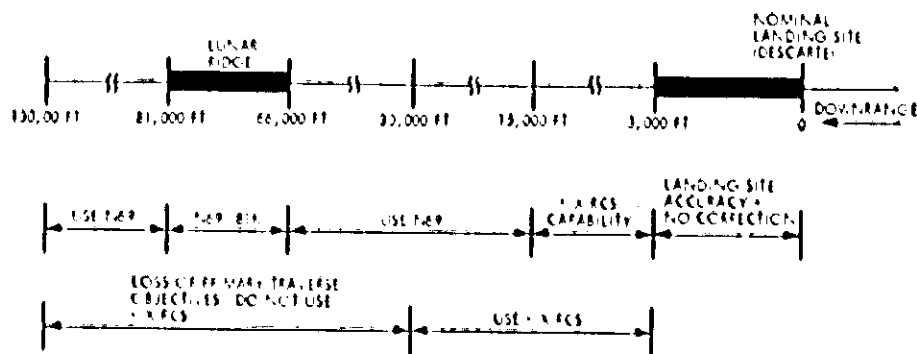


Figure 6-1. N69 and +X RCS Limits for Low DPS Thrust

6.5 AGS ALTITUDE/ALTITUDE RATE UPDATE (Page 4-20 and New Section)

On Apollo missions 16 and 17, the AGS altitude (H) and altitude rate (\dot{H}) will be updated at a predetermined PGNCs/LR-indicated altitude. The H update is performed first and is followed by the \dot{H} update ASAP. The tradeoffs for selecting the update altitude are (1) perform early enough to avoid a crowded timeline near TD, (2) perform late enough to reduce late abort insertion errors, and (3) perform where \dot{H} is near zero to avoid \dot{H} update errors due to inherent time delays. For Apollo 16, this altitude is at 14,000 feet where \dot{H} is essentially constant and can be read to within 1 foot per second (Reference 37). The crew anticipates these values, preloads them, and enters them when the tapemeter shows the anticipated value. Updating the AGS reduces the AGS abort insertion uncertainty and allows the AGS navigation to be used for descent in the event of a PGNCs failure after high gate.

Should the PGNCs fail after high gate, the AGS will be recalibrated with landing radar from the DSKY (raw radar) or tapemeter, if available. It should be noted that the LR \dot{H} may drop out just before landing.

6.6 TERRAIN MODEL (Page 4-22)

The LGC has contained a terrain model since the Apollo 14 mission. The model is used in the approach phase and results in a smoother LPD angle profile and reduced LPD pointing errors (Reference 38). The terrain model is valid for approach azimuth deviations within ± 10 degrees of the nominal (out to a range of 150,000 feet from the landing site). If angular dispersions of these magnitudes occur, the crew will either perform a landing-site redesignate back to the landing site or take over manually. In either case, termination of terrain model in the LGC via V68 is not necessary. Auto landings to other than the planned landing site (outside of the landing ellipse) do require termination of the terrain model. The Apollo 16

lunar terrain should cause LR altitude updates of up to 5,000 feet in the early portion of descent without the terrain model.

6.7 DESCENT ABORT VELOCITY RESIDUAL LIMITS (Page A-1)

The baseline document presents a detailed discussion of the MCC-H techniques for monitoring the PGNCs, AGS, and trajectory performance. The present philosophy of ground descent monitoring is to abort when the monitored parameters indicate that, if above high gate, the PGNCs has degraded to a point where, if an abort was required, the LM could not achieve an orbit with a pericynthion altitude greater than 30,000 feet. The monitoring techniques are based on comparing the three components of velocity residuals of the PGNCs, AGS, and PFP.

Table 6-1 shows the updated descent abort velocity limits for Apollo 16, and Table 6-2 shows how these limits are affected if no LR altitude data are incorporated into the descent solution (Reference 39). It is worth noting that the positive (south) crossrange velocity limit has been increased because there is no terrain obstruction to be avoided at Descartes like there was at Hadley (Apollo 15).

6.8 FAILED CDU TECHNIQUES (New Section)

Although a failed CDU is an extremely low-probability event*, Table 6-3 gives the prime and backup corrective techniques for static and dynamic CDU failures during pre-PDI, PDI, and lunar surface/ascent phases. As indicated, the static failures for any axis and only the dynamic failure in yaw** can be accommodated. No techniques exist for dynamic failure cases after PDI because an erasable memory program cannot be loaded during powered flight. Multiple CDU failures are grounds for abort due to loss of redundant auto attitude control capability.

*Reference 18 shows that there have been 178 PGNCs failures in 7 years with only 13 of them involving the CDU. None of these were static failures.

**A dynamic yaw failure requires an erasable memory program which may not be verified for Apollo 16 because of low priority.

Table 6-1. Descent Abort Velocity Limits (Reference 39)

Axis	Velocity Error Confirmed by: ^a	Limit (fps)	Remarks
Downrange + = long - = short	PFP-PGNCS ^b	+90	Impending accelerometer failure; PGNC auto landing not possible ^d
	and ACS-PGNCS (PGNC accelerometer bias)	-35	Low perilune for late aborts
	PFP-PGNCS only	As above	As above
Crossrange + = south - = north	PFP-PGNCS ^b and ACS-PGNCS (PGNC accelerometer bias)	+90	Impending accelerometer failure; PGNC auto landing not possible ^d
	PFP-PGNCS ^b only (PGNC misalignment, drift)	+200	Gyro drift at impending failure; PGNC auto landing not possible ^e
Radial + = high - = low	PFP-PGNCS ^b and ACS-PGNCS (PGNC accelerometer bias)	+60 ^c	Impending accelerometer failure; low perilune for hover abort; violate APS abort boundary ^d
		-35	Low perilune for late aborts (6:30 to 7:30)
	PFP-PGNCS only (PGNC misalignment, drift)	+45 ^c	Low perilune for hover abort
		-35	As above

^aIf both PFP and ACS are not available, an abort is performed if LR data are not incorporated into the PGNCs by 7:30 (H = 22,000 ft).

^bIf PFP data are not available, ACS-PGNCS data verify a PGNC error.

^cIf LR incorporation is late, maximum ΔH allowable by PGNCs may be exceeded, resulting in violation of APS abort boundary.

^dAccelerometer failure: bias = 0.164 ft/sec² (75g)

^eGyro failure: drift = 1.5 deg/hr (50°).

Table 6-2. Descent Abort Velocity Limits for no LR-H Incorporation^a (Reference 39)

Abort Criteria	PGNCS Altitude to Abort (ft)	Maximum Radial Velocity Residual (fps)	Remarks
PFP-PGNCS ^b and AGS-PGNCS (PGNCS ACC bias)	> 22,000	-35	Low perilune
	≤ 22,000	-10	Low perilune; includes 20-sec information delay
PFP-PGNCS only (PGNCS misalignment, drift)	> 18,000	-35	Low perilune
	≤ 18,000	-20	Intersects lunar surface for APS abort at 10,000 ft with delays
Radial N69 not incorporated	= 10,000		
Minimum safe altitude	= 6,000		

^aFor failures in PGNCS/LR interfaces, incorporation may be satisfied by a PGNCS/LR ΔH less than 1,500 ft.

^bIf PFP is not available, AGS-PGNCS data verify a PGNCS error.

Table 6-3. Failed CDU Techniques (Reference 18)*

Failure Phase	Static CDU Failure			Dynamic CDU Failure	
	X (yaw)	Y (pitch)	Z (roll)	X (yaw)	Y and Z
Pre-PDI	A AGS attitude hold Zero X-axis V21N20E Fly error needles	A AGS attitude hold MIT EMP 103 Fly V16N87 Optional: zero Y-axis V22N20E Fly N22; H and H-DOT	A AGS attitude hold Zero Z-axis V23N20E Fly pitch needle or N22 Optional: EMP 103 Fly N87	A AGS attitude hold MIT EMP 103 Fly error needles	Unacceptable - loss of redundant three-axis attitude control
	B PGNS attitude hold Zero X-axis V21N20E Fly error needles	B PGNS attitude hold Fly N22; H and H-DOT V22N20E	B PGNS attitude hold Zero Z-axis V23N20E Fly pitch error needle or N22	B PGNS attitude hold or AUTO MIT EMP 103 Fly error needles (Disable yaw jets)	
PDI		A AGS attitude hold Zero Y-axis Fly N22; H and H-DOT B PGNS attitude hold Zero Y-axis Fly N22; H and H-DOT			Unacceptable - loss of redundant three-axis attitude control
Lunar surface/ ascent	A AGS AUTO			A AGS AUTO	AGS AUTO
	B PGNS attitude hold Zero appropriate axis Fly N22 or error needles if available			B PGNS (ACS) Attitude hold EMP 103 Fly needles	

A Prime control mode

B Backup control mode

* Landing radar must be inhibited for LGC incorporation when maneuvering in the failed axis.

** PGNS attitude hold consists of two-axis attitude hold (rate command) and one-axis direct.

It should be pointed out that the LR is required for landing and must be inhibited for LGC incorporation when maneuvering in the failed axis. This requires inhibiting the LR for a pitch failure and requires inhibiting the data during roll maneuvers for the out-of-plane correction. For the pitch failure case, a comparison of LR versus PGNS will be made at high gate, and, if the difference is greater than 1,500 feet, the crew will abort.

6.9 RCS FAILURE MANAGEMENT (New Section)

The failure of certain (critical) RCS jets can require a PDI NO-GO or an abort depending upon where in descent the failures occur (Reference 39). Also, the type of failure (mechanical or electrical) contributes to the resulting corrective techniques.

The single and multiple RCS failure cases are given below:

a) Single thruster failure

- 1) If a single vertical thruster mechanically fails off or fails off via the jet driver (can not be fired via PGNS or AGS) this will violate redundant three-axis attitude control and PDI is NO-GO.
- 2) If any single horizontal or vertical thruster mechanically fails on, this will violate redundant three-axis attitude control (RCS system must be isolated); then PDI is NO-GO and will be cause for abort from PDI to high gate.

b) Multiple thruster failures (critical jets)

An abort will be performed from PDI to high gate for the following combinations of thruster failures:

- 1) Any combination of horizontal or vertical thrusters which fail on in critical quads that require isolation.
- 2) Confirmed failed off vertical thrusters in critical quads. To provide confirmation, a PGNS ATT HOLD check will be performed when the critical vertical thruster is suspected to be failed off.
- 3) Any horizontal or vertical thrusters fail on (requiring isolation) and the critical vertical thruster fails off.

The corrective procedures for the failure cases are as follows:

a) Single thruster failure

- 1) Use normal crew procedures
 - If stable, cycle CWEA
 - If unstable, open CB on affected quad and disable LGC thruster pair CMDS
- 2) MSFN will advise crew of type of failure and pass crew critical pair after first failure

b) Critical thruster failures prior to high gate

- 1) MSFN will determine optimum configuration and GO/NO-GO status
- 2) Critical thruster should never be isolated except on call from MSFN (does not apply to first 26 seconds)

c) Critical thruster failures after high gate

- 1) Never isolate second failure
- 2) Control mode should be determined by crew practice. Acceptable control modes are
 - PGNS AUTO
 - PGNS ATT HLD
 - AGS ATT HLD
- 3) Avoid throttle transients
 - Stay in auto throttle, if possible
 - Aborts should be performed with gradual throttle up to soft stop

7. ABORT FROM LUNAR POWERED DESCENT AND SUBSEQUENT RENDEZVOUS

This section discusses two updates to the Abort from Lunar Powered Descent and Subsequent Rendezvous Mission Techniques (Reference 6).

7.1 SUMMARY OF DESCENT ABORT/RENDEZVOUS SEQUENCES (Page 3-2)

Figure 7-1 (Reference 40) presents a summary of the descent abort points and the number of revolutions involved in the subsequent rendezvous. This summary shows that T_3 (concentric) aborts are preferred to T_2 aborts since the rendezvous is two revolutions shorter and the time of rendezvous is one revolution earlier. (Direct rendezvous is two revolutions earlier.) The abort/rendezvous sequences for Apollo 16 are changed from those for Apollo 15 as a result of the interval between CIRC and PDI being shortened from one and one-half revolutions to one-half revolution. (The Apollo 16 sequences are similar to those used for Apollo 11 and 12.) The major changes are

- a) The PDI₀ abort, which is initiated one-half revolution after CIRC, is not applicable on Apollo 16 because of the shift in the time of PDI relative to CIRC (one-half revolution).
- b) The no-PDI + 12-minute abort will be a standard 3-1/4 hour coelliptic rendezvous sequence for Apollo 16. Apollo 15 used the BOOST/HAM/CSI/CDH sequence which required 5-1/4 hours.
- c) The powered descent aborts use the nominal two-revolution CSI/CDH sequence until approximately PDI + 10 minutes. After that, the three-revolution BOOST/HAM/CSI/CDH sequence is used up to and including T_1 . T_1 is at PDI + 17 minutes:30 seconds whereas previously, it was at PDI + 15 minutes.
- d) The T_2 abort is the same as a T_1 abort except that there is one extra revolution required between the BOOST and HAM maneuvers.

The LM-active rendezvous sequences are summarized in Table 7-1. As shown in the table, if PDI is slipped one revolution, the PDI2 sequence will be used. These are essentially the same as the nominal PDI sequence for

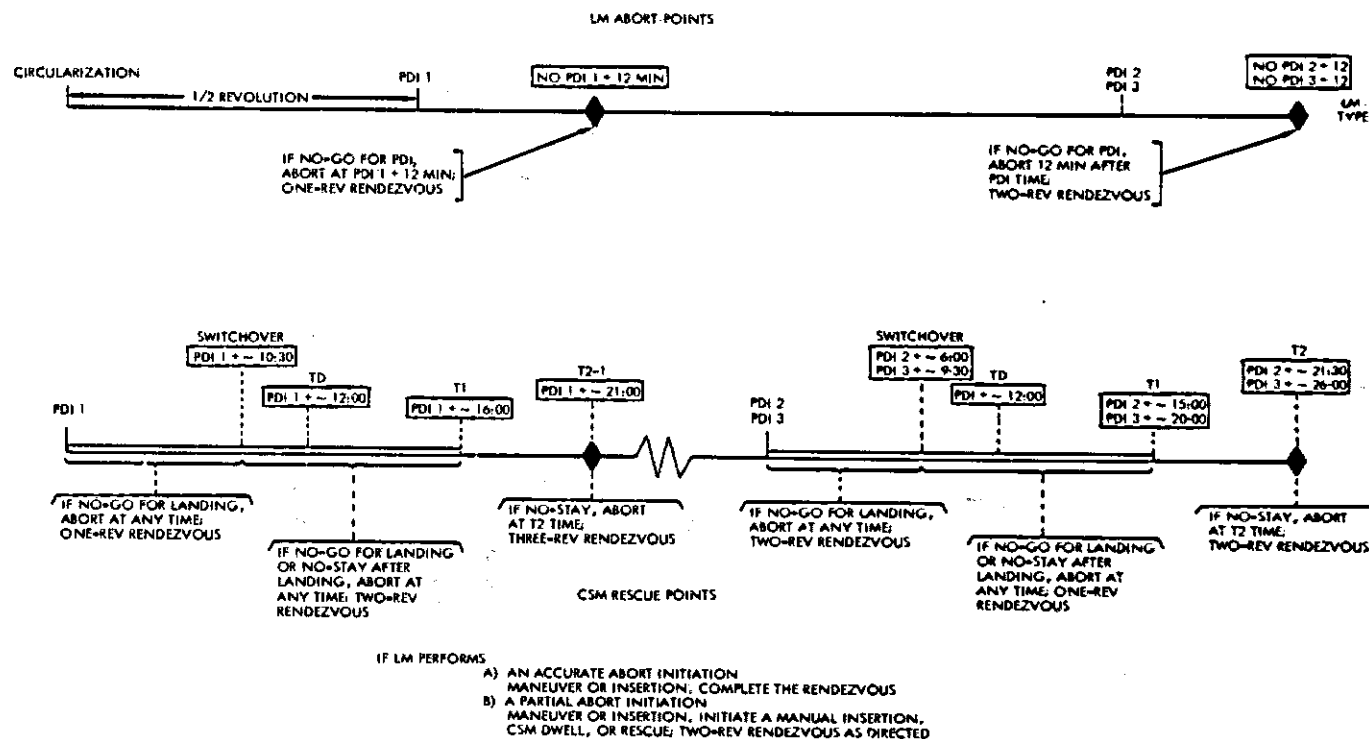


Figure 7-1. Summary of Descent Abort Points (Reference 40)

Table 7-1. Summary of LM-Active Rendezvous Sequence

Fridge, DuPont/PDB/OMAB/MPAD (for LM Timeline)

Data source SCOT

Data confirmed Cont - 0.78

LM 3-1, PDI abort summary data, (March 17 launch date)

Launch month dependent
Mission profile dependent

11/29/71 Final

PAGE	ABORT	INS			BOOST	HAM	CSI		CDM			TPI	AIM		
	TIME PDI+	TIME PDI+	N76	HA/HINS	TIME INS+	TIME INS+	TIME INS+	ΔVX	TIME INS+	ΔVX	ΔVZ	TIME PDI+	TIME PDI+	ΔVX	ΔVZ
	NO 1+12	NA	NA	NA	NA	NA	1+00+00*	47.3	2+02+17*	-119.5	-61.5	2+47+13	12+00	102.3	-50.1
	1+00	2+06	5655.2	137.0/54848	NA	NA	0+55+00	50.2	1+57+16	-120.1	-42.5	2+45+48	NA	NA	NA
	2+00	3+60	5649.9	135.8/60016				48.2	1+57+11	-117.6	-39.1				
	3+00	5+42	5645.8	132.4/60018				47.7	1+57+01	-113.1	-33.1				
	4+00	7+19	5638.9	126.9/60023				47.4	1+56+45	-106.0	-24.6				
	5+00	8+50	5629.2	119.2/60030				47.3	1+56+24	-96.2	-13.5				
	6+00	10+14	5616.6	109.2/60039				47.4	1+55+57	-83.8	-5				
	7+00	12+29	5596.3	96.4/65421				45.9	1+55+22	-67.1	14.2				
	8+00	14+30	5570.0	79.4/71108				44.4	1+54+36	-44.8	30.7				
	9+00	16+07	5545.5	62.5/73870				43.7	1+53+50	-22.2	43.7				
	10+00	17+15	5528.9	49.4/72124				43.9	1+53+15	-4.2	51.3				
	11+00	18+19	5515.8	37.2/66754				44.9	1+52+41	13.0	56.2				
	12+00	19+23	5550.3	59.6/61690	50+00	1+50+00	2+40+00	38.0	3+38+45	-19.5	-22.5	4+44+37	NA	NA	NA
	13+00	20+26	5541.5	52.4/60251				39.9	3+38+29	-11.3	-5.5				
	14+00	21+26	5534.0	46.9/60250				40.6	3+38+15	-4.5	6.7				
	15+00	22+25	5526.4	41.4/60248				41.2	3+38+02	2.5	17.9				
	16+00	23+25	5518.7	35.9/60246				41.6	3+37+48	9.7	28.1				
	17+00	24+24	5511.1	30.5/60244				41.7	3+37+33	17.1	37.3				
	12-1	7+210	5510.6	29.9/60235	50+00	3+50+00	4+40+00	36.6	5+37+21	22.8	54.4	6+43+32	NA	NA	NA
	NO 2+12	NA	NA	NA	1+07+00*	2+07+00*	3+07+00*	38.1	4+09+43*	-132.3	-2.9	4+49+31	12+00	111.4	-50.0
	1+00	2+07	5667.2	146.5/54190	1+00+00	2+00+00	3+00+00	40.9	4+02+43	-132.4	-9.6	4+49+28	NA	NA	NA
	2+00	4+01	5661.7	145.4/60016				38.6	4+02+38	-130.1	-7.1				
	3+00	5+43	5659.6	143.7/60019				38.0	4+02+32	-127.4	-1.6				
	4+00	7+20	5656.3	141.0/60024				37.5	4+02+24	-123.6	6.5				
	5+00	8+51	5651.6	137.2/60031				37.0	4+02+13	-118.5	16.8				
	6+00	10+16	5645.7	132.3/60040				36.6	4+01+59	-112.1	29.4				
	7+00	12+34	5657.2	145.0/65661	NA	NA	0+55+00	43.6	1+57+30	-125.6	-52.7	2+50+39	NA	NA	NA
	8+00	14+35	5632.3	128.1/71266				43.2	1+56+46	-105.9	-26.7				
	9+00	16+11	5609.3	111.3/73944				43.3	1+56+01	-85.6	-4.0				
	10+00	17+19	5593.8	98.2/72172				44.1	1+55+27	-69.3	11.3				
	11+00	18+22	5581.8	86.0/66795				45.6	1+54+54	-53.7	23.6				
	12+00	19+24	5569.6	74.0/61708				46.8	1+54+22	-37.9	34.4				
	13+00	20+26	5551.7	59.9/60253				47.2	1+53+44	-19.0	44.4				
	14+00	21+26	5536.9	49.0/60250				47.0	1+53+14	-4.0	50.6				
	15+00	22+25	5521.8	38.1/60246				46.8	1+52+45	11.3	55.0				
	12-2	7+210	5510.6	29.9/60235	50+00	3+50+00	4+40+00	42.2	5+37+31	17.6	38.5	6+49+33	NA	NA	NA

a INDICATES TIME IS REFERENCED TO LIFT-OFF.

* INDICATES TIME IS REFERENCED TO PDI.

Apollo 15. The CSM-active (rescue) rendezvous sequences have not changed with the exception of the rescue 2 values for the LM partial abort ΔV maneuver. These data are given in Table 7-2.

7.2 SOFTWARE INHIBIT FOR DESCENT ABORT AND AUTO THROTTLE BITS (New Section)

To avoid the descent abort switch malfunctioning difficulties encountered on Apollo 14, the descent abort signal received from the abort and abort stage switch will be masked in the PGNCs on subsequent Apollo missions. Should an abort be required, the abort sequence is

- Attitude Hold
- Full Throttle
- Abort or Abort Stage
- V22N46EE (to unmask the abort bit)

If the switch is failed open, V22N46EE will not work, and the abort program V37E70 or 71E is required. If that fails, the AGS will be used for the abort.

The last two steps of these procedures are reversed for T_1 aborts. In this case, unmasking is done first since the LGC will not command the abort pitch profile until the bit is unmasked.

The auto throttle discrete is also masked on subsequent Apollo missions. Although this protects against failures, it should be remembered that masking the auto throttle discrete negates the possibility of the crew returning to P66 auto throttle from manual throttle.

Table 7-2. Rescue-Two Summary (Reference 40)

<u>Abort Case</u>	<u>Partial Abort Maneuver (fps)</u>	<u>Number of Phasing Revolutions</u>	<u>Revolutions for Rendezvous</u>	<u>Initial Value of N (For Equation in Baseline Document)</u>
No PDI1 + 12	$0 \leq \Delta V < 20$	2	3-2/3	4
	$20 \leq \Delta V \leq 105$	1	2-2/3	2
No PDI2 + 12	$0 \leq \Delta V < 20$	3	4-2/3	6
	$20 \leq \Delta V < 60$	2	3-2/3	4
	$60 \leq \Delta V \leq 123$	1	2-2/3	2
No PDI3 + 12	$0 \leq \Delta V < 3$	5	6-2/3	10
	$3 \leq \Delta V < 20$	4	5-2/3	8
	$20 \leq \Delta V < 45$	3	4-2/3	6
	$45 \leq \Delta V < 95$	2	3-2/3	4
	$95 \leq \Delta V \leq 152$	1	2-2/3	2

8. LUNAR SURFACE PHASE

This section discusses the updates to the Lunar Surface Phase Mission Techniques (Reference 7). Following a brief review of the lunar surface activities, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

The stay time on the lunar surface is open-ended, and the planned maximum will not exceed approximately 73 hours. After checkout of the LM (and alignment of the PGNCs and AGS) to assess its launch capability, the crew will proceed with the first of three 7-hour EVA's. Traverse planning will provide for returning the crew to the LM under each of the following single-failure conditions:

- Use of the buddy-secondary life support system due to an inoperative PLSS anytime during a riding traverse (based on the assumption that the LRV will operate properly during the return to the LM).
- Use of two PLSS's for a walking return to the LM from an inoperative LRV anytime during a riding traverse (based on the assumption that both PLSS's will operate properly during the return to the LM).

Traverse planning does not provide for dual failure conditions such as two PLSS failures or an LRV failure combined with a PLSS failure.

ALSEP deployment operations will be performed, and photography will be obtained to document to EVA's. Television transmission will be provided during the EVA periods.

Ten revolutions prior to the LM lift-off, the CSM will perform the required plane change to permit a nominally coplanar rendezvous.

8.1 POSTTOUCHDOWN ALIGNMENTS (Page 7)

The postlanding IMU alignment sequence has been reduced from that of Apollo 15 to a single Lunar Surface Alignment Program (P57) AT-3 (AOT vector on one star and measured gravity vector). This alignment data will yield a measure of the IMU performance and the LM position and

attitude which are stored in the LGC. The attitude data allow a P57 AT-0 (LM Y- and Z-body axis vectors) to be used for an alignment if a subsequent lunar surface abort should be required. The AT-0 alignment reorients the platforms by an amount equal to the angular difference between the old and the desired REFSMMAT's.

During the AT-3 alignment, the displayed vector difference is the angle between the vehicle X-axis and the measured gravity vector. The AGS pseudo gimbal angles are monitored by the ground during the alignment to determine LM settling during the alignment. LM settling during the alignment will degrade the accuracy and possibly require a realignment. The advantage of the AT-3 alignment is that the IMU is accurately aligned with respect to the vertical, regardless of R_{LS} errors (assuming small gravity anomalies) and yields a correct flight-path angle at insertion. The small misalignment with respect to the basic reference coordinate (BRCS) system is not a problem for rendezvous.

8.2 PIPA UPDATES (New Section)

Reference 26 indicates that the last PIPA bias measurements prior to ascent have always been the most accurate; therefore, the final computed preascent PIPA biases update will be used if they are consistent with other measurements. Small, but significant, bias shifts have been observed on previous missions between IMU power down and ascent preparation. Reference 28 indicates that these PIPA bias shifts can be reduced by having the accelerometer output axes horizontal during the lunar stay. Table 8-1 presents the two methods of computing the angles that will be used to accomplish this after touchdown and before LM powerdown*. Also, the PGNCs will be turned on one hour earlier than previous missions, which gives more time to evaluate any trend in the bias computations.

8.3 PRELAUNCH CSM PLANE CHANGE (Page 15)

During revolution 40 (\approx 154 hours GET), the CSM will execute a lunar-orbit plane-change (LOPC) maneuver, which is targeted to have the CSM pass over the landing site during the nominal lift-off revolution (50).

* The LGC will be in STANDBY and the IMU in OFF until after the sleep period following the third EVA. The LGC will be activated periodically for clock reset and CSM state vector update.

Table 8-1. Gimbal Angles for PIPA Bias Shift Reduction (Reference 28)

Equations \vec{Y}_{SM} Parallel to -G

given (P - 57 gravity determination)

$$-G = \alpha X_{NB} + B Y_{NB} + Y Z_{NB}$$

$$IGA = 0.00 \text{ deg}$$

$$MGA = \sin^{-1} [\alpha] \quad (\alpha > 0)$$

$$OGA = \begin{cases} \cos^{-1} \left[\frac{B}{\cos MGA} \right] \\ \sin^{-1} \left[\frac{-Y}{\cos MGA} \right] \end{cases}$$

Y	B	Quadrant
+	+	4
+	-	3
-	+	1
-	-	2

II (-, +)	I (+, +) (cos, sin)
III (-, -)	IV (+, -)

Simple Equations

Copy CDU angles at touchdown (V06N20)

V41N20

R1: OGA

R2: IGA

R3: MGA + 90 deg where $-90 \text{ deg} < MGA < 90 \text{ deg}$

V41 V33E to fine tune parking

A late revolution of lunar stay is selected to minimize a subsequent LOPC if the CSM should have to support an early LM lift-off. The inplane horizontal residual (Vgy) of the LOPC will be trimmed to within ± 0.2 foot per second to retain the targeted orbital period. Trimming X and Z is not necessary. Keeping the period nominal (± 2 minutes) avoids changing the flight plan. The CSM is rolled so that the downward firing Z-thrusters can be used; this avoids SIM-bay contamination by the RCS. It also minimizes the plane-change yaw steering required by the LM to support a late lift-off. The baseline document discusses the use of MSFN data following LOPC to target the LM for yaw steering into the CSM orbit plane.

The manual LOPC takeover/terminate limits are

- Attitude: 10 degrees
- Attitude rate: 10 degrees/second
- Burn termination: at burntime + 1 second

It should be noted that during the two revolutions before lift-off of lunar stay, the CSM will avoid uncoupled attitude control maneuvers so that the MSFN orbit determination will be free of perturbations.

8.4 PRE-ASCENT ALIGNMENTS (Page 16)

Figure 8-1 shows a new procedure (Reference 28) for obtaining the star data for the first pre-ascent P57 (AT-3 Option 4). This technique is easier and faster for the crew and gives MCC-H more time for IMU evaluation.

Option 4 was selected so a drift check can be performed after the final P57 which occurs 2 hours later. The four-star sighting will be used for the postflight determination of the position of the LLS.

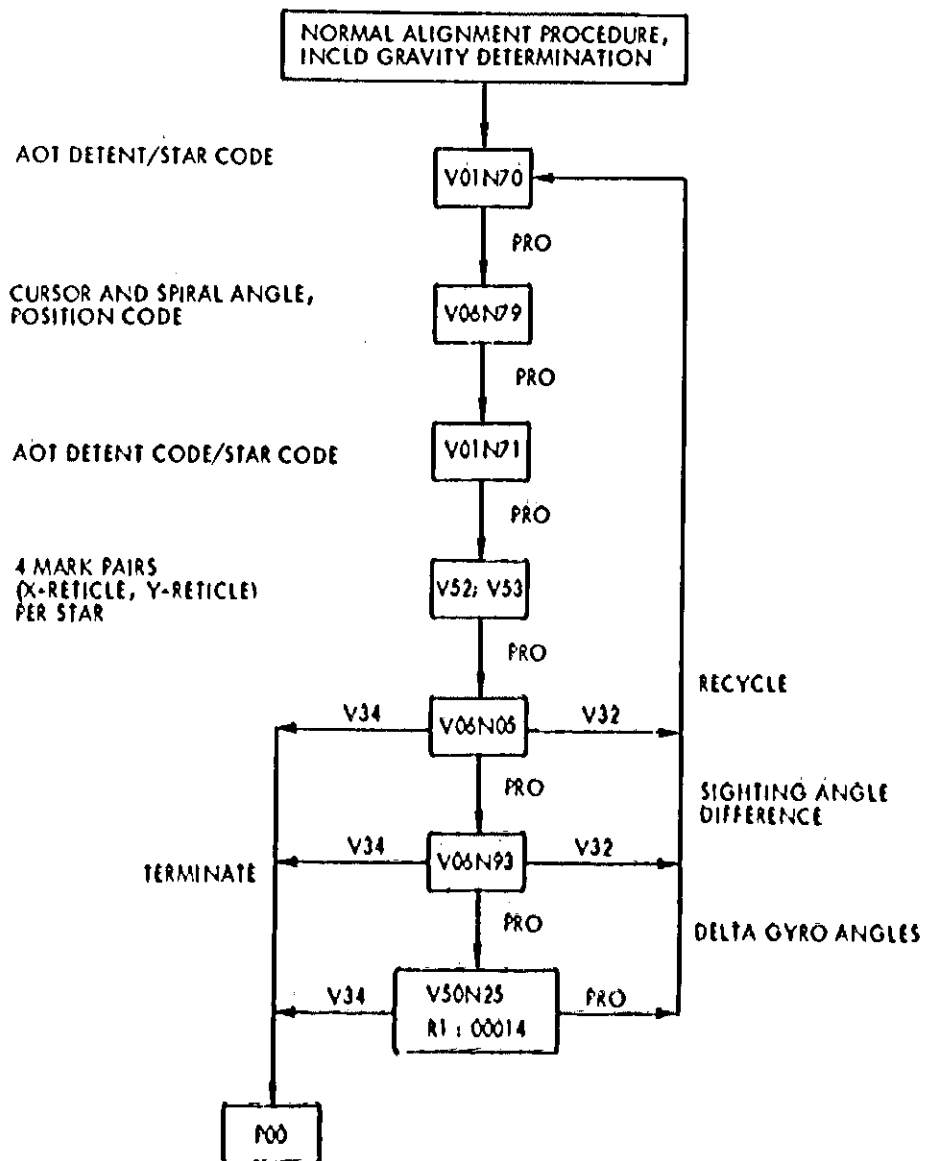


Figure 8-1. P57 AT-3 Option 4 (Four Stars) (Reference 28)

8.5 ASCENT PREPARATION TIMELINE

Figure 8-2 shows the ascent preparation timeline for Apollo 16. The significant changes involve

- a) Moving the PGNC'S TURN ON and the first P57 AT-3 one hour earlier to increase the time between the two P57 AT-3's for the drift check.
- b) Moving APS PRESS from LO - 30 minutes to LO - 10 minutes to improve the trajectory/rendezvous situation. An APS leak at APS PRESS results in an early lift-off, which requires a LM BOOST after insertion to adjust the phasing. At least 20 percent RCS must be reserved for attitude control during rendezvous. Both the ground and crew have charts for this maneuver.
- c) Moving ENGINE ARM from LO - 30 seconds to LO - 10 seconds. Since the possibility exists for some failure to cause an early lift-off at ENGINE ARM, the event was moved closer to the nominal LO time. The allowable error in lunar surface lift-off time is ± 10 seconds (Reference 41) with an appropriate tweak maneuver to establish proper planning for direct rendezvous. The only source for a 10-second early lift-off is probably an error in the clock or ignition at ENGINE ARM. A lift-off of more than 10 seconds early requires a tweak maneuver, which result in a pericynthion altitude of less than 5 nautical miles; therefore, a bailout maneuver would be required. If lift-off does not take place within 10 seconds of the nominal TIG, it is probable that the auto start and manual start failed, and the source is not known. Should this happen, the nominal lift-off attempt will be terminated and a second attempt made one CSM revolution later.
- d) Turning the MSFN RELAY ON before ascent. Prior to ascent, the MSFN RELAY will be turned ON to protect against the no-VHF communication contingency. It should be noted that turning the MSFN RELAY ON (causing up to a 3-second transmission delay) can invalidate the RR/VHF comparison after insertion; however, the Apollo 16 crew requested that it be turned ON, (MSFN RELAY turn-on is discussed in Section 5.14).

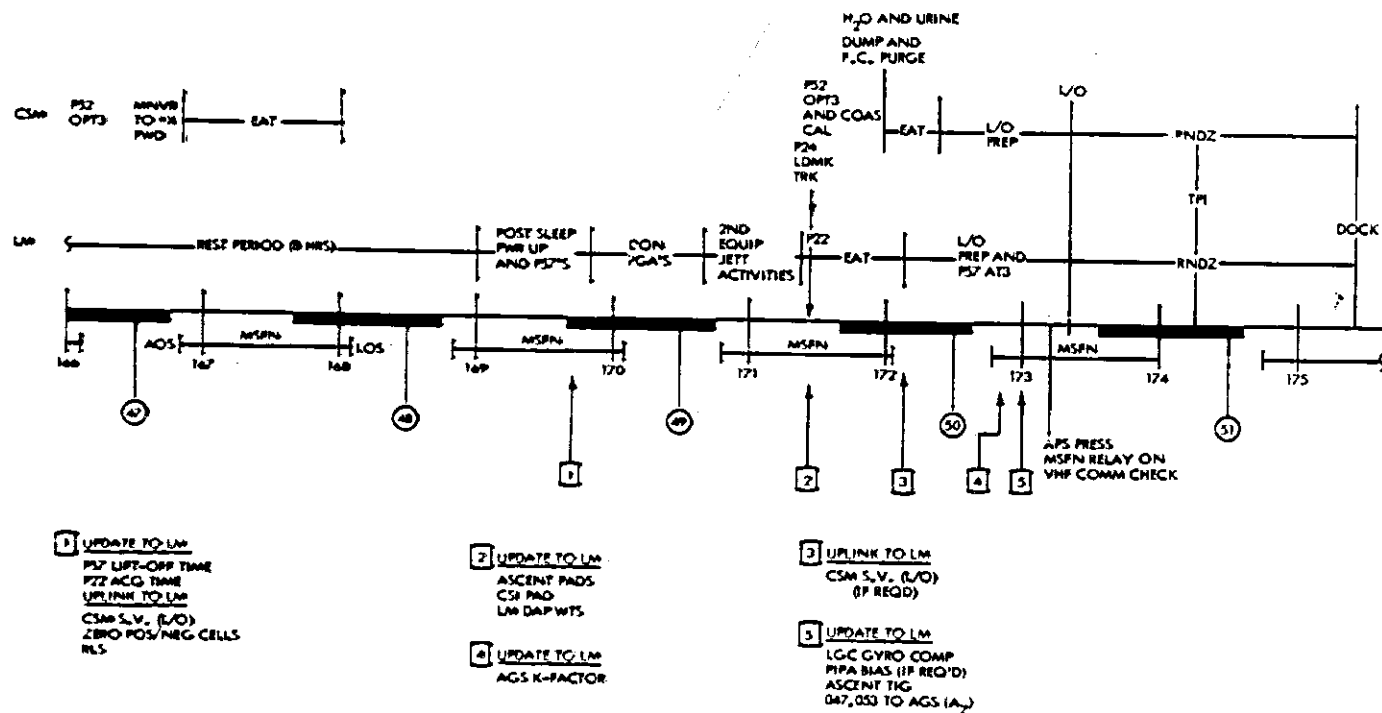


Figure 8-2. Ascent Day Summary Timeline

8.6 TYPE OF RENDEZVOUS FOR PREMATURE LIFT-OFF (New Section)

The Lunar Orbit Activities Section discusses the logic and rationale for a direct rendezvous through the tweak decision. However, for premature lift-off from the lunar surface, two other constraints are imposed:

- a) If there is no communication, the CSI/CDH rendezvous will be used. The no-communication contingency lift-off times sent to the crew while on the lunar surface will always assume a CSI/CDH profile.
- b) For those abort times where the plane change maneuver is small, (just after landing and prior to ascent) the direct rendezvous will be used if one primary G&N system is GO.

9. LUNAR POWERED ASCENT

This section discusses the updates to the Lunar Powered Ascent Mission Techniques (Reference 8). Following a brief review of the powered ascent activities, these changes and additions will be discussed in the order in which the subjects appear in the baseline document.

Ten revolutions prior to LM lift-off, the CSM will complete the required plane change to permit a nominal coplanar rendezvous. Powered ascent will be accomplished using the APS under PGNCs control (nominal). The trajectory consists of a 10-second vertical rise (yawing to the proper azimuth*) followed by a pitchover maneuver. The LM inserts into an approximate 9- by 48-nautical mile elliptical orbit after an 8-minute period of powered flight. LM lift-off will occur about 2-1/2 minutes earlier than the nominal time for a coelliptic rendezvous to permit use of the early rendezvous technique. The mission techniques for ascent are designed to provide safe insertion conditions acceptable for rendezvous. The crew is responsible for detecting and isolating system problems that require immediate action. The ground is responsible for slow-drift malfunctions; therefore, S-band high-bit-rate-communications with the ground are required. Also, a degraded PGNCs will knowingly be used as long as a safe orbit insertion can be achieved before switching to the AGS to protect against a possible switch-over malfunction.

9.1 S-BAND HIGH BIT RATE DATA (Page 2-7)

S-band high-gain antenna communication should be maintained automatically during ascent. On previous missions, a yaw maneuver at approximately 1 minute after lift-off was needed to insure that the steerable antenna was not blocked by LM structures. On Apollo 16, the antenna LOS geometry is such that no yaw maneuver is required.

9.2 RENDEZVOUS RADAR ASCENT PROCEDURES (Page 2-9)

Early RR AOS is desirable for crew monitoring during ascent and to reduce the postinsertion timeline schedule. (The RR is used as a third

* The MCC-II computed value for AGS azimuth will be used for subsequent missions because of an onboard computational error which was discovered on Apollo 15.

vote for disagreements between the PGNCs and AGS). Table 9-1 presents the procedures for manual RR lockon during ascent; however, it should be understood that the ascent procedures will not be used unless the crew is in a situation of no communication. The pre-ascent RR procedures given in Table 9-1 are part of the normal pre-lift-off checklist and will be accomplished to support manual RR lockon should a subsequent communication failure occur.

Table 9-1. Rendezvous Radar Ascent Procedures*

Pre-lift-off procedures

- 1) Select secondary gyros for minimum trunnion drift
- 2) Designate RR antenna (V41N72) to
 Trunnion (R_1) = 180 degrees
 Shaft (R_2) = 333 degrees
- 3) Slew up (high rate) for 3 seconds, antenna should be at
 Trunnion = 180 degrees
 Shaft = 0 degree
- 4) Crew may read RR angles (ground also) if the Normon Flag is set prior to P12

Ascent procedures to be used in the event of a communication failure (at lift-off + 4 minutes)

- 1) Slew up (high rate) for 24 seconds, to ensure antenna against +68 degrees hardstop (mechanical limitation)
- 2) Slew down (high rate) for TBI seconds stopping short of desired LOS
 Example: (Using Apollo 15 LOS for shaft = 275 degrees)

$$\frac{360 \text{ deg} - 275 \text{ deg} + 68 \text{ deg}}{9.18} \cdot 1 = 15.6 \text{ sec}$$
 * 15 sec at high slew rate
- 3) Slew down (low rate) until AGC meter indicates signal strength
- 4) Maximum signal strength if desired, then select auto track
- 5) Possible trunnion drift error of
 57 sec \times 0.58 deg/min = 0.5 deg error

10. MANUAL ASCENT

This section discusses the updates to the Manual Ascent Mission Techniques (Reference 9). Following a brief description of the manual ascent problem, these changes and additions are discussed in the order in which the subjects appear in the baseline document.

The baseline techniques present the guidance and navigation sequence of events which must be performed for a powered ascent using either a manual technique or degraded guidance system. A manual ascent is defined as one in which the attitude control in at least one axis is provided manually and the attitude errors in the manually controlled axes are not automatically computed and displayed (i. e., an ascent in which the astronaut flies the error needles is considered an automatic ascent since the attitude errors are automatically computed by the PGNCs or AGS).

Manual ascent from the lunar surface or from an aborted descent would be necessary in the event of a double failure that disabled automatic guidance steering in both PGNCs and AGS. While the probability of any specific pair of failures occurring is extremely small, there are many combinations of double failures which would disable both PGNCs and AGS automatic steering.

It is generally agreed that

- a) Given a rate command attitude control system, the crew should be able to guide the LM into a safe orbit quite satisfactorily using the lowest priority attitude reference (horizon viewed through the overhead window).
- b) Both the AGS/CES and the PGNCs have a substantial capability, even if the accelerometers are malfunctioning.
- c) The failure of an attitude gyro in either guidance system virtually eliminates the use of that guidance system for vehicle control. The failure of a CES rate gyro causes loss of AGS automatic capability in the disabled axis only.

For a safe manual insertion, it is necessary for the crew to have both an accurate attitude reference and a usable attitude control mode*. An attitude control system is considered unusable if it provides insufficient control authority or requires greater than human (astronaut) manual dexterity. The pulse and minimum pulse modes of attitude control are not considered since it is thought that they lack sufficient control momentum to overcome ascent engine disturbance torques.

10.1 SEVEN-STEP PITCH PROFILE (Entire Document)

The seven-step manual ascent pitch profile shown in Figure 10-1 replaces the eight-step profile discussed in the baseline document. The seven-step profile was selected because it closely follows the nominal \ddot{H} versus \dot{H} profile (Reference 42). Figure 10-1 shows typical FDAI angles and overhead window (OHW) angles of the horizon. The crew prefers to use the FDAI/30-second step profile rather than the seven-step manual ascent pitch profile if a valid FDAI and rate damped controls are available.

10.2 LIFT-OFF TIG PROCEDURES (Entire Document)

The lift-off TIG procedures given in the baseline document erroneously reversed the order of the ARM APS and push ABORT STAGE. The correct ascent TIG procedures are

- Push ABORT STAGE
- ARM APS
- Push START (LGC will insure auto engine ON)

10.3 SUMMARY OF MANUAL ASCENT PROCEDURES (Page 2-6)

The manual ascent procedures shown in Table 10-1 include two changes in the attitude display column and involve

- The eight-step FDAI profile has been replaced by a seven-step profile. This applies to the entire baseline document as discussed in Section 10.1.

*The J-1 Update Document erroneously updated the priority of the control modes to indicate that the PGNCS ATT HOLD mode was preferred to AGS AUTO in the case of a DAP failure. This is not the case, and the priorities given in the baseline document are correct.

F5

DEDA ADDRESSES

R DOT - 440
R - 317
Ha - 315
Hp - 403
VI - 433
H DOT - 367
H - 337
- 227

Y DOT - (Present)(-)270
- (Next Man)(-)263
STORE A - 415 & 1E
STORE R DOT - 503E
STORE R - 316E
VI6N78 RR RNG/RNG RT
VI6N92 %THROT/H DOT/H
V21N69 ΔKLS

ΔV 404, 5,6=0 470,71,72R

8/21/70

MANUAL ASCENT

FDAL 7-STEP

:15 PITCH DN TO 305°
2:00 295
3:00 290
4:00 280
5:00 275
6:00 265
7:00 260

TO APS
DEPLETION

OHV 7-STEP

:15 PITCH DN TO 38°
2:00 31
3:00 28
4:00 20
5:00 18
6:00 11
7:00 8

TO APS
DEPLETION

Figure 10-1. Typical Seven-Step Pitch Profile for Manual Ascent (Apollo 14; Reference 42)

Table 10-1. * Summary of Manual Ascent Procedures

Failure	Attitude Control	Attitude Display	Alignment for Lift-off	APS Cutoff	Comments
PGNCS failed PGNCS or accelerometer failed	PGNCS rate command/ attitude hold	PGNCS FDAI. Fly seven-step FDAI profile	Program 57 AT-2 (two-star) to standard lift-off REFSMAT	MCC-H voice or fuel depletion	The LGC program during ascent is P12.
AGS failed LGC failed	AGS CES rate command	PGNCS FDAI or horizon reference. Fly seven- step profile	Either keep the current alignment or cage the IMU to LH body axis and determine body attitude from two star sightings.	MCC-H voice or fuel depletion	If possible, body axis align AGS
PGNCS failed AGS Y-axis accelerometer failed	AGS rate command/ attitude hold	AGS FDAI. Fly seven- step FDAI profile	AGS laser align and one star shot for azimuth.	MCC-H voice or fuel depletion	AGS will be provided azimuth data from star sighting.
PGNCS failed AGS Z-axis accelerometer failed	AGS rate command/ attitude hold	AGS FDAI if AGS aligned or horizon. Fly seven-step profile.	Current alignment or two star sightings and body axis align if MCC-H recommends	MCC-H voice or fuel depletion	If possible, cage IMU
PGNCS failed AGS Y-axis accelerometer failed	AGS rate in all three axes	AGS FDAI if AGS aligned or horizon. Fly nominal profile.	Current alignment or two star sightings and body axis align if MCC-H recommends	Automatic APS cutoff	Zero scale factor and bias of the broken accel- erometer. Enter 625 + 00000 to orient Y-axis accel- erometer out of the maneuver plane. Provide AGS state vectors and, if possible, cage IMU.
PGNCS failed One AGS rate gyro failed	AGS rate in the two unaffected axes. AGS shows in the axis of the disabled gyro.	AGS FDAI. Fly nominal FDAI profile or disabled axis if required.	AGS laser align and one star shot for azimuth.	Automatic APS cutoff	Provide AGS state vectors and, if possible, cage IMU. AGS will be provided azimuth data from star sighting.

* This table replaces Table 2-4 in Reference 9.

- The last two failure cases have been corrected to show that the crew should fly a nominal profile. These two cases involve a failed PGNCS and either an AGS Y-axis accelerometer failure or an AGS rate gyro failure. In the case of the failed AGS Y-axis accelerometer, the bias and scale factor are set to zero, and the AGS automatic attitude control is available in all three axes. For the failed AGS rate gyro case, two-axis automatic control is available, and the crew will manually steer the nominal profile in the inoperative axis using the FDAI as an attitude reference, if required. These corrections also apply to the detailed discussion presented in Sections 4.3 and 4.4 of the baseline document.

10.4 THREE-STEP MANUAL DESCENT ABORT (Page 2-9)

Table 10-2 shows typical FDAI and overhead window angles for the three-angle technique of manually controlling aborts from powered descent. This three-angle technique is based on a 14-second vertical rise for a DPS abort and a 40-second vertical rise for an APS abort. All pitch maneuver rates are 5 degrees/seconds, and extreme care must be taken in not exceeding this rate since overrates result in low perilunes (Reference 39).

Table 10-2. FDAI and Overhead Window Angles for Manual Descent Abort (Reference 43)

DPS/APS			APS		
1:00 250/0	4:30 0/LV	8:00 - 0/LV	1:00 250/0	4:30 - 0/LV	8:00 - 0/LV
2:26 SD(FDAI)	4:44 300/36	8:14 300/36	2:24 SD(FDAI)	5:10 300/36	8:40 300/36
2:36 SD(OHW)	6:26 270/5	10:16 270/1*	2:36 SD(OHW)	6:22 270/10	12:06 270/14
	7:32 250/0	14:30 250/0		8:26 250/0	14:26 250/0
	8:19 - SD	14:43 - SD		8:31 - SD	15:05 - SD
1:30 0/0	5:00 0/LV	8:30 0/LV	1:30 - 0/LV	5:00 0/LV	8:30 - 0/LV
1:44 300/0	5:14 300/36	8:44 300/36	2:10 250/0	5:40 300/36	9:10 300/36
2:30 250/0	6:58 270/5	11:12 270/15	3:32 SD(FDAI)	7:08 270/10	13:00 270/14
3:20 SD	8:20 250/0	15:18 250/0	3:52 SD(OHW)	9:20 250/0	15:02 250/0
	9:03 - SD	15:31 - SD		9:46 - SD	15:50 - SD
2:00 0/LV	5:30 0/LV	9:00 0/LV	2:00 - 0/LV	5:30 0/LV	9:00 - 0/LV
2:14 300/36	5:44 300/36	9:14 300/36	2:40 300/0	6:10 300/36	9:40 300/36
3:00 300/0	7:34 270/10	12:10 270/16	3:10 250/0	7:54 270/12	13:54 270/14
3:18 250/0	9:04 250/0	16:04 250/0	4:24 SD(FDAI)	10:14 250/0	15:34 250/0
4:16 - SD	9:46 - SD	16:18 - SD	4:34 SD(OHW)	10:42 SD	16:31 - SD
2:30 0/LV	6:00 - 0/LV	9:30 0/LV	2:30 - 0/LV	6:00 0/LV	9:30 0/LV
2:44 300/36	6:14 300/36	9:44 300/36	3:10 300/0	6:40 300/36	10:10 300/36
3:46 300/0	8:04 270/11	12:56 270/16	4:00 250/0	8:44 270/14	14:30 270/14
4:02 250/0	9:50 250/0	16:44 250/0	5:18 SD	11:08 250/0	16:12 250/0
5:12 SD	10:39 SD	16:58 SD		11:37 SD	17:07 - SD
3:00 0/LV	6:30 0/LV	10:00 0/LV	1:00 0/LV	6:30 - 0/LV	10:00 0/LV
3:14 300/36	6:44 300/36	10:14 300/36	3:40 300/36	7:10 300/36	10:40 300/36
4:34 270/0	8:34 270/11	13:16 270/16	4:22 270/0	9:34 270/14	14:30 270/14
5:10 250/0	11:04 250/0	17:32 SD	5:28 250/0	12:02 250/0	17:00 250/0
5:57 SD(FDAI)	11:47 SD		6:09 SD	12:33 SD	17:17 SD
6:06 SD(OHW)					
3:30 0/LV	7:00 0/LV	10:30 0/LV	3:30 - 0/LV	7:00 0/LV	10:30 0/LV
3:44 300/36	7:14 300/36	10:44 300/36	4:10 300/36	7:40 300/36	11:10 300/36
5:24 270/0	9:06 270/14	13:48 270/15	5:00 270/5	10:24 270/14	14:48 270/14
5:54 250/0	12:26 250/0	18:05 SD	6:30 250/0	12:56 250/0	17:50 250/0
6:50 SD(FDAI)	12:53 SD		7:02 SD	13:28 SD	18:09 SD
7:02 SD(OHW)					
4:00 0/LV	7:30 0/LV	11:00 **	4:00 0/LV	7:30 0/LV	11:00 **
4:14 300/36	7:44 300/36	USE	4:40 300/36	8:10 300/36	USE
5:56 270/0	9:38 270/14	MANUAL	5:40 270/5	11:12 270/14	MANUAL
6:38 250/0	13:38 250/0	ASCENT	7:28 250/0	13:48 250/0	ASCENT
7:34 SD(FDAI)	13:53 SD	ANGLES	7:56 SD	14:19 SD	ANGLES
7:40 SD(OHW)					

* The format of the data given above is:

Time of Abort from PDI FDAI/Overhead Window

Time of first step deg/deg
Time of second step deg/deg
Time of third step deg/deg
Time of shutdown (SD) (SD)

All maneuvers are balanced coupled (all pitch rates 5 deg/sec)

** Establish positive HDOT, then abort stage using manual ascent angles.

11. TRANSEARTH INJECTION, MIDCOURSE CORRECTIONS, AND ENTRY

This section discusses the updates to the Transearth Injection, Midcourse Corrections, and Entry Techniques (Reference 10). Following a brief review of the TEI, MCC, and entry activities, the changes and additions will be discussed in the order in which the subjects appear in the baseline document.

TEI is scheduled approximately two revolutions after jettison of the subsatellite discussed in Section 5.19. TEI consists of an SPS burn to boost the CSM out of lunar orbit. The nominal return flight will not exceed 110 hours, and the return inclination will not exceed 70 degrees (relative to the earth's equator).

During transearth coast, midcourse corrections will be made for entry corridor and landing point control, if required. These corrections will use the MSFN for navigation. SIM data retrieval will be accomplished during an EVA. Selected SIM experiments will operate during portions of the transearth coast period.

Prior to atmospheric entry, the CM will be separated from the SM using the SM RCS. Earth touchdown for Apollo 16 will be in the Pacific within ± 35 degrees latitude and will occur within approximately 12.5 days after launch from earth. Immediate recovery of the crew and the CM will be effected.

11.1 MCC MONITORING AND TERMINATION (Page 5)

The transearth MCC-5 burn table (Table 11-1) shows the SPS maneuver monitoring, termination, and completion rules. The rationale for terminating the burn for impact point (IP) control is that, terminating the burn and investigating the cause of possible difficulties does not yield catastrophic results; whereas, for corridor control, immediate return to nominal corridor is necessary. It should be noted that the MCC shutdown time is now BT + 1 second. Previous TEI and the MCC's were shut down at BT + 2 seconds.

Table 11-1. MCC-5 Burn Table

<u>Maneuver</u>	<u>R or Y Rates</u>	<u>Attitude Deviation</u>	<u>Shutdown Time</u>	<u>Residuals</u>
Corridor control	10 deg/sec complete	±10 deg complete	BT + 1 sec and $\Delta V_c = 0$	TRIM X-axis only to 0.2 ft/sec
IP control	10 deg/sec terminate	±10 deg terminate	BT + 1 sec and $\Delta V_c = 0$	TRIM X- and Z-axes to 0.2 ft/sec

11.2 NO-COMMUNICATION CONTINGENCY (Page 14)

In the event of communication loss, the crew will use the Return-to-Earth Program (P37) for midcourse maneuver targeting and entry initialization. If communications are lost between TEI and EI - 24 hours, the W-matrix will be initialized, and navigation sightings will be taken. The current estimated time after which the loss of communication will not present problems for the MSFN navigation (NAV) vector is EI - 24 hours if all subsequent attitude control is coupled. MSFN updates after this time improve the NAV accuracy, since they will be more accurate than onboard navigation. In the event that a MSFN state vector update is obtained at EI - 24 hours or later, onboard navigation (P23)* is not required for subsequent MCC's and entry.

It should be emphasized that P37 must not be used after 4 hours before entry interface when the final scheduled MCC update is uplinked. This would cause the external ΔV target load and the landing point location to be revised based on P37 computations, which is undesirable.

With a steep entry trajectory, it is possible to encounter high-g loading (up to 11 g's) if the lift vector down (after 0.05 g) is maintained until

* Section 13.5 of the Contingency Techniques presents a new technique for initializing P23 in the event there is a loss of communication, a PIPA has failed, and an SCS TEI was performed on block data. For Apollo 16, P23 will not be performed during transearth coast unless communications are lost.

1.5 g. Therefore, in a no-communication instance, the crew will hold the lift vector down until 1.5 g or until the EMS scroll indicates a steeper than, or equal to, nominal trajectory (flight-path angle); whichever occurs first (Reference 44).

In P37 or any program using the INITVEL routine, there are near-polar and high inclination return-to-earth and reentry restrictions on the onboard software (Reference 45). The P37 Return-to-Earth Program used to target returns to earth from outside the sphere of influence of the moon uses the Lambert targeting technique and, therefore, has an inherent infinity of solutions when the target is 180 degrees away from current position. Otherwise, there are two distinct solutions possible where the program automatically selects a trajectory for getting into a posigrade orbit with respect to the earth. If there is degraded knowledge of the state in the onboard computer or the position vector is within 1.5 degrees of the velocity vector, a retrograde solution might be selected. Excessive ΔV changes or no solution resulting from too many iterations on the cotangent of the flight-path angle (605 alarm) will warn the astronaut to that effect.

There is no recovery possible to the large ΔV changes; however, the iteration problem can be solved by specifying a nonzero flight-path angle (γ) as input to R3 of N60. There are no equation-execution or scaling problems associated with the entry onboard software for polar or retrograde return entries.

11.3 TIMELINE AND ATTITUDE UPDATES (Page 18)

The high-speed entry procedures are updated as follows:

- a) The SXT star and boresight star checks which were previously scheduled at EI - 2 hours are now performed at EI - 1:35 (per the P52). To conserve CSM maneuvering, these star checks (to verify the IMU) are performed in the horizon check/separation (no yaw) attitude instead of the entry attitude.
- b) For the nominal case, the CM is maneuvered to the entry attitude after CM/SM separation instead of tracking the horizon. Again, this is done to reduce CM

maneuvering. In the event of no communication, the horizon is tracked from separation to entry as on previous missions.

- c) The spacecraft control is switched from SCS to CMC DAP at 0.05 g rather than EI - 2 minutes (crew preference).

11.4 HIGH INCLINATION ENTRIES (Page 21)

The special takeover techniques for high inclinations (when weather avoidance is required) have been deleted for Apollo 15 and subsequent missions because of modifications (gain improvement) to the CMC entry programs. The modifications also reduced the nominal entry range from 1,250 to 1,190 nautical miles. Failure of the LEWD iteration in P65 to converge cannot occur for nominal or weather avoidance (1,475 nautical miles) entry ranges.

11.5 DUAL-RING RCS ENTRY ROLL CONTROL

The dual-ring RCS will be used for entry roll control beginning with Apollo 15. In the event of a stuck thruster ON, the ring with the stuck thruster will consume propellant three times as fast as the other ring due to the four opposing thrusters (two on each ring) sharing the resultant roll control function. In the worst case situation of a stuck thruster ON at 400,000 feet, the dual-ring configuration provides a 48-pound RCS propellant margin as well as increased roll control authority (over the single-ring configuration) while the stuck thruster is on.

12. TRACKING DATA SELECTION CONTROLLERS PROCEDURES

Following a brief discussion of the Tracking Data Selection Controllers Procedures (Reference 11), one update will be discussed. The baseline document contains detailed information and procedures concerning the responsibilities of data select support, tracking controllers, and the flight dynamics team in the RTCC operations. It also contains the general responsibility, communication interfaces with other mission controllers, tracking site requirements, and input and control of the real-time computer complex. The major function of this effort is the determination of the best state vector to be used for all required trajectory control decisions and information.

Although the basic techniques are unchanged, the tracking data selection controllers functions, which were previously performed during revolutions 12 through 14, are now performed during revolutions 12 and 13 due to deleting a revolution between CIRC and PDI. The current sequence is presented in Figure 5-1.

12.1 LIMITS OF ACCEPTABILITY OF LANDMARK TRACKING DATA FOR INCLUSION IN DESCENT TARGETING (Page 51)

Prior to descent, the landing site will be tracked (CSM P24/SXT) for inclusion in descent targeting. If the revolution 12 and 13 R_{LS} estimates (computed from the CSM P24 sightings) differ from the a priori map values by less than 3 σ limits, the optics data are incorporated into the descent targeting. The 3 σ values (Reference 30) for Apollo 16 are

Downtrack: 3,800 feet

Crosstrack: 7,900 feet

Radial: 4,350 feet

As stated in the updated lunar orbit techniques (Section 5), the landmark tracking is not required for Apollo 16 since the landing ellipse (without landmark tracking) retains all lunar traverse objectives.

13. CONTINGENCY PROCEDURES

This section discusses the updates to the Contingency Procedures Techniques (Reference 12). The baseline document presents the contingency techniques for trajectory and subsystem monitoring for each mission phase (excluding alternate mission possibilities and undocked lunar activities). Since the basic ground rule is for MCC-H to provide the navigation function as long as communications are maintained with the spacecraft, a significant portion of the baseline document is devoted to navigation procedures.

13.1 NO SLA SEP PROCEDURES (New Section)

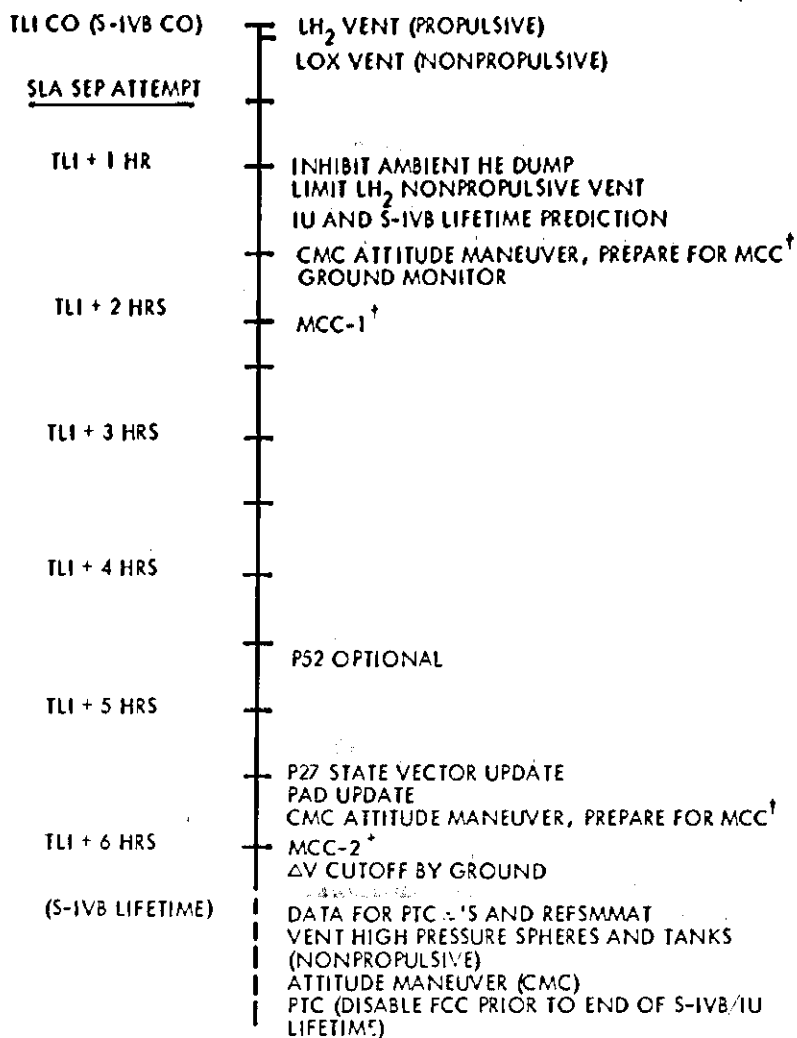
This section presents the techniques to be used in the contingent event that the spacecraft LM adapter (SLA) and the S-IVB fail to separate upon command at approximately TLI + 1/2 hour. Figure 13-1 exhibits the timeline for this contingency, and Figure 13-2 shows the real-time decision logic. A discussion of the ground rules and techniques is given in the following paragraphs.

The philosophy used in the development of the NO SLA SEP techniques include

- Maintain or establish free-return trajectory as soon as practical because postponement of MCC results in a ΔV penalty.
- Use S-IVB LH₂ dump (≈ 19 feet per second)*, LOX dump (≈ 14 feet per second)*, and APS ullage (≈ 23 feet per second)* for the first and second NO SLA SEP MCC's.
- Delay thrusting until the second midcourse correction (TLI + 6 hours) if possible to improve solution.
- RCS ΔV capability is approximately 80 feet per second.
- Perform the second MCC before TLI + 6-1/2 hours**.
- A reasonable period of MSFN tracking should precede each midcourse correction.
- Partially passivate the S-IVB to save the vehicle and minimize trajectory perturbations prior to the first MCC.

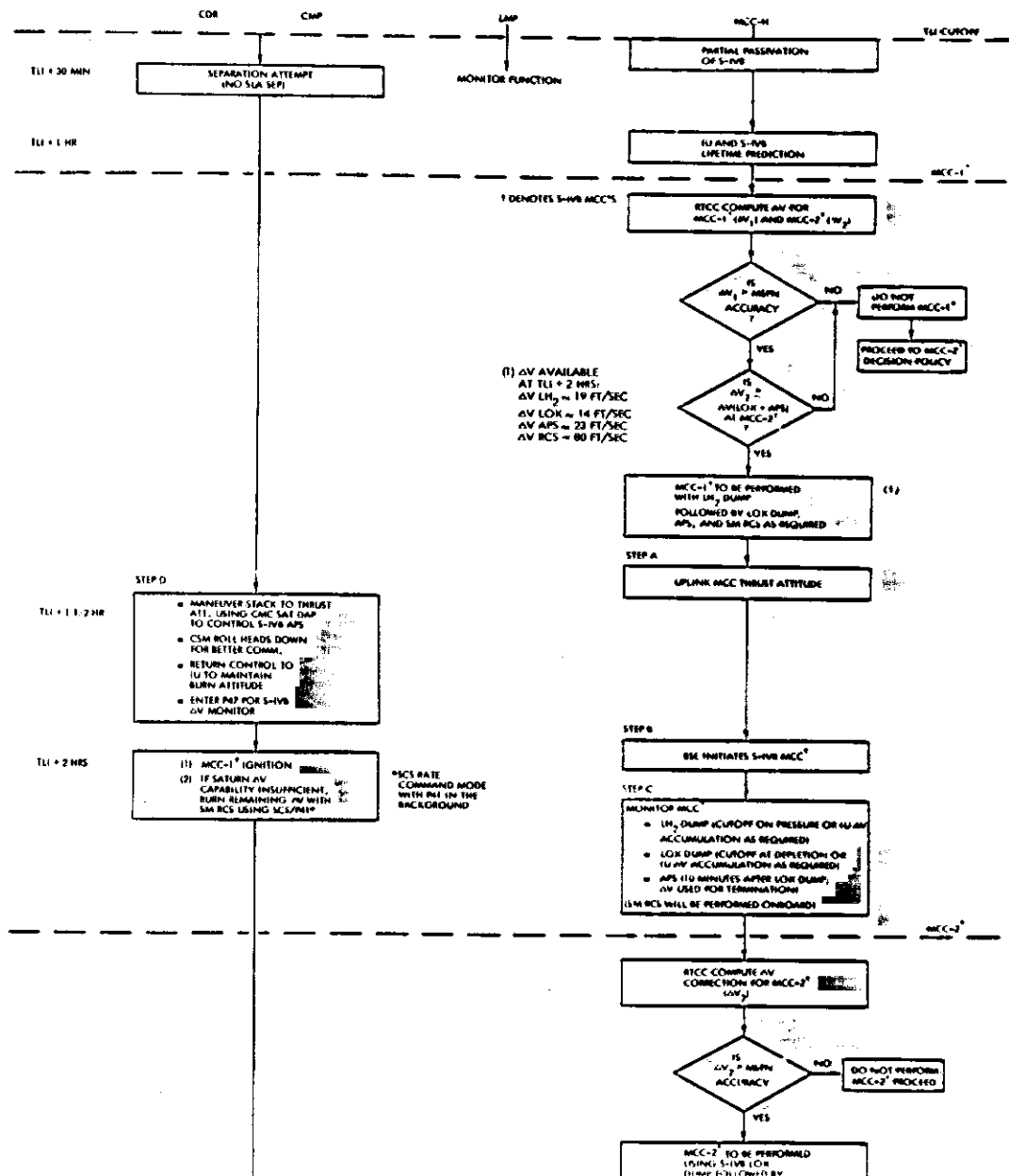
*Values obtained from Reference 46.

**An IU and S-IVB battery lifetime prediction is available at TLI + 1 hour.



[†] THE NO SLA SEP MCC'S ARE MARKED WITH A DAGGER (†) TO AVOID CONFUSING THEM WITH THE NOMINAL MCC'S

Figure 13-1. NO SLA SEP Timeline



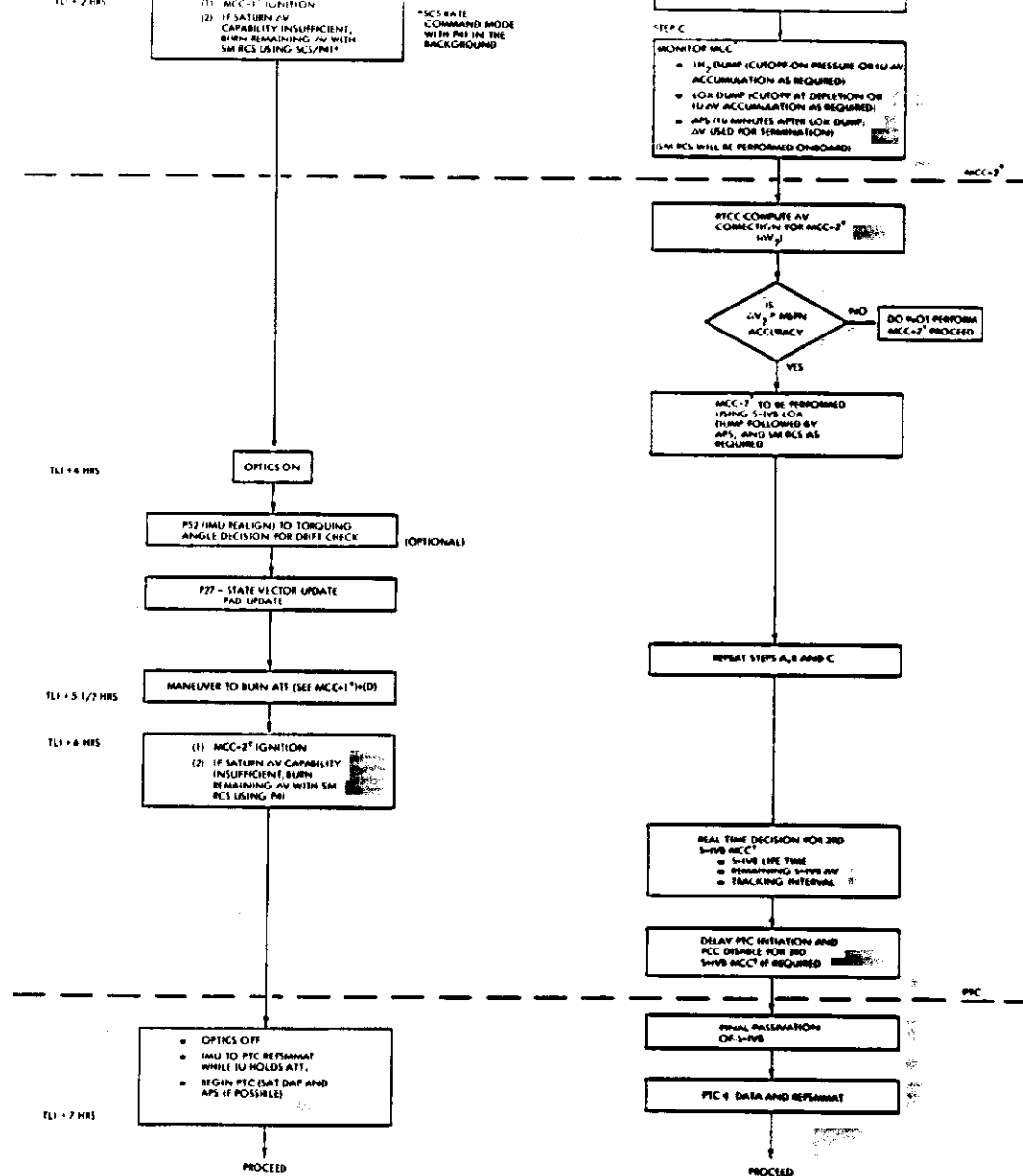


Figure 13-2. NO SLA SEP Flow Chart

13.1.1 Translunar Midcourse Philosophy

The purpose of the translunar (TL) midcourse corrections (MCC) during the NO SLA SEP contingency is to maintain the SC as close as practical to the free-return flyby trajectory. This is the third MCC targeting mode discussed in the Apollo 13 translunar baseline document. The free-return flyby target objectives are a water impact and acceptable entry conditions.

Several MCC's are planned during the TL and transearth (TE) phases to account for TLI maneuver anomalies and MSFN state vector uncertainty. The 3 σ MSFN uncertainties for TLC are shown in Table 13-1. The first two NO SLA SEP MCC's (if required) at TLI + 2 hours and TLI + 6 hours will be performed with the S-IVB propulsion system (LH₂ and LOX dumps and APS) in order to conserve SM RCS propellant for later MCC's. The MCC sequence, using S-IVB propellant, is shown in the decision logic (Figure 13-2). Figure 13-3 shows the typical SM RCS MCC event sequence. This sequence is the same as the sequence normally used, except the P40/SPS capability is not available in the NO SLA SEP case. A typical free-return trajectory and procedural profile for the TL MCC and navigation sightings is shown in Figure 13-4.

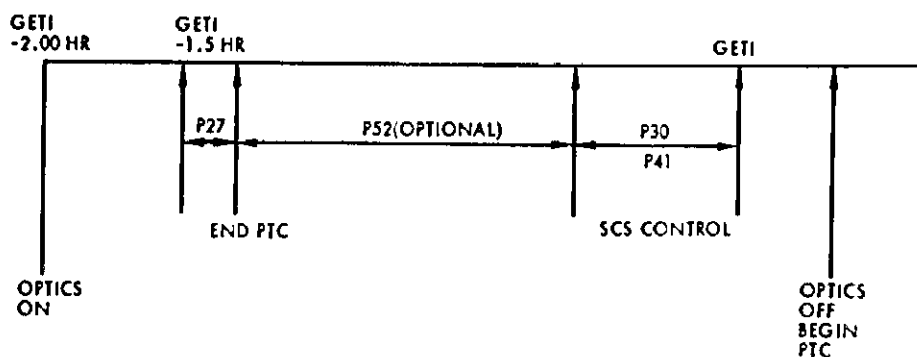


Figure 13-3. SM RCS MCC Sequence

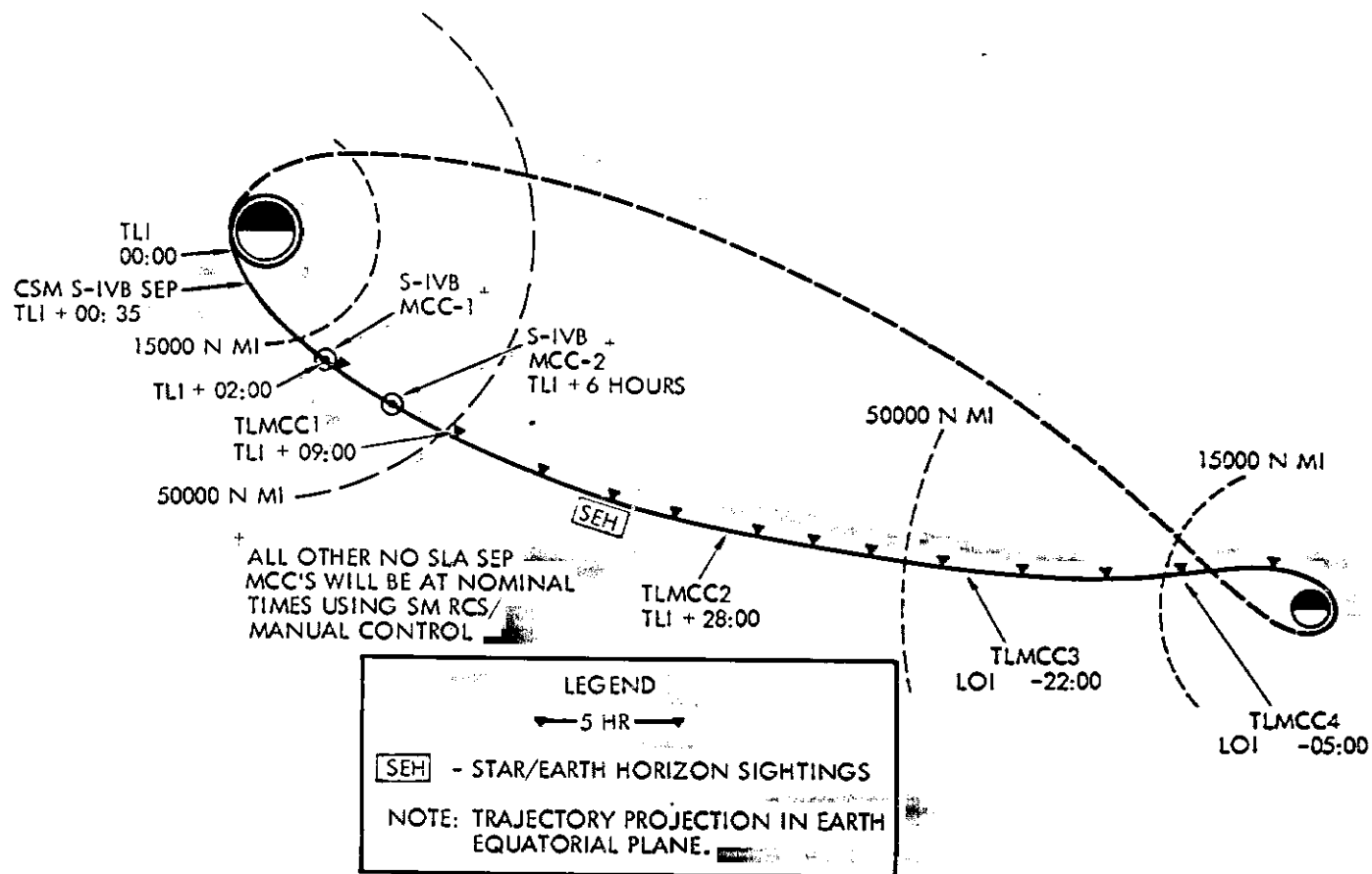


Figure 13-4. Typical Free Return Trajectory and Procedural Profile

Table 13-1. Typical 3 σ Uncertainty of MSFN Velocity Measurement
(Reference 11 of TL MCC and LOI Baseline Document)

Time from TLI Cutoff (hr)	Velocity Uncertainties (fps)		
	\dot{u}^*	\dot{v}^*	\dot{w}^*
2.00	0.42	1.02	2.85
6.00	0.24	0.81	1.59
10.00	0.18	0.72	1.20
15.00	0.15	0.63	0.96
20.00	0.12	0.51	0.72
25.00	0.22	0.24	0.69
30.00	0.09	0.39	0.54
35.00	0.09	0.36	0.50
40.00	0.09	0.33	0.45
45.00	0.07	0.28	0.36
50.00	0.06	0.24	0.32
55.00**	0.06	0.20	0.28
60.00**	0.14	0.06	0.14
65.00**	0.15	0.06	0.33

* \dot{u} is MSFN 3 σ uncertainty in radial direction.

\dot{v} is MSFN 3 σ uncertainty in horizontal direction.

\dot{w} is MSFN 3 σ uncertainty in out-of-plane direction.

** \dot{u} , \dot{v} , and \dot{w} are in moon reference (earth reference prior to 55 hours).

13.1.2 Discussion of Flow Diagram (Figure 13-2)

The following checks and procedures are followed for the translunar midcourse and free-return phase of a NO SLA SEP mission contingency. Nominal periodic alignments (P52) to particular REFSMMAT's will be made. The SCS drift checks, EMS test, CMC updates, PIPA bias updates, and block data messages will also be handled in a similar manner as in the nominal TLC phase.

S-IVB MCC-1[†] and MCC-2[†]

Once the NO SLA SEP has been verified, the S-IVB is partially passivated. At approximately TLI cutoff + 1 hour, the predicted IU and S-IVB (electrical) lifetimes will be available. With these data and the RTCC computed MCC-1[†] ΔV , a real-time decision will be made to determine if the S-IVB can support MCC-1[†] and possibly MCC-2[†]. S-IVB propulsion can be supplied by three sources:

- LH₂ dump (boil-off is fairly rapid; therefore, the LH₂ should be used for MCC-1[†] if a sizable MCC-2[†] ΔV is required) - 19 feet per second
- LOX dump (time limited by S-IVB lifetime) - 14 feet per second
- APS ullage (time limited by S-IVB lifetime) - 23 feet per second

The LH₂ dump will be accomplished for MCC-1[†] only if the ΔV available for MCC-2[†] from LOX and APS is less than required. Should additional ΔV be required during MCC-1[†], the priority of propellant expenditure is S-IVB LOX and APS followed by SM RCS as required. If the LH₂ dump is required, thrust will be terminated by the ground. The IU ΔV accumulation will be used for thrust termination when the required MCC-1[†] ΔV is less than the total LH₂ capability, otherwise a predetermined tank pressure will be used as the criteria for thrust termination. If the combined LOX and APS ΔV capability is sufficient for MCC-2[†], the LH₂ will not be used for MCC-1[†]. The priority of MCC-2[†] propellant expenditure is S-IVB LOX dump and APS followed by SM RCS as required. It is anticipated that no LH₂ will be remaining by MCC-2[†].

The ground will allow a LOX dump to continue through LOX depletion (pull through)[†] if necessary. If LOX depletion is not necessary, the LOX dump thrust will be terminated by the ground monitoring IU ΔV accumulation. Normally, 10 minutes will be required to configure the systems for

[†] Pull through is defined as the time at which the propellant tank ullage gas starts exhausting through the engine and thrust becomes unpredictable (two-phase flow).

APS thrusting. The S-IVB APS thrusting will be terminated by the ground monitoring IU ΔV accumulation.

The crew will maneuver the stack to the thrust attitude using the GMC SAT DAP to control the S-IVB APS 30 minutes prior to MCC. A CSM heads-down attitude is established for better communication; after which, control is returned to the IU to maintain the burn attitude. The crew then enters P47 for S-IVB ΔV monitoring or establishes the SCS rate command mode with P41 (which displays body axis Vg's) for a SM RCS burn.

Following MCC-2[†], the predicted S-IVB lifetime, remaining ΔV , and available MSFN tracking interval will be used to make a real-time decision as to the possibility of a third S-IVB MCC.

Since TLI is performed to meet certain lunar target conditions and is nonfree return, minimum ΔV targeting may be undesirable (e.g., land impact). Table 13-2 shows the ΔV requirements at TLI + 10 hours to obtain a lunar flyby trajectory and mid-Pacific landing (MPL) for several previous missions. Figure 13-5 shows the MCC ΔV for fuel critical flyby maneuvers required for both a minimum flyby to MPL and a minimum flyby to any IP for Apollo 16 (Reference 46).

Substantially more ΔV is required to achieve an MPL than a satisfactory entry at an unspecified landing point; therefore, the IP selection will be determined by the closest body of water. A safe return to earth from both the nominal and dispersed (± 10 feet per second) TLI states is possible with the S-IVB attached using SM RCS only (Figure 13-6). The 10-foot per second underburn case requires that the maneuver be performed before TLI + 30 hours in order to remain within SM RCS capability.

Cislunar MCC's and PTC

The sequence for subsequent MCC's is shown in Figure 13-3. It should be noted that P41 will only be used for monitoring velocity, and the stack will be controlled manually using SM RCS.

Table 13-2. Midcourse Maneuver for Flyby to MPL 10 Hours After TLI for Nominal TLI's
(Reference 47)

Mission	Nominal				Optimum Free Return to MPL						
	H_{00} (m mi)	Inclination, (deg)	Longitude of Free-Return Landing (deg min)	Inclination of Vehicle with Respect to Earth-Moon Plane (deg)	ΔV_{MCC} (g's)	H_{00} (m mi)	Inclination, (deg)	Apog. Ap. (deg min)	External ΔV_a (g's)	External ΔV_v (g's)	External ΔV_z (g's)
Apollo 11	60	+36	174:33E	2.6	6.5	50.0	+34	2:40B, 164:46W	+5.1	0.4	-4.0
Apollo 12	1770	+36	72:15W	57.6	388.4	511.0	+40	7:54S, 164:58W	-127.1	461.9	-95.0
Apollo 13	2040	+33	16:44E	4.8	153.8	899.4	+40	22:48S, 165:00W	134.3	1.5	86.5
Apollo 14	2090	+33	99:23E	28.6	80.9	1700.0	+31	24:16S, 164:22W	-59.9	45.8	-29.3
Delayed free return to Ceres/Pluto	2000	NA	100:00W	NA	61.1	2522.0	+40	25:57N, 169:50W	NA	NA	NA

NA = not available

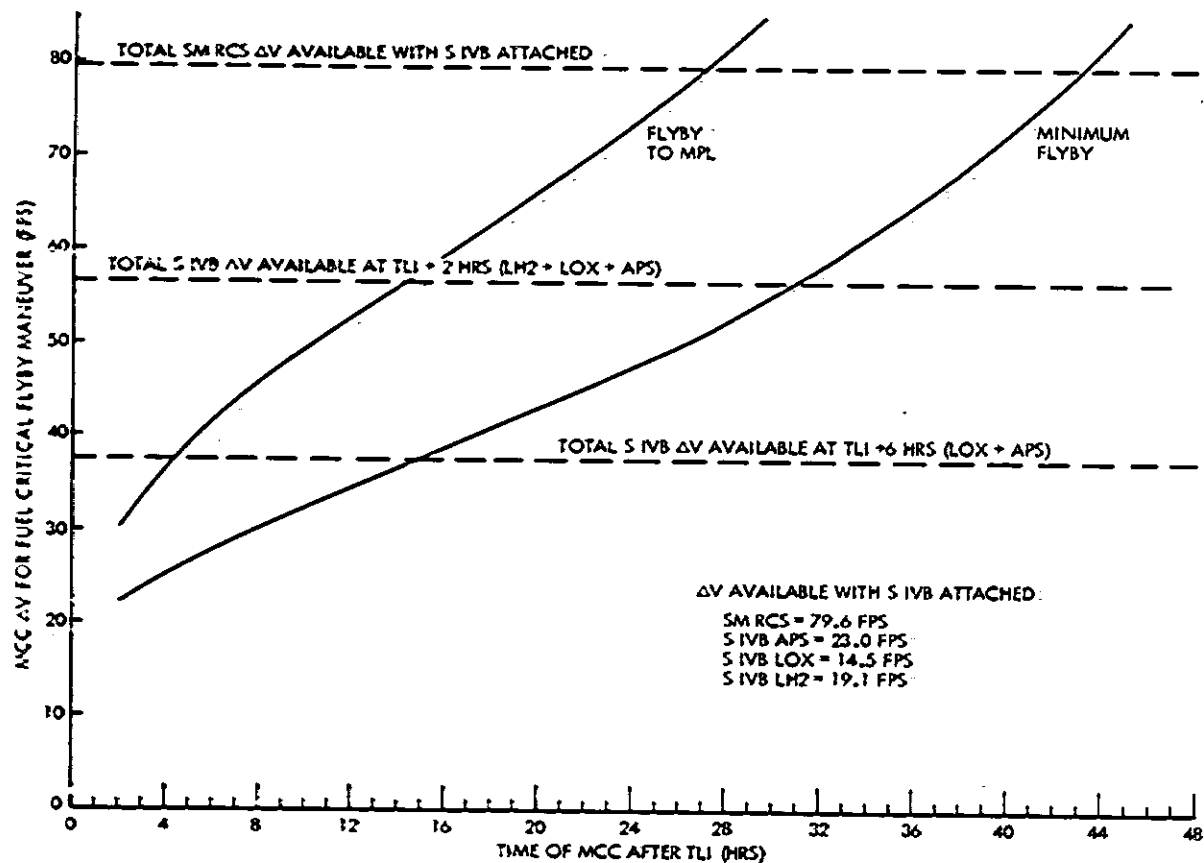


Figure 13-5. Flyby ΔV Available for Apollo 16 with S-IVB Attached (Reference 46)

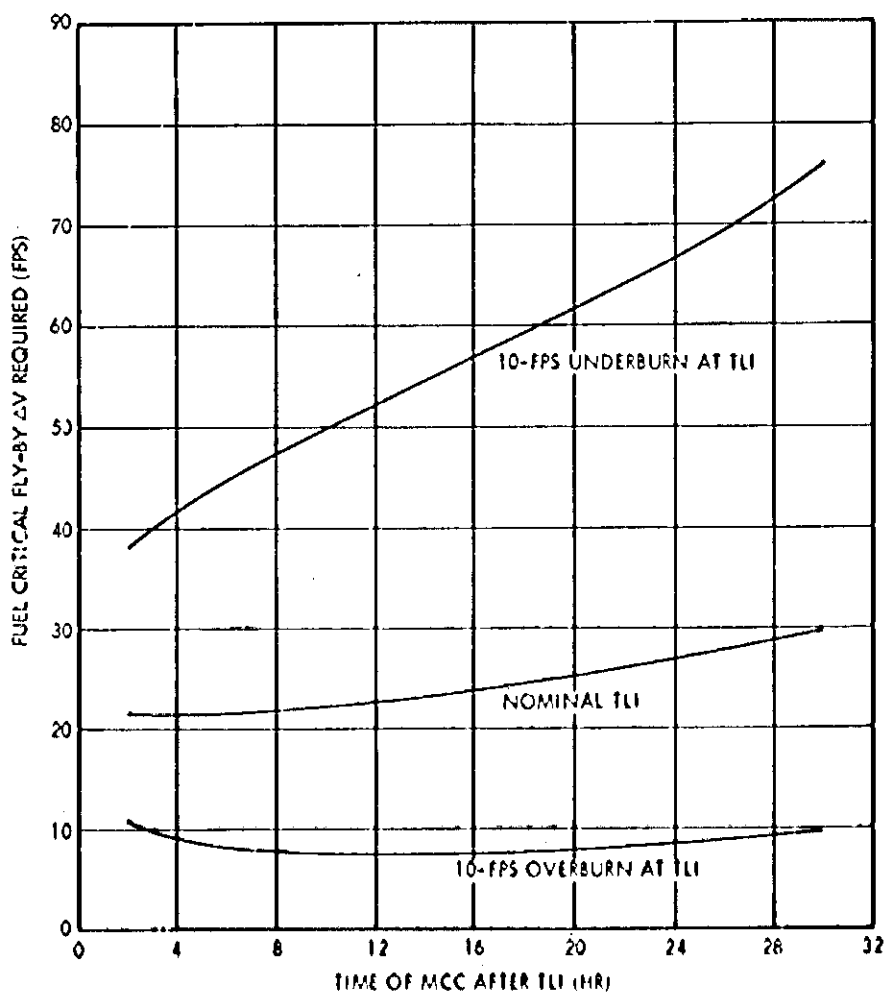


Figure 13-6. Typical Fuel-Critical Flyby ΔV Required for Apollo 15 Nominal and Dispersed TLI's (Reference 48)

Following MCC-2[†], passive thermal control (PTC) will be delayed as long as possible. The IMU will be aligned to the PTC REFSMMAT while the IU holds attitude, and PTC will be established using CMC SAT DAP and APS, if available. Passivation of the S-IVB will be completed, and the IU FCC will be disabled prior to the end of the S-IVB lifetime. MCC-H will uplink the PTC REFSMMAT and attitude data.

To conserve RCS, the "T" handle (THC) will be used to maneuver to the PTC attitude. Table 13-3 shows the results of the two cases considered. Case 1 (the conservative) assumed maneuvering at 0.5 degree per second and switching to DAP control when the rates are reduced to 0.1 degree per second. Case 2, which will be used since it requires less propellant, assumes maneuvers with the "T" handle of 0.1 degree per second and switching over to the DAP at 0.05 degree per second. In the stacked configuration, a 1-second firing with the "T" handle in Y or Z yields approximately 0.083-degree per second rate in the respective axis. A complete cycle, (that is, starting at some attitude initiating PTC, stopping PTC, and going to some attitude) is shown in Table 13-3.

Entry

The NO SLA SEP mission techniques from the first PTC initiation to entry interface (EI) are essentially nominal. At approximately EI - 1:43, the CSM maneuvers to the nominal entry attitude in pitch and roll (X-axis 19 degrees below the local horizontal and 45 degrees out of plane). At EI - 1 hour, the docking ring and probe assembly (DRPA) is jettisoned (nominal $\Delta V = 7 \pm 1$ foot per second). The CM is then separated with an imparted ΔV of approximately 1 foot per second. This sequence was selected to insure sufficient lateral separation. Nominal entry procedures are then followed.

Table 13-3. RCS Expenditure*

	<u>Propellant Expended (lb)</u>	
	<u>Case 1</u>	<u>Case 2</u>
MNVR to PTC ATT	17.3	3.5
Damping with DAP	8.3	4.2
Spin-up for PTC	<u>0.3</u>	<u>0.3</u>
PTC initiate	25.9	8.0
Stop rates (0.3 deg/sec in each axis)	5.2	5.2
MNVR to ATT	17.3	3.5
Damping with DAP	8.3	4.2
Attitude hold (MCG, etc.)	<u>0.0</u>	<u>0.0</u>
PTC stop	30.8	12.9
PTC initiate	<u>25.9</u>	<u>8.0</u>
Total cycle	56.7	20.9
	(4.6% RCS)	(1.7% RCS)

*Reference 49

13.2 TLC FUEL-CRITICAL FLYBY (Page 13 and New Section)

For non-time-critical contingencies during translunar coast a block data solution, at LOI minus five hours which is targeted to the MPL, is available. However, this maneuver requires using the SPS and cannot be performed when the only available propellant is the SM RCS. With the SM RCS ΔV limit of 105 feet per second after transposition and docking, the latest possible minimum flyby to the MPL occurs 37 hours after TLI (LOI minus 36.5 hours). A minimum flyby (not water constrained) is possible as late as 51 hours after TLI (Reference 46). In the event of a post TLI NO SLA SEP (discussed in Section 13.1), the latest opportunities for the two minimum flyby cases becomes TLI plus 27 hours and TLI plus 43 hours, respectively.

13.3 ABORT PROCEDURES FOLLOWING SPS FAILURE AT LOI (Pages 15 and 43)

The LOI maneuver transfers the CSM/LM from the nonfree return translunar trajectory to a 60- by 170-nautical mile lunar orbit (LO). Since the LOI burn occurs behind the moon, the crew must be able to monitor and evaluate the burn without ground support. There are four contingency areas for LOI:

- a) G&C
- b) Non-SPS related: ECS, EPS, etc.
- c) SPS related: sustained pressure decay, low chamber pressure, etc.
- d) Inadvertent SPS shutdown

A G&N-type of failure will result in the violation of a preignition attitude check or a crew takeover of the burn at the nominal ignition attitude. For non-SPS-related failures, the procedure is to complete the burn and later assess the situation with MCC-H. If manual shutdown is necessary because of SPS-related problems or should an SPS shutdown occur, an LOI abort is executed. Table 13-4 defines the LOI abort procedures, and Figure 13-7 shows the return-to-earth LOI abort modes, using the LM propulsion systems as a backup to the three abort modes. Figure 13-8 is the updated flow chart of these techniques. The preferred REFSMMAT for LOI is acceptable for LOI aborts, because the LOI maneuver is essentially inplane.

The Mode 1-30-minute aborts are PONCS controlled at a fixed attitude in P47 and must be initiated 30 minutes after LOI, which results in a crowded timeline for the crew. These two conditions make the Mode 1

Table 13-4. LOI Aborts Modes

Mode I - 2 hours

- 1) Direct abort at LOI + 2 hours
- 2) Minimum delta V to water
- 3) SPS tight limits
- 4) No LOS
- 5) Mode begins at LOI ignition

Mode I - 30 minutes

- 1) Direct abort at LOI + 30 minutes
- 2) Minimum delta V
- 3) SPS loose limits
- 4) May require APS burn to supplement DPS
- 5) No LOS
- 6) Mode begins when Mode I-2-hour abort delta V exceeds DPS delta V available

Mode II

- 1) Two-impulse circumlunar abort
- 2) First burn lowers h_a and raises h_p to stabilize orbit (may be extended to satisfy SHe constraints)
- 3) First burn is radially downward at LOI + 2 hours
- 4) Second burn is basically a TEI near h_p
- 5) SPS loose limits
- 6) LOS at first burn + 13 to 40 hours
- 7) Mode begins when DPS delta V available is sufficient for first and second burns

Mode III

- 1) One-impulse circumlunar abort
- 2) Burn is basically a TEI near h_p
- 3) SPS loose limits until orbital period is less than ~ 7 hours, then tight limits
- 4) LOS at LOI + 1.5 to 15 hours
- 5) Mode begins when Mode II first burn is not required to raise h_p or reduce period to 15 hours

Continuation/PMAS/MPAD (for Contingency C/L_o Flight Plan)
 Data source Delta 16-07
 Data confirmed LB

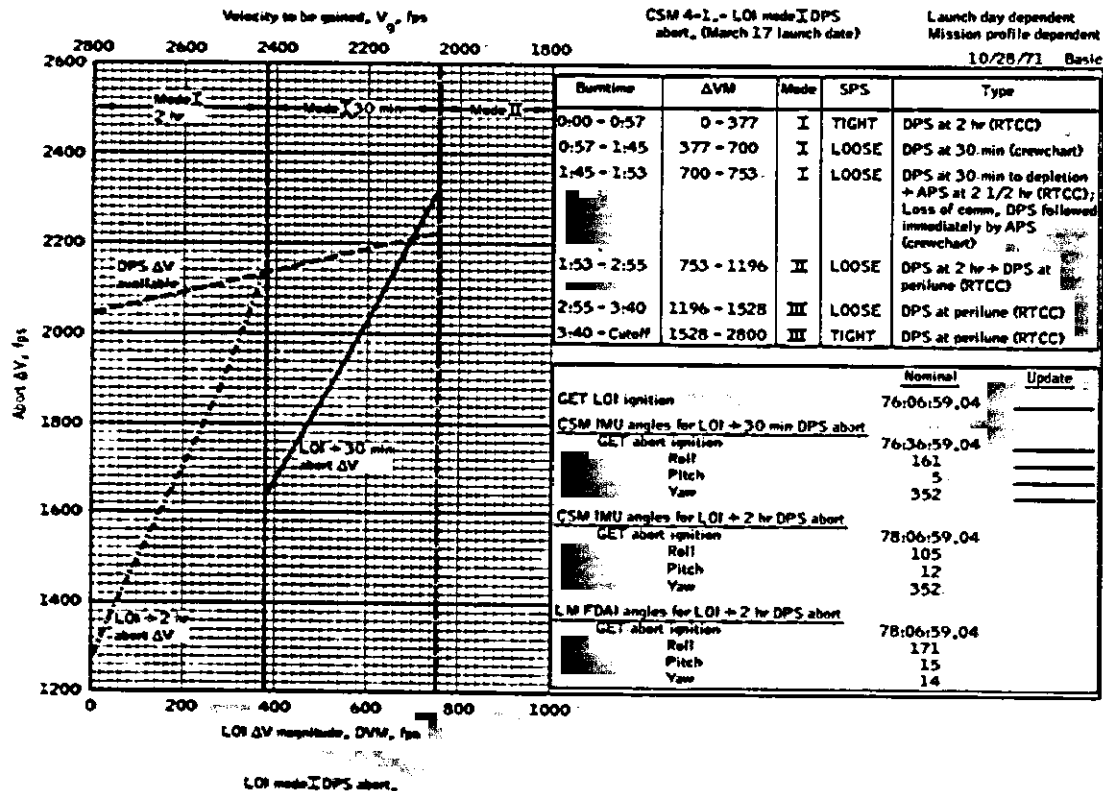


Figure 13-7. Typical LOI Abort

30-minute aborts undesirable; therefore, to minimize the possibility of aborts in this region, Mode I 30-minute aborts are based on loose SPS limits which preclude a subsequent SPS restart.

The SHe tank has been changed for Apollo 16 and has a slower pressure-rise rate; therefore, the requirement on Apollo 15 to extend the Mode I 30-minute DPS + APS region into the nominal Mode II region is not necessary for Apollo 16 although the APS may be required to supplement this mode. (On Apollo 15, the coast time between two burns in conjunction with the duration of the first DPS burn could cause violation of a SHe pressure-rise constraint (Reference 50).)

As discussed in Section 4.5, LOI may now be executed up to 2 minutes late without violating the perilune altitude limit of 5 nautical miles. However, LOI delays introduce errors in the abort chart (Figure 13-7); therefore, a MCC (up to 70 feet per second) is scheduled at LOI + 5 hours to handle these cases.

13.4 TIME FOR RETURN TO EARTH FOLLOWING TIME-CRITICAL ABORTS FROM LUNAR ORBIT (New Section)

In the event of a time-critical abort for Apollo 16, the minimum possible return-to-earth time increases from 43 at LOI to 55 hours (post LOPC2) because of the SPS propellant being used through the stay in lunar parking orbit. Adding dispersions due to the engine, trajectory, weather avoidance, and MCC maneuvers (500 feet per second) increases these times to 47 and 63 hours, respectively (Reference 51).

13.5 TRANSEARTH NO-COMMUNICATION CONTINGENCY

13.5.1 No-Communication Only (Page 20)

In the event of communication loss, the crew will use the Return-to-Earth Program (P37)* for midcourse maneuver targeting and entry initialization. Ideally, MCC-5 should only be performed if it exceeds 3 feet per second (1 σ uncertainty); however, since small errors in MCC-5 do not propagate into substantial errors at MCC-6, all MCC's will be performed as computed.

* Limitations of the software in P37 are discussed in Section 11.2.

Table 13-5 shows a typical sighting schedule for the transearth coast phase. All sets of sighting data consist of three marks on each star. The sightings at EI - 8 hours correspond to the approximate distance at which the single batch of translunar sightings are made. However, should communication be lost before EI - 24 hours (normally the state vectors, MCC-7, and entry PAD's are passed at this time), the onboard vehicle state vector will be used for P37 return-to-earth targeting.

Table 13-5. Typical Sighting Schedule for Transearth Coast (GMT of EI = February 9, 1971, 20 hrs, 46 min, 50 sec)
Apollo 14 - January 31, 1971 Launch Date (Reference 52)

<u>Time</u>	<u>Star</u>	<u>Horizon</u>	<u>R3</u>
TEI + 1	26 (Spica)	MN	00210
	22 (Regulus)	MF	00220
	24 (Gienah)	MN	00210
	151	MF	00220
	156	MN	00210
TEI + 15	212	EF	00120
	33 (Antares)	EF	00120
	40 (Altair)	EN	00110
	77*	EF	00120
	125*	EF	00120
<u>(TEI + 17 MCC-5)</u>			
TEI + 17.5	16 (Procyon)	MF	00220
	22 (Regulus)	MN	00210
	50	MF	00220
	67*	MF	00210
	153*	MN	00210
TEI + 18	212	EF	00120
	33 (Antares)	EF	00120
	40 (Altair)	EN	00110
	201*	EF	00120
	125*	EF	00120

* Alternate stars

Table 13-5. Typical Sighting Schedule for Transearth Coast (GMT of
 EI = February 9, 1971, 20 hrs, 46 min, 50 sec)
 Apollo 14 - January 31, 1971 Launch Date (Reference 52)
 (Continued)

<u>Time</u>	<u>Star</u>	<u>Horizon</u>	<u>R3</u>
TEI + 24	212	EF	00120
	33 (Antares)	EF	00120
	40 (Altair)	EN	00110
	201*	EF	00120
	125*	EF	00120
EI - 28	33 (Antares)	EF	00120
	120	EN	00110
	37 (Nunki)	EF	00120
	77*	EF	00120
	127*	EF	00120
<u>(EI - 26 MCC-6)</u>			
EI - 24	33 (Antares)	EF	00120
	120	EN	00110
	37 (Nunki)	EF	00120
	77*	EF	00120
	127*	EF	00120
EI - 20	33 (Antares)	EF	00120
	120	EN	00110
	37 (Nunki)	EF	00120
	77*	EF	00120
	125*	EF	00120
EI - 8	45 (Formalhaut)	EN	00110
	212	EF	00120
	213	EF	00120
	44 (Enif)	EN	00110
	37 (Nunki)*	EF	00120
	235*	EF	00120
EI - 5	23 (Denebola)	MF	00220
	22 (Regulus)	MF	00220
	16 (Procyon)	MN	00210
	64	MN	00210
	151	MF	00220
<u>(EI - 3 MCC-7)</u>			
EI - 2.5	23 (Denebola)	MF	00220
	22 (Regulus)	MF	00220
	16 (Procyon)	MN	00210
	64*	MN	00210
	151*	MF	00220

* Alternate stars

If TEI is performed at other than the nominal revolution or with a non-nominal TEC time, some of the P23 stars in the no-communication navigation schedule may be occulted by the earth or moon. In this case, the crew will select other stars from onboard charts. The important point is that the nominal number of data batches at the correct times be obtained.

13.5.2 No-Communication and Failed PIPA (Page 53 and New Section)

If communications are lost in lunar orbit, and an SCS TEI is performed using block data (Figure 13-9), additional procedures have been defined to allow the initiation of P23 navigation in the presence of a failure which disallows monitoring the TEI and subsequent maneuvers (e.g., failed PIPA in G&N). The following procedures, which assume a valid onboard preburn vector (last MSFN update at TEI - 10 hours) and TEI PAD, will be used in this no-communication TEI/TEC contingency case:

- a) Following the burn, update the state vector with PAD ΔV 's and TIG time (adjusted for half of the burntime) via P77. This is sufficient to cause P23 convergence.
- b) Initialize the W-matrix. Use extended verb 67 and at the V06N99 display, load
 - $R_1 + 30000$ (30,000 feet)
 - $R_2 + 00300$ (30.0 feet per second)
 - $R_3 + 00003$ (cislunar navigation code)
- c) Initiate P23
- d) Enter NO COMM NAV procedures in contingency check list.

The comparison of the MCC's of nominal TEI and the SCS TEI given in Table 13-6 (Reference 53) show these procedures using P23 data will result in acceptable entry conditions.

P30 MANEUVER

SET STARS	T	E	I	6	4	PURPOSE
	S	P	S	G	N	PROP/GUID
	+			NA		WT N47
R ALIGN _____		0	0	NA		P TRIM N48
P ALIGN _____		0	0	NA		Y TRIM
Y ALIGN _____	+	0	0	2	0	HRS GETI
	+	0	0	0	0	MIN N33
	+	0	3	2	5	SEC
ULLAGE 4 Jet 13 sec	+	3	0	1	4	ΔV_X N81
	-	0	0	4	1	ΔV_Y
	-	0	0	3	8	ΔV_Z
	X	X	X	1	8	R } Lift-off
	X	X	X	1	0	P } Refs
	X	X	X	0	0	Y }
	+					H _A N44
						H _P
	+					ΔVT
HORIZON/WINDOW _____	X	X	X			BT
	X					ΔVC
	X	X	X	X		SXTS
	+				0	SFT
	+			0	0	TRN
	X	X	X			BSS
	X	X				SPA
	X	X	X			SXP
OTHER _____		0				LAT N61
						LONG
	+					RTGO EMS
	+					V10
						GET 0.05G

DATE 3/29/71

Figure 13-9. Typical Maneuver PAD

Table 13-6. Apollo 16 P23 Navigation Accuracy (3 σ) for SCS Controlled TEI^{*}

Event	γ (deg)		\dot{h} (fps)		λ (deg)		ΔV Cost at Next MCC for γ Error	
	Nominal	SCS	Nominal	SCS	Nominal	SCS	Nominal	SCS
MCC-5	1.80	7.20	3,680	8,470	2.10	8.90	7.2	28.8
MCC-6	0.29	0.76	1,150	2,510	0.34	1.30	0.8	1.9
MCC-7	0.21	0.54	450	550	0.25	0.69	-	-
Entry	-	0.53	460	560	0.24	0.68	-	-

* Reference 53 SCS TEI based on loss of communication in lunar orbit
Block data used for TEI burn, and the last MSFN update at TEI - 10 hours

13.6 MINIMUM MSFN TRACKING REQUIREMENTS FOR TRANSEARTH COAST (New Section)*

If the electrical power available during the transearth coast phase has been severely limited, only a limited amount of MSFN tracking will be accomplished. The following conclusions are based on the assumption that pre-MCC-5 tracking begins at TEI + 15 minutes and pre-MCC-7 tracking ends at EI - 4 hours. The conclusions may not be valid if pre-MCC-5 tracking does not begin within TEI + 1 hour, and if pre-MCC-7 tracking ends before EI - 5 hours.

- a) The minimum tracking time required to support MCC-5 is 20 minutes. This 20 minutes should be taken in two 10-minute tracking arcs separated by 4 hours or enough time to allow for a tracking geometry change. The uncertainty in determining perigee altitude with two 10-minute arcs separated as described above is less than 50 nautical miles (3 σ).
- b) Ten minutes of pre-MCC-7 MSFN tracking data are enough to determine the return perigee latitude to within 10 nautical miles provided the tracking is done no earlier than EI - 5 hours. (A single 10-minute error analysis solution was obtained at EI - 16 hours (not shown in results*) and the resulting uncertainty in perigee altitude was 33.0 nautical miles (3 σ).)
- c) The absolute minimum MSFN tracking requirements for a survival transearth trajectory is 30 minutes of tracking taken in 10-minute blocks as described in (a) and (b) above. However, it is recommended that at least three 10-minute tracking arcs be scheduled prior to MCC-7 if it is at all possible.
- d) It is better to track over two separate short arcs that represent two tracking geometries than it is to track over a much longer arc with a single tracking geometry.
- e) The tracking stations should be chosen so that a maximum longitudinal separation exists between the primary tracker and each of its co-trackers.

*Reference 54

14. ATTITUDE REFERENCE INITIALIZATION TECHNIQUES

The Attitude Reference Initialization Techniques (Reference 13) present the alignment techniques and accuracies for the CSM inflight, LM inflight, LM lunar surface, CSM/LM docked, and external alignments. Only one update is required for Apollo 16 and is given below.

14.1 COAS CALIBRATION/FAILED SEXTANT (New Section)

A calibrated COAS is an acceptable substitute for a failed (frozen and occulted) SXT where utilized properly in direct rendezvous. These failures are defined as follows:

- a) Frozen - either the SXT cannot be moved in shaft and trunnion or the SXT/CMC interface has failed.
- b) Occulted - optical sighting cannot be made because of moisture condensation, broken lens, etc.

The best scheme employing the COAS during direct rendezvous involves VHF ranging only until 6 minutes prior to the end of the tracking interval, followed by COAS marking. The procedures for calibrating the COAS when the CMC has failed are as follows:

Min DB SCS or G4N (LOAD)

Boresight star in COAS

Record gimbal angles

Add 180 degrees to OGA

Maneuver to new attitude (roll)

Record SPA and SXP of star in COAS

Alternate LOS for COAS is

$$\text{TRUN} = 57.48 \text{ degrees} - \text{SPA}/2$$

$$\text{SHAFT} = -0.6 \text{ SXP}$$

The 57.48-degree value is the angle between the SC X-axis and the SXT LOS. The SXP angle is multiplied by 0.6 rather than 1/2 to approximate the effect of the COAS LOS not being exactly aligned with the SC X-axis.

APPENDIX
NOMENCLATURE

AEA	abort electronics assembly
ACA	attitude controller assembly
ACRA	Atlantic continuous recovery area
ADRA	Atlantic discrete recovery area
AGC	automatic gain control
AGS	abort guidance system
AK	apogee kick
ALT	altitude
AOA	angle of attack
AOS	acquisition of signal
AOT	alignment optical telescope
APS	ascent propulsion system
ASAP	as soon as practical
ASC	ascent
ASPO	Apollo Spacecraft Program Office
ATT	attitude
AT-2	P57 alignment technique 2
AUTO	automatic
Az	azimuth angle
BSS	boresight star
BT	burntime
CAL	calibration

NOMENCLATURE (Continued)

CCW	counterclockwise
CDH	constant delta height
CDR	commander
CES	control electronic system
CDU	coupling data unit
CIRC	circularization burn
CM	command module
CMC	command module computer
CMP	command module pilot
CO	cutoff
C/O	checkout
COI	contingency orbit insertion
COAS	crew optical alignment sight
CSI	concentric sequence initiation
CSM	command service module
CWEA	caution and warning electronic assembly
DAP	digital autopilot
DB	dead band
DEDA	data entry and display assembly
DES	descent
DET	detent or digital event timer
D _L	skip-out drag acceleration (predicted by CMC)
DOI	descent orbit insertion

NOMENCLATURE (Continued)

DPS	descent propulsion system
DRPA	docking ring and probe assembly
DSKY	CMC display and keyboard
E	erasable (E-memory) or enter
EDS	emergency detection system
EI	entry interface
EMP	erasable memory program
EMS	entry monitoring system
ENG	engine
EPO	earth parking orbit
EOI	earth orbit insertion
ESS	early S-IVB staging
EST	eastern standard time
ET	elapsed time
EVA	extravehicular activity
FCC	flight control computer
FDAI	flight director attitude indicator
FDO	flight dynamics officer
FOV	field of view
G, g	acceleration measured in g (gravity)
G&N	guidance and navigation
GET (or g. e. t.)	ground elapsed time
GETI	ground elapsed time of ignition

NOMENCLATURE (Continued)

GMT	Greenwich mean time
GMTLO	Greenwich mean time of lift-off
GNCS	CSM primary guidance, navigation, and control system
GTC	guidance thrust command
GUID	guidance
h, H	altitude
\dot{h}, \dot{H}	altitude rate
\ddot{H}	rate of change of \dot{H}
$\Delta H, \Delta h$	change in height or horizon bias
H_a, h_a	altitude of apogee (or apolune)
HAM	height adjustment maneuver
He	helium
HOSC	Huntsville Operations Support Center
H_p, h_p	height of perigee or perilune
ICDU	inertial CDU gimbal angles
INC	orbit inclination
INCL _{FR}	orbit inclination (free+return)
INS	insertion
IMU	inertial measurement unit
IP	impact predictor or impact point
IU	instrument unit (S-IVB)
IVT	intravehicular transfer
Jett	jettison

NOMENCLATURE (Continued)

K'	1,000 feet
KM	kilometer
KSC	Kennedy Space Center
LAT	latitude
LDMK	landmark
LES	launch escape system
LET	launch escape tower
LEV	launch escape vehicle
LGC	LM guidance computer
LGC/RR	LGC and RR interface
LH ₂	liquid hydrogen
LLS	LM landing site
LM	lunar module
LMP	lunar module pilot
LMK	landmark
LO	lunar orbit or lift-off
L/O	lift-off
LOI	lunar orbit insertion
LONG	longitude
LOPC	lunar orbit plane change
LOS	loss of signal or line of sight
LOX	liquid oxygen
LPD	landing point designator

NOMENCLATURE (Continued)

LPO	Lunar parking orbit
LR	landing radar
L/S	landing site
LV	launch vehicle
LVLH	local vertical, local horizontal
MCC	midcourse correction
MCC [†]	NO SLA SEP MCC
MCC-H	Mission Control Center - Houston
MFCO	main fuel cutoff
MIT	Massachusetts Institute of Technology
MNVR	maneuver
MPL	mid-Pacific landing
MSC	Manned Space Flight Center
MSFN	Manned Space Flight Network
MTO	manual take over of booster steering by crew
N	noun or north
NAV	navigation
NDB	narrow dead band
OCDU	optics CDU gimbal angles
OGA, MOA, IGA	outer, middle, and inner gimbal angles
OHW	overhead window
ORDEAL	orbital rate display-earth and lunar
P	program
PAD	data voiced to crew from ground and recorded for future use

NOMENCLATURE (Continued)

P&FS	Particle and Fields Satellite
PC	pericynthion
PDI	powered descent initiation
PFP	Powered Flight Processor
PGA	pressure garment assembly
PGNCS (PGNS)	LM primary guidance, navigation, and control system
PIPA	pulse integrating pendulous accelerometer
PLSS	portable life support system
PTC	passive thermal control
P, Y, R	pitch, yaw, roll
Q	aerodynamic loading
R	roll gimbal angle, range
ΔR	delta range
RCS	reaction control system
RDZ	rendezvous
REFSMMAT	transformation matrix from basic reference coordinate system to CSM stable member inertial coordinates
REQD	required
RETEI	time of retrofire entry interface (free fall)
REV	revolution
RFO	retrofire officer
RIP	range of impact point
RL	roll left

NOMENCLATURE (Continued)

\underline{R}_{LS}	landing-site position vector
RR	rendezvous radar
RSDP	remote site data processing
RSO	range safety officer
RTCC	Real-Time Computer Complex
S	south
SAT	Saturn
SC	spacecraft
SCS	stabilization and control system
SCT	scanning telescope
SEA	sun elevation angle
SEL	selection
SEP	separation
SIM	Scientific Instrumentation Module
SLA	spacecraft LM adapter
SM	service module
SPA	COAS BSS pitch angle (PAD data)
SPS	service module propulsion system
SV	state vector
SXP	COAS BSS X-position (PAD data)
SXT	sextant
S-1C	first stage of Saturn launch vehicle
S-II	second stage of Saturn launch vehicle
S-IVB	third stage of Saturn launch vehicle

NOMENCLATURE (Continued)

T	time or nominal time of launch
ΔT	delta time
$T \pm 24$	T plus or minus 24 hours
TB	time base
t_B	burn duration
TD	touchdown
TD&E	transposition, docking, and LM extraction
TE	transearth
TEI	transearth injection
t_{ff} T_{FF}	estimated time of free fall from the current position to an altitude of 300,000 feet
THC	translational hand controller
t_{ig} , TIG	time of ignition
TGT	target
TL	translunar
TLC	translunar coast
TLI	translunar injection
TPF	terminal phase final
TPI	terminal phase initiation
TPMR/RR	tape meter/rendezvous radar interface
TRK	tracking
TRUN	trunnion
V	verb
V	velocity

NOMENCLATURE (Continued)

ΔV	velocity increment
ΔV_{INS}	delta velocity at insertion
V_G, V_{GO}, V_g	velocity to be gained
VHF	LM-CSM voice communication and ranging device
VHF/CMC	VHF/CMC interface
VHF/EMS	VHF/EMS interface
V_I	inertial velocity (DSKY display)
WDB	wide dead band
y	yaw
σ	standard deviation
ϕ	latitude
λ	longitude
α	angle
γ	flight path angle

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