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APOLLO EXPERIENCE REPORT - ABORT PLANNING

by Charles T. Hyle, Charles E. Foggatt, and Bobbie D. Weber

Manned Spacecraft Center
Houston, Texas 77058

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16. Abstract

Definition of a practical return-to-earth abort capability was required for each phase of an Apollo mission. A description of the basic development of the complex Apollo abort plan is presented in this paper. The process by which the return-to-earth abort plan was developed and the constraining factors that must be included in any abort procedure are also discussed. Special emphasis is given to the description of crew warning and escape methods for each mission phase.

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ACRONYMS

AGS	abort guidance system
APS	ascent propulsion system
CDH	constant differential height rendezvous maneuver
CM	command module
CSI	coelliptic sequence initiation rendezvous maneuver
CSM	command and service module
DOI	descent orbit insertion
DPS	descent propulsion system
ΔH	altitude difference between PGNS and LR altitude determinations
Δh	differential height
ΔV	characteristic velocity change
EDS	emergency detection system
EMS	entry monitoring system
EPO	earth parking orbit
g	load factor (force divided by weight)
G&N	guidance and navigation
GTC	guidance-determined thrust command
IMU	inertial measurement unit
LES	launch escape system
LM	lunar module
LOI	lunar-orbit insertion
LPO	lunar parking orbit
LR	landing radar
MCC	midcourse correction

MSFN	Manned Space Flight Network
PDI	powered descent initiation
PGNCS	primary guidance, navigation, and control system
PGNS	primary guidance and navigation system
P37	command module computer program number 37, abort program
RCS	reaction control system
RR	rendezvous radar
RTCC	real-time computer complex
S-IC	Saturn IC, first stage of the Saturn V launch vehicle
S-II	Saturn II, second stage of the Saturn V launch vehicle
S-IVB	Saturn IVB, third stage of the Saturn V launch vehicle
SPS	service propulsion system
t_B	burn time
TEC	transearth coast
TEI	transearth injection
TLC	translunar coast
TLI	translunar injection

APOLLO EXPERIENCE REPORT

ABORT PLANNING

By Charles T. Hyle, Charles E. Foggatt,*
and Bobbie D. Weber
Manned Spacecraft Center

SUMMARY

To attain the high confidence level required of the safety aspects of the Apollo Program, a practical return-to-earth abort capability was provided for each of the various mission phases. The development of such an abort capability is especially complex because of the myriad of potential mission contingencies that can be identified. Trajectories that satisfy the requirements and constraints of a safe return to earth and that extend to all possible preabort conditions are analyzed. The spacecraft system capabilities, ground-support-equipment capabilities, flight-crew performance, and other operational considerations and limitations are then superimposed. The resultant interaction has led to the development of several distinct abort techniques and powered-flight monitoring procedures. These techniques and procedures ensure that a safe return-to-earth capability exists throughout the spectrum of anticipated off-nominal mission conditions.

INTRODUCTION

Although the major objectives of the Apollo Program are to land men on the moon, explore the lunar surface, and return the men safely to earth, the safety of the flight crew has always been of paramount importance. The stringent requirement for crew safety dictates the necessity of as much or more contingency planning for abort situations as is provided for the nominal mission. The contingency planning to ensure that the crew can always abort the mission and return safely to earth is accomplished when an adequate crew warning technique and a method of escape have been defined. An adequate crew warning technique is difficult to achieve during powered flight but has been accomplished by having the crew or ground-control personnel (or both) monitor critical spacecraft systems and parameters. The method of escape from a contingency situation is often determined by the amount of parameter deviation from the nominal allowed by the monitoring techniques and parameter-deviation limits. A method of escape is referred to as an abort mode.

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*Presently a management trainee with the Wicks Corporation, Saginaw, Michigan.

Definition of a practical monitoring technique and abort mode for the realm of potential contingencies is the result of an iterative analysis cycle in which the effects of various constraints are considered. Such constraints may be caused by either hardware or operational limitations. The characteristics of earth or moon trajectories (or of combinations of earth and moon trajectories) determine to a great extent the set of initial conditions upon which the hardware and operational constraints are superimposed. However, the trajectory characteristics are also functions of operational constraints (for example, a specific lunar-landing-site requirement). The achievement of a sound contingency return-to-earth abort plan is therefore obtained by considering the interaction of many constraints and by satisfying the objectives of an adequate crew warning and escape capability.

Because of the diversity of the constraints encountered during the various phases of a lunar mission, independent abort plans must be prepared for each of several discrete mission parts. Basically, these mission parts include the powered-flight phases (launch, translunar injection (TLI), lunar-orbit insertion (LOI), lunar descent and ascent, and transearth injection (TEI)) and the coast phases (earth parking orbit (EPO), translunar coast (TLC), lunar parking orbit (LPO), and transearth coast (TEC)).

Acknowledgment is made to W. Bolt for contributing the section on lunar descent and ascent and to R. Becker and J. Alexander for the section on lunar module abort to rendezvous and command and service module rescue.

ABORT PLANNING PROCESS

For this paper, an abort is defined as the recognition of an intolerable situation and the performance of the activities necessary to terminate the mission and return the crew to earth. An alternate mission is a continuation of the flight, usually with less ambitious objectives than were originally planned.

The process used to formulate a sound abort plan is shown in figure 1. In the abort planning process, the spacecraft and launch-vehicle hardware capabilities and the mission operational constraints and objectives are used as a basis for simulated abort trajectories to investigate display adequacy, entry conditions, time requirements, landing points, and so forth. Throughout this process, the flight crew, flight-control personnel, safety personnel, and other responsible groups conduct reviews and discuss and modify the simulation results until all groups are satisfied. The total abort plan for a particular mission eventually consists of several detailed documents. Each document is concerned with a unique responsibility, but all are based on the same assumptions and are consistent

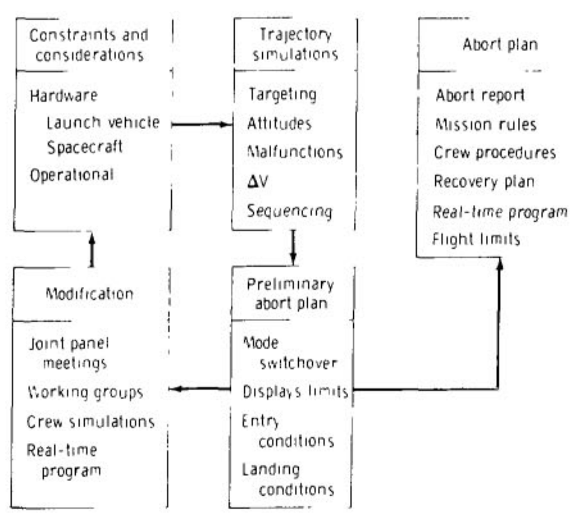


Figure 1. - Abort planning process.

with the other documents. Among the documents that most closely reflect the total abort philosophy are the Abort Report, the Mission Rules, the Crew Procedures Manual, the Data Priority Contingency Techniques Document, and the Flight Limits Document.

Development of the abort plan requires that detailed consideration be given to two types of constraints. Although these constraints are generally classified as either hardware or operational, each category is composed of a variety of considerations (table I).

TABLE I. - ABORT PLANNING CONSTRAINTS

Vehicle	System or category	Constraints
		Hardware
Launch vehicle (S-IC, S-II, and S-IVB)	Propulsion	Emergency detection system (EDS) and redundant capabilities
	Computer	Redundancy, update, and targeting
	Guidance and control	EDS, gimbal lock, and hardover
	Structural	EDS and shutdown sequence
Spacecraft (command and service module (CSM), and lunar module (LM))	Displays	Caution and warning, switch configuration, attitude, and entry-monitoring-system (EMS) quantities
	Propulsion	Type, performance, backup capabilities, and duty cycles
	Computer	Targeting, display, storage, and navigation
	Guidance and control	Performance, procedures, backup systems, autopilots, inertial platforms, gimbal lock, and hardover
	Optics	Navigation, sextant, telescopes, and field of view
	Structural	Couch supports, landing, and CSM/LM interface
	Thermal	Protection limitations, pyrotechnics, and heat shield
	Aerodynamics	Stability and trim and lift/drag characteristics

TABLE I. - ABORT PLANNING CONSTRAINTS - Concluded

Vehicle	System or category	Constraints
	Hardware - Concluded	
	Window/crew geometry	Crew visibility to horizon, manual take-over, reticles, and CSM and LM blockage
	Consumables	Electrical power, environmental system, and propulsion
	Sequencing	Attitude requirements, activation times, and procedures
	Operational	
Spacecraft (CSM and LM)	Trajectory	Free-return mission, orbital altitude, launch windows, flight times, and entry corridor
	Lunar-landing sites	Launch windows, lunar-orbit inclinations, earth/moon geometry, and lighting
	Alternate missions	Objectives, lunar operations, and photography
	Landing and recovery	Geography, lighting, communications, logistics, and medical support
	Communications and tracking	Systems monitoring and ground targeting and command capability
	Environmental surroundings	Atmospheric properties, winds, lighting, weather conditions, radiation, and meteorites
	Human factors	Crew schedules, crew acceleration and deceleration tolerances, crew and ground-control response times
	Procedures	Separation techniques, recontact avoidance, simple and reliable for training proficiency, and mission-to-mission carryover
	Range safety	Land-impact avoidance and launch windows

Malfunctions in any of the hardware items must be considered as additional constraints. Obviously, some of these constraints are meaningful only during one particular mission phase; however, other constraints always exist. Because of the large number of constraints that must be considered, abort plans and techniques must be kept as simple as possible.

The constraint of time criticality has meaning both in monitoring (crew warning) and in abort (escape) mode analysis. Some contingency situations allow more time than others for making the abort decision (monitoring) and for making the return to earth (abort method).

LAUNCH PHASE

Constraints

Because of potential launch-vehicle breakup and subsequent explosion possibilities, the launch phase is probably the most dangerous part of any manned space flight. The atmospheric environment is a major operational constraint that requires special escape-hardware design considerations. The launch escape system (LES) is the mechanism that provides this atmospheric escape capability. The LES is shown with other major Saturn-Apollo vehicle components in figure 2. The planned or nominal launch trajectory also determines the kind of environment from which the available hardware must escape. After the spacecraft exits the significant part of the atmosphere, malfunctions that might occur are significantly less time critical; that is, structural breakup with the associated overpressure and fire hazard is less probable because of the reduced aerodynamic loading. Every major system or category listed in table I influences the launch abort plan in some way.

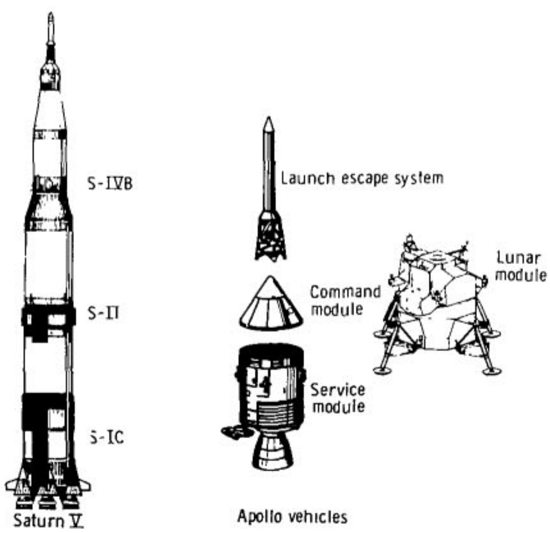


Figure 2. - Saturn-Apollo vehicle components.

Maneuver Monitoring

Ground-control and crew-monitoring activities must include distinction between nominal and off-nominal flight performance in real time. The degree to which nominal flight conditions may be allowed to deteriorate is determined by the escape capability available at the time. Therefore, the monitoring requirements and abort methods are closely related. The interaction of these two facets determines both the limits for maneuver monitoring and the adequacy of a given escape technique. Three types of contingencies are of concern, and all may be considered the result of launch-vehicle problems.

The first type of contingency concerns premature or late (either actual or apparent) thrust termination of the Saturn II (S-II) or

S-IVB. The premature launch-vehicle shutdown situation is more hazardous in that the vehicle trajectory may still be suborbital. The necessity of better tracking and trajectory monitoring capabilities than are available to the crew makes the abort mode decision in this situation primarily a ground-control responsibility.

The second type of contingency includes those malfunctions that rapidly lead to catastrophic results. The time-critical nature of these malfunctions requires that the crew or the automatic system — emergency detection system (EDS) — initiate the abort, based on an observed or sensed violation of preestablished limits.

Vehicle attitude rates and thrust-chamber pressures are the primary quantities used to activate an automatic abort. The crew initiates a manual abort after receiving two abort cues, one of which is usually excessive attitude rates, attitude errors, total attitude dispersions, or angle of attack. In addition to the spacecraft computer displays, other display devices that are available for providing abort cues are listed in table II.

TABLE II. - SPACECRAFT DISPLAYS USED FOR ABORTS

Display	Description
Abort request light	Red light; commanded ON or OFF by ground control
Engine status lights	Yellow lights; ON when engine is not operating (one per engine)
Flight director attitude indicators	Indicate vehicle attitude, attitude error, and attitude rates (two indicators)
Launch-vehicle overrate light	Red light; ON when rates are exceeded
Guidance failure light	Red light; ON when attitude reference is lost
S-II separation light	Red light; ON if S-II first-plane separation occurs; OFF if second-plane separation occurs
Angle-of-attack indicator	Indicates a combination of angle of attack and dynamic pressure as measured by the Q-ball
G-meter	Longitudinal accelerometer
Altimeter	--
Digital event timer	--
Caution and warning master alarm	--
Lift-off and no-automatic-abort lights	--
Propellant-tank pressure indicators	Indicate pressure in fuel and oxygen tanks

The automatic systems — complemented by the crew who are aided by onboard displays, window views, and physiological cues — comprise an adequate system for contending with rapid launch-vehicle deviations. The adequacy of this system is ensured by computer simulations of the most probable vehicle failures. These simulations are made to establish the required limits for an automatic or manual abort. For the most part, rapid deviations, such as an engine hardover, would result in an LES abort. Typical results of this type of contingency are shown in figure 3 in the form of rate and attitude excursions as a function of time from lift-off.

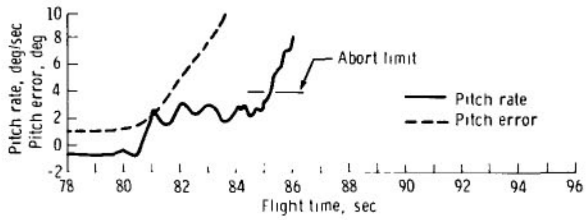


Figure 3. - Typical time history of launch-vehicle dynamic variables following an actuator failure.

The third type of contingency consists of those malfunctions with effects that are not immediately obvious to the crew; these malfunctions may be referred to as slow deviations. Slow deviations are caused by attitude-reference problems or guidance failures and are not as time critical as the malfunctions of the first or second contingency types. The results of such slow deviations usually appear on the ground-monitoring displays as deviated flight conditions, when compared to nominal trajectory conditions. To provide the crew with the required abort decision, the ground controller must know the extent to which the trajectory can be allowed to deviate before a subsequent abort procedure would violate a crew or spacecraft constraint. Such constraints as the 100-second crew-procedure time allotment or the 16g human-endurance limit for entry deceleration forces are used to establish these slow-deviation trajectory monitoring limits (fig. 4).

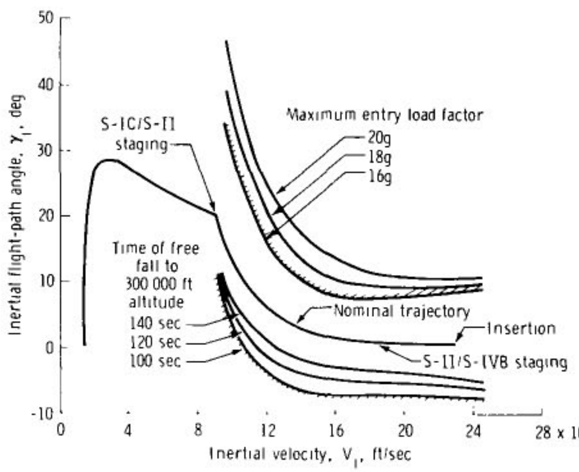


Figure 4. - Ground-control trajectory limits.

Parameters for displays such as those shown in figure 4 were developed after many studies. The parameters selected were generally the most significant variables relating the equations of motion to known constraints. This development task has not been simple because the motion described by orbital and flight mechanics requires more than the two dimensions available for displays. In some instances, two displays were required for completeness (for example, velocity compared with flight-path angle (or altitude rate) and altitude compared with range). The desire to have a reasonably common terminology in the decisionmaking displays for ease of understanding was another factor used to select the parameters and constraints on the display shown in figure 4 as well as on most other displays. Constraints such as the LES performance limits were studied and discarded on the basis that the validity in their use as constraints was ambiguous.

Return to Earth

For an Apollo spacecraft launch, the abort modes that have evolved from the contingency planning process can be briefly described as follows. In a mode I abort, the LES tower performs the time-critical escape from an impending launch-vehicle explosion during atmospheric flight. A mode II abort may occur after the vehicle leaves the atmosphere, where there is little chance of aerodynamic loads that can lead to launch-vehicle breakup and explosion. A mode II abort consists simply of a separation of the spacecraft and the launch vehicle, followed by spacecraft orientation to entry attitude and a subsequent landing in the Atlantic Ocean.

As the flight progresses and the inertial velocity V_i increases, controlling the time increment between launch-vehicle cut-off and a retrograde maneuver was determined to be the most effective means available of controlling the spacecraft landing point. This launch abort technique is referred to as a mode III abort.

If a contingency should arise in the last 2 minutes of the launch phase, when trajectory conditions are still suborbital, the spacecraft propulsion system can provide the transition to an orbital trajectory. Such a procedure, called a mode IV abort, is not strictly an abort in that the procedure does not produce an immediate return of the crew to earth. Because initiation of either of the other two abort modes that can occur in this flight regime (mode II or III) could result in a wide range of undesirable landing locations, a primary advantage of the mode IV procedure is that landing-site selection opportunities are provided. That is, after an orbital state is attained, the spacecraft may travel through part of a revolution until a desired landing area is approached and the crew may then perform the usual entry maneuver. If sufficient propellant remains after the transition maneuver and if the original contingency does not require flight termination, some mission objectives may still be accomplished. The four launch abort modes are shown in figure 5.

The last few seconds of a launch are particularly critical because the ground-control personnel must advise the crew at launch-vehicle cut-off whether the trajectory is satisfactory (a status of go) or whether a mode III or mode IV abort procedure is required (a status of no-go). The important go/no-go decision is facilitated by the use of a ground-monitoring display such as that shown in figure 6.

After an EPO is achieved, emergency entry requirements may be sent to the spacecraft by the ground-control personnel. Also, in this situation, the crew may use previously prepared deorbit data cards.

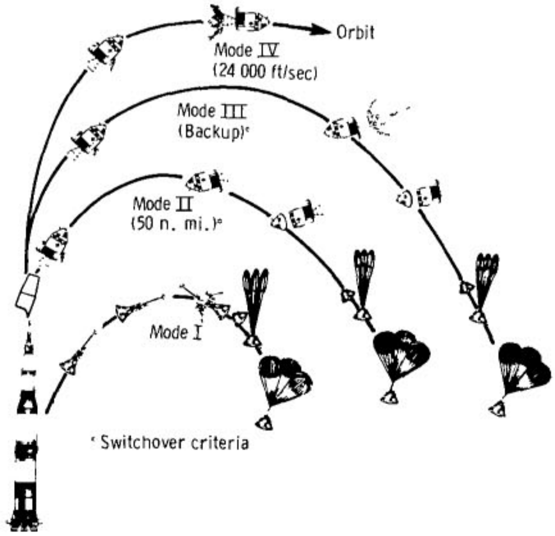


Figure 5. - Apollo spacecraft launch abort modes.

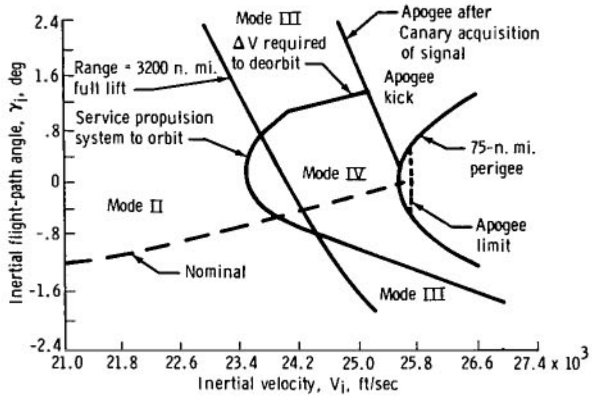


Figure 6. - Ground-monitoring display of near-insertion abort decisions for Apollo missions.

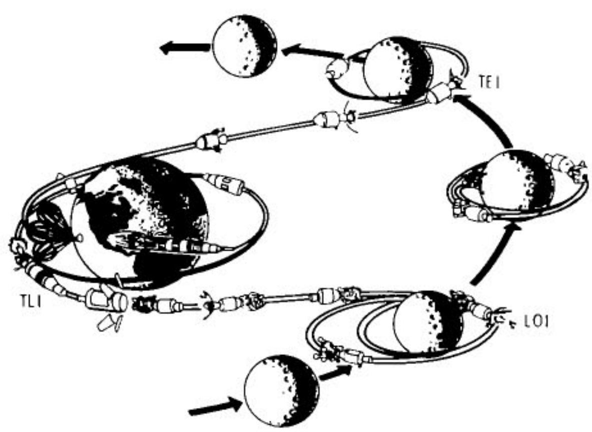


Figure 7. - Apollo mission profile.

TRANSLUNAR INJECTION

Constraints

Translunar injection is the powered-flight maneuver made by the S-IVB that sends the command and service module and lunar module into a translunar trajectory. This event is noted as a part of the mission profile shown in figure 7. Because TLI begins in an EPO at an altitude of 100 nautical miles, the previously discussed launch-vehicle-type contingencies that would have required an abort during launch may now, because of more flexible constraints, be reduced to alternate-mission situations. That is, off-nominal S-IVB performance is likely to require thrust termination but is unlikely to require an immediate return to earth. Therefore, the CSM has its entire propulsion capability intact and available for other uses.

Because the spacecraft is in a passive role during TLI, potential problems are less likely to occur in the spacecraft than in the S-IVB. However, much premission effort is expended in making the warning and escape techniques completely inclusive, even for unlikely situations. Although all constraints in table I are generally applicable, certain constraining factors for possible contingencies apply only during the TLI mission phase. These constraints are shown in table III.

TABLE III. - PRIMARY CONSTRAINTS DURING TRANSLUNAR INSERTION

Vehicle	Constraint
	Hardware
Launch vehicle	Guidance and control, inertial references, and gimbal lock
	Computer
	Communications
Spacecraft	Guidance and control, inertial references, gimbal lock, manual steering, and displays
	Computer — displays and programs
	Optics — window visibility
	Propellants
	Separation and sequencing
	Operational
--	Alternate missions
	Minimum orbital altitude
	Entry corridor
	Recovery areas

Maneuver Monitoring

Within the context described, the primary objective (after crew safety) when a malfunction develops is to perform an alternate mission. Therefore, allowable deviated flight conditions must be determined in advance to ensure that the desired alternate-mission capability will exist. Consideration must also be given to the provision of reasonable initial conditions for performance of an abort maneuver. These requirements have been fulfilled by the development of a crew-monitoring procedure that includes appropriate S-IVB shutdown limits.

The crew must be able to monitor and evaluate TLI without ground support, because the S-IVB second burn can occur out of the Manned Space Flight Network (MSFN) tracking range. Generally, TLI occurs near Australia in the West Pacific Ocean. A schematic of the basic crew-maneuver-monitoring technique (fig. 8) shows that an abort can be performed for attitude-rate problems, for attitude-deviation problems, and for spacecraft system problems. Because S-IVB failures normally result in an alternate mission, only a critical spacecraft system problem is likely to require an abort. All four of these monitoring rules, as well as numerical limits, were established after studies showed that the alternate objectives and possibly crew safety could not be ensured without them.

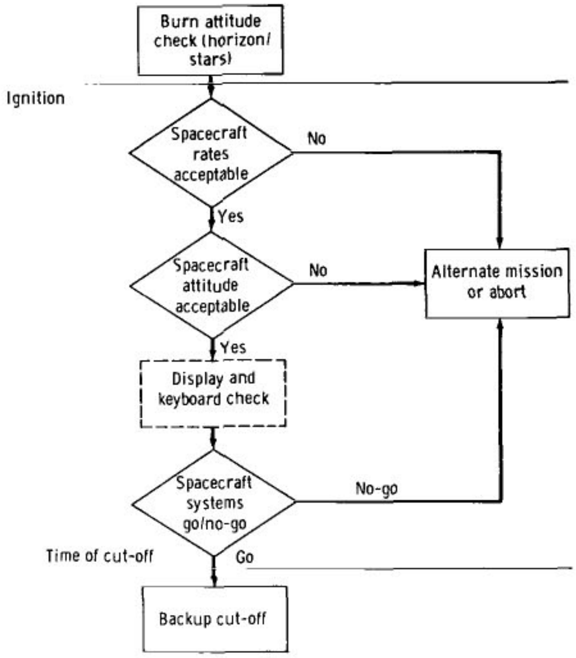


Figure 8. - Basic crew-maneuver-monitoring technique.

The following items can be noted about the TLI monitoring technique.

1. The TLI ignition will be inhibited if the launch-vehicle attitude before ignition is more than 10° from nominal, as determined by horizon reference.

2. The S-IVB engine will be cut off by the crew for pitch or yaw rates of 10 deg/sec or greater.

3. The S-IVB engine will be cut off by the crew with the abort handle for attitude deviations of 45° or more from the nominal attitude.

4. A backup to the S-IVB guidance cut-off signal will be performed by the crew if the S-IVB has not shut down at the end of the predicted burn time plus a 2σ dispersion of 6.0 seconds and if the nominal inertial-velocity display by the spacecraft computer has been achieved.

The crew is provided with preflight tables of attitude and computer-display parameter values at discrete times during the TLI burn. These tables provide for both nominal monitoring and crew manual steering. If a launch-vehicle inertial-platform failure occurs before TLI or if attitude-reference signals are lost during TLI, the crew may assume manual control of the burn with the hand controller. Typical problems that the monitoring concept can prevent or avert are shown in figure 9.

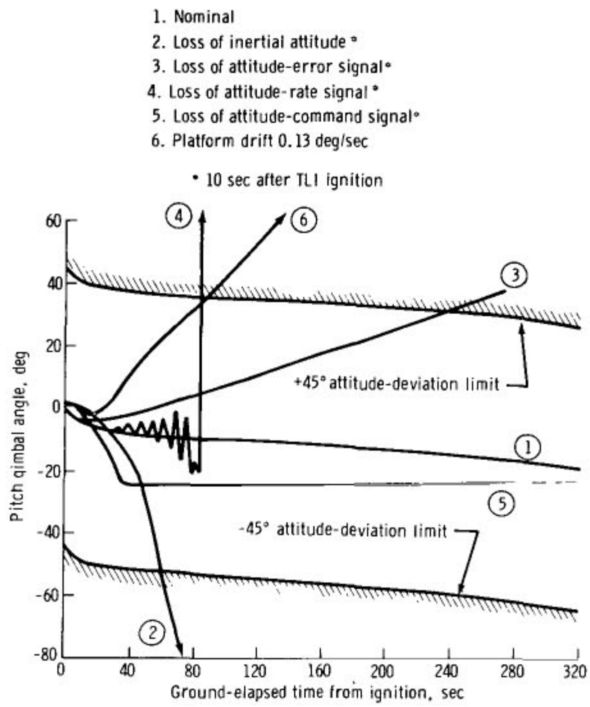


Figure 9. - Malfunctions during TLI.

Return to Earth

After the investigation of possible spacecraft system failures that might occur during TLI, no known single-point failures or constraint factors were found that might require the crew to abort the mission during TLI. However, trajectory analyses have been made that indicate the feasibility of performing a simple abort maneuver by using the horizon of the earth as a reference. The decision was reached to develop this method of aborting during TLI to protect the crew in the event future system-failure analyses dictated a need for a TLI abort capability.

A constant attitude of 5° was selected as optimum by trading off return times for aborts at either the start or the end of the TLI maneuver, that is, providing sufficient time from abort to entry for aborts at the beginning of TLI and decreasing the return time as much as possible for aborts at the end of TLI. A delay time of 10 minutes was selected for the following two reasons.

1. The 10-minute delay provided the crew with sufficient time to orient the spacecraft for the maneuver.

2. The inertial attitude of the vehicle remained constant at the abort point for the full range of TLI cut-off velocities. The operational technique resulting from these analyses requires the crew to carry a chart similar to figure 10 on board the spacecraft for use in conjunction with onboard displays because an abort during TLI would probably be made without ground-controller assistance. Although this abort procedure was eliminated as unnecessary on the later Apollo missions, it was discussed here to indicate the degree to which abort plans were developed.

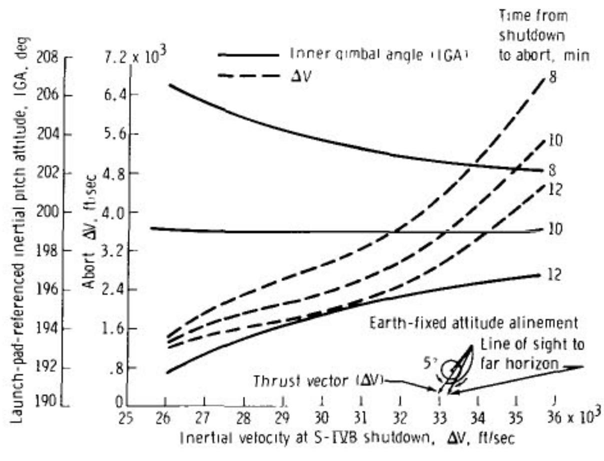


Figure 10. - Aborts during TLI using an earth-fixed thrust-vector attitude alinement.

A more practical technique was also pursued that would provide ground confirmation that systems suspected by the crew to be malfunctioning were in fact malfunctioning and that an abort was required.

This technique is partially facilitated by the fact that tracking ships are placed such that the acquisition of signal always occurs within 10 to 20 minutes after TLI.

Trajectory analyses indicated that, for any TLI position (geographic), communications could be established sooner by completing the TLI maneuver than by shutting down the S-IVB and coasting in an EPO. Thus, a systematic check of the spacecraft systems can be accomplished earlier by continuing the TLI maneuver. This procedure was adopted at the risk of increasing the return-to-earth time in order to maximize mission success probabilities if the failure indication should be negative.

After the decision was made concerning the best possible course of action following failure indications during TLI, the abort planning then consisted of determining, through trajectory analysis and consideration of the operational constraints, the earliest practical time that the mission could be aborted should the failure indication be confirmed. Among the factors considered were the time required to perform the malfunction check, S-IVB separation, orbit determination, abort-maneuver computations, and inertial-platform alinement. The resultant abort time is 90 minutes following nominal TLI cut-off. Implementation of this abort technique requires that ground controllers compute, during the EPO, a return-to-earth solution timed for 90 minutes following TLI. This solution results in a return-to-earth time of less than 18 hours and a return to one of five contingency landing sites that are geographic lines located near convenient recovery-staging areas throughout the world. The 18-hour return was consistent with the available time limit established for an assumed pressure-suit compressor malfunction. The probable recovery area would be the Atlantic Ocean because of the operational constraint that TLI should occur over the Pacific Ocean.

TRANSLUNAR COAST

Constraints

The abort planning for TLC (if a nominal TLI and a nominal transposition and CSM docking with the LM are assumed) consists primarily of determining which of the available propulsion systems should be used for the abort. With the LM and CSM docked, the total vehicle is capable of firing two independent main propulsion systems, with each vehicle having a primary guidance, navigation, and control system (PGNCS). In addition, the service module reaction control system (RCS) is available for maneuvering, and two independent communications systems are available. Also, the command module (CM) computer contains an abort program (P37) capable of computing targeting parameters for the CM PGNCS. With transformations, the CM abort program can be used in the LM. The following constraints are considered in planning abort trajectories.

1. The maximum total flight time considered is within the spacecraft system lifetime.

2. The maximum entry velocity considered is limited by the spacecraft entry heating constraints.

3. The abort maneuver magnitude is within the service propulsion system (SPS) capability.

4. The abort maneuver is targeted to conditions within which the crew and spacecraft are expected to operate at an optimum (that is, at design limits).

Three methods of effecting a return to earth from TLC involve maneuvers that are performed before the spacecraft reaches pericynthion and that provide a direct return to earth (direct-return aborts); maneuvers that are performed before the spacecraft reaches pericynthion and that provide a return to earth after the spacecraft passes behind the moon (circumlunar aborts); and maneuvers that are performed after the spacecraft reaches pericynthion and that provide a return to earth at that time (post-pericynthion aborts).

Because the flexibility afforded by multiple systems and maneuver choices does not allow for a simple contingency plan, other guidelines were developed. The following guidelines were among those developed.

1. Circumlunar aborts are not to be performed outside the lunar sphere of influence.

2. The LM descent propulsion system (DPS) will not be used instead of the SPS for a direct-return abort. (Also, disposing of the LM before the burn would be a real-time decision.)

Return to Earth

With continuous ground tracking available during TLC, the contingency warning decision can be made by the ground-control personnel. With key factors and potential problems already identified, abort plans for TLC are reduced essentially to real-time decisions. These decisions are facilitated by use of the type of data display shown in figure 11. Investigations helped reduce candidate displays to the simple necessities shown in figure 11. The TLC variables for SPS aborts to the primary contingency landing site located in the middle of the Pacific Ocean are also shown in figure 11. From trade-offs, possible abort solutions are selected for arbitrary times (usually coinciding with crew awake times) throughout TLC. These solutions are also analyzed before the mission to provide the crew with navigation sighting schedules that would assist them (in the event an abort is necessary) in performing the required navigation maneuvers to allow a safe return within the entry corridor at the desired landing site.

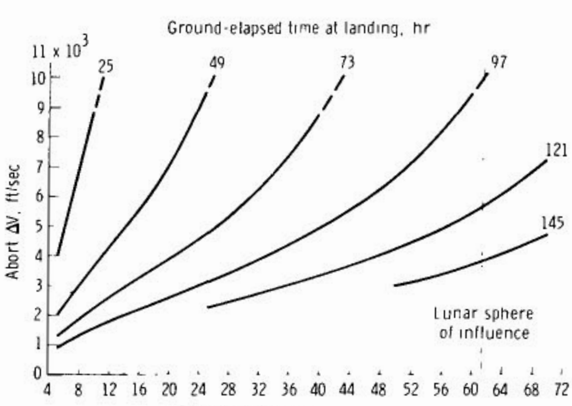
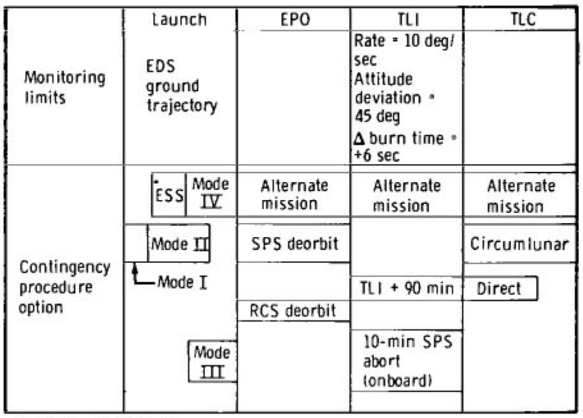


Figure 11. - Apollo 11 trade-off display.

The abort maneuver information provided to the crew in real time for guidance-computer targeting consists of abort time, ΔV, longitude of the earth landing site, and entry time. This information is used only in the event of a total loss of ground-to-air communications.

The outbound portion of the abort plan (from launch to lunar arrival) can be concisely represented as shown in figure 12.



*Early S-IVB staging

Figure 12. - Operational abort plan from launch to TLC.

Of particular interest are the relationships of the various abort modes to each other and to nominal mission events. As an example, maneuver monitoring occurs nominally during TLI, and an alternate mission is the preferred procedure rather than an abort at TLI plus 90 minutes, depending on the severity of the contingency.

Apollo 13 Translunar Coast Abort

It is appropriate to mention in this section the Apollo 13 contingency that incapacitated the electrical power and maneuvering capability of the CSM during the translunar phase of that mission. As a result, the TLC postpericynthion abort previously described was required to effect the safe return of the flight crew to earth. The abort maneuver was initiated 2 hours after pericynthion passage and was performed with the LM DPS as previously established by the abort plan.

In addition to providing a previously prepared and rehearsed return-to-earth technique, other aspects of the preflight abort planning were used during this emergency situation. For example, backup (to the CSM) life-support procedures and limitations using the LM had been identified. Also, the use of window views of the celestial sphere to obtain prescribed spacecraft attitudes for performing abort as well as midcourse correction (MCC) maneuvers had been developed, and this concept was used because the guidance and navigation (G&N) system was unavailable because of electrical power limitations.

LUNAR-ORBIT INSERTION AND LUNAR ORBIT

Constraints

The LOI burn, which is performed by the SPS, transfers the spacecraft from TLC to an LPO. Premature termination of the LOI burn places the vehicle on an off-nominal trajectory from which either an abort or an alternate mission may result. In the event an SPS failure occurs, the LM is required to return the CSM to earth.

The development of feasible abort procedures for the LOI mission phase must take into account many hardware and operational constraints. The major constraints that were included in the definition of an operational LOI abort philosophy are summarized in table IV.

TABLE IV. - PRIMARY CONSTRAINTS DURING LUNAR-ORBIT INSERTION

Constraint category	Constraint
Hardware	Engine duty cycles
	Optics — window visibility
	CSM and LM autopilots
	Inertial reference frames
	Propellants available
	Propulsion pressurization system
	Computer displays
	No onboard abort processor within lunar sphere of influence
	Real-time computer complex (RTCC) processor (single impulse only)
Operational	Minimum LM activation times
	MSFN tracking and RTCC solution requirements
	Maximum total mission time — consumables

Maneuver Monitoring

Because LOI always occurs behind the moon, the crew must be able to evaluate the progress of the maneuver without ground support. The recommended LOI crew-monitoring technique is shown in figure 8.

The preignition spacecraft attitude check, which is illustrated in figure 13, is made more difficult by the presence of the LM. However, the horizon and several stars should be visible from the CM rendezvous window, and these references may be used as a backup to the optics for the orientation check before ignition. If the spacecraft attitude is not within $\pm 5^\circ$ of nominal, the LOI should be no-go, because larger attitude-reference errors could result in more serious problems during this critical maneuver.

Although maintenance of crew safety is always the primary objective of the monitoring procedures, another important objective is the assurance that adequate abort capability is provided and that the capability is compatible with possible results of the

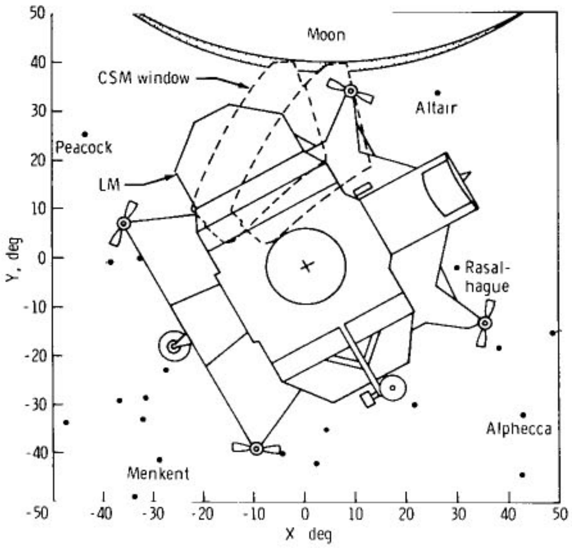


Figure 13. - Visual LOI attitude check.

monitoring procedures. This objective is accomplished for LOI by definition of sound procedures for the four types of problems possible during LOI. The four problem types are guidance and control, non-SPS systems (other than guidance and control), the SPS, and inadvertent SPS shutdowns.

A solution to the guidance and control problem is for the crew to assume manual control of the LOI maneuver, which is normally controlled by the PGNCS, and to complete the LOI at the original ignition attitude. One of the most dangerous possibilities associated with guidance and control problems is spacecraft inertial measurement unit (IMU) drift during LOI. The crew cannot detect a small drift until an attitude deviation builds up and appears on the secondary inertial-attitude reference system. Because the drift could occur in either the secondary reference system or the IMU, the crew is unable to distinguish the erroneous system without using the backup attitude error needles (a third inertial reference system). Detection of the error makes possible a manual takeover and completion of the burn so that the spacecraft can enter an LPO. Because uncorrected IMU drifts in pitch can produce impact trajectories, attitude limits for which a takeover should be initiated were developed (fig. 14). As is the case for TLI, the rate limit for LOI is 10 deg/sec because larger rates are not within normal system operation. If this limit is exceeded, a crew takeover is initiated, and manual completion of LOI at ignition attitude is performed. Non-SPS problems require completion of LOI, because it is advantageous to be in the planned lunar orbit rather than in a possibly undesirable lunar orbit in the event an abort is required.

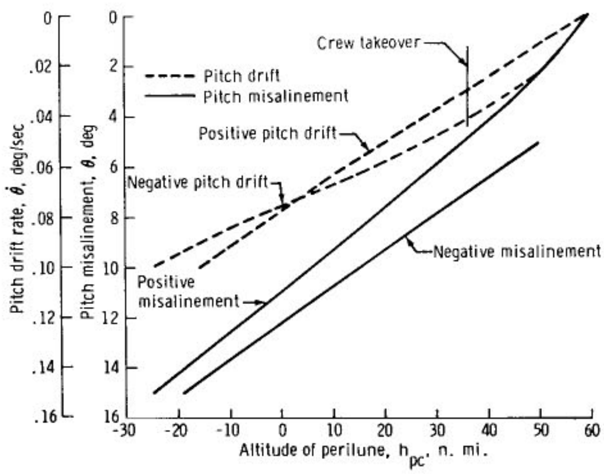


Figure 14. - Effects of attitude-reference failures on pericynthion altitude.

Manual SPS shutdown occurs only if critical SPS subsystem problems arise that would severely restrict the future performance of the engine or jeopardize the safety of the crew. If an inadvertent SPS shutdown occurs and SPS limits are not exceeded, the recommended procedure is to initiate an immediate SPS restart. If the restart is unsuccessful and an abort situation exists, the LM DPS is used for the abort maneuver.

In summarizing LOI monitoring, an important objective is completion of the LOI burn. Only when burn completion is not possible should the SPS burn be terminated. For this situation, the LM DPS is the primary source for the return-to-earth maneuver.

Return to Earth

Lunar orbits that result from premature LOI thrust termination range from highly energetic escape hyperbolas (which result from early SPS shutdown) to stable lunar ellipses (which result from late SPS shutdown). These lunar orbits fall into three general classes.

1. Escape trajectories (class I)

2. Unstable or impacting ellipses (class II)

3. Stable or nonimpacting ellipses (class III)

The three classes of lunar orbits are shown in figure 15. Escape trajectories (class I) are orbits with energies sufficiently high to cause escape from the lunar sphere of influence. Escape is sometimes caused by earth (third body) perturbations acting on a spacecraft in a highly elliptical orbit at or near apocynthion. Unstable ellipses (class II) are orbits with apocynthion altitudes high enough for the spacecraft to undergo large perturbations by the earth, but not high enough for the spacecraft to escape the lunar sphere of influence. Impacting ellipses (class II) are trajectories that are sufficiently perturbed for the spacecraft to impact the moon during its first approach to pericynthion. Stable nonimpacting ellipses (class III) result if SPS shutdown occurs during the final portion of the LOI burn. Stable ellipses are defined as orbits with pericynthion altitudes greater than 40 nautical miles. The regions of the typical lunar-mission LOI burn that will produce each trajectory class are illustrated in figure 16.

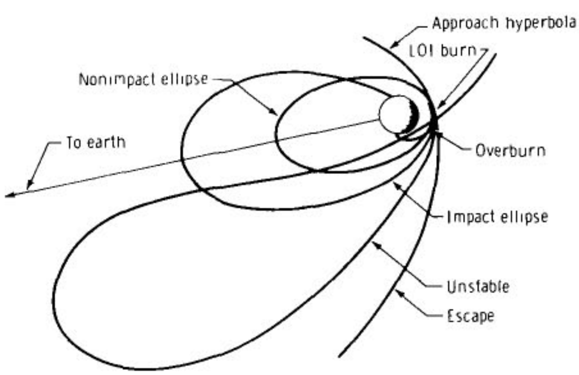


Figure 15. - Premature LOI shutdown trajectories.

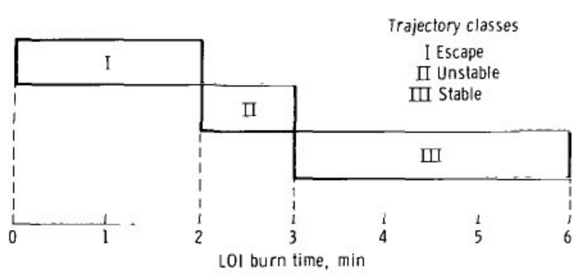


Figure 16. - Premature LOI shutdown trajectory classes.

Because the class III trajectories are stable lunar orbits with a pericynthion altitude in excess of 40 nautical miles, either an alternate mission or an abort may result. In the event an abort situation exists, the return-to-earth maneuver would be similar to the normal TEI burn and would occur on the far side of the moon. In this region, the abort maneuver (an LOI mode III abort) consists of a single burn that is generally within the ΔV capability of the LM DPS engine.

The abort ΔV requirements for a typical stable lunar ellipse are shown in figure 17. The abort ΔV is a function of the delay time and desired transearth flight time. The ΔV that is required rapidly increases as the time of the abort ignition

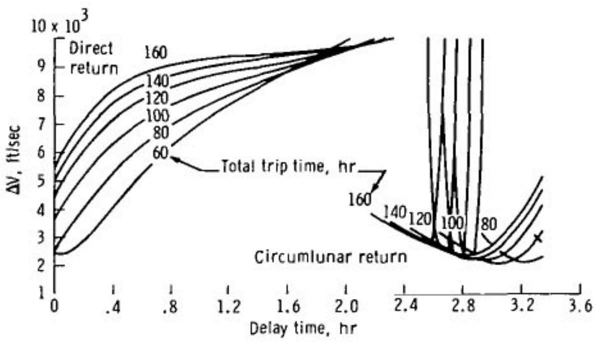


Figure 17. - Abort ΔV requirements for a stable ellipse.

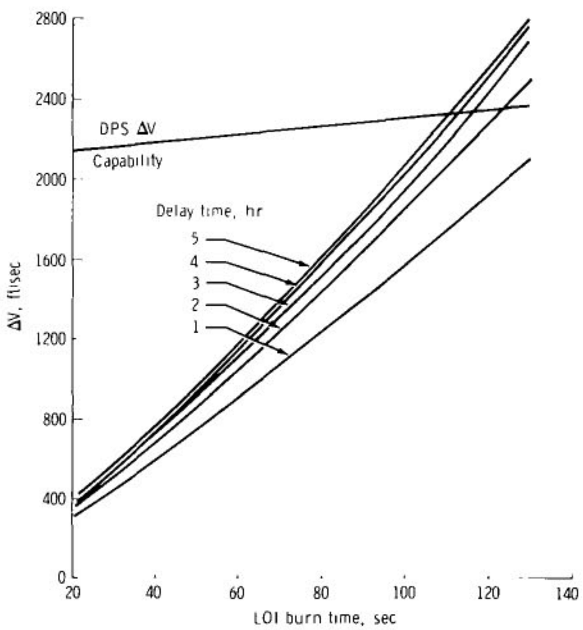


Figure 18. - Minimum ΔV requirements for LOI mode I aborts.

is delayed past the LOI burn shutdown. As apocynthion is passed in the ellipse, however, the more optimum circumlunar abort solutions become available, and the ΔV increases to a minimum value just before pericynthion is reached.

During the LOI burn, if an SPS shutdown occurs before a class III trajectory is reached, an abort is necessary because a stable lunar orbit has not been achieved. The single-impulse abort requirements for shutdowns early in the LOI burn are shown in figure 18 for a typical mission. This type of abort maneuver (an LOI mode I abort) is a function of delay time and desired transearth flight time (or desired landing-site longitude). The ΔV that is required increases proportionally with the time delay before execution of the abort maneuver. By superimposing on figure 18 the LM DPS ΔV that is available, it is shown that the LOI mode I abort capability decreases as the abort maneuver is delayed.

The development of operational abort techniques for the early Apollo lunar missions was greatly influenced by the constraints summarized in table IV. The ΔV capability of the LM was based on the use of the LM DPS engine only, because the use of the LM ascent propulsion system (APS) engine results in control problems. In addition, the time constraints of LM activation for a primary guidance and navigation system (PGNS) DPS burn, combined with the time constraints for ground trajectory-solution preparation, result in a minimum delay time of 2 hours for an abort using the LM PGNS DPS. The delay time has no effect on LOI mode III aborts because multiple abort opportunities exist. (That is, there are multiple revolutions in lunar orbit.) However, as can be seen from figure 18, the LOI mode I abort capability is directly affected.

As indicated in the previous discussion, the limitations caused by the real-time computer complex (RTCC) single-impulse abort solution capability (table IV) bound a region of the LOI burn for which an abort capability does not exist. In the absence of single-impulse abort solutions, an operational multi-impulse abort technique that used the existing ground computation programs had to be developed. The technique considered was a two-impulse procedure that would use the existing real-time single-impulse abort processor and thus avoid the development of a specialized first-impulse processor. Moreover, the first impulse could be specified in a convenient manner by using premission data.

Use of the DPS for multi-impulse operation necessitates the consideration of additional hardware restrictions. For example, multiburn operation of the DPS engine introduces constraints on the engine duty cycles, specifically with regard to burn duration and coast periods between the burns. A maximum coast-time limit between DPS burns exists because of the pressure buildup in the LM supercritical-helium storage tanks. In addition, the duration of the first burn must not exceed engine restrictions and thus inhibit the DPS restart capability. Crew activities and rest cycles also were considered.

The first maneuver of the two-impulse procedure must accomplish several goals. The maneuver must be small enough to allow a restart of the DPS and must leave enough propellant to complete the return-to-earth maneuver. The initial maneuver must also result in a safe intermediate lunar orbit that allows satisfactory delay time to the second impulse. Following an extensive analysis, the first maneuver was planned as a variable ΔV burn (depending on the LOI burn time) directed down the radius vector.

Selection of the ΔV value to be used for the corrective maneuver is based on a trade-off between total fuel expenditure and intermediate delay time. For LOI burn cut-offs early in the class II trajectory region, a large value of ΔV is necessary to reduce the apocynthion altitude (and thereby earth perturbations) to the extent that an adequate pericynthion for a stable ellipse (class III) results. Conversely, for late cut-offs in the class II trajectory region, where the preabort ellipse approaches a class III stable ellipse, a correspondingly small corrective maneuver is required.

The total ΔV of the two maneuvers increases almost linearly with the magnitude of the first burn ΔV_1, and a trade-off must be made between propellant costs and time between DPS burns. Increases in delay time before initiation of the first maneuver reduce the total ΔV requirements at the cost of increased delay time before the second maneuver and generally lower pericynthion altitudes caused by earth perturbations for the longer period orbits.

In view of these considerations, the final technique for determination of the corrective maneuver is to define the minimum allowable ΔV_1 necessary to obtain a trajectory just inside the class III trajectory region. This value of ΔV_1 will provide a pericynthion of approximately 60 nautical miles and a time between burns that is within the helium pressurization limits. The value of ΔV_1 will thus decrease linearly to 0 ft/sec at the start of the LOI mode III region.

The final two-impulse abort procedure (a mode II abort) for shutdowns in the class II trajectory region is as follows. The corrective maneuver is directed down the radius vector. (The RTCC targeting is determined from premission data.) The corrective maneuver is performed as soon as possible after LOI SPS shutdown (nominally at 2 hours for LM activation and ground-based tracking). The magnitude of the corrective maneuver decreases linearly with LOI burn time for simplification of real-time requirements.

Because the abort capability during LOI is a function of earth-moon geometry, LOI geometry, and so forth, the abort requirements must be determined for each particular mission. The total abort ΔV for minimum-fuel returns that would have been required following an SPS failure during the Apollo 11 LOI burn is summarized in figure 19. The value of ΔV_1 is shown in figure 20. The pericynthion altitude and the time between burns for the LOI mode II intermediate ellipse are shown in figure 21.

During the lunar-orbit phase, a return-to-earth maneuver (a mode III abort) similar to the nominal TEI burn can be initiated on each revolution.

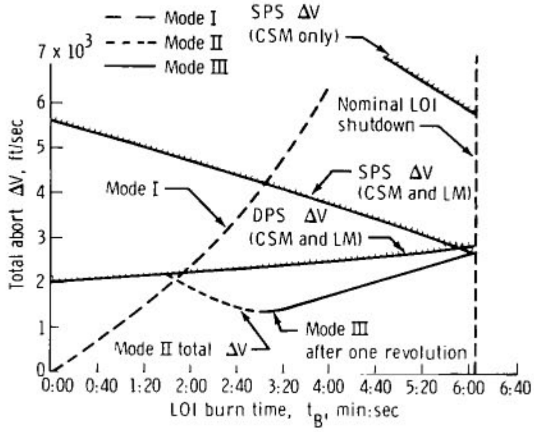


Figure 19. - Summary of minimum-fuel abort capability as a function of LOI burn time.

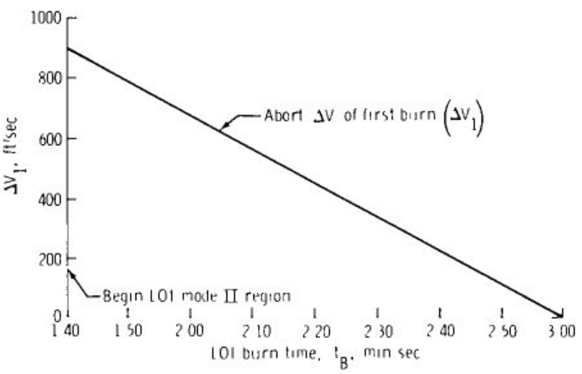


Figure 20. - The LOI mode II first-burn ΔV as a function of LOI burn time.

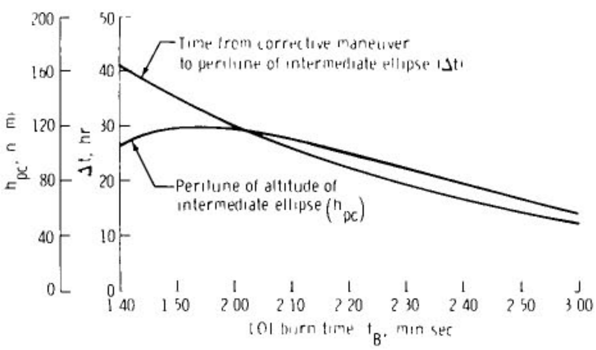


Figure 21. - Time between burns and the pericynthion altitude following a nominal first burn for an LOI mode II abort.

LUNAR DESCENT AND ASCENT

If an abort decision should be made after CSM/LM separation for the lunar-landing phase, rendezvous of the two vehicles must be effected before the return-to-earth maneuver. If descent procedures have also been initiated, the descent maneuvers are monitored to maintain rendezvous capability.

Descent-Orbit-Insertion-Maneuver and Powered-Descent-Maneuver Monitoring

The descent-orbit-insertion (DOI) maneuver is the first of the two descent maneuvers and occurs on the far side of the moon. The DOI, a short retrograde maneuver of approximately 75 ft/sec, is performed with the descent engine and efficiently reduces the orbit altitude from approximately 60 nautical miles to 50 000 feet for the powered descent initiation (PDI).

The PDI maneuver is initiated at the perilune (50 000-foot altitude) of the DOI orbit, which is targeted approximately 260 nautical miles up range of the landing site. The powered descent requires a continuous thrusting of the descent engine for 12 minutes 36 seconds. During this maneuver, the thrust direction and magnitude are modulated as necessary to bring the LM to a hovering condition over the desired landing site. The pitch attitude profile is designed to allow the crew visual assessment of the lunar surface during the terminal phases of the maneuver (from high gate, which is at approximately 7000 feet altitude).

The DOI maneuver is monitored exclusively by the crewmen because of the position of the maneuver. An overburn of 12 ft/sec (or 3 seconds) will cause the LM to be on an impacting trajectory before PDI. The maneuver is monitored for this impacting condition by comparing the PGNS performance with that of the abort guidance system (AGS) during the burn and by range/rate tracking with the rendezvous radar (RR) immediately after the burn. If the maneuver is unsatisfactory, an immediate rendezvous with the CSM is performed using the AGS.

The powered descent is a complex maneuver that is demanding on both crew and systems performance. Therefore, as much monitoring as possible is performed on the ground to reduce crew activities and to use complex computing techniques not possible on board the spacecraft. Obviously, time-critical failures and near-surface operations must be monitored on board by the crew for immediate action.

The ground monitoring to detect G&N system problems includes direct comparison of telemetered data from the two guidance systems on board the LM and data derived from the MSFN. The primary guidance-monitoring source is a comparison of the velocity components, the AGS-minus-PGNS velocities, and the MSFN-minus-PGNS velocities. In this manner, an erroneous system can be isolated by reference to the three sources. Limits are established on the MSFN-minus-PGNS velocity comparison such that a degrading guidance system can be detected early enough that a maneuver can be completed on the PGNS into a safe orbit (height of perigee $\geq$ 30 000 feet) without impacting the surface.

The performance of the total G&N system is evaluated by monitoring the commanded thrust magnitude (guidance-determined thrust command (GTC)). Nominally, the GTC decreases (approximately parabolically) from an initial value near 160 percent to the throttleable level of 57 percent approximately 2 minutes before high gate. If the DPS engine produces off-nominal low thrust, the GTC drops to 57 percent later to guide to the desired position and velocity. If the thrust becomes excessively low, the targets will not be satisfied and the guidance solution for the GTC can diverge. This divergence can result in an unsafe trajectory, one from which an abort cannot be satisfactorily performed because of excessive altitude rates. Hence, the GTC is monitored for divergence, and an abort on the PGNS is performed at the time of the detection of divergence.

The landing radar (LR)/PGNS interface is another area of prime concern during the descent. Without LR altitude updating, systems and navigational errors are such that the descent cannot be safely completed. In fact, it is unsafe to try to achieve high gate (the point where the crew can visually assess the approach) without altitude updating. Thus, a mission rule for real-time operation was established that calls for aborting the descent at a PGNS-estimated altitude of 10 000 feet if altitude updating has not been established. In addition to the concern for the time the initial altitude updating occurs, there is also concern for the amount of altitude updating (that is, the difference between PGNS and LR altitude determinations ΔH). If the LM is actually higher than the PGNS estimate, the LR will determine the discrepancy and update the PGNS. The guidance then tries to steer down rapidly to achieve the targets. As a result of the rapid changes, altitude rates may increase to an unsafe level for aborting the descent; that is, should an abort be required, the altitude rates could not be nulled by the ascent engine in time to prevent surface collision. The initial ΔH is monitored for acceptability before incorporation into the PGNS navigation. If the ΔH is unacceptable, it will not be incorporated and an abort is required.

The trajectory is monitored for flight safety at all times. The prime criterion for flight safety is the ability to abort the descent at any time until the final decision to commit to touchdown. Thus, flight dynamics limits are placed on altitude and altitude rate to ensure that the vehicle maintains the capability to abort on the APS until the last possible moment. The altitude and altitude rates are monitored by both the crewmen and the ground; however, because of communications delays, the ground only advises, based on projected trends, and the crewmen are responsible for protecting against flight-safety violations.

Lunar Module Abort to Rendezvous and CSM Rescue

From the beginning of the development of the procedures for the LM abort to rendezvous and the CSM rescue of the LM, the primary emphasis was placed on the previously discussed powered-descent maneuver and immediate-postlanding phase, which are the most probable phases for an LM abort. A considerable amount of planning was also done for failures associated with DOI, for cases of no PDI, and for correct-phasing LM ascents before the nominal lift-off time.

The original (beginning in 1964) LM-abort and CSM-rescue plans were extremely complex because of the limited onboard capabilities. For example, for an abort at any time during the powered-descent maneuver, the LM was targeted for a constant insertion orbit. Therefore, several abort regions existed, and the rendezvous techniques varied for each. For aborts early in the burn, the final approach of the LM to the CSM was from above; for a later region, one and one-half revolutions were required between the coelliptic sequence initiation (CSI) and constant differential height (CDH) rendezvous maneuvers instead of the normal one-half revolution; and, for late aborts, the LM approached from below the CSM. Because of this complexity in the LM abort plan, the rescue plan was also complex. The CSM did not have the CSI/CDH logic on board, and the CM pilot had to depend either on the ground or on the "mirror-image" technique (that is, the method whereby the CSM applies the LM-computed maneuver in the opposite direction). The primary rescue technique for bad-phasing situations was the six-impulse technique, in which the CSM transferred to a 20-nautical-mile circular orbit with the first two maneuvers and then adjusted the phasing, became coelliptic, and executed the terminal phase (theoretically with two impulses) with the last four maneuvers.

In early 1968, analyses were begun for the incorporation of several powered-descent-abort insertion orbits (to vary as a function of abort time regions). By late 1968, this work evolved into the variable-targeting concept, whereby the correct insertion orbit for an LM approach from a coelliptic differential height Δh of 15 nautical miles below the CSM could be targeted for all abort times during the first 10 minutes of powered descent. For an abort after 10 minutes, a constant 30-nautical-mile apolune orbit was targeted; however, an in-orbit phasing maneuver (derived from onboard programs in conjunction with onboard charts) permitted the standard LM approach from below, although one additional revolution was required. For the Apollo 12 mission, a second variable-targeting region (through a two-revolution rendezvous) replaced this post-10-minute phasing region. The variable-targeting concept was originally thought to be unfeasible because of the software requirements involved; however, after a detailed analysis of the precise requirements, the technique was deemed feasible and implementation began in early 1969.

The variable targeting led to much simplification and standardization of the abort and rescue plan. The same basic technique was now applicable for almost all cases in which the LM performed the rendezvous maneuvers. The rescue techniques, therefore, were standardized; for example, for a CSM-active terminal phase, the CSM would always approach the LM from above. By this time, the CSI/CDH logic had been placed on board the CSM, and an independent onboard rendezvous solution for the coelliptic sequence could be determined in the CSM. This technique was a great improvement in the CSM support of any rendezvous sequence using CSI/CDH logic. Emphasis was placed on spacecraft independence because of the uncertainty in the lunar potential and because nearly all rescue plans involved only one externally computed (ground) maneuver. When correct phasing existed initially, no external maneuver was required. The addition of very-high-frequency ranging capability to the CSM ensured further independence and confidence.

The current abort and rescue plan has been changed somewhat because of the change to the nominal plan (landing one revolution later relative to the main CSM/LM separation). However, the order of the occurrence of the regions (one-revolution or two-revolution rendezvous) is the only significant change; the basic techniques are the same. The current plan is by no means a simple one; but, compared to the plan in use approximately 1 year before the Apollo 11 flight, the current plan is considerably simpler and more standard.

Powered Ascent

During ascent from the lunar surface, the LM ascent stage is steered by the PGNS to effect the planned rendezvous with the orbiting CSM. As described in the descent monitoring section, the guidance system is evaluated to determine proper operation. Detection of errors or malfunctions could be cause for ascent-maneuver completion on the AGS. A nonnominal orbit after cut-off caused by an early engine shutdown could require the CSM to perform the rendezvous to rescue the LM if the LM maneuvering capability were lost. The limiting altitude on CSM rescue of the LM is 30 000 feet.

TRANSEARTH INJECTION AND TRANSEARTH COAST

Constraints

The TEI burn is intended to transfer the spacecraft from its nominal LPO to an earth-return moon-centered hyperbola. At this time, the spacecraft consists solely of the CSM combination. Therefore, the sole propulsion source is the SPS because the service module RCS is incapable of performing a burn as large as that required for the TEI maneuver. The constraints (both hardware and operational) that were considered for TEI aborts are similar to those for LOI aborts, with the exception of LM-related limitations.

Maneuver Monitoring

Both TEI and LOI occur behind the moon, and the monitoring procedures and techniques for both maneuvers are basically the same. Preignition attitude checks from the CM windows are performed the same as for LOI. The major difference is that the guidance and control and system problems during TEI require a continuation of the maneuver; that is, guidance and control problems result in crew takeover and burn completion at the ignition attitude, and SPS or spacecraft system problems are ignored until the important TEI maneuver has been completed. A backup to the CM PGNCS TEI cut-off is performed by the crew at 2 seconds past the nominal cut-off time, following confirmation that the desired cut-off velocity has been achieved (as shown by the entry-monitoring-system ΔV counter). If inadvertent termination of the SPS occurs during TEI, the engine is restarted, if possible, within approximately 30 seconds, or a ground solution will be required for a later abort attempt. Manual takeover of the TEI maneuver occurs if, as explained previously for LOI, the crew confirms by use of two independent reference systems a 10° deviation from the fixed inertial burn attitude or if the TEI rate limit of 10 deg/sec is exceeded. The attitude-deviation limit was selected with the aid of the data presented in figure 22. The midcourse correction required following a TEI maneuver that has been made relative to a drifting inertial-reference platform is also shown in figure 22.

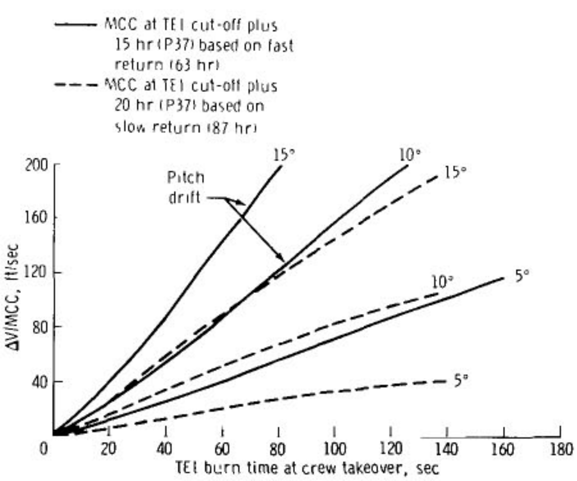


Figure 22. - The ΔV for midcourse corrections as a function of TEI burn time at crew takeover for various pitch drifts.

In summary, the philosophy of TEI burn monitoring is that completion of the

TEI burn is mandatory; that is, a manual shutdown is not to be initiated for any CSM system problem. If an early automatic SPS shutdown occurs, an immediate restart is to be attempted. Only if immediate reignition is not possible will an RTCC abort solution be required.

Return to Earth

The orbits that result from premature TEI thrust terminations are similar to the orbits that result from LOI underburns; however, the orbits that result from premature TEI thrust terminations occur in reverse order and as a function of TEI burn time. Therefore, the TEI phase of the mission has abort characteristics similar to those of the LOI phase, with the addition of the following two facts peculiar to the TEI phase.

1. The increased abort ΔV capability exists because of the use of the SPS and the lack of the LM weight.

2. There is no backup propulsion system for a TEI abort.

During TEC, abort maneuvers are initiated only if a much-faster-than-nominal earth return is required or if a change in landing position is necessary. A concise form of the abort plan from lunar arrival to TEC is presented in figure 23.

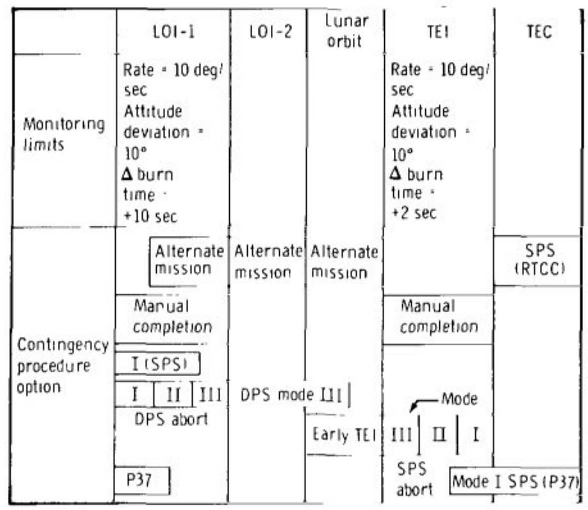


Figure 23. - Operational abort plan from lunar arrival to TEC.

CONCLUDING REMARKS

Some of the planning performed to ensure the safe return to earth of an Apollo crew in the event of a contingency situation during a lunar-landing mission has been described. In particular, the development of crew warning and escape methods for each mission phase has been emphasized. The development was accomplished primarily by providing powered-flight monitoring procedures and abort modes that are compatible with hardware and operational constraints. Because of the interaction of these constraints and because so many systems and components can malfunction, much premission effort is required to ensure that abort techniques are available during all mission phases. Although probabilities are low that an abort plan will be put into effect, the high confidence level in the Apollo Program is, in part, because of the fact that a safe, simple, well-rehearsed abort plan exists.

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