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APOLLO EXPERIENCE REPORT - MISSION PLANNING FOR LUNAR MODULE DESCENT AND ASCENT

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16. Abstract

The premission planning, the real-time situation, and the postflight analysis for the Apollo 11 lunar descent and ascent are described in this report. A comparison between premission planning and actual results is included. A navigation correction capability, developed from Apollo 11 postflight analysis, was used successfully on Apollo 12 to provide the first pinpoint landing. An experience summary, which illustrates typical problems encountered by the mission planners, is also included.

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APOLLO EXPERIENCE REPORT

MISSION PLANNING FOR LUNAR MODULE DESCENT AND ASCENT*

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SUMMARY

Premission planning, real-time analysis of mission events, and postflight analysis are described for the lunar module descent and ascent phases of the Apollo 11 mission, the first manned lunar landing, and for the Apollo 12 mission, the first pinpoint lunar landing. Based on the Apollo 11 postflight analysis, a navigation correction capability was provided for the Apollo 12 descent. Flight results for both missions are shown to be in agreement with premission planning. A summary of mission-planning experience, which illustrates typical problems encountered by the mission planners, is also included in this report.

INTRODUCTION

Premission planning for Apollo lunar module (LM) descent and ascent started in 1962 with the decision to use the lunar orbit rendezvous (LOR) technique for the Apollo lunar-landing mission (ref. 1). The LOR concept advanced by Houbolt and others is defined in references 1 and 2. The technique allowed optimization of both the design of LM systems and trajectories for orbital descent to and ascent from the lunar surface.

The LM descent was designed to be accomplished in two powered-flight maneuvers: the descent orbit insertion (DOI) maneuver and the powered-descent maneuver. The DOI maneuver, a short or impulse-type transfer maneuver, is performed to reduce the orbit altitude of the LM from the command and service module (CSM) parking orbit to a lower altitude for efficiency in initiating the longer, more complex powered-descent maneuver. The basic trajectory design for the powered descent was divided into three operational phases: an initial fuel-optimum phase, a landing-approach transition phase, and a final translation and touchdown phase. The initial trajectory analysis which led to this design was performed by Bennett and Price (ref. 3). In reference 4, Cheatham and Bennett provided a detailed description of the LM descent

*The material presented in this report, with the exception of the section entitled "Mission-Planning Experience," was previously published in NASA TM X-58040.

design strategy. This description illustrates the complex interactions among systems (guidance, navigation, and control; propulsion; and landing radar), crew, trajectory, and operational constraints. A more detailed description of the guidance, navigation, and control system is given by Sears (ref. 5). As LM systems changed from design concept to hardware, and as operational constraints were modified, it became necessary to modify or reshape the LM descent trajectory; however, the basic three-phase design philosophy was retained.

The LM ascent was designed as a single powered-flight maneuver to return the crew from the lunar surface, or from an aborted descent, to a satisfactory orbit from which rendezvous with the CSM could be performed. The basic trajectory design for the powered ascent was divided into two operational phases: a vertical-rise phase for surface clearance and a fuel-optimum phase for orbit insertion. Thus, the ascent planning was more straightforward than the descent planning and, because of the lack of a lunar atmosphere, less complex than earth-launch planning.

The purpose of this report is to describe the premission operational planning for LM descent and ascent; that is, to describe the bridge from design planning to flight-operation status. A discussion of the primary criteria which precipitated the plan for the Apollo 11 mission, a comparison of the real-time mission events with this plan, a discussion of the postflight analysis of the Apollo 11 mission and its application to the Apollo 12 and subsequent missions, and a brief postflight discussion of the Apollo 12 mission are included in this report. In addition, a section on mission-planning experience is included to provide insight into typical problems encountered by the mission planners and the solutions that evolved into the final operational plan.

The author wishes to acknowledge the assistance of the members of the Lunar Landing Section of the Landing Analysis Branch (Mission Planning and Analysis Division), particularly, W. M. Bolt, J. H. Alphin, J. D. Payne, and J. V. West, who contributed to the generation of the data presented in this report.

PREMISSION PLANNING

Premission planning entails the integration of mission requirements or objectives with system and crew capabilities and constraints. The integration is time varying because neither mission requirements nor system performances remain static. This has been particularly true of the LM descent and ascent maneuvers, which were in design and planning for 7 years.

In this section, the final evolution of the planning for the descent and ascent maneuvers for the Apollo 11 mission will be described. A brief description of the pertinent systems, the guidance logic, the operational-design phases, the trajectory characteristics, and the ΔV and propellant requirements for each maneuver is provided.

Descent Planning

The LM descent from the CSM parking orbit (approximately 62 by 58 nautical miles) is illustrated in figure 1. After the LM and the CSM have undocked and separated to a safe distance of several hundred feet, the LM performs the DOI, which is the first and simplest of the two descent maneuvers. The DOI, which is a short retrograde maneuver of approximately 75 fps, is performed with the LM descent engine and is made at a position in the orbit 180° from powered descent initiation (PDI), which is the second descent maneuver. The purpose of the DOI is to reduce efficiently (with Hohmann-type transfer) the orbit altitude from approximately 60 nautical miles to 50 000 feet in preparation for PDI. Performance of continuous powered descent from altitudes much greater than 50 000 feet is inefficient, and a PDI at lower than 50 000 feet is a safety hazard (ref. 3). The DOI is described in the operational trajectory documentation at the NASA Manned Spacecraft Center and is discussed further in the section entitled "Real-Time Analysis." Powered-descent planning is discussed in the remainder of this section.

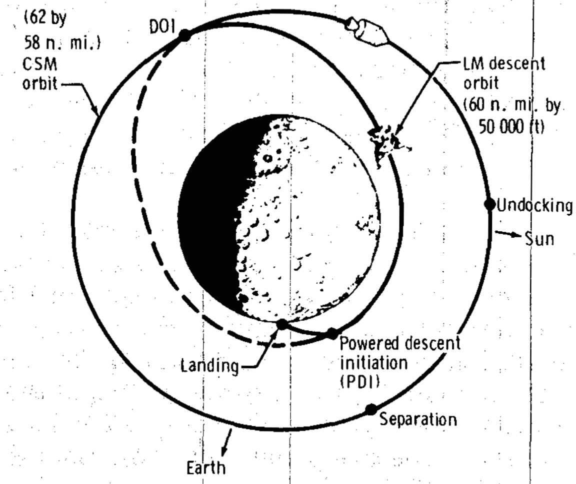
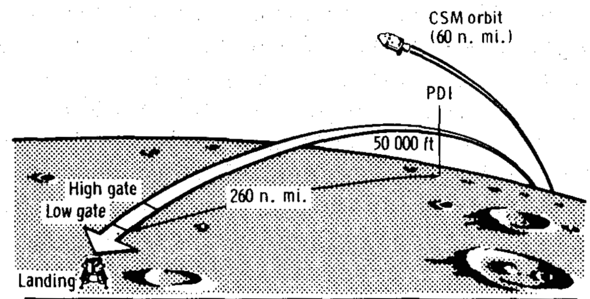


Figure 1. - Lunar module descent.

Operational phases of powered descent. - The LM powered-descent trajectory design was established (ref. 1) as a three-phase maneuver (fig. 2) to satisfy the operational requirements imposed on such a maneuver. The first operational phase, called the braking phase, is designed primarily for efficient propellant usage while the orbit velocity is being reduced and the LM is guided to high-gate conditions for initiation of the second operational phase, called the approach phase. The term "high gate" is derived from aircraft-pilot terminology and refers to beginning the approach to an airport. The approach phase is designed for pilot visual (out of the window) monitoring of the approach to the lunar surface. The final operational phase or landing phase, which begins at low-gate conditions (again from aircraft-pilot terminology), is designed to provide continued visual assessment of the landing site and to allow pilot takeover from automatic control for the final touchdown on the lunar surface. A brief description of the systems and the guidance and targeting logic required for achieving these operational phases is given in the following sections. A detailed description of each phase is also given in the operational trajectory documentation.



Phase	Initial event	Design criteria
Braking	PDI	Minimize propellant usage
Approach	High gate	Crew visibility
Landing	Low gate	Manual control

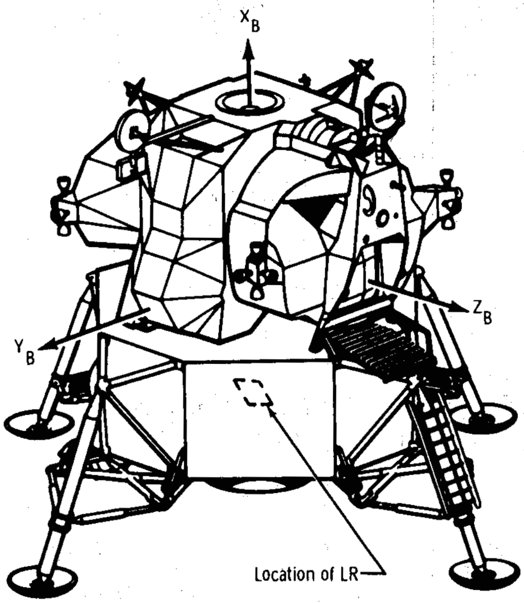
Figure 2. - Operational phases of powered descent.

System descriptions. - The success of the LM powered descent depends on the smooth interaction of several systems. The pertinent systems are the primary guidance, navigation, and control system (PGNCS); the descent propulsion system (DPS); the reaction control system (RCS); the landing radar (LR); and the landing point designator (LPD). A detailed description of each system and its performance characteristics is given in reference 6. A brief description of each system follows.

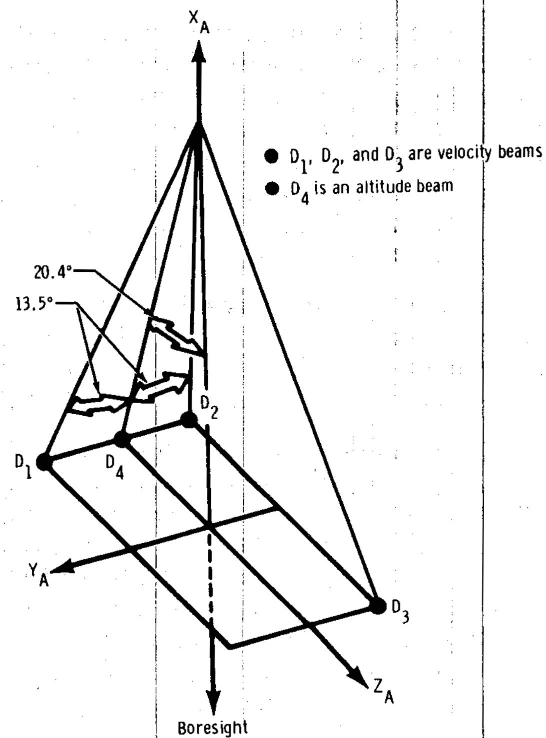
The PGNCS consists of two major subsystems: an inertial measurement unit (IMU) and a computer. The IMU is the navigation sensor, which incorporates accelerometers and gyros to sense changes in LM velocity and attitude. The IMU sends this information to the computer, which contains preprogramed logic for navigation, for calculation of guidance commands, for sending steering commands (by means of the digital autopilot (DAP)) to the DPS and the RCS, for processing LR measurements of LM range and velocity relative to the lunar surface, and for display of information to the crew. The crew controls the mode of computer operation through a display and keyboard (DSKY) assembly. A description of the guidance logic is given in a subsequent section, and a complete description of the guidance, navigation, and control logic can be found in reference 7.

The DPS, which contains the rocket engine used for lunar descent and its controls, consists of a throttle and a gimbal drive capable of $\pm 6^\circ$ of motion. The engine has a maximum thrust of approximately 10 000 pounds (nominal engines varying from 92.5 to 95.5 percent of the design thrust of 10 500 pounds). The maximum thrust level is referred to as the fixed throttle position (FTP) and is used for efficient velocity reduction during the braking phase. The throttle can be controlled automatically by the PGNCS guidance commands or by manual controls. The descent engine is throttleable between 10 and 60 percent of design thrust for controlled operations during the approach and landing phases. The gimbal drive is controlled automatically by the DAP for slow attitude-rate commands. For high-rate changes, the DAP controls the RCS, which consists of four groups of four small control rockets (100 pounds of thrust each) mounted on the LM to control pitch, roll, and yaw.

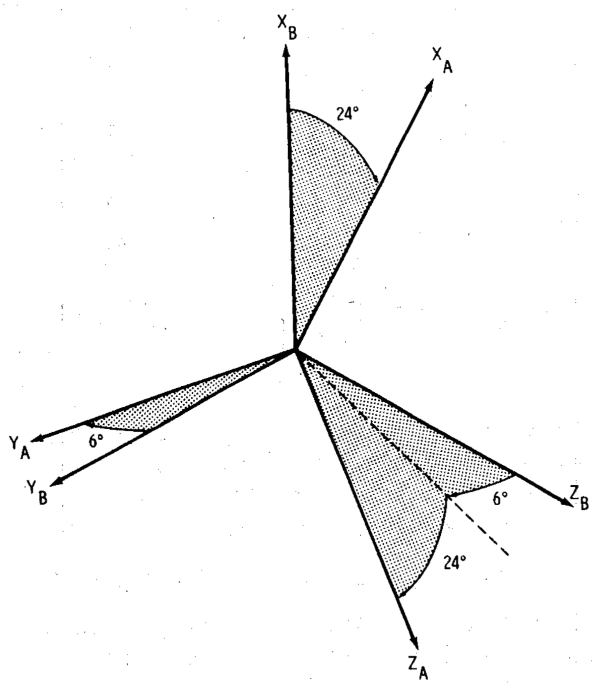
The LR, mounted at the bottom rear of the LM, is the navigation sensor which provides ranging and velocity information relative to the lunar surface. The LR consists of four radar beams, one beam to provide ranging measurements and three beams to provide velocity measurements. This beam pattern, which is illustrated relative to the LM body axis system in figures 3(a) and 3(b), can be oriented in one of two positions, as shown in figures 3(c) and 3(d). Position 1 (fig. 3(c)) is used in the braking phase of the descent when the LM is oriented near the horizontal. Position 2 (fig. 3(d)) is used during the approach and landing phases of descent when the LM nears a vertical attitude. The guidance computer converts the ranging information to altitude data and updates its navigated position every 2 seconds. The guidance computer also converts the velocity measurement along each radar beam to platform coordinates and updates a single component of its navigated velocity every 2 seconds; thus, 6 seconds is required for a complete velocity update. The LR data are weighted before they are incorporated into the guidance computer (ref. 7).



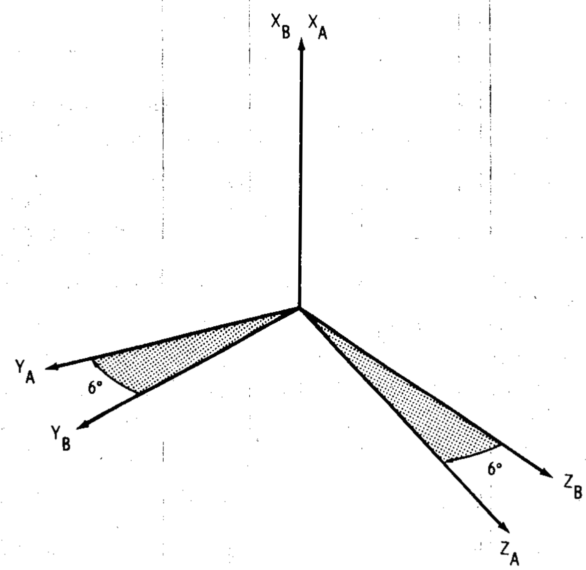
(a) Lunar module body axes.



(b) Landing radar antenna axes.



(c) Landing radar position 1 (used in braking phase).



(d) Landing radar position 2 (used in approach and landing phases).

Figure 3. - Lunar module body axes and LR antenna axes.

The final system to be described is a grid on the commander's forward window called the LPD (fig. 4). The window is marked on the inner and outer panes to form an aiming device or eye position. During the approach and landing phases, the computer calculates the look angle (relative to the forward body axis Z_B) to the landing site and displays it on the DSKY. The commander can then sight along the angle on the LPD (zero being along body axis Z_B) to view the landing area to which he is being guided. If the commander desires to change the landing area, he can make incremental changes inplane or cross range by moving the hand controller in the appropriate direction to provide input to the computer. Cross-range position is changed in 2° increments, and inplane position is changed in 0.5° increments. A detailed description of the guidance logic is given in references 7 and 8.

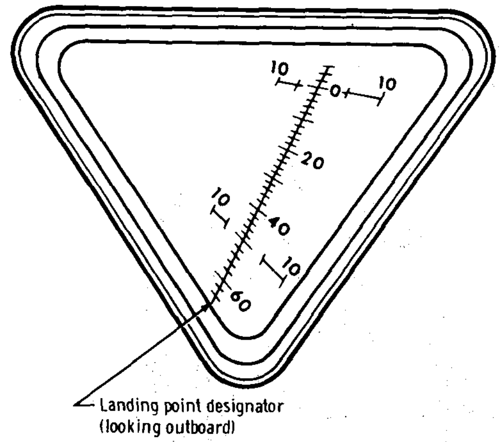


Figure 4. - Lunar module forward window.

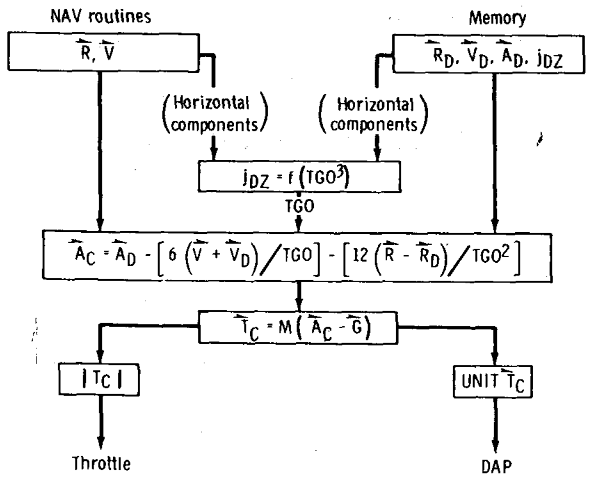


Figure 5. - Basic LM descent guidance logic.

Guidance logic. - The basic LM descent guidance logic is defined by an acceleration command which is a quadratic function of time and is, therefore, termed quadratic guidance. A simplified flow chart of quadratic guidance is given in figure 5. The current LM position and velocity vectors $\vec{R}$ and $\vec{V}$ are determined from the navigation routine. The desired (or target) position vector $\vec{R}_D$, velocity vector $\vec{V}_D$, acceleration vector $\vec{A}_D$, and down-range component of jerk j_{DZ} are obtained from the stored memory. (Jerk is the time derivative of acceleration.) The down-range (horizontal) components of these state vectors (current and desired) are used in the jerk equation to determine the time to go (TGO); that is, the time to go from the current to the desired conditions. If the TGO, the current state vector, and the desired state vector are known, then the commanded acceleration vector $\vec{A}_C$ is determined from the quadratic guidance law. Note that the acceleration-command equation yields infinite commands when the TGO reaches zero. For this reason, the targeting is biased such that the desired conditions are achieved prior to the TGO reaching zero. By using spacecraft mass M, calculating the vector difference between the commanded acceleration and the acceleration of lunar gravity $\vec{G}$, and applying Newton's law, a commanded thrust vector $\vec{T}_C$ can be found. The magnitude of $\vec{T}_C$ is used to provide automatic throttling of the DPS. When the throttle commands exceed the throttle region of the DPS (10 to 60 percent of design thrust), maximum thrust (FTP) is applied. The vector direction is used by the DAP to orient the DPS thrust by either trim gimbal attitude commands or RCS commands to reorient the entire spacecraft.

During the powered descent, the guidance computer provides several sequential programs (P63 to P67) for guidance and control operations. A description of each program follows. A complete description of the descent guidance logic and guidance modes is given in references 7 to 9. The first program is P63, entitled "Braking Phase Guidance." Program 63 contains an ignition algorithm and the basic guidance logic. The ignition logic, which determines the time for the crew to ignite the DPS for PDI, is based on a stored, preselected surface range to the landing site. After descent-engine ignition, the basic guidance logic is used to steer the LM to the desired conditions for the beginning of the approach phase. As stated previously, the targets are selected with a bias such that the desired conditions are achieved prior to the TGO reaching zero. When the TGO reaches a preselected value, the guidance program switches automatically from P63 to P64, which is entitled "Approach Phase Guidance." Program 64 contains the same basic guidance logic as P63, but a new set of targets is selected to provide trajectory shaping throughout the approach and landing phases and to establish conditions for initiating an automatic vertical descent from a low altitude to landing. In addition, P64 provides window-pointing logic for the LPD operation. That is, the landing point will be maintained along the LPD grid on the commander's window. During this time, the crew can make manual inputs with the attitude hand controller to change incrementally (down range or cross range) the intended landing site and remain in automatic guidance. (See the section entitled "System Descriptions.")

When the TGO reaches a preselected value, the guidance program switches automatically from P64 to P65, which is entitled "Velocity Nulling Guidance." Program 65, which nulls all components of velocity to preselected values, is used for an automatic vertical descent to the surface, if desired. No position control is used during this guidance mode. The sequencing for automatic guidance is illustrated in figure 6.

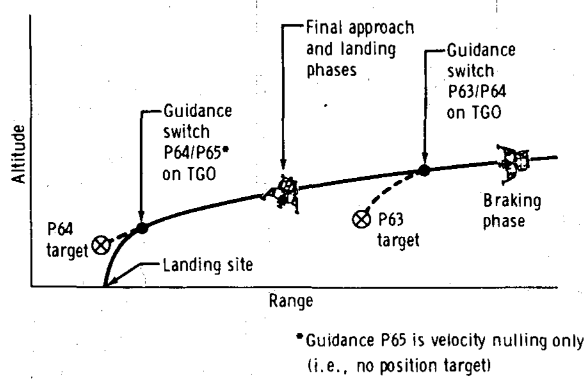


Figure 6. - Target sequence for automatic-descent guidance.

Program 66, entitled "Rate of Descent," and program 67, entitled "Manual Guidance," are optional modes which can be used at crew discretion (manually called up through the DSKY) at any time during the automatic guidance modes (P63, P64, or P65). During P66 operation, the crew control spacecraft attitude, and the computer commands the DPS throttle to maintain the desired altitude rate. This rate can be adjusted by manual inputs from the crew and is normally entered late in P64 operation (near low gate) prior to P65 switching for manual control of the final touchdown position. Program 67 maintains navigation and display operations for complete manual control of the throttle and altitude. Normally, this mode is not used unless P66 is inoperative.

Braking phase. - A scale drawing of the LM powered descent for the Apollo 11 mission is given in figure 7. The intended landing area, designated Apollo site 2, in the Sea of Tranquility is centered at latitude 0.6° N and longitude 23.5° E. The major events occurring during the braking phase (illustrated in figure 7 and tabulated in table I) are discussed as follows. The braking phase is initiated at a preselected range approximately 260 nautical miles from the landing site near the perilune of the descent transfer orbit (altitude of approximately 50 000 feet). This point is PDI, which coincides with DPS ignition. Ignition is preceded by a 7.5-second RCS ullage burn to settle the DPS propellants. The DPS is ignited at trim (10 percent) throttle. This throttle setting is held for 26 seconds to allow the DPS engine gimbal to be alined (or trimmed) through the spacecraft center of gravity before throttling up to the maximum or fixed throttle position. The braking phase is designed for efficient reduction of orbit velocity (approximately 5560 fps) and, therefore, uses maximum thrust for most of the phase; however, the DPS is throttled during the final 2 minutes of this phase for guidance control of dispersions in thrust and trajectory. As stated earlier, the DPS is throttleable only between 10 and 60 percent; therefore, during FTP operation, the guidance is targeted such that the commanded quadratic acceleration, and consequently the command thrust, is a decreasing function. When the command decreases to 57 percent, a 3-percent low bias, the DPS is throttled as commanded (illustrated by the time history of commanded and actual thrust shown in fig. 8(a)). The thrust attitude (pitch) profile is shown in figure 8(b). Early in the descent, orientation about the thrust axis is by pilot discretion. The Apollo 11 crew oriented in a windows-down attitude for visual ground tracking as a gross navigation check. Rotation to a windows-up attitude is performed at an altitude of approximately 45 000 feet, so that the LR can acquire the lunar surface to update the guidance computer estimates of altitude and velocity. Altitude updating is expected to begin at an altitude of approximately 39 000 feet; velocity updating is expected to begin at approximately 22 000 feet.

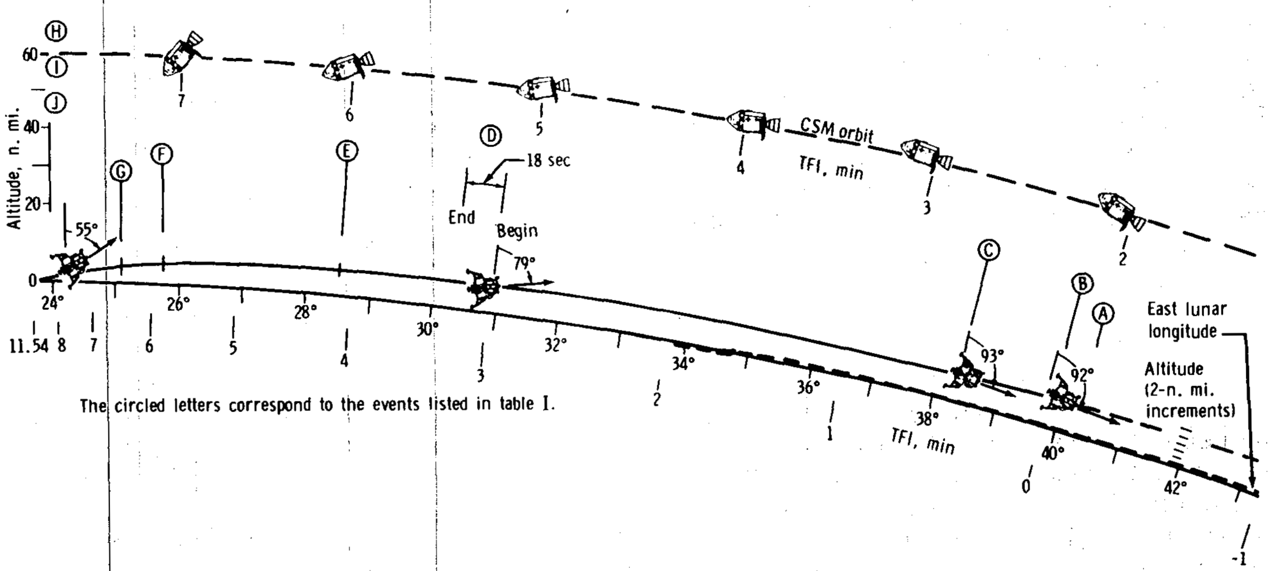


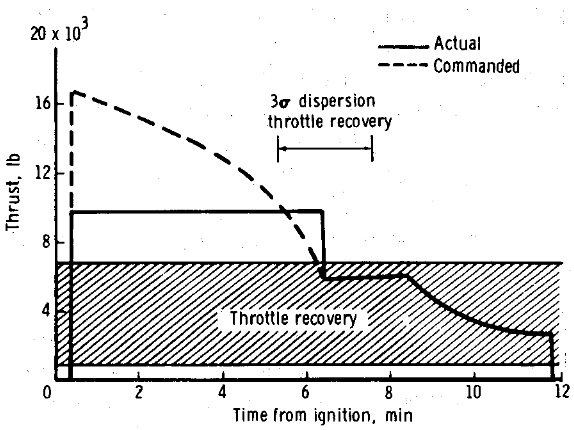
Figure 7. - Premission Apollo 11 LM powered descent.

TABLE I. - APOLLO 11 PREMISSION POWERED-DESCENT EVENT SUMMARY

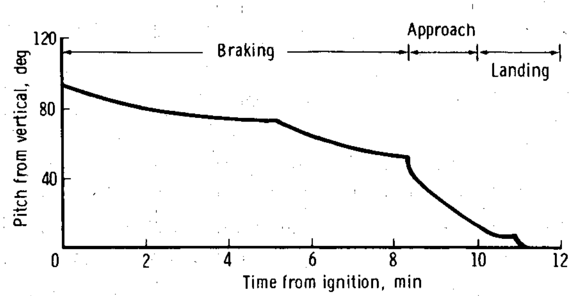
Event	TFI, min: sec (a)	Inertial velocity, fps	Altitude rate, fps	Altitude, ft	ΔV, fps
Ullage	-0: 07				
Powered descent initiation	0: 00	5560	-4	48 814	0
Throttle to maximum thrust	0: 26	5529	-3	48 725	31
Rotate to windows-up position	2: 56	4000	-50	44 934	1572
LR altitude update	4: 18	3065	-89	39 201	2536
Throttle recovery	6: 24	1456	-106	24 639	4239
LR velocity update	6: 42	1315	-127	22 644	4399
High gate	8: 26	506	-145	7 515	5375
Low gate	10: 06	[b]55(68)	-16	512	6176
Touchdown (probe contact)	11: 54	[b]-15(0)	-3	12	6775

[a]Time from ignition of the DPS.

[b]Horizontal velocity relative to the lunar surface.



(a) Thrust.



(b) Attitude.

Figure 8. - Premission Apollo 11 time history of thrust and attitude.

The braking phase is terminated when the guidance-calculated TGO to achieve targets is reduced to 60 seconds. Termination occurs at an altitude of approximately 7000 feet, a range of approximately 4.5 nautical miles from the landing site, and a time from ignition (TFI) of 8 minutes 26 seconds. The guidance computer automatically switches programs and targets from P63 to P64 to begin the approach phase, as explained in the previous section.

Approach phase. - The approach phase (fig. 9) provides visual monitoring of the approach to the lunar surface. That is, the guidance (P64) is targeted to provide spacecraft attitudes and flight time adequate to permit crew visibility of the landing area through the forward window throughout the approach phase. At high gate, in addition to the guidance-program switch, the LR antenna is changed from position 1 to position 2 for operation near the lunar surface. (See the section entitled "System Descriptions.") The trajectory-approach angle (glide angle) is shown to be approximately 16° relative to the surface. This angle allows the crew visual line of sight to the landing area to be above the sun angle (10.9° nominal to 13.6° maximum) even in dispersed (up to 3σ) situations. The angle above the sun line is desirable because surface features tend to be washed out when looking along or below the sun line. (See reference 10.) The LM attitude, LPD angle, and LR beam geometry are also shown in figure 9. During the approach phase, the altitude decreases from 7000 to 500 feet, the range decreases from approximately 4.5 nautical miles to 2000 feet, and the time of flight is approximately 1 minute 40 seconds. Although no guidance changes or other transients are made, operationally, the approach phase is considered to be terminated at an altitude of 500 feet (low gate), at which point the landing phase begins.

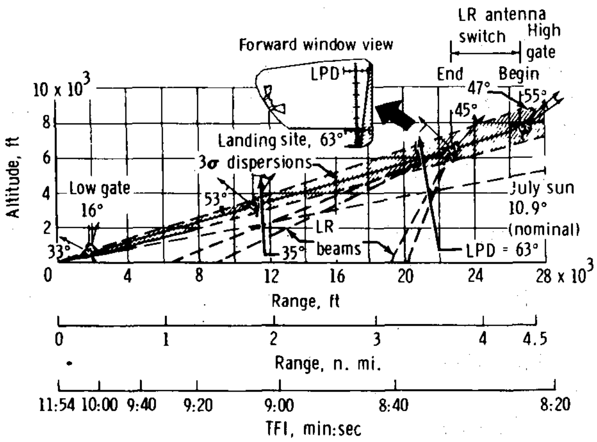


Figure 9. - Approach phase.

Landing phase. - The landing phase is designed to provide continued visual assessment of the landing site and to provide compatibility for pilot takeover from the automatic control. No change occurs in guidance law or targets at this point (low gate) because the approach-phase targets have been selected to satisfy the additional constraints. The approach- and landing-phase targets (P64) yield conditions for initiating the automatic vertical descent from an altitude of approximately 150 feet at a 3-fps altitude rate. These conditions, along with the selected acceleration and jerk targets, yield trajectory conditions of 60 fps of forward velocity, 16 fps of vertical descent rate, and an attitude of approximately 16° from the vertical at a 500-foot altitude. These conditions were considered satisfactory by the crew for takeover of manual control. Should the crew continue on automatic guidance, at a TGO of 10 seconds, P65 (the velocity-nulling guidance) is automatically called to maintain the velocities for vertical descent to the lunar surface. Probes that extend 5.6 feet below the LM landing pads, upon making surface contact, activate a light which signals the crew to shut down the DPS manually, whether automatic or manual guidance is being used. The landing-phase trajectory is shown under automatic guidance in figure 10.

Premission estimates of dispersions in landing position are shown in figure 11. These dispersions, which are based on a Monte Carlo analysis, include all known system performances as defined in reference 6. Based on this analysis, the 99-percent-probability landing ellipse was determined to be ± 3.6 nautical miles in-plane by ± 1.3 nautical miles cross range.

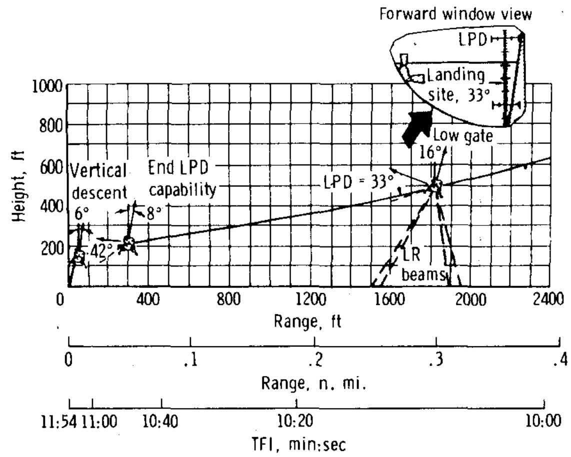


Figure 10. - Landing phase.

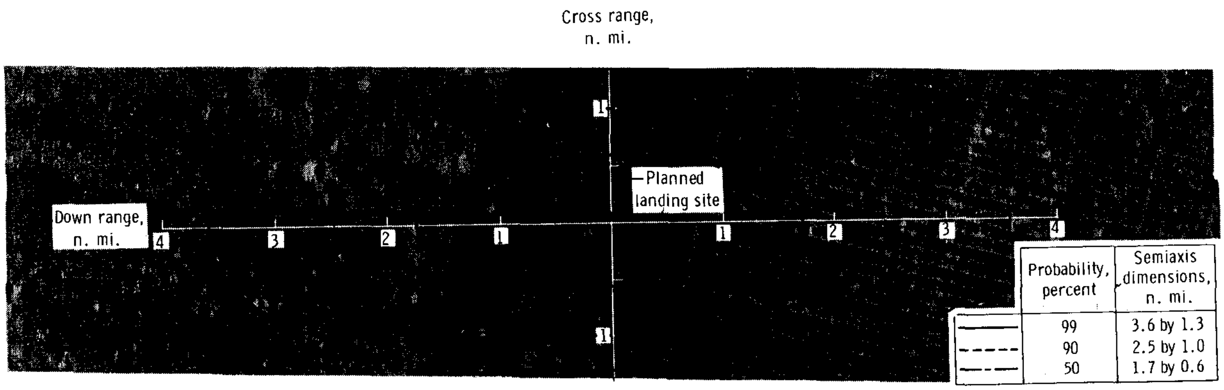


Figure 11. - Predicted Apollo 11 landing dispersions.

The ΔV and propellant requirements. - The ΔV and propellant requirements are determined by the nominal trajectory design, contingency requirements, and dispersions. Consequently, these requirements have undergone continual change. The final operation requirements are given in table II. The required 6827-fps ΔV is established by the automatically guided nominal. In addition, 85 fps is added to assure 2 minutes of flying time in the landing phase, that is, below an altitude of 500 feet. The automatic guidance required only 104 seconds of flying time for the landing phase. Also, a 60-fps ΔV is added for LPD operation in the approach phase to avoid large craters (1000 to 2000 feet in diameter) in the landing area. Contingency propellant allotments are provided for failure of a DPS redundant propellant flow valve and for bias on propellant low-level-light operation. The valve failure causes a shift in the propellant mixture ratio and a lower thrust by approximately 160 pounds, but otherwise, DPS operation is satisfactory. The low-level light signifies approaching propellant depletion; therefore, a bias is used to protect against dispersions in the indicator. If the low-level light should fail, the crew uses the propellant gage reading of 2 percent remaining as the abort decision indicator. The light sensor provides more accuracy and is therefore preferred over the gage reading. The ground flight controllers call out time from low-level light "on" to inform the crew of impending propellant depletion for a land-or-abort decision point at least 20 seconds before depletion. This procedure allows the crew to start arresting the altitude rate with the DPS prior to an abort stage to prevent surface impact. The allowance for dispersions is determined from the Monte Carlo analysis mentioned previously. As can be seen in table II, the ΔV and propellant requirements are satisfied by a positive margin of 301 pounds. This margin can be converted to an additional hover or translation time of 32 seconds.

TABLE II. - APOLLO 11 PREMISSION DESCENT ΔV AND

PROPELLANT REQUIREMENTS

Item	Propellant required, lb	Propellant remaining, lb
System capacity[a]	--	18 260.5
Offloaded[b]	75.4	18 185.1
Unusable	250.5	17 934.6
Available for ΔV	--	17 934.6
Nominal required for ΔV (6827 fps)	16 960.9	973.7
Dispersions (-3σ)	292.0	681.7
Pad	--	681.7
Contingencies		
Engine-valve malfunction	64.7	617.0
Redline low-level sensor	68.7	548.3
Redesignation (60 fps)	102.9	445.4
Manual hover (85 fps)	144.0	301.4
Margin	--	301.4

[a]7051.2 pounds of fuel and 11 209.3 pounds of oxidizer.

[b]Fuel offload of 75.4 pounds to minimize malfunction penalty.

Ascent Planning

A sketch of the LM ascent from the lunar surface is given in figure 12. The ascent has a single objective, namely, to achieve a satisfactory orbit from which rendezvous with the orbiting CSM can subsequently be performed. Nominally, insertion into a 9- by 45-nautical-mile orbit, at a true anomaly of 18° and an altitude of 60 000 feet, is desired. The time of lift-off is chosen to provide the proper phasing for rendezvous. A description of the powered ascent, not the choice of targeting for rendezvous, is the subject of this section.

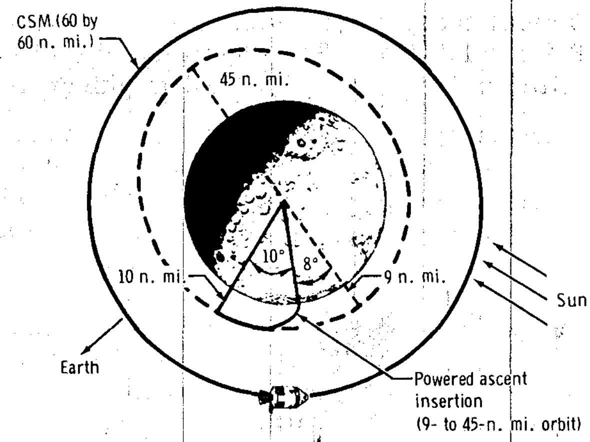


Figure 12. - Premission Apollo 11 LM ascent.

System descriptions. - Only three pertinent systems are required for ascent — the PGNCS and RCS, which have already been described, and the ascent propulsion system (APS). The APS, unlike the DPS, is not throttleable and does not have a trim gimbal drive, but provides a constant thrust of approximately 3500 pounds throughout the ascent (ref. 6). Engine throttling is not required during ascent, because down-range position control is not a target requirement; that is, only altitude, velocity, and orbit plane are required for targeting. This thrust can be enhanced slightly (by approximately 100 pounds) by the RCS attitude control. The ascent DAP logic is such that only body positive X-axis (along the thrust direction) jets are fired for attitude control during ascent.

A fourth system, the abort guidance system (AGS), should also be mentioned. The AGS is a redundant guidance system to be used for guidance, navigation, and control for ascent or aborts in the event of a failure of the PGNCS. The AGS has its own computer and uses body-mounted sensors instead of the inertial sensors as used in the PGNCS. A detailed description of the AGS is given in references 11 and 12.

Operational phases. - The powered ascent is divided into two operational phases: vertical rise and orbit insertion. The vertical-rise phase is required to achieve terrain clearance. The trajectory for propellant optimization takes off along the lunar surface. A description of trajectory parameters and LM attitude during the vertical-rise phase and during the transition to the orbit-insertion phase is shown in figure 13. The guidance switches to the orbit-insertion phase when the radial rate becomes 40 fps. However, because of DAP steering lags, the pitchover does not begin until a radial rate of approximately 50 fps is achieved. This delay means that the vertical-rise phase is terminated 10 seconds after lift-off. Also, during the vertical rise, the LM body Z-axis is rotated to the desired azimuth, which is normally in the CSM orbit plane.

The orbit-insertion phase is designed for efficient propellant usage to achieve orbit conditions for subsequent rendezvous. The orbit-insertion phase, the total ascent-phase performance, insertion orbit parameters, and onboard displays at insertion are shown in figure 14. The onboard-display values reflect the computer-estimated values. If required, yaw steering is used during the orbit-insertion phase to maneuver the LM into the CSM orbit plane or into a plane parallel with the CSM orbit. In the nominal case, no yaw steering is required. The nominal ascent burn time is 7 minutes 18 seconds with a 3σ dispersion of ± 17 seconds. The trajectory dispersions are plotted in figure 15. The ascent guidance logic is discussed in the following section.

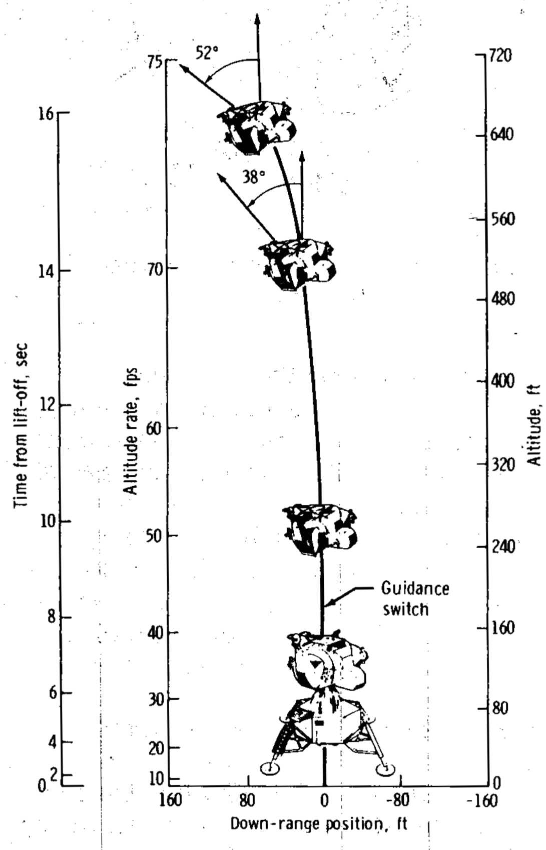


Figure 13. - Premission Apollo 11 vertical-rise phase.

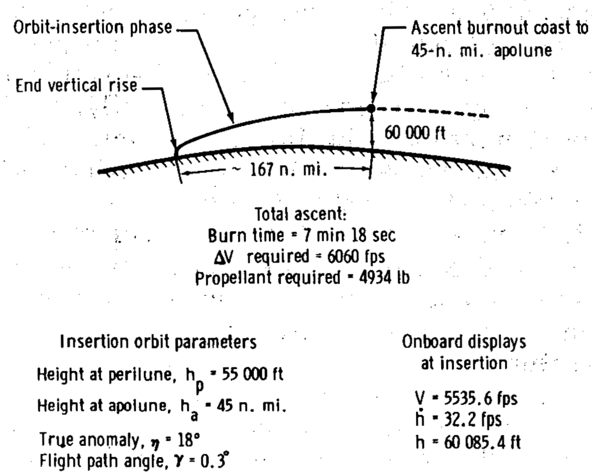


Figure 14. - Premission Apollo 11 orbit-insertion phase.

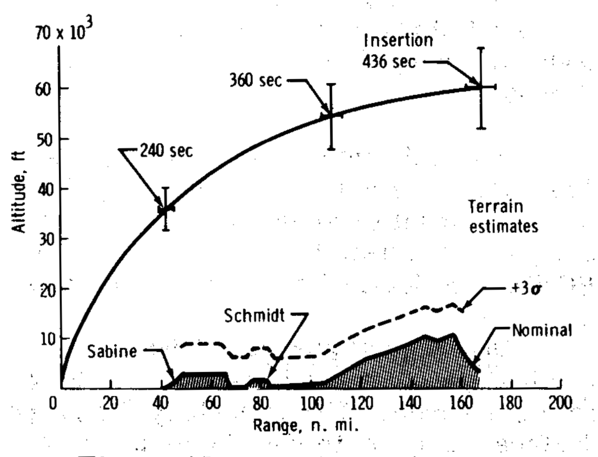


Figure 15. - Predicted Apollo 11 ascent dispersions.

Guidance logic. - The ascent-guidance logic commands only attitude, because no engine throttling is required. For the vertical-rise phase, the logic is simple. The initial attitude is held for 2 seconds in order to clear the LM descent stage, the attitude is pitched to the vertical while rotating to the desired azimuth, and vertical-rise-phase termination occurs when the altitude rate is greater than or equal to 40 fps upward, or when the altitude is greater than 25 000 feet (used for aborts from descent.

The insertion-phase guidance logic is defined by an acceleration command which is a linear function of time and is, therefore, termed linear guidance. The TGO is determined as a function of velocity to be gained; that is, the difference between the current and the desired velocity. This TGO, along with the current state and the desired state, is used to determine acceleration commands in radial and cross-range directions. The acceleration available from the APS is oriented by firing the RCS according to the DAP logic to satisfy these commands, with any remaining acceleration being applied in the down-range direction. Cross-range steering is limited to 0.5°. Out-of-plane maneuvering greater than 0.5° is combined with the subsequent rendezvous sequencing maneuvers. When the TGO becomes less than 4 seconds, a timer is activated to cut off the APS at the desired time.

Three ascent guidance programs are used: P12 for ascent from the surface, P70 for ascent aborts during descent to be performed with the DPS, and P71 for ascent aborts during descent to be performed with the APS. All the programs use the vertical-rise and insertion logic described previously. The programs differ only by the targeting logic used to establish the desired orbit-insertion conditions. For aborts at PDI and through the braking phase, the LM is ahead of the CSM, as a result of the DOI maneuver. During the approach and landing phases, the CSM moves ahead of the LM. Therefore, the desired orbit-insertion conditions targeted by P70 and P71 vary as a function of the phase relationship between the LM and the CSM to establish rendezvous sequencing. Reference 7 contains a complete description of the ascent guidance logic.

The ΔV and propellant requirements. - The ΔV and propellant requirements are determined by the nominal trajectory design, contingency requirements, and dispersions. Consequently, the requirements for ascent, as for descent, have undergone continual change. The final operation requirements are given in table III. The

TABLE III. - APOLLO 11 PREMISSION ASCENT ΔV AND PROPELLANT REQUIREMENTS

Item	Propellant required, lb	Propellant remaining, lb
System capacity[a]	--	5244.4
Offloaded[b]	20.7	5223.7
Unusable	56.3	5167.4
Available for ΔV	--	5167.4
Nominal required for ΔV (6055.7 fps)	4966.7	200.7
Dispersions (-3σ)	66.7	134.0
Pad	--	134.0
Contingencies		
Engine-valve malfunction	18.8	115.2
PGNCS to AGS switchover (40 fps)	23.8	91.4
Abort from touchdown (ΔW = +112.9 lb, $\Delta(\Delta V)$ = -14.3 fps)	43.2	48.2
Margin	--	48.2

[a]2026.0 pounds of fuel and 3218.4 pounds of oxidizer.
[b]Fuel offload of 20.7 pounds to minimize malfunction penalty.

required 6056-fps ΔV is established by the nominal insertion into a 9- by 45-nautical-mile orbit. In addition, a 54-fps ΔV is provided for two contingencies. A 40-fps ΔV is provided for the first contingency, which is a switchover from PGNCS to AGS for inserting from an off-nominal trajectory caused by a malfunctioning PGNCS. A 14-fps ΔV is provided for the second contingency, in which the thrust-to-weight ratio is reduced in an abort from a touchdown situation wherein the LM ascent stage is heavier than the nominal ascent-stage lift-off weight. Some weight is nominally off-loaded on the lunar surface. Also, 19 pounds of propellant is allotted for contingency engine-valve malfunction, as in the descent requirements. The allowance for dispersions is determined from the Monte Carlo analysis. As can be seen in table II, the ΔV and propellant requirements are satisfied with a positive margin of 48 pounds.

REAL-TIME ANALYSIS

During the real-time situation, monitoring of the spacecraft systems and of the trajectory is performed continually both on board by the crew and on the ground by the flight controllers. The real-time monitoring determines whether the mission is to be continued or aborted, as established by mission techniques prior to flight. The real-time situation for the Apollo 11 descent and ascent is described in the following section.

Descent Orbit Insertion

The DOI maneuver is performed on the farside of the moon at a position in the orbit 180° prior to the PDI and is, therefore, executed and monitored solely by the crew. Of major concern during the burn is the performance of the PGNCS and the DPS. The DOI maneuver is essentially a retrograde burn to reduce orbit altitude from approximately 60 nautical miles to 50 000 feet for the PDI and requires a velocity reduction of 75 fps. This reduction is accomplished by throttling the DPS to 10-percent thrust for 15 seconds (center-of-gravity trimming) and to 40-percent thrust for 13 seconds. An overburn of 12 fps (or 3 seconds) would cause the LM to be on an impacting trajectory prior to PDI. Thus, the DOI is monitored by the crew with the 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 with the AGS. For Apollo 11, this maneuver was nominal. Down-range residuals after the burn were 0.4 fps.

Powered Descent

The powered descent is a complex maneuver which is demanding on both crew and system performances. Therefore, as much monitoring as possible is performed on the ground to reduce crew activities and to use sophisticated computing techniques not possible on board. Obviously, however, time-critical failures and near-surface operations must be monitored on board by the crew for immediate action. Pertinent aspects of guidance, propulsion, and real-time monitoring of flight dynamics during the powered descent are given as follows.

The PGNCS monitoring. - To determine degraded performance of the PGNCS, the ground flight controllers continually compare the LM velocity components computed by the PGNCS with those computed by the AGS and with those determined on the ground through Manned Space Flight Network (MSFN) tracking. That is, a two-out-of-three voting comparison logic is used to determine whether the PGNCS or the AGS is degrading. Limit or redlines for velocity residuals between the PGNCS and the MSFN computations and between the PGNCS and the AGS computations are established before the mission, based on the ability to abort on the PGNCS to a safe (30 000-foot perilune) orbit.

In real time, the Apollo 11 PGNCS and AGS performance was close to nominal; however, a large velocity difference in the radial direction of 18 fps (limit line at 35 fps) was detected at PDI and remained constant well into the burn. This error did not indicate a systems performance problem, but rather an initialization error in down-range position. This effect is illustrated geometrically in figure 16. The PGNCS position $\vec{R}_E$ and velocity $\vec{V}_E$ estimates are used to initiate the MSFN powered-flight processor. The MSFN directly senses the actual velocity $\vec{V}_A$ at the actual position $\vec{R}_A$, but, having been initialized by the PGNCS state, the MSFN applies $\vec{V}_A$ at $\vec{R}_E$. Thus, a flight-path-angle error Δ_γ is introduced by a down-range position error and shows up as a radial velocity difference $\Delta\vec{V}_{DIFF}$. The magnitude of the velocity difference indicates that the Apollo 11 LM down-range position was in error by approximately 3 nautical miles at PDI and throughout the powered descent to landing. The reason for the down-range navigation error was attributed to several small ΔV inputs to the spacecraft state in coasting flight. These inputs were from uncoupled RCS attitude maneuvers and cooling system venting not accounted for in the prediction of the navigated state at PDI.

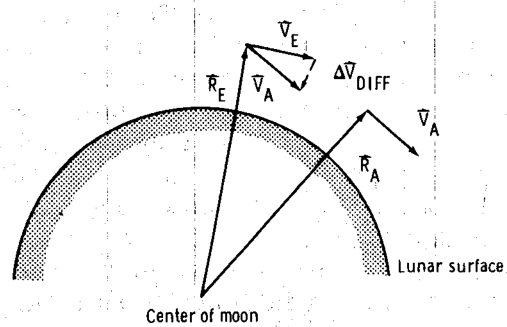


Figure 16. - Effect of position error on velocity comparison.

The LM guidance computer (LGC) also monitors the speed at which it is performing computation tasks: navigation, guidance, displays, radar data processing, and auxiliary tasks. If the computer becomes overloaded or falls behind in accomplishing these tasks, an alarm is issued to inform the crew and the flight controllers, and priorities are established so that the more important tasks are accomplished first. This alarm system is termed "computer restart protection." During real time, because of an improperly defined interface, a continuous signal was issued to the LGC from the RR through coupling data units. These signals caused the LGC to count pulses continually in an attempt to slew the RR until a computation time interval was exceeded. As a result, the alarm was displayed and computation priorities were executed by the computer. The alarm was quickly interpreted, and flight-control monitoring indicated that guidance and navigation functions were being performed properly; thus, the descent was continued. In spite of the initial position error and the RR inputs, the PGNCS performed excellently during the Apollo 11 powered descent.

The DPS and PGNCS interface. - To determine in real time if the DPS is providing sufficient thrust to achieve the guidance targets, the flight controllers monitor a plot of guidance thrust command (GTC) as a function of horizontal velocity, as shown in figure 17. Nominally, the GTC decreases almost parabolically from an initial value near 160 percent of design thrust to the throttleable level of 57 percent, approximately 2 minutes (horizontal velocity being 1400 fps) before high gate (horizontal velocity being 500 fps). If the DPS produces off-nominal high thrust, horizontal velocity is being reduced more rapidly than desired to reach high-gate conditions. Therefore, the GTC drops to 57 percent earlier with a higher-than-nominal velocity to guide to the desired position and velocity targets. This early throttledown results in propellant inefficiency. If the DPS produces off-nominal low thrust, horizontal velocity is not being reduced rapidly enough. Therefore, the GTC drops to 57 percent later at a lower velocity to guide to the desired position and velocity. This later throttledown results in increased propellant efficiency (i. e., longer operation at maximum thrust). However, if no throttledown occurs prior to high gate (program switch from P63 to P64), the targets will not be satisfied, and the resulting trajectory may not be satisfactory from the standpoint of visibility. In fact, for extremely low thrust, the guidance solution for the GTC can diverge (fig. 17); as TGO becomes small, the guidance calls for more and more thrust in order to achieve its targets. This divergence can result in an unsafe trajectory, one from which an abort cannot be satisfactorily performed. The 2-minute bias for throttle recovery before high gate provides sufficient margin for 3σ low thrust even with propellant valve malfunction. However, the flight controllers monitor the GTC to assure satisfactory interface between DPS and PGNCS operation. A mission rule was established that called for an abort based on the GTC divergence. During the Apollo 11 landing, the DPS thrust was nearly nominal (fig. 17); thus, no DPS and PGNCS interface problems were encountered.

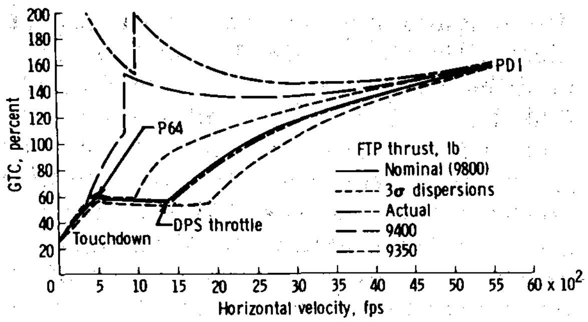


Figure 17. - Guidance thrust command as a function of horizontal velocity.

The LR and PGNCS interface. - Normally, the LR update of the PGNCS altitude estimate is expected to occur by crew input at an altitude of 39 000 ± 5000 feet (3σ dispersion). Without LR altitude updating, system and navigation errors are such that the descent cannot be safely completed. In fact, it is unsafe to try to achieve high gate where the crew can visually assess the approach without altitude updating. Thus, a mission rule for real-time operation was established that called for aborting the descent at a PGNCS-estimated altitude of 10 000 feet, if altitude updating had not been established.

In addition to the concern for the time that initial altitude updating occurs is the concern for the amount of altitude updating (i. e., the difference between PGNCS and LR altitude determinations Δh). If the LM is actually higher than the PGNCS estimate, the