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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.

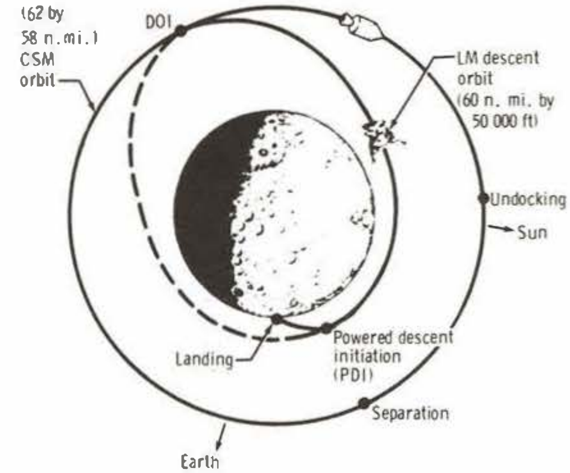
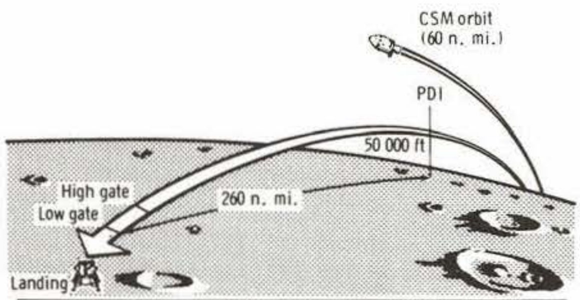


Figure 1. - Lunar module descent.

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

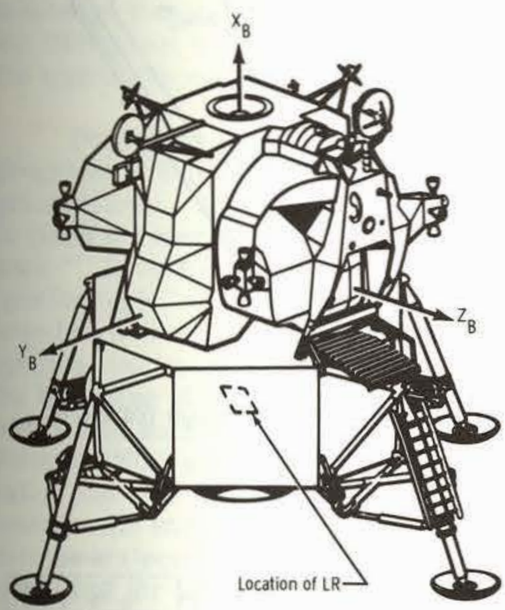
Figure 2. - Operational phases of powered descent.

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.

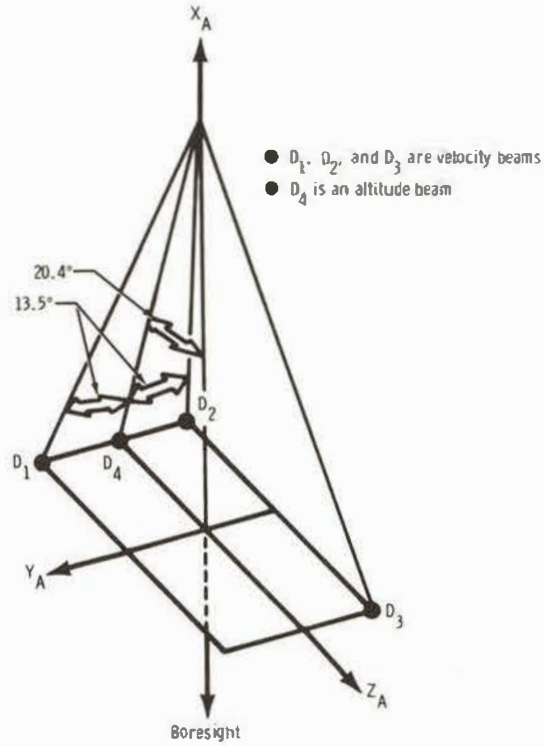
The PGNCS consists of two major subsystems: an inertial measurement unit (IMU) and a computer. The IMU is the navigation sensor, which incorporates accelerometers and gyros to sense changes in LM velocity and attitude. The IMU sends this information to the computer, which contains preprogramed logic 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.

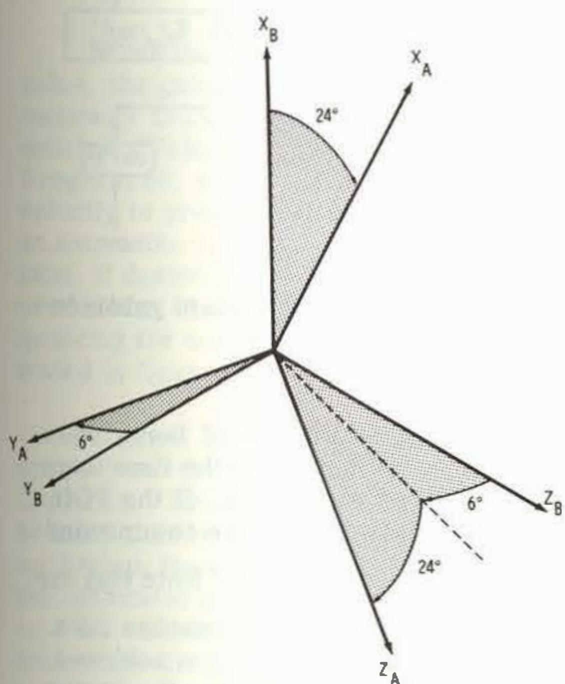
The LR, mounted at the bottom rear of the LM, is the navigation sensor which provides ranging and velocity information relative to the lunar surface. The LR consists of four radar beams, one beam to provide ranging measurements and three beams to provide velocity measurements. This beam pattern, which 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).



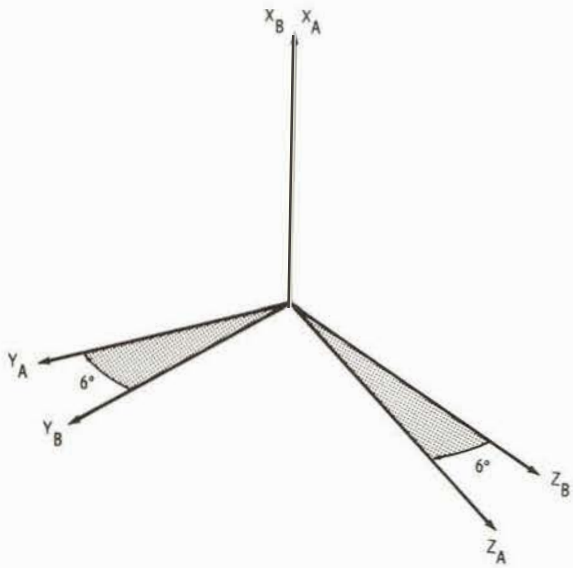
(a) Lunar module body axes.



(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.

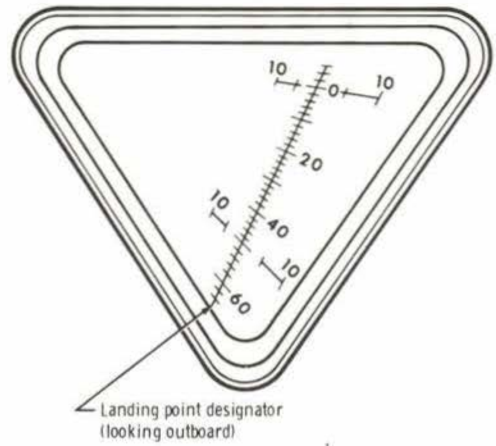


Figure 4. - Lunar module forward window.

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

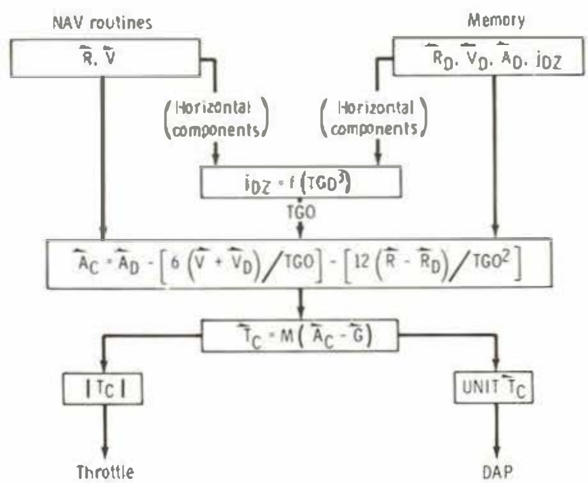


Figure 5.- Basic LM descent guidance logic.

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.

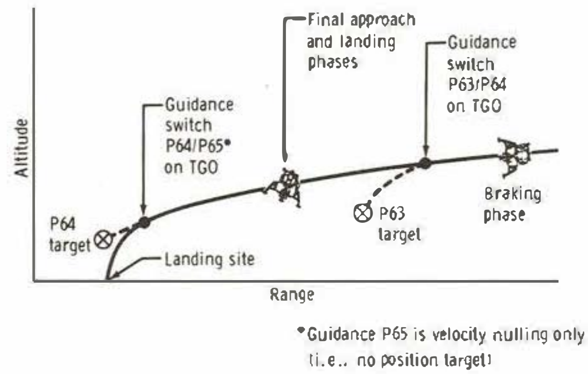


Figure 6.- Target sequence for automatic-descent guidance.

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.

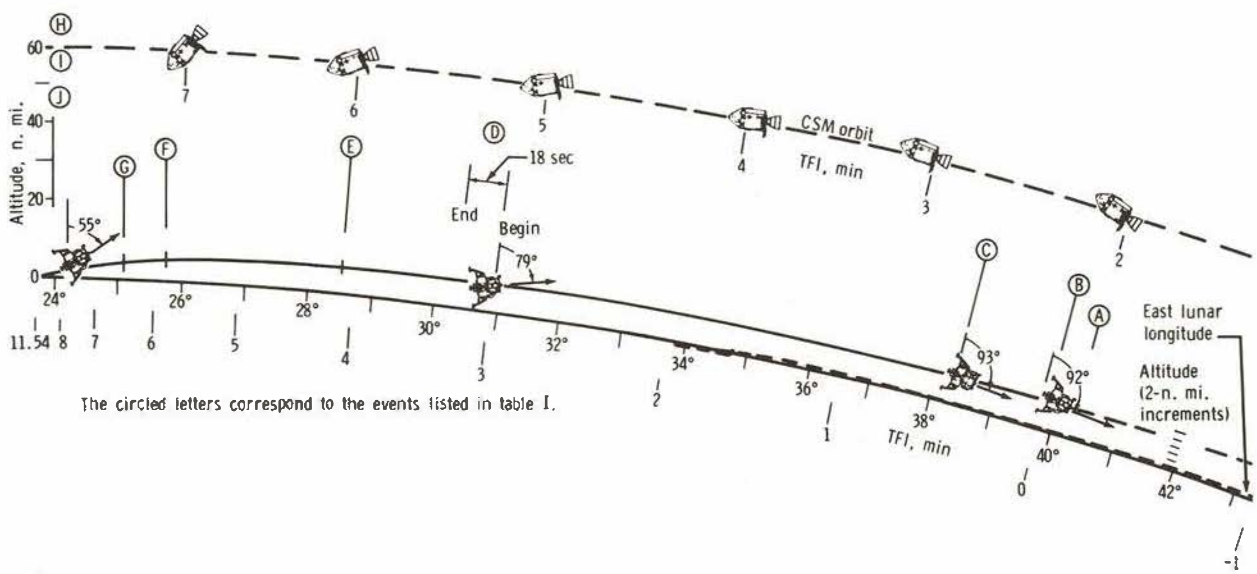


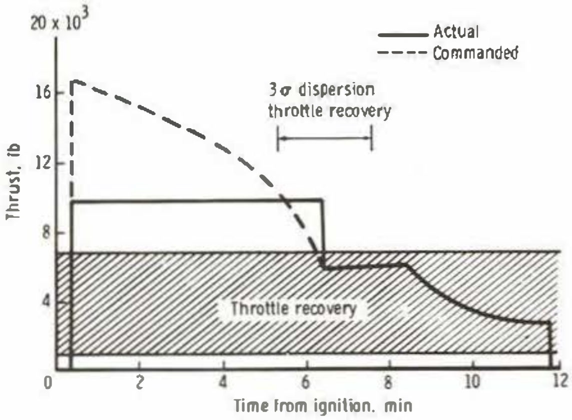
Figure 7. - Premission Apollo 11 LM powered descent.

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

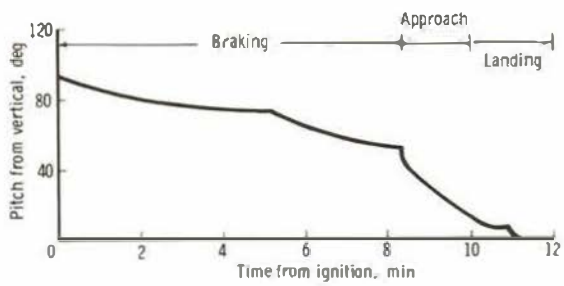
Event	TFI, min: sec (a)	Inertial velocity, fps	Altitude rate, fps	Altitude, ft	ΔV, fps
Ullage	-0: 07				
Powered descent initiation	0: 00	5560	-4	48 814	0
Throttle to maximum thrust	0: 26	5529	-3	48 725	31
Rotate to windows-up position	2: 56	4000	-50	44 934	1572
LR altitude update	4: 18	3065	-89	39 201	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.

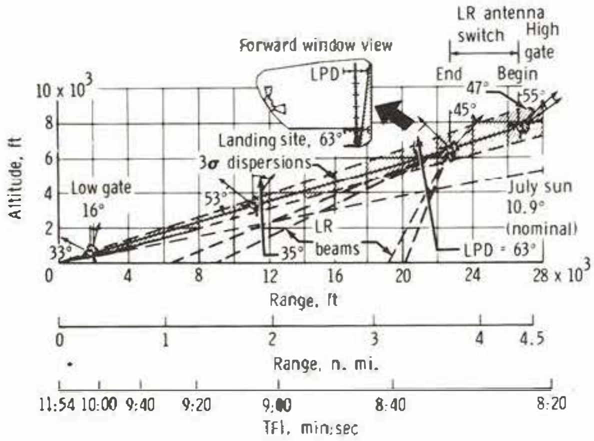


Figure 9. - Approach phase.

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.

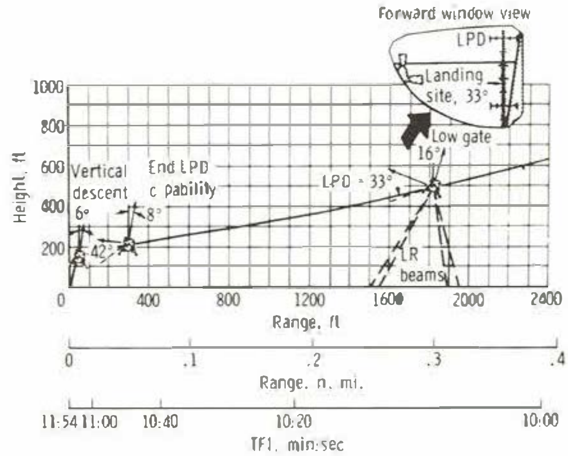


Figure 10. - Landing phase.

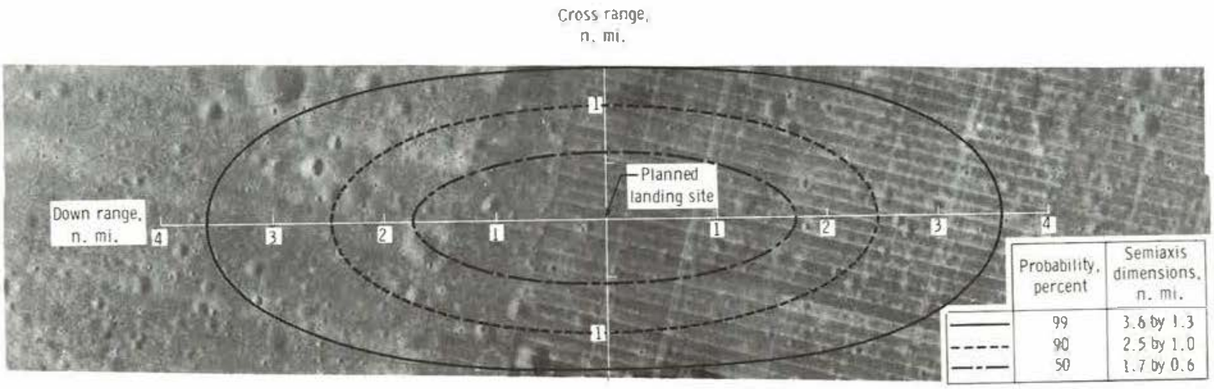


Figure 11. - Predicted Apollo 11 landing dispersions.

<u>The ΔV and propellant requirements.</u> - 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.

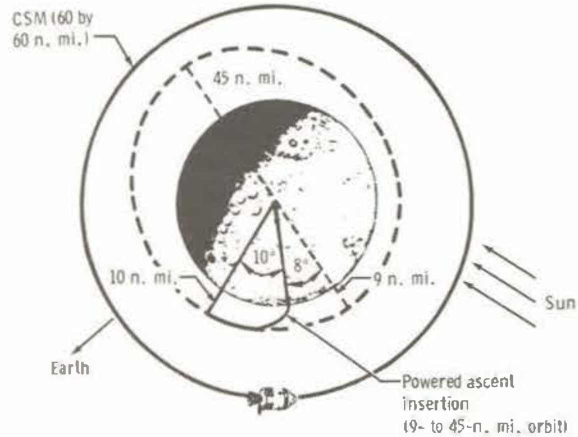


Figure 12. - Premission Apollo 11 LM ascent.

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 downrange 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.

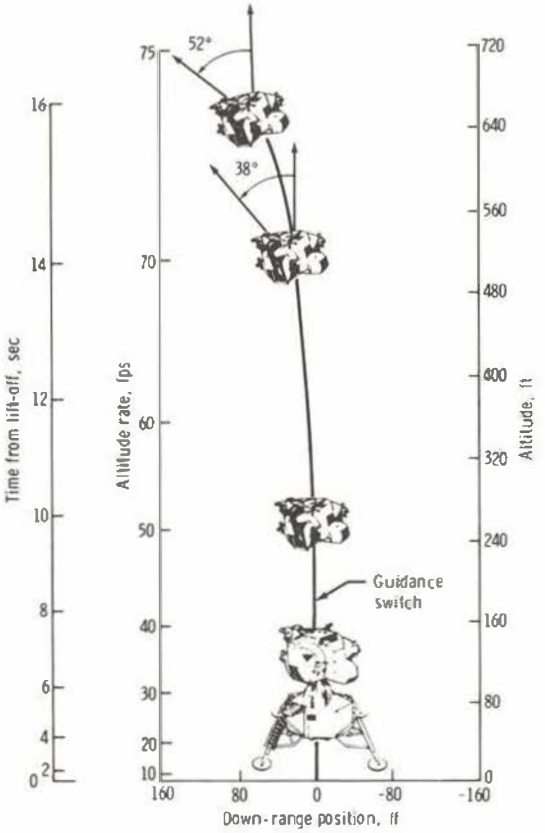


Figure 13. - Premission Apollo 11 vertical-rise phase.

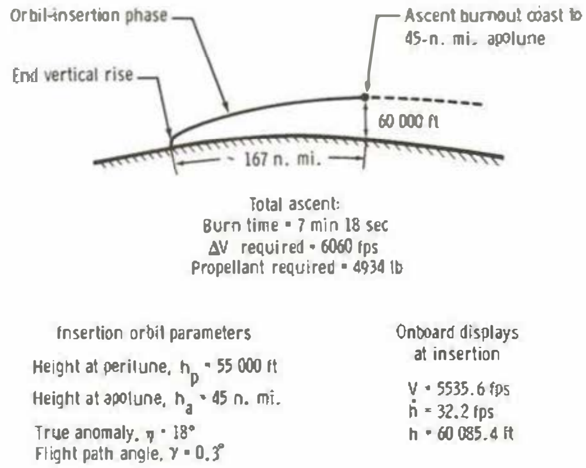


Figure 14. - Premission Apollo 11 orbit-insertion phase.

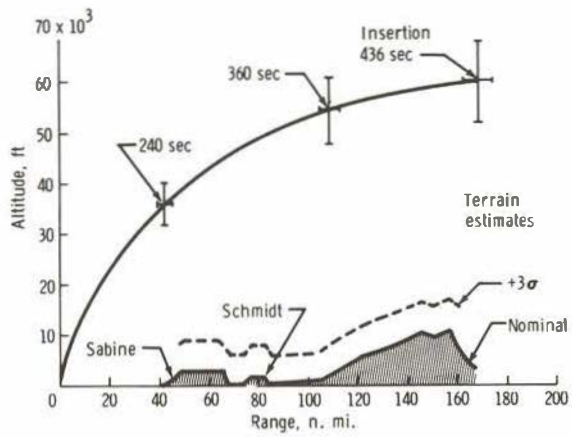


Figure 15. - Predicted Apollo 11 ascent dispersions.

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.

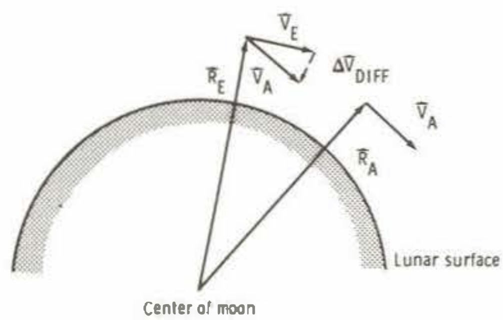


Figure 16. - Effect of position error on velocity comparison.

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.

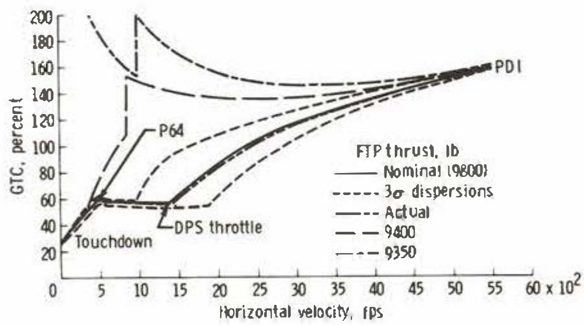


Figure 17. - Guidance thrust command as a function of horizontal velocity.

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

LR will determine the discrepancy and update the PGNCS. 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 Δh limits necessary to avoid these rates are shown in figure 18. Notice that over the estimated 3σ region of LR initial updating (which at the time of that analysis was centered at an altitude of only 35 600 feet instead of 39 000 feet), the Δh limits are much greater than the $+3\sigma$ navigation estimates of Δh. However, the flight controllers, as well as the crew, monitor Δh to assure that the boundary is not exceeded before incorporation of the LR altitude updating. If the boundary is exceeded, then the data are not incorporated, and an abort is called. When the LM is actually lower than estimated, no excessive rates are encountered upon LR updating. It is necessary only that the LM altitude and altitude rate be above the abort limits, defined in the section entitled "Trajectory Limits."

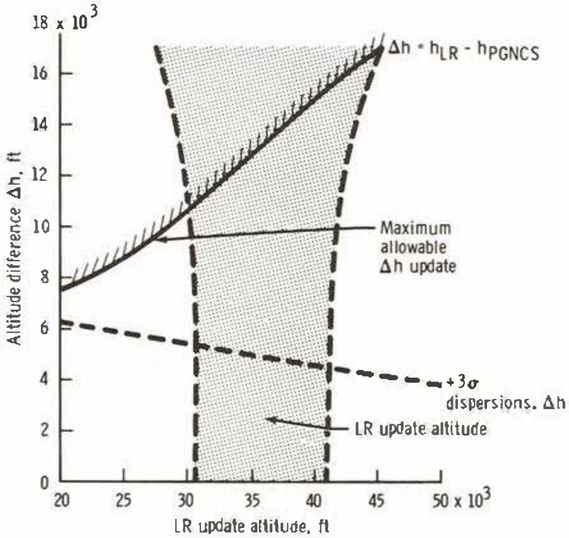


Figure 18. - Landing radar altitude updates.

During the Apollo 11 mission, the LR acquired lock-on to the lunar surface during the rotation to face-up attitude at an altitude of 37 000 feet. The Δh was -2200 feet, indicating that the LM was actually low. This small amount of Δh can readily be attributed to terrain variations. Because no limits were violated, the data were incorporated after a short period of monitoring at an altitude of 31 600 feet. The Δh readily converged to a small value of 100 feet within 30 seconds. The LR velocity updates were incorporated nominally, beginning at a 29 000-foot altitude. As expected, LR signal dropouts were encountered at low altitudes (below 500 feet) but presented no problem. When the velocity becomes small along the LR beams, depending on the attitude and approach velocity, zero Doppler shift is encountered; hence, no signal occurs.

Trajectory limits. - During real time, trajectory limits are monitored for flight safety. 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, as shown in figure 19. Notice that the nominal trajectory design does not approach the limits until late in the descent, after the crew has had ample time for visual assessment of the situation. The limits shown are based on APS abort with a 4-second free fall for crew action delay or on DPS abort with a 20-second communications delay for ground notification. The flight controllers and the crew monitor altitude and altitude rate, but because of communication delays with the ground, the flight controllers only advise, based on projected trends. The Apollo 11 altitude and altitude-rate profile shown in figure 19 was near nominal.

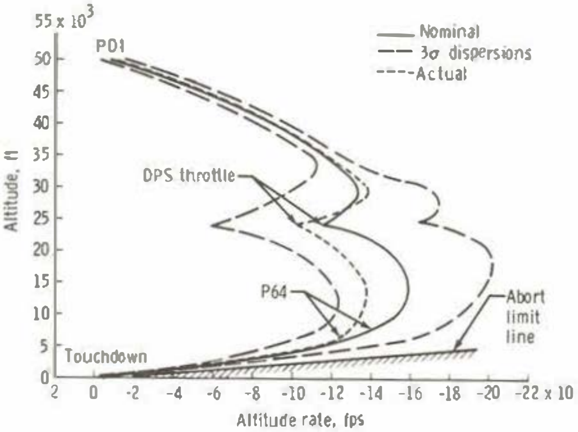


Figure 19. - Altitude as a function of altitude rate during powered descent.

Crew visual assessment. - As stated previously, the approach and landing phases have been designed to provide crew visibility of the landing area. This provision allows the crew to assess the acceptability of the landing area, to decide to continue toward the landing area, or to redesignate a landing away from it with LPD or manual control. During the Apollo 11 mission, because of the initial navigation errors, the descent was guided into the generally rough area surrounding West Crater (fig. 20 and the section entitled "The PGNCS Monitoring"). West Crater is inside the premission mapped area, approximately 3 nautical miles west of center. Unfortunately, because of the guidance program alarms, the commander was unable to concentrate on the window view until late in the descent (near low gate). Thus, crew visual assessment during the approach phase was minimal, which resulted in continued approach into the West Crater area. This problem is discussed further in the subsequent section entitled "Postflight Analysis."

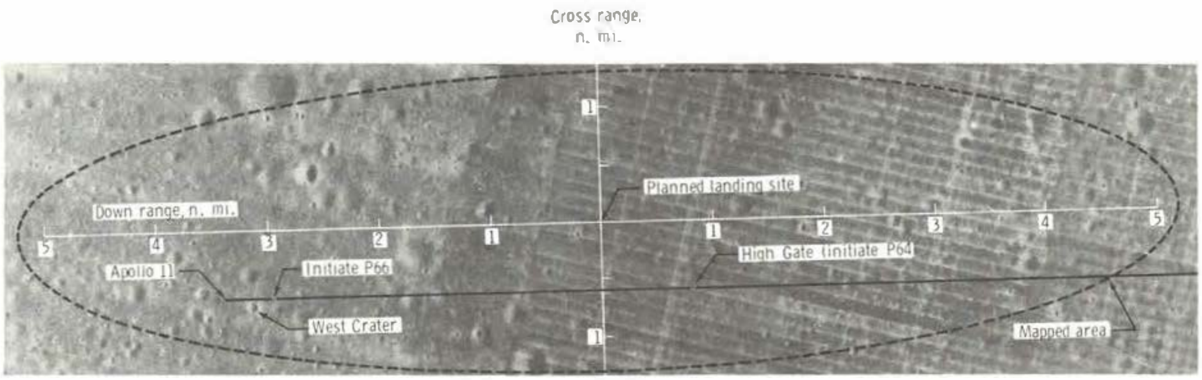


Figure 20. - Apollo 11 landing site.

Ascent

During the real-time situation, the crew and flight controllers continually monitor the LM systems and trajectory for detection of off-nominal performance. Of primary concern is the performance of the APS and the PGNCS. The APS must perform because no backup propulsion system is provided. Should the APS fail during the final 30 seconds of ascent, the RCS can complete the insertion. The PGNCS performance is monitored by the AGS and powered-flight processor, using MSFN tracking in the same manner as in the descent-guidance monitoring. The limit lines are set for completion of the ascent on the AGS should the PGNCS performance degrade.

In real time, the PGNCS and AGS performance was excellent, and guidance switchover was not required. The APS performance was also excellent. Insertion occurred at 7 minutes 15 seconds from lift-off, with 7 minutes 18 seconds being the operational trajectory prediction.

POSTFLIGHT ANALYSIS

A postflight analysis is conducted to determine how the actual flight performance compared with the premission planning. The purpose of a postflight analysis is to determine if the premission planning was adequate and, if it is not, to determine the changes required for subsequent flights. A brief description of the Apollo 11 postflight results for LM descent and ascent, application of these results to the Apollo 12 planning, and a preliminary postflight analysis of the Apollo 12 mission are given.

Apollo 11 Descent

The DOI maneuver was performed nominally, as discussed in the preceding section. The events during powered descent are tabulated in table IV. The braking phase events were near nominal (table I). Rotation to a windows-up attitude was delayed slightly because of the selection of a slow rotational rate by the crew. This delay resulted in the slight delay in LR acquisition, which took place prior to completion of the rotation. The approach phase, as shown in figure 21, also was consistent with premission planning. As shown previously (fig. 20), the descent headed into the area near West Crater because of an initial navigation error, approximately 3 nautical miles down range. During the approach phase, the LPD indicated to the commander that the automatic system was guiding to a landing up range of West Crater. Later on, the landing appeared to be heading into the rock field just beyond West Crater. This uncertainty was caused by several factors: the time rate of change in LPD angle, errors introduced by terrain variations (primarily slope), and the lack of time for visual assessment because of crew diversion to guidance-program alarms. (Refer to the section entitled "Real-Time Analysis.") Therefore, not until the beginning of the landing phase did the commander try to avoid the large area of rough terrain by assuming manual control (P66 guidance) at an altitude of 410 feet when the forward velocity was only 50 fps. An LPD input was made, as shown in table IV; but in discussions with the crew, it was determined that this input was inadvertent. The landing phase is illustrated in figure 22, and the ground track is shown in figure 23. The landing site is shown to have been moved, through manual maneuvering, approximately 1100 feet down rante and 400 feet cross range from where the automatic guided descent (under P64 and P66 control) would have landed. The attitude and altitude-rate profile are shown in figures 24 and 25, respectively. The somewhat erratic behavior of these profiles can be best explained by Commander Neil A. Armstrong's comments to the Society of Experimental Test Pilots meeting in Los Angeles on September 26, 1969. "I [was] just absolutely adamant about my God-given right to be wishy-washy about where I was going to land."

TABLE IV. - APOLLO 11 LUNAR-DESCENT EVENT TIMES

g. e. t.,[a] hr: min: sec	Event
102: 17: 17	Acquisition of data
102: 20: 53	LR on
102: 24: 40	Alinement of abort guidance to primary guidance
102: 27: 32	Yaw maneuver to obtain improved communications
102: 32: 55	Altitude of 50 000 feet
102: 32: 58	Propellant-setting firing start
102: 33: 05	Descent-engine ignition
102: 33: 31	Fixed throttle position (crew report)
102: 36: 57	Face-up yaw maneuver in process
102: 37: 51	LR data good
102: 37: 59	Face-up maneuver complete
102: 38: 22	1020 alarm (computer determined)
102: 38: 45	Enabling of radar updates
102: 38: 50	Altitude less than 30 000 ft (inhibit X-axis override)
102: 38: 50	Velocity less than 2000 fps (start LR velocity update)
102: 39: 02	1202 alarm
102: 39: 31	Throttle recovery
102: 41: 32	Program 64 entered
102: 41: 37	LR antenna to position 2
102: 41: 53	Attitude hold (handling qualities check)
102: 42: 03	Automatic guidance

[a]Ground elapsed time.

TABLE IV.- APOLLO 11 LUNAR-DESCENT EVENT TIMES - Concluded

g.e.t.,[a] hr:min:sec	Event
102:42:18	1201 alarm (computer determined)
102:42:19	LR low scale (less than 2500 ft)
102:42:43	1202 alarm (computer determined)
102:42:58	1202 alarm (computer determined)
102:43:09	Landing point redesignation
102:43:13	Attitude hold
102:43:20	Update of abort guidance altitude
102:43:22	Program 66 entered
102:44:11	LR data not good
102:44:21	LR data good
102:44:28	Propellant low-level sensor light on
102:44:59	LR data not good
102:45:03	LR data good
102:45:40	Landing
102:45:40	Engine off

[a]Ground elapsed time.

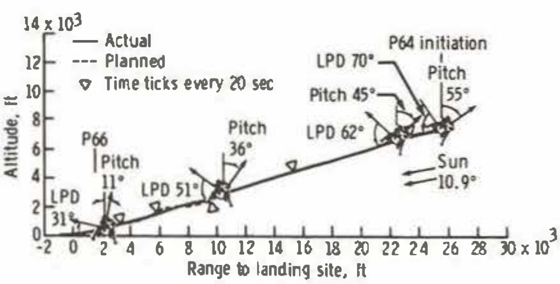


Figure 21. - Apollo 11 approach phase.

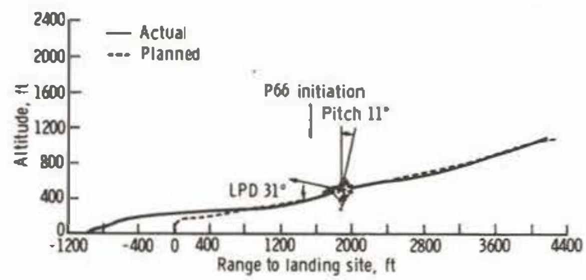


Figure 22. - Apollo 11 landing phase.

The propellant situation during the landing phase is summarized in figure 26. The actuals shown are based on low-level sensor indications. Touchdown is shown to have occurred 40 to 50 seconds prior to propellant depletion, only 20 to 30 seconds from the land-or-abort decision point and approximately 52 to 62 seconds longer than predicted for an automatic landing. The flying time below 500 feet was approximately 2 minutes 28 seconds.

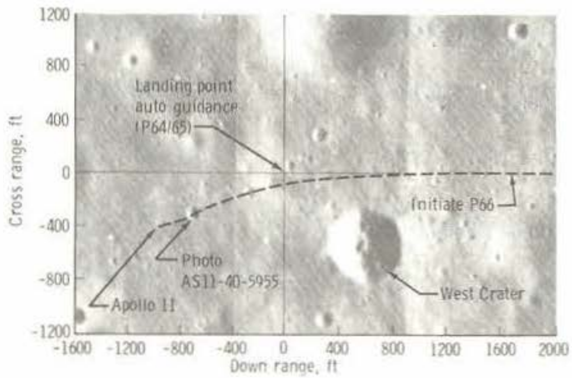


Figure 23. - Apollo 11 groundtrack for the landing phase.

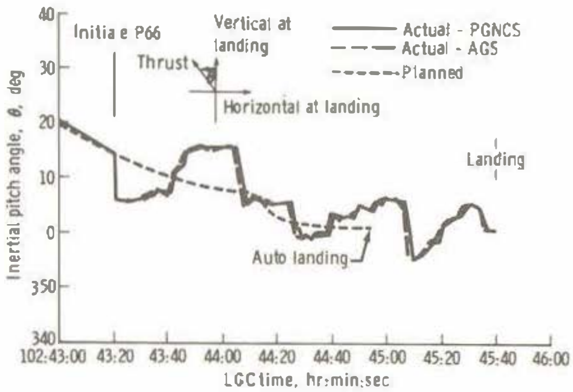


Figure 24. - Apollo 11 attitude profile for the landing phase.

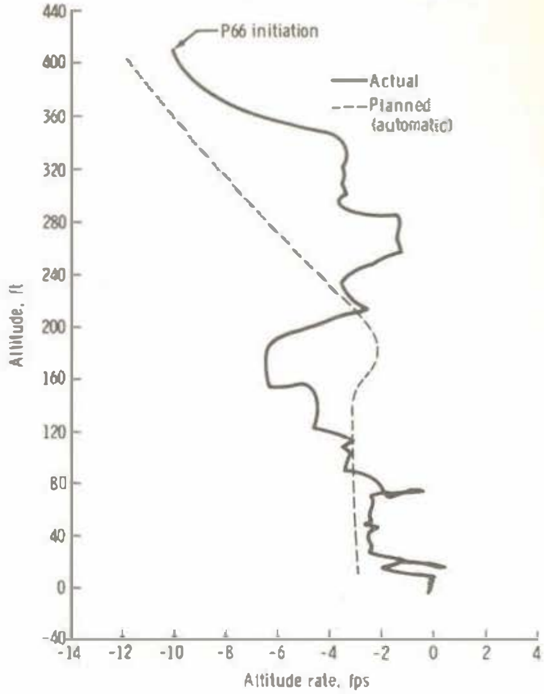


Figure 25. - Apollo 11 altitude as a function of altitude rate for the landing phase.

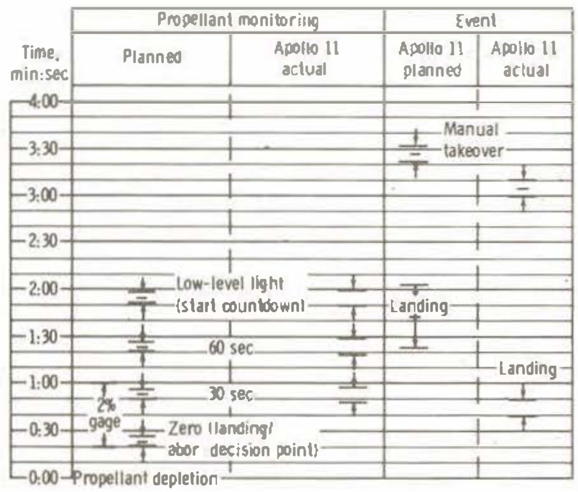


Figure 26. - Apollo 11 landing-phase events.

The Apollo 11 landing was an unqualified success. The descent was nominal until the beginning of the landing phase (an altitude of approximately 410 feet), when the commander was required to avoid a large area of rough terrain with manual control. The size of the area was such that the crew should have been able to detect and avoid it efficiently during the approach phase, if sufficient attention could have been devoted to visual assessment. Adequate visual assessment was not possible during the Apollo 11 descent because of the guidance-program alarms. The problem causing these alarms has been corrected.

Apollo 11 Ascent

A summary of ascent is given in table V and is compared with premission estimates. This comparison indicates that no anomalies occurred during the ascent burn and that the insertion objectives were closely satisfied. The 3-second difference in

TABLE V. - APOLLO 11 ASCENT SUMMARY

(a) Events

Event	TFI, min:sec	
	Premission	Actual
End of vertical rise	0:10	0:10
Insertion	7:18	7:15
Beginning of velocity residual trim	--	7:33
Residual trim complete	--	8:37

(b) Insertion conditions

Measurement type	Altitude, ft	Radial velocity, fps	Down-range velocity, fps
Premission	60 085	32	5535.6
PGNCS (real time)	60 602	33	5537.0
AGS (real time)	60 019	30	5537.9
MSFN (real time)	61 249	35	5540.7
Postflight	60 300	32	5537.0

TABLE V.- APOLLO 11 ASCENT SUMMARY - Concluded

(c) Parameters

Ascent targets	
Radial velocity, fps	32.2
Down-range velocity, fps	5534.9
Cross range to be steered out, n. mi.	1.7
Insertion altitude, ft	60 000
PGNCS velocity residuals (LM body coordinates)	
V_{gx}, fps	-2.1
V_{gy}, fps	-0.1
V_{gz}, fps	1.8
Resulting orbit after residual trim	
Apolune altitude, n. mi.	47.3
Perilune altitude, n. mi.	9.5

burn time is attributed to a slightly higher actual thrust-to-weight ratio than predicted. No means are available to determine whether the difference resulted from high thrust or less weight. Usable APS propellant at cut-off was estimated to be approximately 250 pounds.

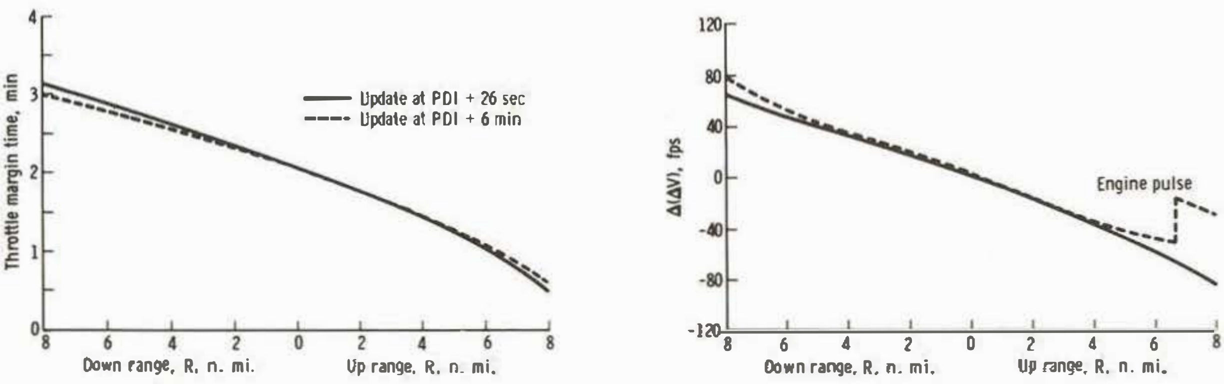
APOLLO 12 MISSION

Apollo 12 Planning

The Apollo 12 mission had the same major mission objective as the Apollo 11 mission; namely, to land men on the moon and return them safely to earth. In addition, a secondary objective for the Apollo 12 flight was to demonstrate pinpoint landing capability, which is required for future scientific missions, by landing within a 1-kilometer (0.54 nautical mile) radius of the target, near the Surveyor III spacecraft located at Apollo site 7 (latitude 3.0° S, longitude 23.4° W). Basically, the planning philosophy for the Apollo 12 descent and ascent remained the same as the philosophy for the Apollo 11 mission. However, because the Apollo 11 LM landed approximately 3 nautical miles off target and consumed more propellant for terrain avoidance than anticipated, several minor changes were considered for the Apollo 12 descent. These changes were concerned with alleviating ΔV and propellant requirements and with more efficiently correcting position errors during the descent.

Two methods for alleviating propellant requirements were proposed. The first method was to perform DOI with the CSM before undocking the LM, perhaps even combining DOI with the lunar orbit insertion maneuver. By using this method, the LM ΔV and propellant requirements can be reduced by 75 fps and 190 pounds of propellant, which increases hover or translation time available in the landing phase by 20 seconds. The planning time for analysis and the crew-activity time line did not permit incorporation of this method for the Apollo 12 mission. However, the method was determined to be feasible and was planned for use on the Apollo 13 and subsequent missions. The second method was to modulate the DPS thrust 10 to 12 times between FTP (maximum) and 57 percent (upper throttle region) to correct thrust dispersions. In using this method, the 2-minute throttle recovery region prior to high gate could be eliminated, resulting in about the same propellant savings as with the first method. This modulation required a change in the basic guidance logic, considerable system dispersion analysis, and DPS testing over this duty cycle before incorporating the logic. The second method also could not be incorporated in the Apollo 12 planning, but is being considered for future missions. Thus, the Apollo 12 ΔV and propellant requirements for descent remained the same as the Apollo 11 ΔV and propellant requirements.

Two methods for providing more efficiency in position correction during descent were proposed. The first method was to take advantage of the detection of down-range position error by the powered-flight processor during the braking phase. (See the section entitled "The PGNCS Monitoring.") Analysis showed that large updates in down-range or up-range target position could be made for small changes in ΔV and throttle recovery time (fig. 27). In addition, dispersion analysis using this update indicated that down-range dispersions would be reduced to approximately 11.3 nautical miles, as shown in figure 28. A minor change to the guidance logic to allow the crew to enter manually (through the DSKY) updates to the landing-site coordinates sent from the ground was required. The guidance change was made, and this proposed technique was approved for use on the Apollo 12 mission. The second method proposed was to change the guidance targeting for the approach and landing phases (P64 guidance) to enhance redesignation (LPD) and manual maneuvering capabilities. Use of these capabilities would be required to reduce the 3σ dispersions shown in figure 28 to a 1-kilometer radius for pinpoint landing. The results of a limited study for varying horizontal and vertical velocities at low gate (500 feet) with vertical descent targeted to a 100-foot altitude are shown in figure 29. It was determined that by increasing forward velocity at 500 feet from 60 to 80 fps, significant gains in redesignation capability (fig. 30) were achieved while altitude rate was maintained at 16 fps. In addition, this trajectory resulted in a slowly changing or more constant LPD time history during approach, as shown in figure 31. Therefore, this proposal was also accepted for the Apollo 12 operational-trajectory planning.



(a) Throttle margin time.

(b) Change in characteristic velocity.

Figure 27. - Landing site update capability during braking phase.

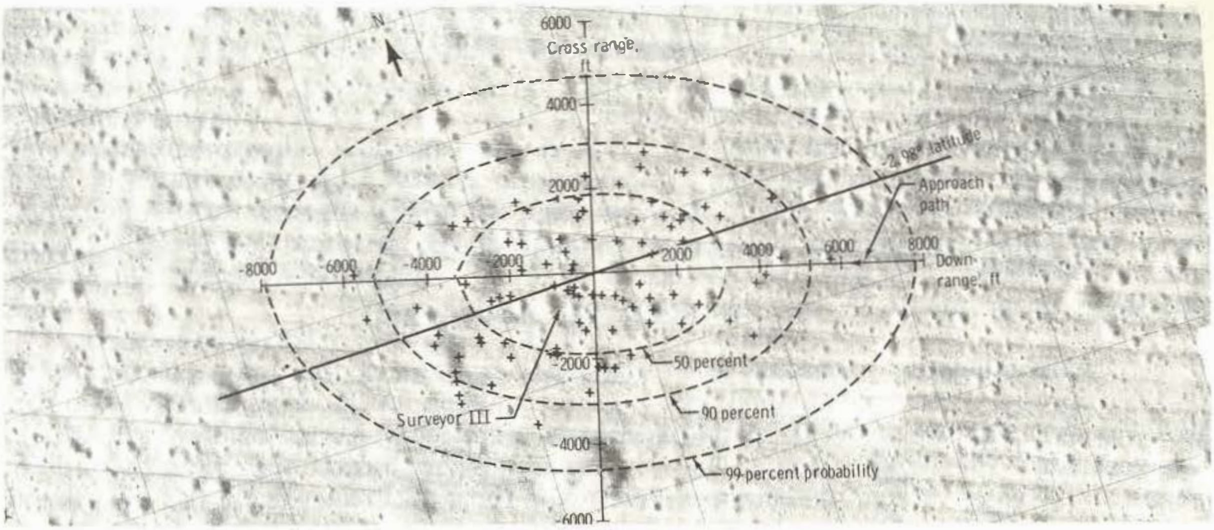


Figure 28. - Predicted Apollo 12 landing dispersions.

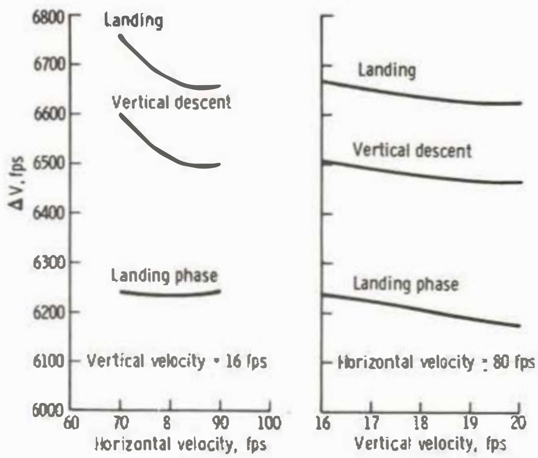


Figure 29. - Variation of ΔV with landing-phase velocities.

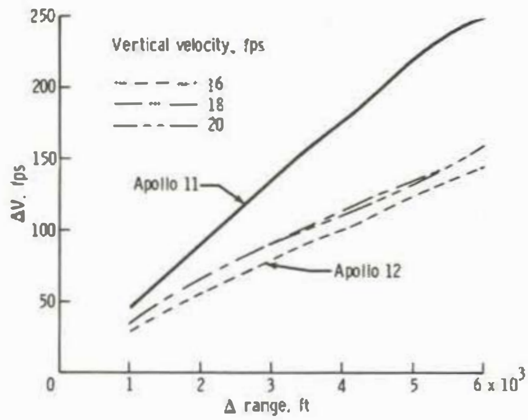


Figure 30. - The ΔV requirements for down-range redesignations at a 4000-foot altitude.

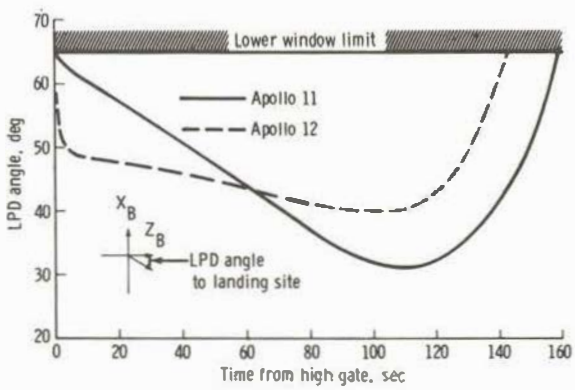


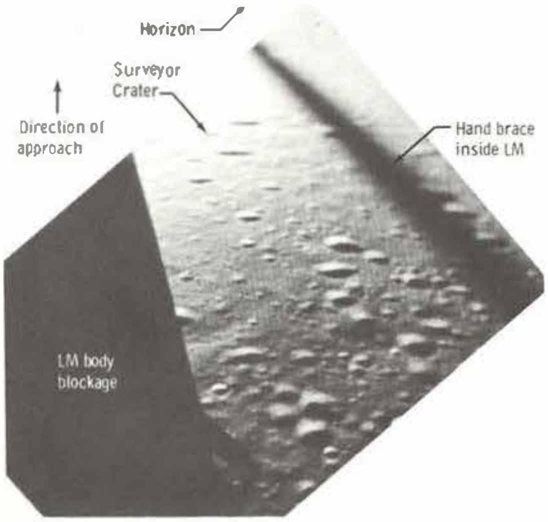
Figure 31. - Comparison of Apollo 11 and Apollo 12 LPD profiles.

In summary, the Apollo 12 descent and ascent used the same design as the Apollo 11 descent and ascent. The descent approach and landing-phase trajectory were speeded up slightly. The capability to update the landing site position during the braking phase was added. Finally, reduction in the descent ΔV and propellant requirements for missions subsequent to the Apollo 12 flight was planned.

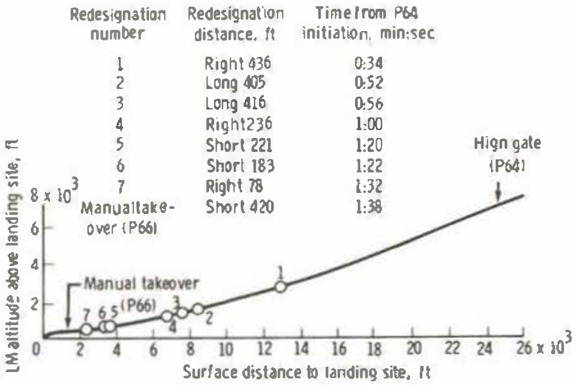
Apollo 12 Postflight Analysis

The second manned lunar landing occurred on November 19, 1969, at Apollo site 7 in the Ocean of Storms, adjacent to the crater containing Surveyor III. Throughout powered descent, all systems performed excellently, with not even a program alarm. The PDI occurred 5 nautical miles north of the nominal groundtrack. This cross-range distance was known to the guidance and was steered out during the braking phase for a minimal ΔV of approximately 10 fps. Also, at PDI, an up-range position error of 4200 feet was determined by the powered-flight processor. Thus, the landing-site position was updated (moved down range) by that amount early in the braking phase. This correction resulted in a 5-second-early throttle recovery and a slight ΔV penalty (fig. 27). A down-range redesignation of 4200 feet could have been performed in the approach phase, if necessary — however, not as cheaply as the braking-phase update (figs. 27 and 30).

During the approach phase, the commander was able to determine that the guidance was very close (approximately 600 feet, which is the diameter of Surveyor Crater) to being on target, as illustrated in figure 32. Figure 32(a) shows the view from the right-hand window (the lunar module pilot's window) taken in real time by the onboard 16-millimeter camera 20 seconds after high gate. Based on this view and with trajectory reconstruction, the view as seen by the commander from the left window was determined from an analytical computer program, as shown in figure 32(b). The commander performed several redesignations late in the approach, as indicated in figure 33, to land in a more acceptable area. A plot of the guidance-targeted landing site as a result of these redesignations is shown in figure 34, along with a groundtrack of the landing-phase trajectory under P66 (manual) control. The commander switched to manual control to land closer to the Surveyor III, maneuvering the LM some 420 feet closer (short) than would have



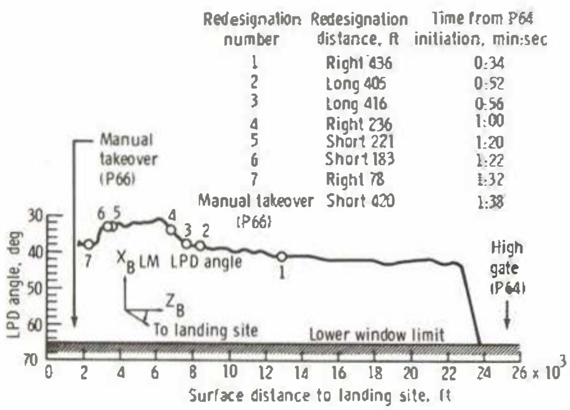
(a) Right-hand window view taken with onboard 16-millimeter camera (camera tilted 41° to the horizon).



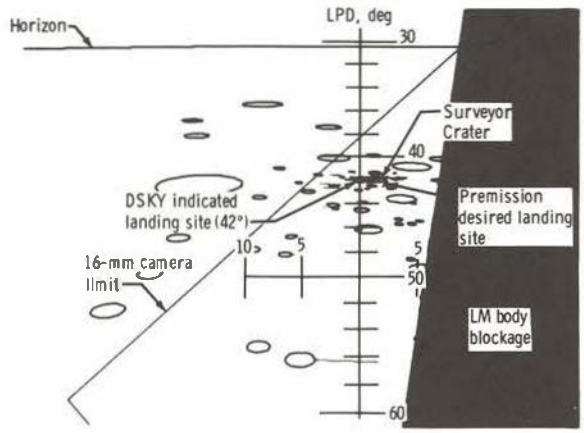
(b) Lunar module altitude above the landing site as a function of surface distance to the landing site.

Figure 32. - Apollo 12 window views 30 seconds after high gate (altitude, 4000 feet).

occurred by continuation of automatic guidance control. The altitude-range profile under manual control is illustrated in figure 35. The time of flight in the landing phase below 500 feet was 2 minutes (1 minute 50 seconds under manual control). This is considered nominal for a manual landing. Total powered descent took 12 minutes 26 seconds. Premission automatic nominal descent was 11 minutes 20 seconds.



(a) Landing point designator angle as a function of surface distance to the landing site.



(b) Computer reconstruction of commander's view.

Figure 33. - Apollo 12 approach phase.

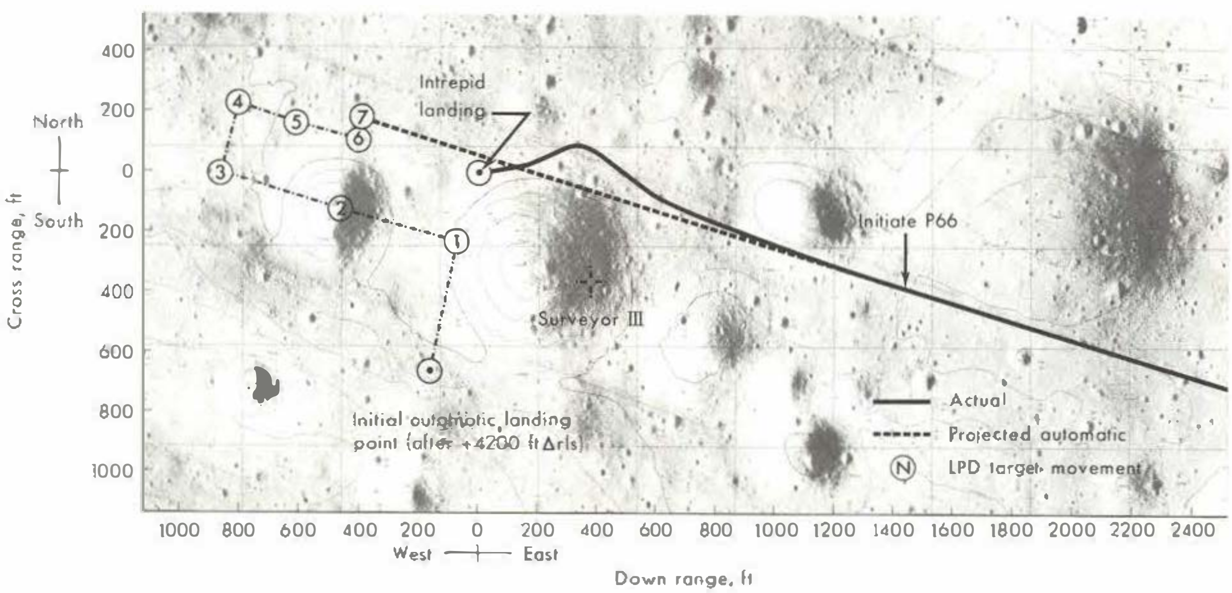


Figure 34. - Apollo 12 groundtrack for the landing phase.

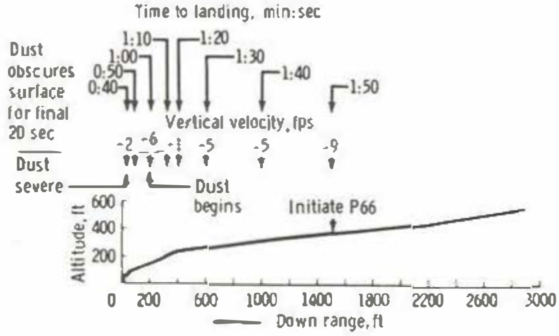


Figure 35. - Attitude as a function of up-range distance for the Apollo 12 approach and landing phases.

Touchdown occurred 35 seconds after low-level light "on," or approximately 60 seconds before the land-or-abort decision point. This margin is almost twice the Apollo 11 margin. However, postflight analysis has shown that the low-level sensor was actuated early (20 seconds on the Apollo 11 mission, 25 seconds on the Apollo 12 mission) because of propellant sloshing. This problem is expected to be solved for future missions by (1) increasing the quantity measurement samples on each propellant tank from 1 to 100 samples per second to define the dynamic effects and (2) installing baffles to decrease slosh.

The Apollo 12 landing stirred up more dust than the Apollo 11 landing during final touchdown, which resulted in considerable loss of visibility. (See notes on figure 35.) This visibility problem has led to a modification to the landing-guidance program (P66) for future missions. In addition to the current manual control mode, the capability has been added for the commander to be able to select automatic horizontal velocity nulling. That is, should visual cues be lost near or during vertical descent, the automatic mode could be selected to null any horizontal velocity components while the commander maintains control of vertical descent rate to touchdown.

In summary, the Apollo 12 mission, the second highly successful manned lunar landing, achieved the first pinpoint landing. The achievement of pinpoint landing greatly enhanced the possibilities for lunar exploration into the rougher mountainous areas of particular interest to scientists.

MISSION-PLANNING EXPERIENCE

Mission planning entails the development of trajectories and associated software logic for accomplishing defined objectives within the capabilities and constraints of the spacecraft systems and the crew, when operating in a specified environment. Thus, the mission planners' task is primarily one of integration to achieve the proper balance among performance, constraints, and objectives. The soundness of the plan is based on the ability to achieve mission success with at least 99.7-percent (3σ) probability while maintaining crew safety.

As stated previously, the basic mission-design philosophy for LM descent and ascent remained unchanged throughout the 7 years of planning. However, as LM systems changed from design concept to reality and as operation constraints were modified, it became necessary, particularly for the descent, to modify or reshape the trajectory and software logic accordingly.

In the preceding sections, it has been shown that the final premission plan was sound, leading to two highly successful manned lunar landings. The purpose of the following sections is to provide some insight into typical problems (not intended to be all inclusive) encountered by the mission planners and the solutions that evolved into the final operation plan. Because most of these problems involved changing system capabilities and constraints, the discussion of typical problems is divided into system design specifications, system performance definitions, system interfaces, and mission-planning flexibility.

System Design Specifications

The DPS will be used as an example of problems associated with design specifications, because it presented many problems to the mission planners. The original design requirements specified a throttle range of 10 500 to 1050 pounds, a range beyond the state of the art at that time. This range of thrusting provided three capabilities. First, the maximum thrust level provided near-optimum propellant efficiency with an initial thrust-to-weight ratio T/W_0 of 0.42 (ref. 3). Second, the minimum thrust provided translation and hover capability in a vertical attitude near the lunar surface. Third, the continuous throttle capability provided the PGNCS the means for achieving the desired final position (altitude, cross range, and down range) and velocity vectors. (See the section entitled "Guidance Logic.") Difficulties encountered in the development of the DPS resulted in achieving a nominal maximum thrust of only 9800 pounds and not achieving the full design range of throttle capability. The reduced maximum thrust coupled with a weight growth from 25 000 to approximately 34 000 pounds yielded a T/W_0 of only 0.29. These changes resulted in a loss of efficiency amounting to 160-fps ΔV increase or 600 pounds of additional propellant required. However, only 30 percent of this penalty is attributed to the reduction in maximum thrust; the remainder is charged to the weight growth.

More serious, however, was the reduced throttle capability. The throttle capability (as defined in the section entitled "System Descriptions") was reduced to a range of 10 to 60 percent (100 percent being defined as 10 500 pounds) with a fixed throttle position at maximum thrust. The propulsion-system designers were satisfied because this capability solved the hardware design problems and still achieved the nominal-design mission duty cycle. However, to the mission planners, the reduced throttle capability was a severe constraint which meant that the means to satisfy PGNCS commands for achieving targeted conditions were not available during FTP operation. Because flight safety could be impaired if desired altitude and velocity targets were not achieved, consideration was given to relaxing the down-range target constraint. However, the down-range dispersions from thrust errors alone would be ±9 nautical miles, which was considered unacceptable even for mare-type (smooth) landing areas. Modulation of the down-range thrust vector by out-of-plane thrusting (roll about body Z-axis) similar to lift vector modulation for atmospheric entry could theoretically provide range control. However, this maneuvering was incompatible with LR operation and stable conditions for crew monitoring. Thus, it was not given further consideration.

Attempts to regain some throttle control by (1) shallow throttling and (2) throttle pulsing were investigated. Shallow throttling refers to a small (±3 percent) throttle capability about FTP. Throttle pulsing refers to modulating the thrust several times between FTP and 57 percent (upper throttle region). The cost estimates by the propulsion-system designers for development and test were too high and confidence of success too low to warrant use of either of these proposals. However, the second technique was reviewed and considered for later Apollo missions, as discussed in the section entitled "Apollo 12 Planning."

The solution finally adopted was to target the braking phase inefficiently, such that the guidance would command thrust levels to drop within the throttle capability (less than 57 percent) before high gate. In this manner, the desired target conditions could be achieved within the throttling capability of the DPS. Nominally, 2 minutes of inefficient throttling before high gate was required to accommodate dispersions in thrust and navigation. This resulted in a ΔV penalty of approximately 100 fps or approximately 380 pounds of additional required propellant. This solution also resulted in what at first seemed to be a contradiction. A high FTP thrust performing engine (higher than nominal) was less efficient, and a low thrust engine was more efficient than nominal. This was because higher thrust resulted in early throttle recovery with a longer inefficient throttle region, and low thrust resulted in late throttle recovery with a shorter inefficient throttle region. (See the section entitled "The DPS and PGNCS Interface.")

Definition of System Performances

Many of the mission planners' problems were associated with a proper definition of system operation performance capabilities and constraints. The DPS, LR, and PGNCS are used as typical examples of this type of problem.

To meet the mission objective of landing on the moon with 99.7-percent probability of success, the guidance system had to be able to correct for off-nominal initial-condition errors, for system errors during the descent, and for uncertainties of the lunar terrain on approach to the landing area. Thus, a variety of DPS duty cycles could be commanded in addition to the nominal to achieve this objective. The mission planners, then, needed a definition of thrust and specific impulse as a function of commanded throttle to perform trajectory analyses. Because the DPS was an ablative-cooled engine, the amount of time spent at a given throttle setting affected throat erosion and, consequently, affected subsequent performance at given throttle settings. Therefore, to provide the mission planners with performance data, the system designers needed to know the specific duty cycle for each trajectory. Thus, the iteration began. This iteration resulted in much confusion and many investigative false starts before the mission planners and system designers realized the extent to which the inputs of one affected the other. The problem was then solved by including the system designers' sophisticated DPS model (temperature, pressure relations for determining appropriate thrust, and specific impulse) in the mission planners' simulations for trajectory generation. This simulation included closed-loop guidance and other pertinent systems models. This allowed the system designers to incorporate the latest test results rapidly into the mathematical model of the DPS. In this manner, a true and updated knowledge of the best trajectory and system-design requirements was obtained.

The LR problems are analogous to the DPS problems. Again, the mission planners' problems involved a proper definition of the operational performance and constraints. The mission planners needed to know the answers to such questions as the following.

1. At what maximum altitude was the LR expected to operate for updating the PGNCS estimate of altitude and velocity?

2. What was the accuracy of the updating?

3. What was the best orientation for positioning the LR beam?

4. Where would loss of signal occur because of zero Doppler shift? (The velocity is normal to the beam; therefore, no signal return occurs.)

5. How close to the lunar surface would the LR operate effectively?

To deal with these questions, the system designers needed to know several answers themselves.

1. What trajectory (acceleration, velocity, and position profile) was to be flown?

2. What were the attitude and attitude-rate profiles?

3. What was the terrain profile that the LR was to track?

4. What were the lunar-surface reflectivity characteristics?

Thus, the iteration began. The LR updates changed the PGNCS estimate of the trajectory and caused the guidance to change commands and fly a trajectory other than the nominal. The new commands and trajectory changed orientation of the LR beams, which resulted in different LR performance. Again, both the mission planners and the system designers were underestimating the extent to which the inputs of one affected the other. That is, the system designers had been tying the design to a nominal trajectory as opposed to a flight regime. The mission planners were again using an oversimplified system-performance model. The resulting confusion was not cleared until the system designers' sophisticated LR model was included in the mission planners' simulations for trajectory generation, as was done with the DPS model. The LR model included acquisition and performance determined from calculations of signal-to-noise ratio for each beam as a function of the trajectory conditions (beam incidence angle, range, and velocity) and electronic characteristics (bandwidth, preamplifier slope, tracker gains, et cetera).

Even with the sophisticated modeling of system performances, the outputs were still no better than the inputs. Unfortunately, the inputs provided by the system designers were often overly conservative; that is, the performance inputs were gross underestimates of the actual system performances. For example, the system analysis for providing inputs to the DPS model was initially conducted on a worst-case basis. That is, all error sources were considered unrealistically to be linearly additive. This led to large uncertainties in performance and, consequently, required large allocations of propellant to be held in reserve, which resulted in gross inefficiency. If this type of analysis had been used on all systems (and it was tried on many), the flight would, literally, never have gotten off the ground. With the integrated system models in the trajectory simulations, the mission planners and system designers were able to conduct appropriate statistical analyses that deleted the unnecessary overconservatism and still provided 99.7-percent mission-success planning.

Another example of overconservatism involved altitude navigation before PDI and during the braking phase of powered descent. The preliminary estimate of altitude uncertainty at PDI was 7800 feet (3σ). Initial estimate for beginning LR altitude updating was 20 000 feet (nominal). By the time the LM reached this altitude, the 3σ altitude uncertainty had grown to approximately 11 000 feet. From figure 18, it can be seen that the maximum allowable Δh (altitude difference between PGNCS and LR) could be exceeded for the nominal (20 000 feet) initiation of LR altitude updating with better than 3σ PGNCS performance. That is, 99.7-percent mission-success probability could not be assured. Thus, the mission planners were faced with either changing the mission plan in some manner or seeking improvement in system performances from the system designers. Because changing the mission plan could impact all elements involved (system operations, crew training, flight control, et cetera), it was first decided to investigate system performances. The systems of concern were the MSFN navigation (PGNCS initialized with the MSFN), PGNCS errors (primarily accelerometer bias), and LR performance. However, no recognized improvement was to be found. The MSFN uncertainties were involved in the newly discovered and little understood mascon theories. The PGNCS hardware was tested and found to be considerably better than design specifications; however, only the specification performance showed up in the official data book. Finally, although the mathematical model of the LR (which was supplied by the system designers) indicated that LR altitude updating should be expected nominally at 35 600 feet instead of 20 000 feet, the system designers' official data still did not reflect this capability. It was not until the Apollo 10 flight (the dress rehearsal for the lunar landing) demonstrated the LR operational capability to be above 60 000 feet, as predicted by the mathematical model (LR beams pointed nearer vertical than in descent, thus the increase in performance altitude), that the system designers agreed to upgrade the LR performance estimates. Also, the Apollo 8 and Apollo 10 missions provided increased understanding of lunar-orbit navigation, which resulted in an improvement by a factor of 2 in altitude uncertainty for PDI. (See PDI dispersions in figure 18.) It was not until after the Apollo 11 mission that the PGNCS performance estimates were upgraded, again primarily accelerometer bias, by a factor of 2.

Thus, with the recognized improvement in LR capability and orbit navigation, the altitude navigation problem was finally solved, but not until after the Apollo 10 mission. In the meantime, considerable manpower was being devoted to crew-monitoring techniques for trying to estimate altitude. These included (1) RR tracking of the CSM before PDI and (2) tracking surface features with window markings and a stopwatch. These techniques had gross accuracies of approximately 10 000 feet; however, because of a lack of confidence in orbit navigation and LR capability at that time, the RR tracking technique was planned for and used on the Apollo 11 flight. Also, a face-down attitude was planned for and used (after much controversy) during the first portion of the descent braking phase. In this attitude, the crew planned to monitor surface features visually for gross down-track and cross-track errors as well as altitude errors. There were two major controversies.

1. The effect of the face-down attitude on S-band antenna pointing limits for communications with the earth

2. The length of time the face-down attitude could be maintained without interfering with LR altitude updating

First, the effect on S-band coverage was considered. It was determined that communications would be blocked in a face-down attitude. Thus, the choices were as follows.

1. Do not allow the face-down attitude so that ground communications can be maintained.

2. Allow face-down attitude and give up communications.

3. Allow face-down attitude and modify the S-band antenna limits.

Because of the operational uncertainties of the systems involved (LR, MSFN, and PGNCS), it was decided that the attitude flexibility be provided to the crew for surface monitoring and that ground communications be maintained to take advantage of earth-based monitoring capabilities. Thus, the face-down attitude was allowed after the associated penalties in hardware cost, manpower, testing, and schedule impacts for modifying the S-band antenna were accepted.

Second, the effect of the face-down attitude on LR altitude updating was considered. It was determined that the crew could yaw face down or face up at their discretion; however, the guidance computer would command (X-axis override) face-up attitude at an indicated altitude of 30 000 feet if the crew had not already done so. This provided the crew with nearly 6 minutes for surface monitoring and still allowed sufficient margin for the LR update to correct altitude dispersions. On the Apollo 11 mission, the crew completed surface monitoring and began yawing at approximately 4 minutes into the descent. The LR acquired lock-on to the lunar surface during this rotation at an altitude of 37 000 feet. With this added confidence in LR, PGNCS, and MSFN capabilities, the RR tracking of the CSM and surface features was not deemed necessary for the Apollo 12 flight. This change allowed a face-up attitude throughout descent, simplifying crew procedures, simplifying S-band antenna pointing, and maximizing LR use. Before this change, because of overly conservative estimates or lack of confidence in system test-development programs, an extensive expenditure of manpower and money was made, which in some cases was unnecessary or at least overemphasized.

The preceding problem also had considerable influence on mission rules; in particular, the rule calling for an abort at a PGNCS-estimated altitude of 10 000 feet if LR altitude updating had not been established. (See the section entitled "The LR and PGNCS Interface.") It was desirable to be able to proceed to high gate without LR to enhance mission success, because manual control of the descent with out-the-window visibility of the surface was possible from that point. However, with the estimates of systems performance, it was not safe (on a 99.7-percent basis) to do so. As estimates of system performances change, this mission rule is subject to change. Thus, mission rules are another pertinent reason for having the proper definition of system performances. Without this definition, mission rules could be quite arbitrary, because they would be based on an unreal situation.

System Interfaces

Perhaps the most difficult problems facing both the mission planners and the system designers are those associated with interfacing one system with another. The difficulty arises from trying to achieve and maintain compatibility between output from one system, which is input to the other. Maintaining this compatibility is often lost. As system development evolves, a necessary modification in one system may result in a subtle change in output or input format (hardware or software). If this change is not properly analyzed and tested, it usually will cause problems in some phase of the process. Certainly, the responsibility rests upon the system designers to define all such changes clearly. However, it is also the responsibility of the mission planners, as integrators of system capabilities and constraints for accomplishing mission requirements, to understand the ramifications of the change and to communicate these effects to program management for final resolution of the change.

Many interface problems had to be resolved in planning LM descent and ascent. Most of these problems concerned interfaces between the guidance computer and each of the other systems. This was to be expected because the guidance computer is the real-time integrator of all other (as well as its own) system performances and constraints to achieve the desired target objectives. In the real-time situation, no system interface problems occurred during ascent, and only one interface problem occurred during descent. The descent problem concerned the interface between the guidance computer and the RR and has been discussed in the section entitled "The PGNCS Monitoring." Although this interface problem went undetected during system design and premission simulations, the possibility of this type of problem had been anticipated (computer restart protection). Thus, when the problem occurred in real time, its effect was minimized and continuation of the mission was possible.

Other interface problems that could have been encountered in real time, but were not, have also been discussed in the section entitled "Real-Time Analysis." These problems were the subject of the real-time monitoring limits and the rules for aborting the mission. Next, some additional problems encountered in premission planning of the LM descent are discussed to illustrate further the difficulties of system interfaces.

A typical example was mission design for the DOI maneuver. It was decided to reduce the lunar parking-orbit altitude from 100 nautical miles (original design) to 80 nautical miles (later reduced to 60 nautical miles) to reduce propellant requirements that allowed increases in the system and spacecraft dry weights. The problem was as follows. The ΔV requirement for DOI was reduced; thus, burn time on DPS at maximum thrust (FTP) was reduced. Unfortunately, it was reduced to the point where the PGNCS guidance did not have sufficient time to command an accurate cut-off, which resulted in dispersions too large for continuation of the mission. Obviously, the use of a lower thrust level, which the DPS was capable of, would result in a longer burn time and would solve the problem. Because the DPS was always ignited at 10-percent thrust for trimming the gimbal, consideration was given to performing the entire DOI maneuver at 10-percent thrust. Unfortunately, the DPS could not perform this type of burn and still assure temperature and pressure (supercritical helium pressure-feed system) conditions necessary to perform PDI 1 hour later. After many iterations, much cost, and extensive testing, it was determined that the solution was to throttle the DPS manually, after trim, to 40-percent throttle. Auto-throttle logic would unduly complicate the software logic. The PGNCS and DPS interface for this maneuver was once again compatible.

An example of the interface problems between the PGNCS and the LR was side-lobe lockup. The LR design was such that the beams had a main lobe (strongest pattern of signal return) and side lobes (weaker patterns of signal return). This could present a problem at low altitudes (below high gate). In the event of LR signal dropout (zero Doppler shift, antenna switching), it was possible for reacquisition to occur on the side lobe rather than on the main lobe. These situations would result in erroneous updates to the PGNCS. Rather than change the hardware, it was determined that a satisfactory and inexpensive solution would be to modify the guidance software logic to include a reasonability test on the LR data before they were incorporated. This solution has worked; however, it has further complicated and constrained the PGNCS and LR interface and continues to be analyzed. This is one of the many examples where software logic has been added to solve hardware problems. This type of solution should be exercised judiciously because of the additional complexity and the limits of the guidance computer: fixed and erasable memory, timing requirements, and logic changes.

Mission-Planning Flexibility

Many problems were encountered because original system designs were molded too closely to a nominal trajectory. Therefore, it is imperative that the mission planners define a flight regime for the system designers, as opposed to only a reference or nominal profile. Just as the system designers should not be arbitrarily conservative in defining performance, neither should the mission planners do so in defining the flight regime. This would place unnecessary requirements on system designs and result in either degrading the performance where it is really needed or increasing the development and test costs, or both. The mission planners must define the flight regime which optimizes the balance between mission objectives (including crew safety) and system capabilities. This regime will change when either objectives or capabilities change. Thus, the mission planners' design must maintain the capability or flexibility to accommodate reasonable changes in both. This capability must exist not only during system development but also after the systems become operational.

For example, in the initial development of the guidance-computer software, the descent targets were allocated to fixed (hardwired) memory. Erasable memory was and still is quite limited and reserved primarily for system performance coefficients that might change because of final test results reported after computer rope manufacture. Position of the landing site was the only descent-trajectory-dependent parameter in erasable load. This completely destroyed the mission planners' capability for operational flexibility after manufacture of computer ropes (which can occur several months before launch). After it was pointed out several times that system capabilities as well as mission objectives would be enhanced by targeting changes based on latest system test results, the targets were placed in erasable memory. Without this capability, the efficiency and adequacy of mission planning would have been severely hampered.

Another example of limiting mission-planning flexibility is the technique for LR altitude updating of the guidance computer. (See the section entitled "Premission Planning.") The original objective of the manned lunar landing was to land safely and return. Therefore, the relatively large, smooth mare basins were selected for flight safety. For these landing areas, the surface-tracking (assuming a spherical moon) altitude-updating concept was and still is quite adequate. However, the objectives for future missions have changed (to maximize scientific return) to landings in areas characterized by considerably more rough-approach terrain. The LR measurements reflect the erratic nature of the surface, and when these measurements (although weighted) are incorporated into the navigation system, degradation in the guidance performance occurs. This process also results in errors in the LPD pointing accuracy which thus degrades the pinpoint landing capability. To gain sufficient capability to land in these rough areas, the mission planners incorporated a simplified model (linear segments) of the terrain over the range of LR updating into the guidance computer. This concept is not a cure-all for rough terrain, because of model limitations and accuracy of the knowledge of the terrain. Terrain modeling is a complete science in itself (perhaps an art), dedicated to generating accurate profiles for sites of interest from Lunar Orbiter, Surveyor, and Apollo photography. Terrain modeling is a very complex and also limited (to availability and type of photography) science. As new accuracies are obtained for terrain characteristics, the descent trajectory must be reanalyzed and modified as necessary. What is needed is a system of navigation updating pointed (during approach) directly at the landing site, which thus divorces the dependency of the trajectory design on approach-terrain variations and uncertainties. Such a system is not available for Apollo; therefore, the mission planners' flexibility in selecting landing sites in areas of rough terrain will be limited by the present LR navigation technique.

Experience Summary

From the preceding discussions of typical problems associated with mission planning of LM descent, what can the mission planners and designers learn? First, it is imperative that both the mission planners and system designers understand the objectives and requirements of the other group. The system design cannot be limited to a nominal trajectory; at a minimum, the design must be capable of operating over trajectories that result from its own performance dispersions. The mission planners must be provided with realistic system models for the generation of a trajectory design that satisfies the mission objectives. Likewise, the mission planners must provide the system designers with a realistic flight regime to assure a compatible system design. Overconservatism on the part of either group can cause as much difficulty as would a total lack of conservatism. The flight regime must provide a reasonable amount of flexibility to adjust to changes in system design developments and mission objectives. The mission planners must protect the capability to provide mission-planning flexibility through computer software design, both on board the spacecraft and at the flight control center.

Both the mission planners and the system designers must be alert to system-interface problems, which often go unnoticed for long periods because of the interfacing of technical disciplines. After an awareness has been established, generally, the problem is readily solved.

One recommendation for changing the design concept for future landing programs (beyond Apollo) is offered. The navigation concept based on surface tracking along the approach can severely limit the flexibility of landing-site selection. This concept demands the generation of considerable data from previous missions or programs and can be the constraining factor in deleting some scientifically desirable sites. A navigation technique based on direct ranging to the landing site during approach provides greater flexibility for site selection.

CONCLUDING REMARKS

The premission planning for the lunar descent and ascent phases of the Apollo 11 mission has been presented and compared with actual flight results. The Apollo 11 lunar module descent and ascent compared excellently with premission planning. An initial navigation error caused the landing to be approximately 3 nautical miles down range from the target, but the landing was still within the premission mapped area. The original three-phase descent design and contingency planning afforded the crew the opportunity, late in the descent, to maneuver out of an area of rough terrain to a successful touchdown.

As a result of the Apollo 11 postflight analysis, only two minor changes were incorporated in descent planning for the Apollo 12 flight. The first change was the provision for a navigation update of the landing site early in the braking phase to enhance the pinpoint landing capability. The second change was a slight modification to the descent targeting to enhance the landing-site redesignation and manual translation capability in the approach and landing phases.

The Apollo 12 lunar module descent and ascent data also correlated well with premission planning. During lunar module descent, the landing-site navigation update and redesignation capabilities were used, along with manual maneuvering, to achieve the first pinpoint lunar landing. The landing, within 600 feet of the Surveyor III spacecraft, has provided confidence for premission planning of future manned lunar-exploration missions.

From the Apollo experience, it has been shown that many mission-planning problems were encountered as a result of changing system capabilities and constraints. These problems were solved in the Apollo Program and can be avoided in future programs by (1) proper understanding by the mission planners and the system designers of all objectives and requirements; (2) proper definition and modeling of system performances; (3) awareness and understanding of system interfaces; (4) definition of a design flight regime, not just a nominal trajectory; (5) maintenance of a capability for mission-planning flexibility; and (6) avoidance of false conservatism in defining system performances and flight regimes.

Only one recommendation for changing the design concept for future lunar-landing programs (beyond Apollo) is offered. It is recommended that a navigation technique based on direct ranging to the landing site be investigated to replace surface tracking along the approach. This would provide greater flexibility for site selection in areas of rough-approach terrain.

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