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LUNAR MODULE PILOT CONTROL CONSIDERATIONS

by Clarke T. Hackler, James R. Brickel,
Herbert E. Smith, and Donald C. Cheatham

Manned Spacecraft Center
Houston, Texas

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ABSTRACT

The lunar module mission and the role of the pilot in spacecraft control during the lunar mission are discussed in this paper. A brief description is made of the lunar module guidance and control systems, the methods of guidance in various mission phases, and the interfaces between the pilot and the guidance and control systems. The primary areas of pilot control in the various phases are examined in detail and are related to such areas as guidance monitoring, landing site inspection and change, manual updating of the guidance system, landing, and terminal rendezvous. Pilot control through the guidance system is discussed, and manual backup procedures in certain phases of the mission are examined.

CONTENTS

Section | Page

TABLES

Table | Page

FIGURES

Figure | Page

LUNAR MODULE PILOT CONTROL CONSIDERATIONS

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SUMMARY

The role of the pilot in control of the lunar module during the lunar landing mission is discussed. A brief description is given of the lunar module mission phases and the lunar module and its primary and backup guidance, navigation, and control systems. The pilot interfaces with the control system are also discussed. The various simulation studies and flight tests that have been conducted are reviewed. The reasons for the departure from a near-fuel-optimum trajectory and the several aspects of shaping the trajectory for visibility and other piloting considerations following this departure are analyzed. Finally, the primary areas of pilot control in the various phases are examined in detail and are related to such areas of concern as guidance monitoring, landing site inspection and change, landing, engine shutdown, terminal rendezvous, and docking.

INTRODUCTION

The Apollo lunar module (LM) must provide the crew with the means for retromaneuvering out of lunar orbit, decelerating to a soft landing, and then, after a period of time on the lunar surface, accelerating back into orbit for rendezvous and docking with the command and service module (CSM). One of the principal reasons for selecting a lunar-orbital rendezvous concept is that the LM may be designed to optimize pilot control of its flight phase in a lunar environment. Detailed analysis of the system requirements, coupled with operational procedures for performing these maneuvers, has led to a two-stage design configuration. Staging, under normal mission operation, occurs on the lunar surface so that the weight of the descent engine does not have to be carried back into orbit.

An early design decision made in the interest of saving weight was to utilize a single set of reaction jets to serve both stages. With a single set of jets, the response of the control system changes because the moments of inertia of the spacecraft change by an order of magnitude during the LM mission, mainly as a result of staging. Although the landing maneuver takes place about halfway through the powered portion of the LM mission, it occurs before the moments of inertia have changed appreciably. As a result, extreme care must be used in selecting control powers and control system parameters that will provide satisfactory qualities for manually controlled maneuvers and also will provide good system response for the automatically controlled phases of the mission.

Except for rendezvous, the mission cannot be completely evaluated in earth orbit; therefore, the ability of the crew to perform the mission satisfactorily will depend largely on the ability of the designer to anticipate the nature of the control task, to judge operational procedures correctly, and to provide the guidance and control system required to meet mission demands. Since the mission cannot be evaluated except under simulated conditions, the success of anticipating systems design and operational procedures will be ascertained only after the first lunar mission has been completed.

A list of the abbreviations used in the text is supplied in the appendix.

THE LM LUNAR MISSION

The mission of the LM (fig. 1) begins after the CSM-LM combination is inserted into a nearly circular lunar orbit of 80 nautical miles. The mission is divided into two major phases: the descent to the lunar surface, and the ascent to the orbiting CSM. These mission phases can be separated into several well-delineated subphases (fig. 2).

Lunar Descent

The descent to the lunar surface is initiated after the LM subsystems are checked out and the onboard inertial references are alined. Separation of the LM from the CSM is controlled manually by the crew using the reaction control subsystem (RCS) jets to orient the LM for injection into the 180° transfer orbit. The coasting descent transfer starts at the end of injection firing (executed with the descent engine) and terminates at a pericynthion altitude of 50 000 feet, about 240 nautical miles from the programed landing site. A braking descent is initiated at pericynthion by thrusting with the descent engine. This phase is terminated about 450 seconds after the start of the braking descent firing, at an altitude of approximately 10 000 feet, and a distance from the landing site of some 7 nautical miles. The final approach phase starts with the LM being rotated from 70° pitch back to about 45° pitch back, and the descent engine throttle is reduced from a near-maximum thrust level to about 50 percent of full thrust. The velocity at this time is approximately 730 fps, and the flightpath angle is -14°; these are the high-gate conditions. While the braking descent is designed for nearly optimum expenditures of velocity change ΔV, the final approach trajectory is shaped so that the crew can begin an initial visual assessment of the preselected landing site. Provisions have been made for the crew to change landing sites, using the primary navigation, guidance, and control system (PNGCS), if the preselected site is judged unacceptable for landing. For an automatically controlled landing, the final approach phase terminates with the LM hovering above the landing site; for a manually controlled landing, this phase

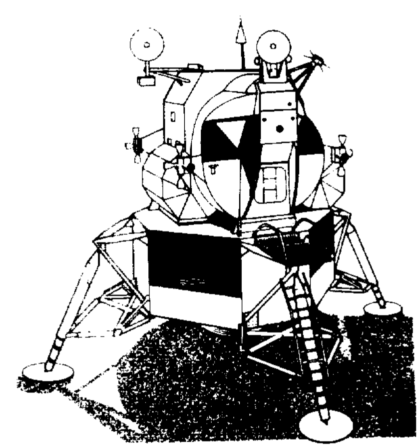


Figure 1. - Artist concept of LM spacecraft.

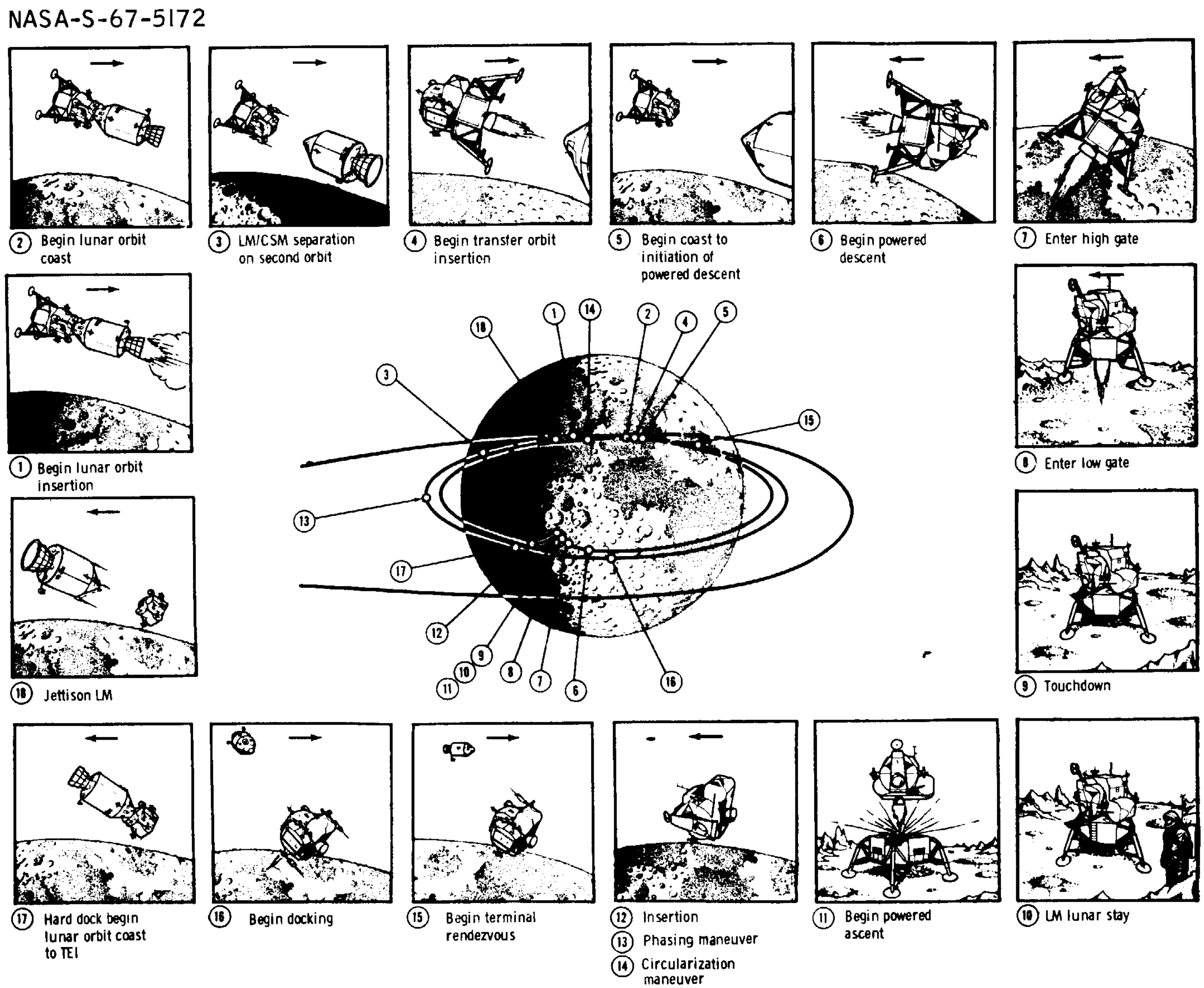


Figure 2. - LM mission events.

terminates when the crew believes that the velocity and position of the spacecraft are correct for starting the landing approach.

The manual landing approach maneuver is initiated with the LM being rotated to a nearly vertical attitude (low gate) to allow the crew a better assessment of the landing site. Using present concepts, at the end of the final approach phase the altitude is approximately 500 feet; the forward velocity, 60 fps; the descent rate, 15 fps; and the distance to the landing site, about 0.2 nautical mile. The spacecraft is maintained at a slight (10° to 15°) pitch-back attitude to reduce the forward velocity, and the descent rate is gradually reduced. When the landing site has been judged satisfactory, the translational velocity is controlled to arrive over the landing site with forward and lateral velocities of zero and a descent rate of about 5 fps. The inertial guidance system, the computer in particular, is now updated with landing radar data for the final time. This updating occurs at an altitude of approximately 100 feet prior to the final descent to touchdown. During the descent to the surface, the command pilot maintains the forward and lateral velocities at zero and gradually reduces the descent rate to 4 fps. At an altitude of about 4 feet, a signal is obtained from probes attached to the landing gear pads, and the descent engine is shut off by the crew. Touchdown on the lunar surface completes the descent phase of the mission.

Lunar Ascent

The lunar ascent of the LM mission is initiated after the lunar-stay objectives are completed and the inertial reference system is alined. Powered ascent is made with the 3500-pound-thrust ascent engine; the descent stage is used as the launch platform. This powered phase ends at orbit insertion (60 000 feet), with supercircular velocity conditions prevailing. Coasting ascent takes the LM from the 60 000-foot altitude to a 90° phasing maneuver; a coelliptic maneuver is performed at apocynthion, followed by a transfer maneuver. Terminal rendezvous starts with the LM about 20 nautical miles from the CSM and ends with the LM approximately 500 feet from the CSM and closing at about 5 fps. After the last midcourse correction, the crew manually controls the LM through the terminal phase of rendezvous and completes the hard docking maneuver to the CSM.

With transfer of the crew to the CSM, the LM mission is completed. After crew transfer, the LM is jettisoned in lunar orbit. The description of the lunar mission has been greatly abbreviated, but a detailed discussion can be found in reference 1.

NAVIGATION, GUIDANCE, AND CONTROL SYSTEMS

The primary navigation, guidance, and control functions of the LM spacecraft are performed through an aided inertial system (in the PNGCS) utilizing radar and optical sensor data. The PNGCS is a control path which meets all mission requirements including abort. As a backup to the primary system, the LM also contains an abort guidance system (AGS) for control and guidance to a safe rendezvous with the CSM in the event of PNGCS failure. Figure 3 illustrates the basic structures and interfaces of the two guidance systems.

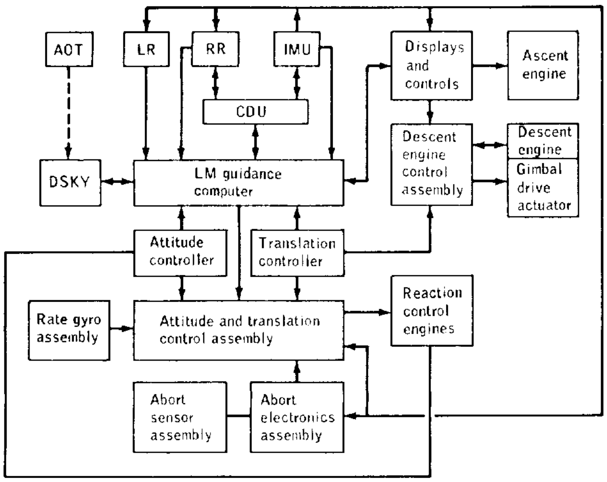


Figure 3. - Block diagram of LM guidance and control systems.

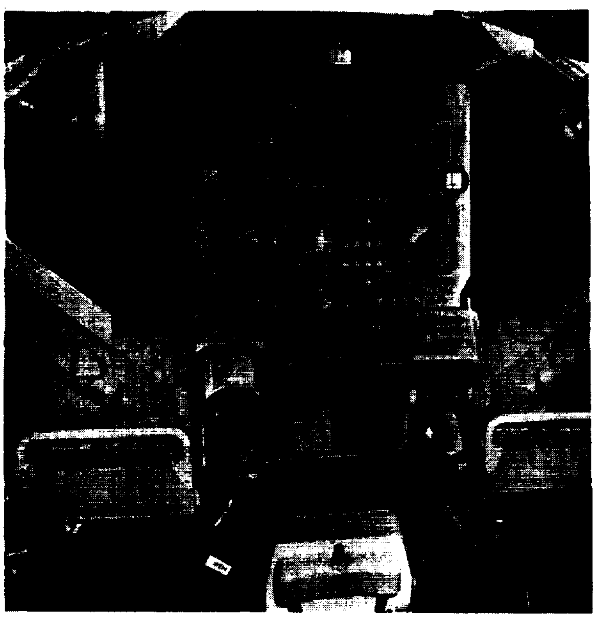


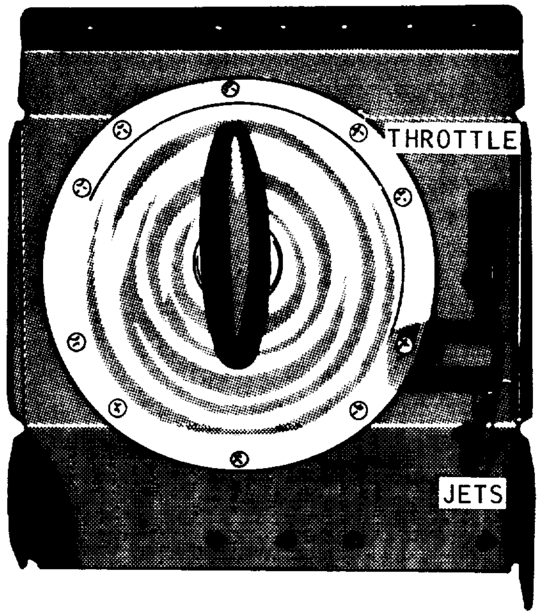
Figure 4. - LM flight displays.

Pilot Interfaces

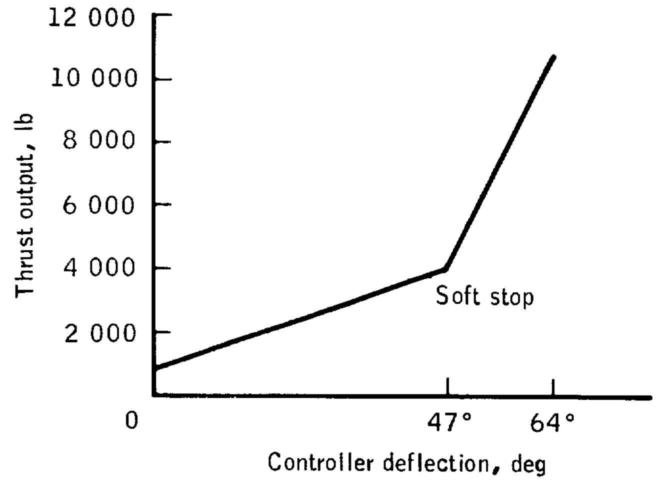
Pilot interface with the PNGCS is through display and keyboard (DSKY), rotational hand controllers, descent engine throttles, translational controllers, and alinement optical telescope (AOT). Other interfaces are through the displays (fig. 4) showing the state of the spacecraft and its various subsystems, including the landing and rendezvous radars. The interfaces for the backup control path are the same as in the PNGCS except that the data entry and display assembly replaces the DSKY, and the AOT cannot be used for abort guidance alinement. These interfaces permit the crew to monitor the guidance systems for correct operation and to perform manual control of the spacecraft using the selected control path.

Flight Control Modes

Attitude control in the PNGCS control path is obtained by digital stabilization by means of the primary computer. The backup attitude control path is obtained by an analog stabilization through the control electronics section. The maximum normal maneuvering rate available to the crew is 20 deg/sec about any axis. An electronic deadband of either 0.3° or 5.0° can be selected by the crew. In the PNGCS control mode, the crew also has available a rate-of-descent (ROD) control by means of the computer which is available to the crew through the PNGCS for altitude rate control during descent. The control is a discrete-level type in which a change of ±1 fps in the descent rate is commanded each time the ROD controller is actuated. The crew can disengage the rate-of-descent mode automatically by moving the throttle controller past the throttle soft stop (fig. 5), by changing the descent engine control from automatic to manual, or by switching the control path from the PNGCS to the AGS.



(a) Controller.



(b) Throttle characteristics.

Figure 5. - Integrated translational and descent engine throttle controller.

PNGCS Attitude Control

The attitude control mode generally used by the crew for rotational maneuvering is rate-command/attitude-hold (RCAH). In the PNGCS mode, the attitude-hold feature will not engage until the sum of the absolute values of the rotational rates is equal to or less than 1 deg/sec. This method of mechanization is employed to eliminate undesirable overshoots in attitude caused by low control powers in the descent configuration. Loss of the attitude reference, in this case the inertial-measurement-unit gimbal angles, also results in the loss of the primary attitude control mode.

AGS Attitude Control

Through the backup attitude control path, the crew can use one of three control modes — rate-command/attitude-hold, pulse, or direct — for rotational maneuvering.

Rate-command/attitude-hold mode. - The RCAH mode in the backup control path differs from that in the PNGCS control path; when the backup control path is selected, the attitude-hold feature engages whenever the rotational controller is returned to detent.

Pulse mode. - With the pulse control mode, the crew can obtain a single impulse by moving the controller to a 2.5° deflection or beyond and then returning it to detent, or the crew can command a stream of pulses at a rate of 1.5 times per second by keeping the controller out of detent. This control mode, which can be selected by the crew on a per-axis basis, is actuated by signals from the detent switches located at the 2.5° controller deflection point.

Direct mode. - The direct control mode is also selected by the crew on a per-axis basis and operates through the 2.5° controller deflection point. However, in this case the detent switch outputs go directly to the secondary coils of the reaction jets and fire two jets continuously for as long as the controller is out of detent.

X-Axis Override

The X-axis override feature allows the crew to disengage the X-axis steering (attitude commands) and to manually yaw the spacecraft about the thrust vector during periods of powered flight. The override can be engaged during automatic flight by taking the X-axis of the controller out of the detent position. This feature is primarily used for terrain observation and landing site changes during powered descent. It is also available to the crew during powered ascent.

Emergency Direct Mode

The final attitude control mode is emergency direct, available in either the PNGCS or backup control paths. This function is actuated whenever the rotational controller is placed against the controller hard stop. Direct connection to the secondary coils of the reaction jet is made through the attitude controller soft-stop switches.

Attitude Controller

The crew controls the attitude of the LM through a three-axis attitude controller (fig. 6). The approximate angular motions of the controller about these axes are ±13°. The sending of inadvertent thruster firing signals to the control systems is prevented by a mechanical deadband of ±0.25° about the center position and by detent switches at ±1.25°. In addition, direct switches have been incorporated about all three axes at ±2.5° and ±11.0°; the 2.5° switches are used for normal direct-pulse control and for operation of the landing point designator (LPD), and the 11.0° switches are used for emergency-direct thruster control. Rate command transducers are also provided in each of the three axes to provide signals proportional to controller deflection.

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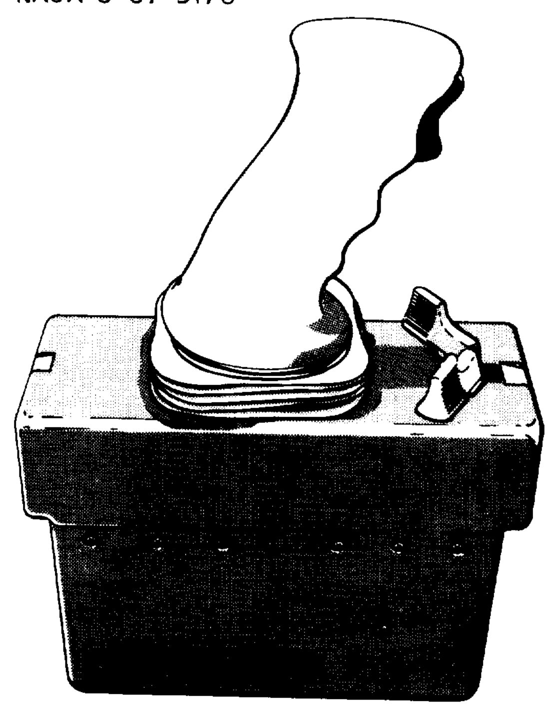


Figure 6. - Rotational hand controller.

Translational Controller

The LM utilizes the integrated translational and descent engine thrust controller (fig. 5). The T-handle controls firing of the reactions jets along the X-, Y-, and Z-axes except during powered descent when the X-axis (vertical firing) jets are mechanically prevented from operating. During powered descent, an up-and-down motion of the T-handle controls (in conjunction with PNGCS commands) the thrust output of the descent engine. The controller can also be used independently of PNGCS commands. The relationship between thrust output and controller position is also shown in figure 5.

Reaction Control Subsystem

The RCS is composed of sixteen 100-pound thrusters and a fuel supply arrangement as shown in figure 7. As indicated, the thrusters are arranged in quads mounted at 45° to the spacecraft axes to provide redundant attitude control thruster couples.

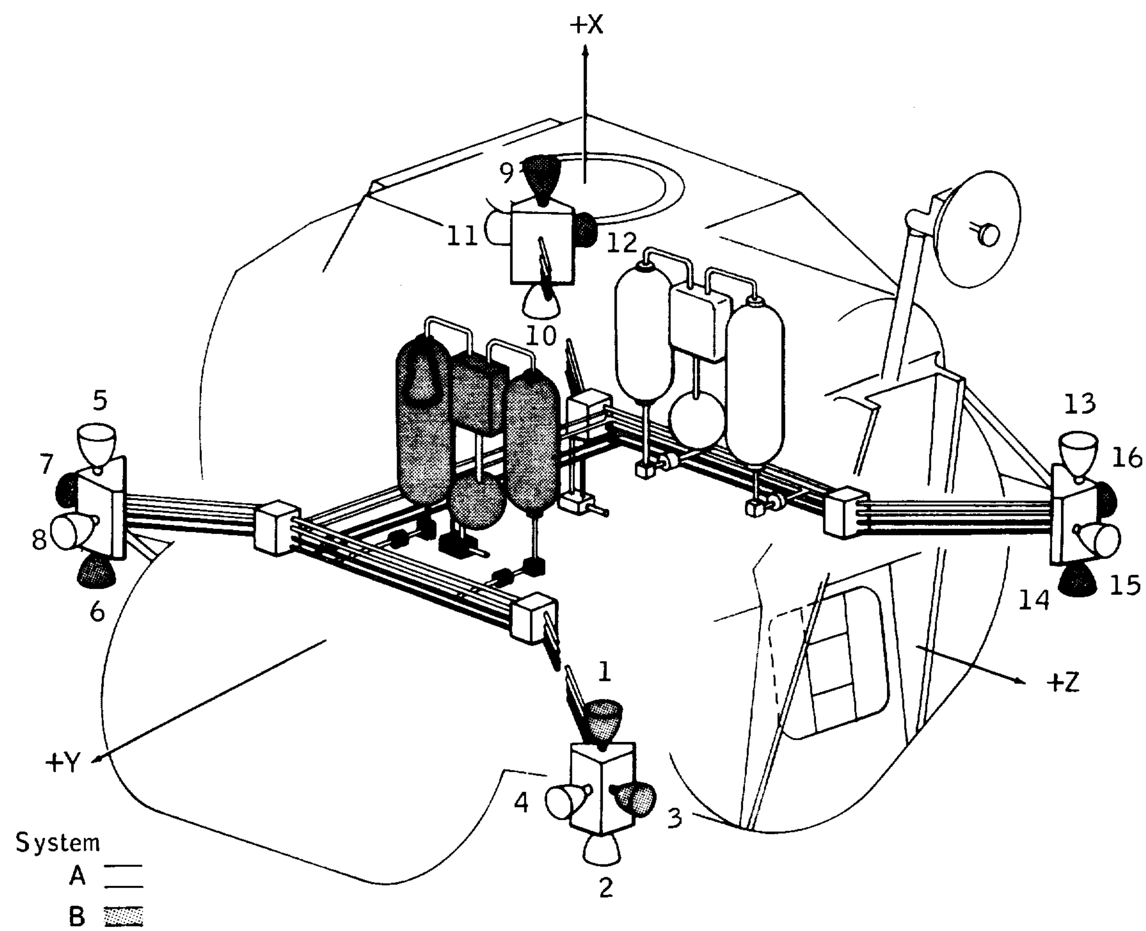


Figure 7. - RCS system.

Propulsion System

The main propulsion system of the LM consists of a descent engine and an ascent engine; each engine has an independent fuel system. The restartable descent engine has a thrust range of 1050 to 10 500 pounds. The engine is throttleable between 1050 and 6000 pounds, but above 6000 pounds it operates at a constant throttle setting giving a thrust of approximately 10 000 pounds. This engine, including the fuel system, remains on the lunar surface. A 3500-pound-thrust, restartable, constant-throttle engine is used as the ascent stage propulsion.

SIMULATION STUDIES AND TESTS OF LM CONTROL

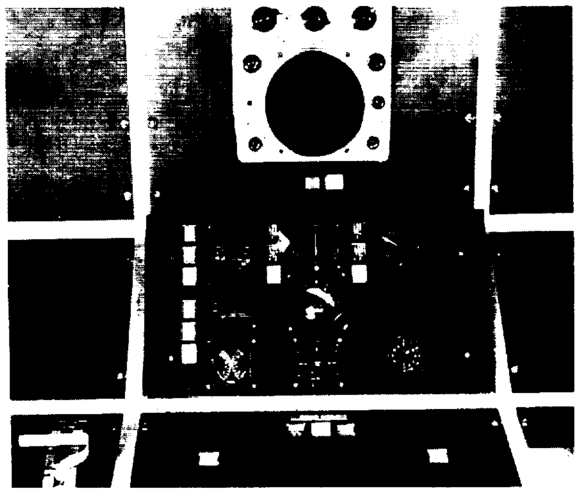
Background

The Manned Spacecraft Center (MSC), along with other elements of NASA, recognized the need for knowledge of LM control problems even before contract proposals for the LM were evaluated, almost 5 years ago. At that time, research facilities such as the lunar landing research facility (LLRF) of the NASA-Langley Research Center (LRC) and the lunar landing research vehicle (LLRV) of the NASA Flight Research Center were both in the conceptual stage. The Gemini flights, which could be used to study and verify docking and terminal rendezvous concepts, were not to be operational for several years thereafter. Thus, there were no flight vehicles available for even limited tests, and actual operational experience was too far in the future to be of benefit in the initial design of the LM.

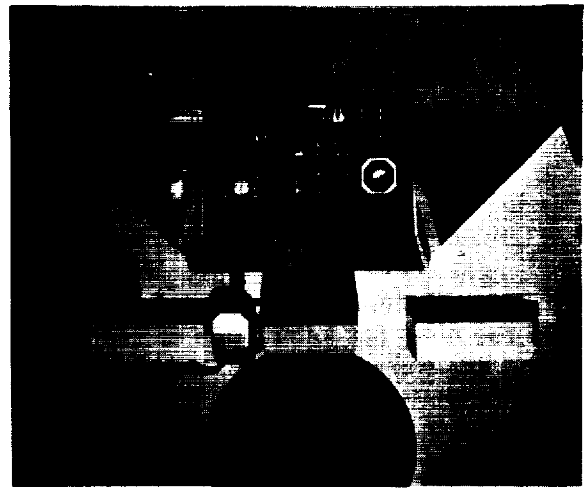
Because of these factors, the decision was made to obtain the needed information through simulation studies. A general simulation plan was outlined for investigating known areas of concern and for determining unknown problem areas. The simulations required by this plan were and are still being conducted at MSC and at LRC. Operational tests to supplement simulation results and to evaluate system designs were planned and are being implemented. In addition, provisions were made for the LM mission to be studied by contractors in the aerospace industry and by the eventual contractor of the LM spacecraft. The results of these studies form the basis for most of the discussion that follows.

Simulation Studies

The MSC and contracted simulation studies have examined, and are continuing to assess, pilot control problems associated with the final approach, landing approach, and touchdown phases of the powered descent. In other simulations, problems and concepts for terminal rendezvous, lunar orbit docking, and thrust vector control of the ascent stage have been studied. These studies have been a series in which both the simulation facilities and the fidelity of the simulated problem have improved as the knowledge of control and piloting requirements allowed better definition. In general, they have been implemented by coupling analog and analog-digital solutions of the equations to fixed-base, partial simulations of the LM cockpit (fig. 8). These simulations of the cockpit have ranged from the functional arrangement shown in figure 8(a) to the arrangement illustrated in figure 8(b), which is essentially identical to the current LM. Flight displays used have varied in both type and arrangement from study to study, but all have included the displays considered necessary to the control of the LM. Earlier simulations used oscilloscope displays for indicating the position of the LM with respect to a body or a place, but later studies have used a virtual image display for out-the-window cues.



(a) Early study configuration.



(b) Recent study configuration.

Figure 8. - Simulator flight displays.

The LRC, at the request of the MSC, conducted a study of LM-active docking using a movable-base simulator containing a partial mockup of the LM cockpit (fig. 9). The objectives of this study were to examine the pilot control problems associated with overhead docking and to determine the best docking aid for use by the pilot during the maneuver.

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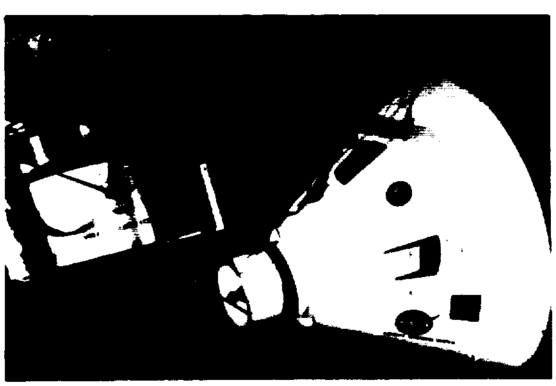

(a) LM and CSM docked.

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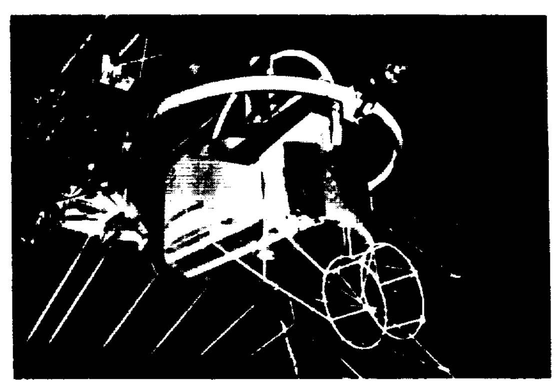

(b) LM simulation.

Figure 9. - Movable-base simulator used in docking studies.

Flight Test Programs

The purpose of the flight test programs conducted to date has been to investigate various aspects of the landing maneuver of the LM. The earliest of these was a study of the effect of lighting on landing approach trajectories. This test was conducted at Pisgah Crater, California, in 1963, using a Marine Corps H-34 helicopter. Another trajectory study was conducted by MSC in 1964 at Craters of the Moon, Idaho, using a T-33 aircraft. In this study, the trajectory was initiated at a high-gate altitude of 15 000 feet and terminated when the T-33 velocity profile was outside the expected trajectory limits. The purpose of the study was to determine the effects of earthshine lighting on visibility of the terrain and ability of the pilot to detect off-reference trajectory parameters during early phases of the final approach. The final test program to date in this series was performed at MSC with an H-34 helicopter; the purpose was to determine the ability of the pilot to null translational velocities at various altitudes using visual cues.

Lunar Landing Flight Test Vehicles

The LLRV under development and test at NASA Flight Research Center has been supported by MSC to provide a test bed for evaluating the system design on handling qualities, the operational problems associated with lunar landing, and the data for the training of pilots. The LLRF at LRC has been used to fill out portions of the test program unsuitable for the LLRV.

Earth-Orbital Flight Tests

Partial flight testing of the LM will be accomplished through a simulation in earth orbit of the rendezvous maneuver. The earth-orbital flights will give the crews an opportunity to assess the control system handling qualities of the LM during rendezvous and docking, and to evaluate the rendezvous and docking procedures.

Pilotage Analysis

A feasibility study of pilotage, which can be defined as navigation by determining position relative to known landmarks, surface characteristics, or apparent relative motion, has been completed under contract. The objectives of this study, confined to the descent phase, were to investigate methods of verifying position and elevation of landmark features for updating or correcting onboard lunar navigation maps; to evaluate devices, methods, and procedures for estimating distance and directions (and rates of these variables) to terrain features during coasting descent; and to evaluate devices, methods, and procedures for estimating distance and directions (and rates of these variables) to terrain features during powered descent, with emphasis on problems generated by the short time period.

LM CONTROL CHARACTERISTICS

From the previous discussion, in particular that of guidance operation in the mission phases, it is apparent that control of the spacecraft can be divided into two broad areas: (1) the crew controls flight through the automatic commands generated by the primary or abort guidance; (2) the crew controls the LM directly through either the primary or backup control paths. However, it is significant to note that the crew exercises complete control at all times inasmuch as they initiate key operations, monitor progress, and can override automatic commands at any time.

The LM control system has been designed to take advantage of the best attributes of the automatic guidance systems as well as the flying and decision-making capabilities of the crew. That is, automatic guidance commands for spacecraft control are used in those mission phases wherein spacecraft control is executed most effectively by this means; manual control is used in those phases where it appears that the crew can most effectively maneuver the spacecraft. Because of the limited knowledge of the control problems associated with the lunar landing mission, the simulations and analytical studies of the preceding section were conducted to obtain answers to some of the fundamental design and procedural problems. The subsequent discussion is based upon the results obtained from these studies.

Control During Descent

The investigations related to control of the LM during descent to the lunar surface have involved analytical studies, simulations by a pilot, flight tests, and digital program studies. The analytical studies have dealt primarily with the generation of guidance laws for midcourse correction, powered descent, and abort to rendezvous. Digital studies have been made to verify these guidance laws and, in general, have been confined to point-mass analyses. Previous simulation studies conducted, using pilots, have been concerned with manual control during the terminal landing maneuver and with the investigation of control problems from high gate to and including touchdown. Flight tests to date have examined specific problem areas such as lighting and velocity nulling. These simulations have also been heavily supported by detailed analysis of guidance techniques for both descent and ascent. Little work has been done in the investigation of control during separation and coasting descent. As the more immediate problems are solved, more study effort will be given to those phases which have received less attention.

LM-CSM Separation

While separation from the CSM has not been examined in simulations to date, it appears, based on test data and experience during the Gemini flights, that unlatching and separation using the RCS translational jets pose no control problems. The control powers and translational accelerations available to perform the maneuver are lower than at any other time in the mission. The maneuver is not time critical, however, and monitoring of precise attitude control following separation is not a requirement; neither are high separation velocities. Velocities of 1 fps or less can be used and then nulled after separation to allow the pilots to check the control system operation.

Descent Transfer Maneuver

Simulations of maneuvers similar to the descent transfer thrust maneuver have been conducted at MSC. Based on these simulations, no pilot control problems are anticipated, and the crew should have no difficulty in manually performing the maneuver. However, it appears best to have the crew command the thrusting maneuver through the primary guidance system since this procedure allows the crew to check out the automatic guidance system in a maneuver that can be aborted close to the CSM.

Coasting Descent

During coasting descent from injection to pericynthion, the primary tasks of the pilots are to monitor the primary and abort guidance systems for compatibility and to conduct those checks required to verify trajectory parameters. Simple pilotage techniques, which may provide the crew with a position reference with respect to lunar landmarks, have been developed analytically. One device being considered for use in pilotage techniques is a binocular viewer, as shown in figure 10.

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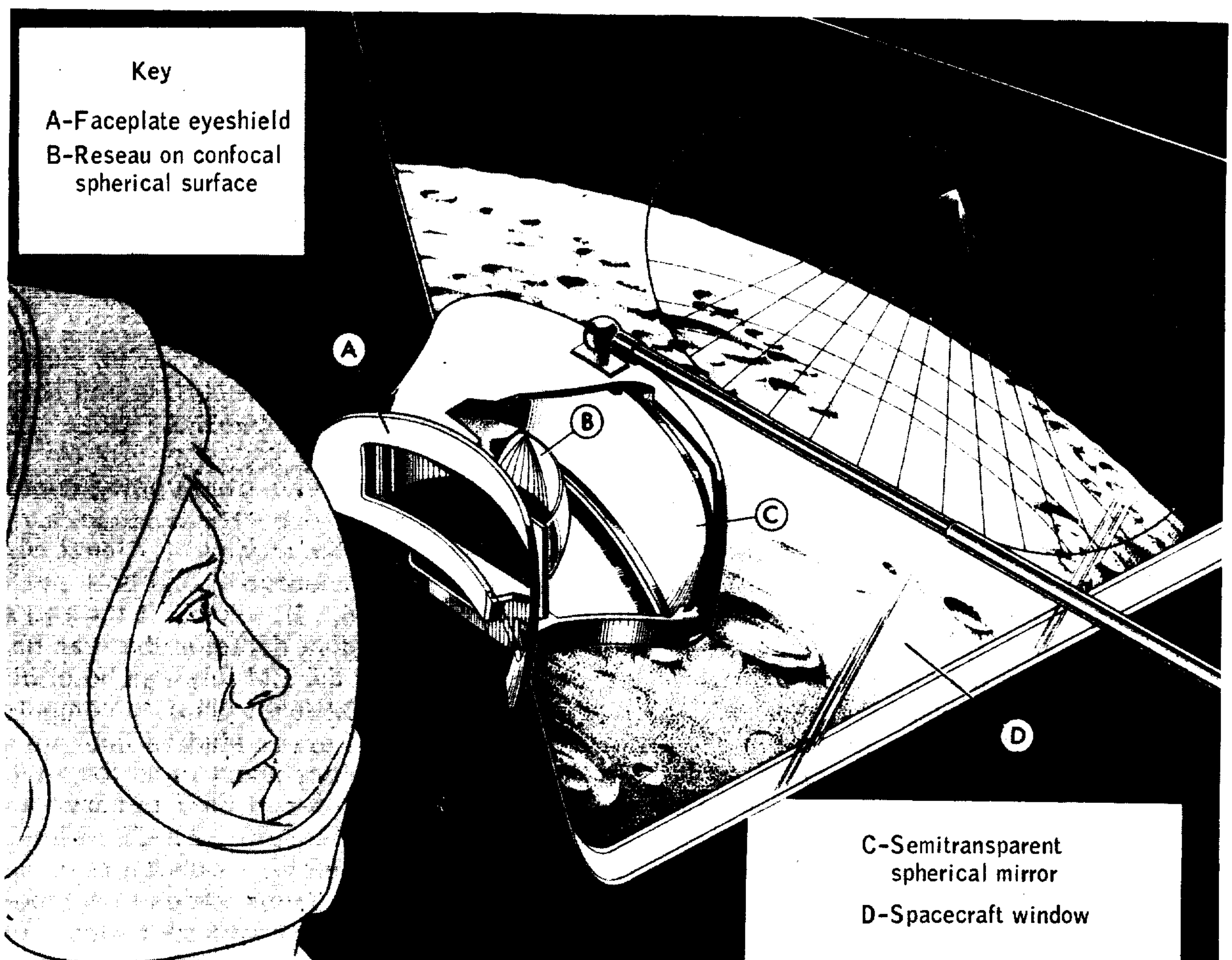


Figure 10. - Binocular viewer.

Studies have indicated that a pilot using the reseau grids shown in figure 10 can estimate altitude to within 1.3 to 1.9 nautical miles during the coasting descent. In addition, velocity can be determined to a reasonable degree of accuracy by measuring the time interval of passage between known landmarks. The rendezvous radar (RR) is also used to check position and velocity relative to the command module. Midcourse corrections to the trajectory, if required, are commanded by the crew using automatic signals from the primary guidance systems.

Powered Descent to High Gate

The crew initiates the powered descent at pericynthion using the commands from the primary guidance system. To preclude relatively large attitude changes at the start of firing, the descent engine thrust is initially set at a level of approximately 1050 pounds for a short period of time and then rapidly increased to nearly the full thrust of 10 500 pounds. This procedure allows the trim gimbal system (which can move at only 0.2°/sec) to accommodate any center-of-gravity offsets that might be present.

The LM pitch attitude at the initiation of the firing is approximately 95° back from local vertical (measured at pericynthion); thus the crew does not have the visibility necessary to determine the spacecraft trajectory except from the flight displays and the guidance and navigation computer displays. By using the X-axis override, the crew can achieve visibility by yawing the LM about the thrust vector to examine the terrain directly below and somewhat to the front of the spacecraft. This maneuver, however, could cause an undesirable loss of communications with the CSM and the earth unless caution is exercised.

Guidance Monitoring

One function that the crew performs during the guidance monitoring phase is determining that the primary system is operating correctly and detecting and isolating guidance systems degradations before adverse flight safety conditions arise. A high degree of confidence that the PNGCS and AGS are operating satisfactorily can be achieved by comparing the trajectories (in particular the velocities) being computed by the primary and abort guidance systems to the nominally expected trajectory. As long as the two trajectories agree with one another and with the nominal (at least within some predictable difference), the crew has reasonable assurance that both systems are operating properly and that the descent can be continued. However, if the two systems disagree, the crew must decide which system has failed, a decision that can be made correctly only by using a third independent system. In the LM this can be either the RR in the early phases of descent, the landing radar (LR) (at least prior to computer update), or perhaps the deep space flight network. Initial error studies indicate that extremely large trajectory deviations are necessary before abort is no longer possible. For example, it requires a trajectory deviation of the order of 30σ to cause an unsafe flight condition as late as 300 seconds into the powered descent. At 425 seconds into the descent, a trajectory deviation of 10σ is required before a safe abort is no longer possible. Generally, trajectory deviations of 30σ arise from complete component failure, and trajectory deviations of 10σ represent highly degraded operation. In contrast to this, the analysis showed that the probability of detection was over 99 percent for component degradations causing 5σ trajectory deviations. The results of the analysis have been verified in simulations which indicated that the crew could detect those failures leading to unsafe abort conditions long before safety of flight was compromised.

Final Approach to Low Gate

By the time the LM reaches the high-gate attitude, the pitch attitude is of the order of 70°. Rotation from this attitude to approximately a 45° pitch back initiates entry into high gate and into the final approach phase of the powered descent. While the first phase (braking) covers most of the powered descent trajectory and is designed to provide a nearly optimum reduction in velocity for the least expenditure of fuel, the final phase is shaped so that the landing site is visible to the crew. Entry into this phase is attained through explicit guidance and, in addition to providing visibility of the landing site, is planned to allow the approach to the landing site to be made at a deceleration level considerably lower than the maximum descent engine thrust capability. The advantage of the lower deceleration, obtained by reducing the throttle level of the descent engine, is that the rate of velocity change becomes more in line with the ability of the crew to keep track of the descent as it occurs. The phase covers 6 to 8 nautical miles, and the velocity decreases from about 700 fps to approximately 50 fps at low gate (entry to the landing-approach phase). In addition to monitoring the large changes in velocity and altitude in this short phase, the crew must also begin evaluating the suitability of the landing area, choosing a desired landing position, and preparing to take over and manually fly the final phase of the descent maneuver if it appears advisable. All of these events take place in a time period roughly equivalent to the time available to a jet aircraft pilot between the final checkpoint and touchdown during an instrument approach.

Various trade-offs are associated with selection of a trajectory for the final-approach phase of the powered descent. One of these is obviously that of fuel expended versus visibility afforded the pilot during this maneuver. Early studies of this problem (ref. 2) produced the results shown in figure 11, which illustrates look angle (depression angle between landing site and straight-ahead-design eye position) to the landing site as a function of time-to-go for four different trajectories. Although the terminal conditions are somewhat different from the ones being considered at this time, the general picture remains essentially correct. The dotted line at the 26° look angle indicates the lower viewing limit of the present LM window. Obviously, the fuel-optimum (ΔV of 5630 fps to low gate) trajectory provides the crew with zero visibility of the landing site until low gate is reached and thus is unacceptable for the primary purpose of final approach. Trajectory A in figure 11 shows the visibility afforded for a high gate of 15 000 feet, and trajectory B shows the visibility for a high gate of 10 000 feet. Both roughly afford the same degree of visibility, but trajectory A requires a characteristic velocity of 5800 fps to low gate, and trajectory B requires 5760 fps. The visibility of trajectory C gives a good look angle, but requires a characteristic velocity of almost 5850 fps to low-gate conditions. Hence, of the trajectories shown, the best choice for fuel expenditure is the fuel optimum, and the best choice for visibility is trajectory C, which is also the worst choice for fuel expenditure. The selection lies between trajectories A and B with visibility and fuel as the determining criteria.

NASA-S-67-5183

Trajectory	Transfer altitude, ft	Pitch angle, deg	ΔV, ft/sec
A	15 000	50	5800
B	10 000	50	5760
C	10 000	40	5854
Fuel optimum	--	--	5630

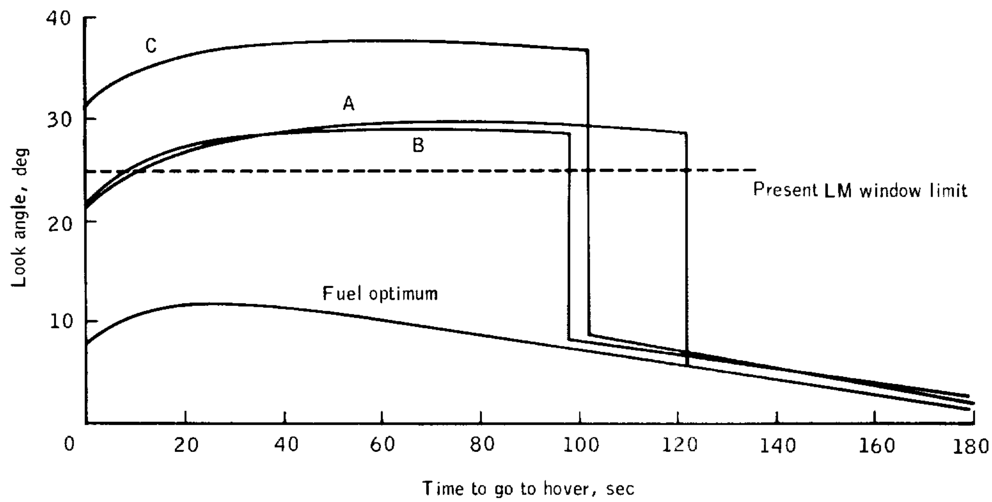
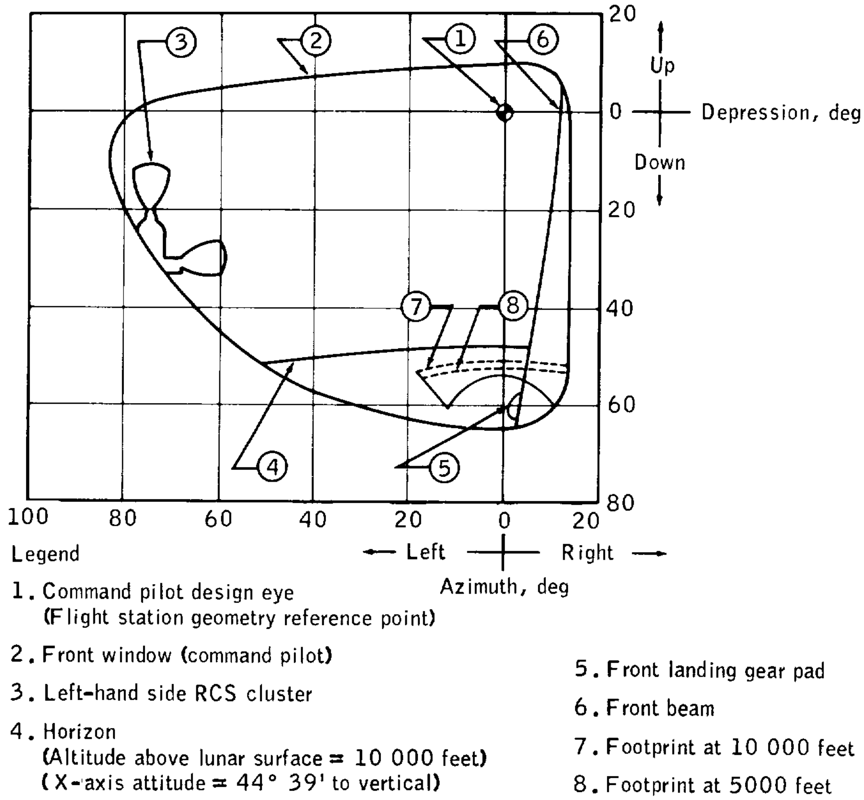


Figure 11. - Effect of transition altitude on look angle and viewing time of landing site.

The second trade-off to be considered is that of viewing time during final approach versus ΔV expended. In this case, the high gate of 15 000 feet for trajectory A gives about 25 seconds more visibility to low gate than trajectory B, but it also requires 40 fps more ΔV to low gate than does trajectory B. Experience in fixed-wing aircraft indicates that the 10 000-foot altitude is probably the earliest that the crew can make an efficient assessment of the landing area.

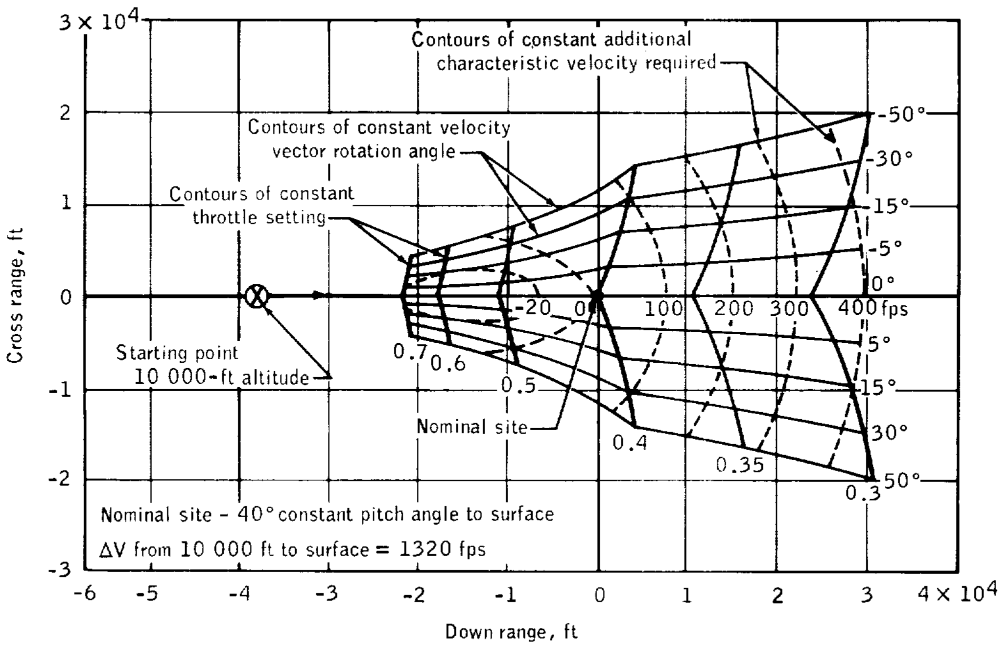
The constraints placed on crew visibility by the design of the LM window and by trajectory parameters make the viewing of the programed landing site a major problem. As shown in figure 11, the landing site lies in the lower 5° or 6° of the window for the reference trajectory. This is shown even more graphically in figure 12(a), which describes the angular view afforded the command pilot for the design eye position. To complicate the problem, the LM is tilted back from vertical on the order of 40°, and the command pilot is required to view the landing site with his eyes down 60° from the straight-ahead eye position. This is a difficult position from which to determine the suitability of the landing site area; further, the command pilot must also monitor the flight instruments to evaluate the trajectory. Because the landing area is located 60° down from the straight-ahead eye position, the command pilot is required to scan continuously up and down between the flight displays and landing area. Acquisition of either viewing area requires time, which is extremely critical when the short time period of the final-approach phase is considered.



(a) Pilot view during final approach phase.

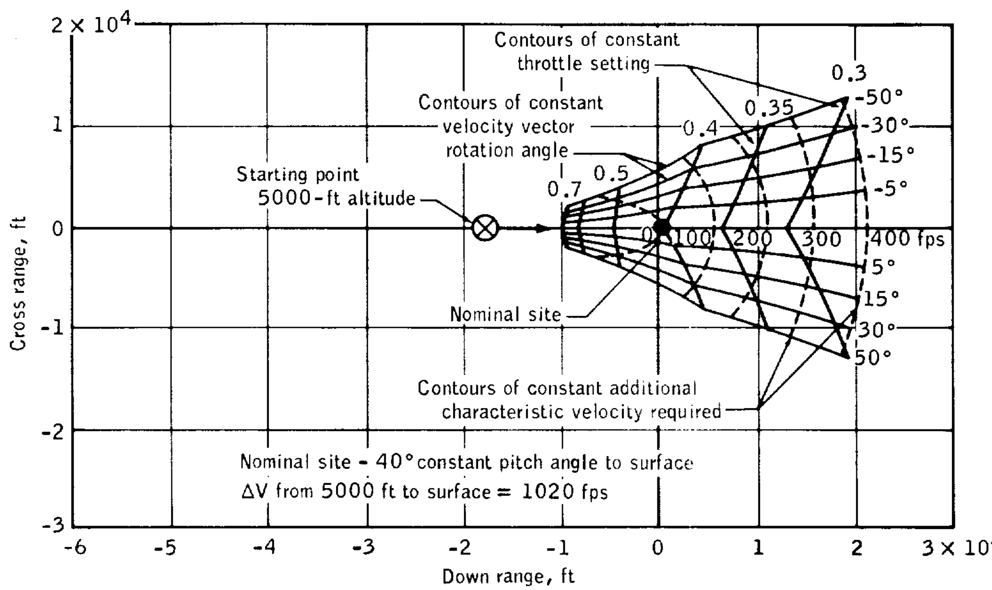
Figure 12. - Factors influencing changes of landing site.

During this time, the command pilot is also confronted with the possibility that the landing area targeted into the guidance system may be unsuitable for landing. In the event this occurs, the command pilot must also assess the available footprint area for a satisfactory landing area, determine its location relative to the spacecraft, retarget the computer, and initiate a maneuver to transfer to the new landing area. If time and fuel were not critical, these events would be handled with ease by using standard piloting procedures. As an idea of the complexity of this particular task, figure 12(a) illustrates the available footprint for an expenditure of the 100-fps characteristic velocity.for landing site changes at altitudes of 10 000 and 5000 feet referenced to the window of the command pilot. As shown in figure 12(a), the 100-fps contour lines for landing site changes at 10 000 and 5000 feet differ perhaps by 1°, and both lie only 3° to 4° above the referenced landing site at the straight-ahead view. This situation, coupled with the fact that the spacecraft is pitching during descent, indicates that the command pilot must be extremely careful in his selection of a new landing site in order not to exceed the ΔV allotment for the maneuver. The situation relaxes somewhat for out-of-plane changes, but the time required to assess the site, address the computer, and initiate the maneuver limits the choice. As shown in figures 12(b) and 12(c), the changes in range for the 100-fps expenditure are very nearly the same for each altitude. The changes are 7000 feet and 5000 feet for the maneuver at altitudes of 10 000 and 5000 feet, respectively. However, the change at the 10 000-foot altitude is largely academic because the crew is quite unlikely to attempt a landing area redesignation at that point since very little time is available to assess the targeted site. Even if an immediate assessment of the site showed it to be unacceptable, it is likely the command pilot would wait a period of time so that a better assessment of alternate landing areas could be made. From the tests and simulations made to date, site redesignation, if required, is not likely to take place at an altitude higher than approximately 5000 feet.



(b) Landing footprint at a 10 000-foot altitude during the final approach phase.

Figure 12. - Continued.



(c) Landing footprint at a 5000-foot altitude during the final approach phase.

Figure 12. - Concluded.

The crew normally will perform the maneuver for a change of landing site with the automatic guidance even though simulation studies of this portion of the descent show that the crew can perform this task manually. In this manner, the ability of the pilot to visually assess the terrain for a suitable landing site and the ability of the computer to rapidly correct the trajectory to go to the landing site may be used to full advantage. This is done by providing the pilot with a lubber line called the landing point designator, which consists of two lines, one on the inside of the window and the other on the outside of the window. The LPD (fig. 13) is marked in 2° increments showing the depression angle below the body +Z-axis. At the same time, the guidance computer keeps track of the depression angle, which can be read by the pilot from the computer display panel. Operation of the LPD and LPD-depression-angle reading consists of having the right-hand pilot note the depression-angle reading and call out the reading to the left-hand pilot, who alines his eye such that the inner and outer portions of the LPD line up. The left-hand pilot then locates the landing site by looking through the LPD depression angle. If the landing site appears satisfactory, the left-hand pilot merely monitors the computed LPD reading (being called out by the right-hand pilot at discrete time intervals) and the visual LPD/landing-site lineup until the point of manual takeover. If, however, it becomes necessary to change the landing site, the left-hand pilot changes the trajectory through the rotational controller. Each discrete forward or backward movement of the controller changes the vector between the LM and landing site by 0.5°, and each right or left movement of the controller changes the vector by 2.0°. Azimuth corrections are made only to aline the window LPD with the site. Corrections in this plane are estimated, since the LPD computer reading is always zero regardless of where the landing site is located.

NASA-S-67-5187

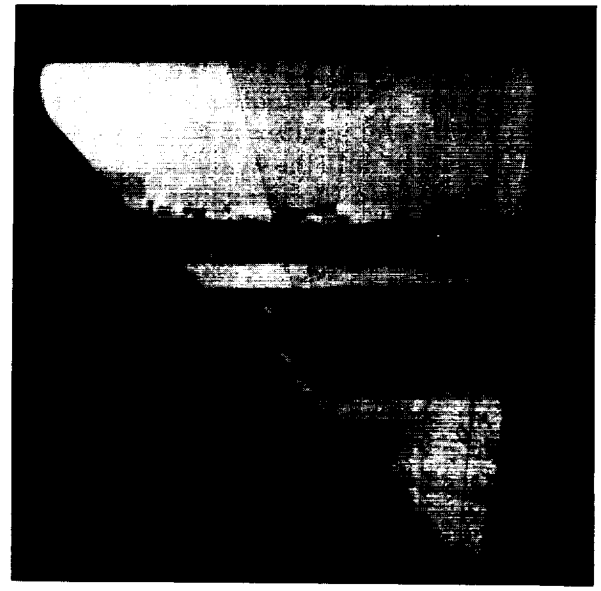


Figure 13. - Landing point designator lubber lines.

The crew has the option of either completing the landing with the automatic guidance through the LPD or disengaging the automatic system and continuing under manual control. At this time, the primary landing mode is assumed to be a manual function. However, the guidance system is programed such that the crew can complete the landing using the LPD and guidance combination. Simulation studies indicate that the pilot should have no difficulty in controlling the LM to the desired landing site even though the semiautomatically controlled landing results in somewhat faster velocities and decelerations than the pilot would normally attempt in the manual mode. The velocities and decelerations, however, still remain well within the range in which the crew can take over control of the spacecraft at any time in the landing.

LM Landing

After taking control of the LM from the guidance system and rotating the spacecraft to the vertical, the crew must, in a period of less than 2 minutes, make a final assessment of the landing site, maneuver to it, probably conduct a final update of the LM guidance computer (LGC) velocities and altitude with the landing radar, perhaps even visually null the translational velocities, and descend to the lunar surface. This must be done in the presence of relatively unknown terrain features, uncertain lighting conditions, and possible dust obscuration while descending to the surface. Added to these factors is the possible danger of fuel depletion.

The landing maneuver starts with entry into low gate. At this time, the procedure is for the pilot to rotate the LM forward to about a 15° pitch-back attitude when the forward velocity is nearly 60 fps. The LM at this time has a flightpath angle of -14°, an altitude of 500 feet, and is approximately 1200 feet downrange from the landing site. These conditions are not firm and could change as knowledge of the problems associated with the landing maneuver become better defined. However, the takeover conditions cited appear to afford the crew the best chance of completing the maneuver with a minimum of risks. The particular conditions selected were taken from the following considerations:

1. The crew is allowed adequate time to assess the landing site.

2. The footprint capability is larger with higher forward velocities.

3. The landing radar can be used to update the computer with a maximum of accuracy.

4. The crew has excellent visibility throughout the maneuver, barring obscuration.

5. The landing site change is relatively simple.

6. The spacecraft is in an excellent attitude for abort, if this becomes necessary.

7. The trajectory is shaped to remain above the minimum abort altitude for a maximum length of time.

Handling Qualities

One of the first problems encountered in studying the LM was that of determining what constituted satisfactory attitude control system handling qualities for the pilots to perform the landing maneuver. Initial studies of the landing approach and touchdown maneuver (ref. 3) indicated that satisfactory handling qualities for a rate-command control mode, operating with on-off thruster firing logic, could be obtained over a wide range of controller sensitivities with the control powers available. The area of interest relative to the LM handling qualities determined in simulation studies of the landing maneuver is shown in figure 14.

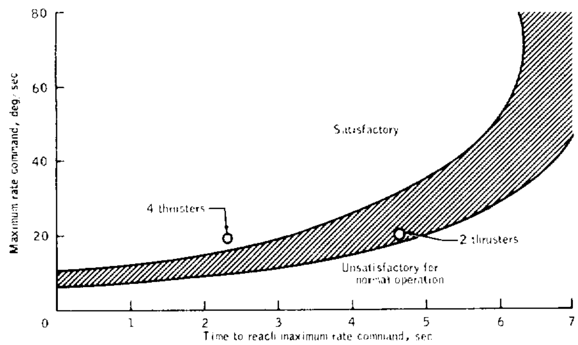


Figure 14. - LM attitude control system handling qualities.

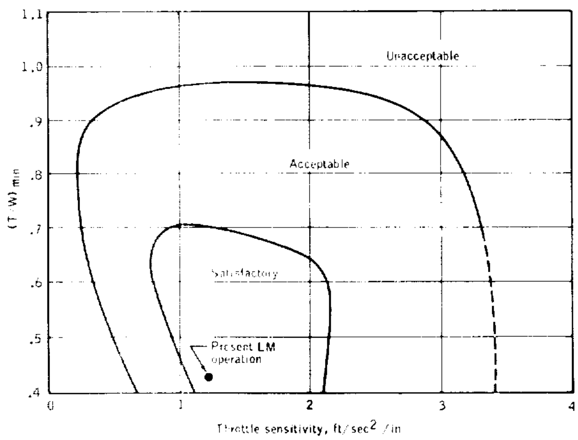
While the rate-command mode provided satisfactory handling qualities with the 100-pound reaction jet thrusters, it appeared that better control could be exercised by the crew using an RCAH mode of attitude control because the command pilot would not be required to compensate continuously for rate deadband drifts while holding a given attitude. As long as the control powers remained on the order of 10 deg/sec^2, this assumption proved correct, although with the small rate deadbands (0.2 deg/sec), the pilots had little difficulty in maintaining good control with the rate-command mode. However, increases in the inertias of the LM caused a decrease in control effectiveness and consequently tended to move the operating points nearer the boundary of the satisfactory area. The result was that the RCAH mode began exhibiting undesirable overshoot characteristics about the command attitude. With the primary system, the overshoot was correctable to some extent by programing the digital autopilot to inhibit the engagement of the attitude-hold feature until the rotational rates were less than 1 deg/sec. The same arrangement could be used in the backup control path, but in the interests of maintaining simple circuitry and high system reliability, the decision was made not to incorporate the inhibiting circuitry. Should the backup path be used for landing, the crew either must use the RCAH mode and accept the overshoot, or must abort the mission.

The present operating point for both single- and double-couple thruster operation for the LM is indicated in figure 14. As can be seen, the control system has marginally satisfactory handling qualities with a single couple, but good handling qualities with double-couple operation. These operating points have been verified by the LLRV and LLRF. At this time, it appears that the LM control system is satisfactory for the task, if the pilot exercises caution in making attitude maneuvers.

Descent Engine Throttling Characteristics

The range over which the descent engine should be throttled also presented a problem during the design stage of the LM. It was necessary to provide an upper thrust level that would accommodate the deceleration required during the braking phase, and yet have a low enough thrust to give satisfactory handling qualities for the landing maneuver. The chief problem encountered by the designers was that the LM weight decreases by a factor of about 2 during descent. Because the descent engine is constrained to a finite throttling ratio, the possibility existed that the thrust-to-weight ratio (T/W) demands in early descent would be incompatible with the T/W requirements at landing. A secondary problem was that of throttle sensitivity during landing. If the throttle were too sensitive, it would be extremely difficult to establish a precise descent rate; if the throttle were too insensitive, the pilot would be required to make large motions with the throttle controller to control descent rate. A series of tests was conducted at a contractor facility, using a fixed-base simulator, to examine these problems and to resolve the question of throttle sensitivity and thrust-to-weight ratios.

The satisfactory boundary for descent engine control as a function of minimum engine T/W and throttle sensitivity is shown in figure 15(a). The satisfactory region (Cooper scale rating) is roughly bounded by a minimum T/W of 0.7 for throttle sensitivity ranges between 1.0 and 2.0 fps^2/in. However, the satisfactory boundary in figure 15(b) indicates that the T/W should be less than 0.7 and the throttling ratio should be 3 to 6. These tests indicated that both the sensitivity and the throttling ratio for landing were at odds with the engine requirements for early powered descent. The solution to this problem was to apply a nonlinear thrust output to the controller deflection curve shown in figure 5(a). With this arrangement, the controller sensitivity and throttling ratio below the soft stop give the desired handling qualities for landing and also provide the thrust level required during early powered descent. The throttle is quite sensitive above the soft stop, but this is of small consequence because the thrust is controlled by the primary guidance system in this region; normal landing maneuvers do not require throttle operation above the soft stop. In any event, even if the pilot were required to operate the throttle above the soft stop, the thrust level is not critical.



(a) Throttle sensitivity versus minimum thrust-to-weight ratio.

Figure 15. - Descent engine throttle handling qualities.

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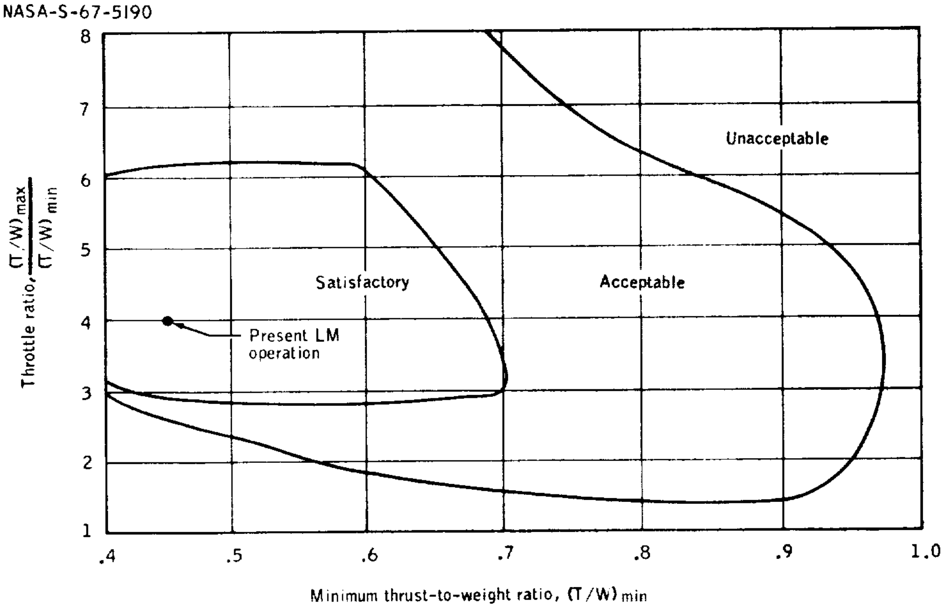


(b) Throttle ratio versus minimum thrust-to-weight ratio.

Figure 15. - Concluded.

Present operating points in the LM below the soft stop are at a minimum T/W of about 0.44, a throttling ratio of 4, and a throttle sensitivity of approximately 1.2 ft/sec^2/in., all of which are within the satisfactory region of figure 15(a). Both the LLRV and LLRF have verified this throttle design.

Selection of Landing Trajectory

One consideration in selecting the landing trajectory after entry to low gate is that of the minimum altitude at which staging and abort are possible. If an abort is initiated with the descent engine, the T/W capability of the engine is large enough so that almost any reasonable combination of altitude and descent rate can be accommodated. However, if the ascent engine must be used for abort, the staging time and T/W of the ascent engine clearly define the permissible altitude-descent rate profile for safe abort. This profile is shown in figure 16, which plots the minimum abort altitude as a function of descent rate for two different staging times (the staging times shown include the time required for the command pilot to decide whether an abort is necessary). The present altitude-descent rate profile for the last part of the landing-approach trajectory is shown by the dashed line. The proposed trajectory crosses the 4-second staging time profile slightly below 200 feet and the 2-second staging time between 70 and 80 feet. Some care must be exercised in selecting the altitude-descent rate profile: if the descent rate is too high, a landing commitment must be made at a high altitude because of the staging limitations; if the descent rate profile selected is too low, the trajectory is expensive with aspect to fuel consumption. With the present knowledge of staging times and operational considerations, the profile being considered appears to meet the known requirements for either manual or automatic landing. An additional point is that once the minimum abort altitude has been penetrated and the sooner the spacecraft is landed, the better the chance for crew safety and mission success. The limitation to the permissible descent rate is that the landing gear criteria should not be exceeded at touchdown.

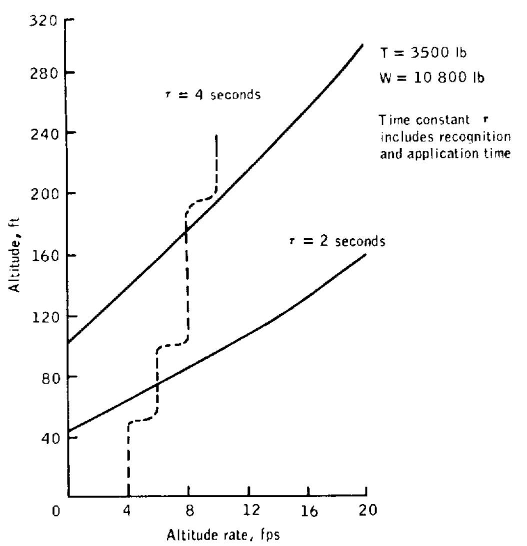


Figure 16. - Minimum abort attitude using ascent engine.

Manual Velocity Update Prior to Touchdown

The antenna for the landing radar is located on the bottom of the descent stage near the descent engine nozzle. It has two tilt positions — one directed back from the X-axis by some 25°, and the other parallel with the X-axis. The first position is used during the braking phase, while the second position is used throughout the remainder of the descent. Radar data are used to drive the cross-pointer velocity meter and the altitude/altitude-rate tape indicator. The radar information is also sent to the computer where it is used to update the inertially derived information and is processed for display (at the discretion of the crew) on the velocity and altitude indicators.

Accuracy of the radar is of importance during the entire descent phase because the crew also uses the data to monitor guidance performance. Although a velocity error of 5 or 10 fps may be of little consequence during a large portion of the descent phase, a velocity uncertainty of the order of 1 fps during the final descent and touchdown may affect both crew safety and mission success. This is because the radar errors, added to processing, display, and pilot control errors, may be sufficient to cause the design velocity envelope to be exceeded. Studies are presently underway to define the radar uncertainties so that landing and control procedures can be adapted to obtain the best possible radar data for as long as possible during the descent.

As a backup procedure which can be used in the event of landing radar failure or simply because the landing radar cannot operate satisfactorily near the lunar surface, it is possible that the crew can look out the window and manually update the computer inertial data by visually nulling translational velocities. Tests conducted using an H-34 helicopter indicate that the accuracy to which the velocities can be nulled is excellent, and the time necessary to perform the nulling is compatible with the LM fuel budget. The results of several runs made during the H-34 tests are contained in figure 17. As indicated, the average velocities are fairly low, and the spread between runs is relatively small. It may be noted that if one velocity is high, the other two are generally low. The time required to null the maneuver averaged about 15 seconds, which appears to be reasonable for the task. However, other data from this particular task indicated that performing the maneuver at an altitude of 200 feet seriously degraded performance

NASA-S-67-5192

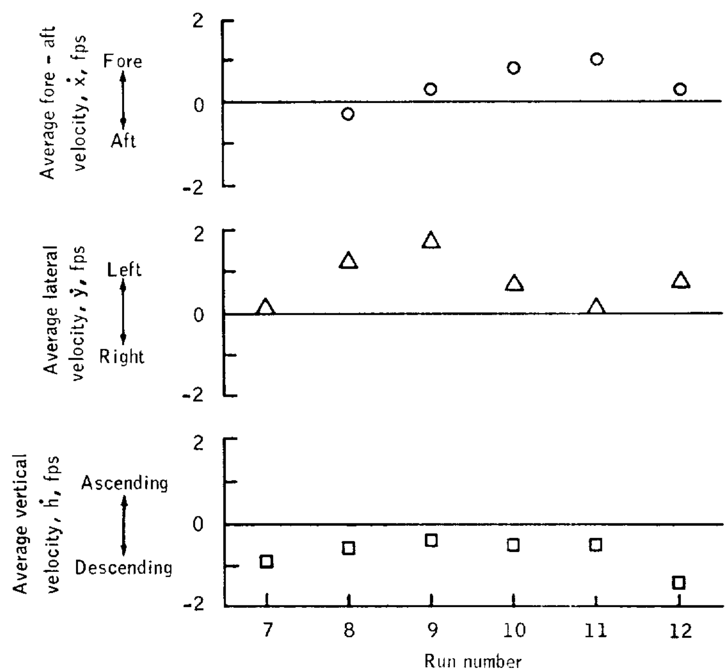


Figure 17. - H-13 helicopter hovering test results.

and increased the maneuvering time. A number of test factors influenced the test results: (1) the H-34 helicopter affords much greater visibility than the LM; (2) the test area (Ellington Air Force Base, Texas) contains familiar terrain features and thereby provided visual cues that could not be expected in the actual mission; and (3) the tests were flown in a 15-knot crosswind, which complicated the control problem. The results of these tests were verified using the LLRV with the correct window view. The PNGCS is, therefore, programed so that the pilot can manually update the LGC inertial data, if the situation becomes necessary. The remaining unknown is that of whether visibility conditions near the lunar surface will permit the technique to be used at an altitude low enough to provide information of the quality necessary for updating the computer.

Final Vertical Descent

Descent to the lunar surface must be accomplished within a very short time after the final update of the computer inertial data to prevent the accumulation of errors in altitude and velocity data. The descent rates used in the maneuver must be carefully selected to prevent the use of excess fuel and yet allow the pilot time to control the forward and lateral velocities, monitor altitude, sense and react to contact indication, and shut down the descent engine before the landing gear contacts the lunar surface. The control task is expected to be no more difficult than landing a vertical-takeoff-and-landing aircraft if satisfactory contact (visual) conditions can be assured. The LLRV landing experience has indicated this assumption to be correct. However, considering the uncertainties associated with the present knowledge of the lunar surface, it is almost imperative that the procedures used be able to accommodate the worst problem control case, probably a complete instrument landing through the final vertical descent. The landing, of course, must be made within the landing gear design envelope, which is presently set as ±4 fps horizontally for a vertical velocity between 0 and 7 fps, decreasing linearly to 0 fps horizontally for a vertical velocity of 10 fps (fig. 18). Attitude and attitude-rate constraints are set at 6° and 2 deg/sec, respectively.

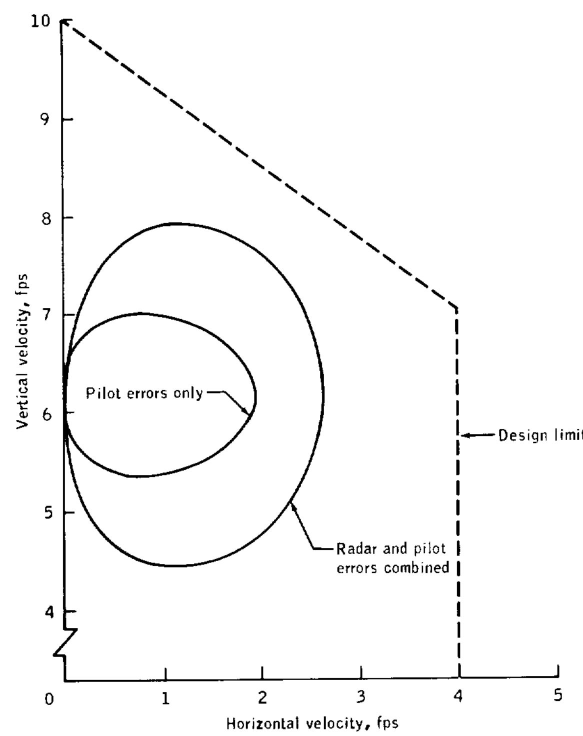


Figure 18. - Expected touchdown velocity errors.

Touchdown Velocity Control

At the present contemplated descent rate in the order of 4 fps, the time allowed for manual control of horizontal velocities appears adequate. The simulation studies conducted to date indicate that control of velocities is generally satisfactory if the flight displays and their locations are compatible with the task. One of the factors influencing velocity control at touchdown is that the critical displays are time shared during other portions of the mission. The cross-pointer velocity display, for example, is used to indicate forward and lateral velocity during landing and line-of-site angular rates during rendezvous radar operation. Consideration must be given to the requirements of each mission phase when selecting the scaling and resolution of this flight display. At the present time, the indicator has a symmetrical scale about the null point with scales of ±200 and ±20 fps. The tapes for altitude and altitude rate are time shared with range and range rate. However, the sharing of instruments does not appear to be a critical problem at this time.

With the present control powers and error-free velocity sources used to drive the displays, the manual control of touchdown velocity does not appear to be a difficult task. The more recent simulation studies conducted at MSC indicate that the pilots can control the spacecraft within the design envelope by using either the primary or the backup control path, although control is a little more difficult with the backup system, and the vertical velocity data spread is somewhat greater. The studies have not included velocity errors in the equation mechanizations, and the only errors of any kind existing in the simulation were those associated with the velocity display, which were of the order expected in the actual LM display. The expected radar errors, however, can be added statistically to the simulation test data, and a reasonably good approximation of expected performance in the presence of these errors can be obtained.

The results of a recent simulation study were subjected to an analysis to determine the statistical characteristics of these data. The analysis, which included combining the expected radar velocity errors with the test data, indicated that the variables were generally symmetrical about some mean value, but were not, except for pitch and roll attitude, normally distributed. Adding the radar error to the translational velocity did, however, change the distribution of these variables to normal. The reason for the velocity data not being normally distributed without radar errors is that the test data were grouped for analysis. Individual test data (for each pilot) generally tested normal. Additional tests also showed that the variables under examination were, in general, statistically independent of one another.

The most important result of the analysis was that the forward, lateral, and vertical velocities were essentially statistically independent. The fact that the translational velocities exhibited this characteristic was not entirely unexpected because it had been noticed that the pilots generally tended to control these velocities independently. The independence of roll and pitch attitudes and rates and of forward and lateral velocities was unexpected because the horizontal velocity is controlled entirely by pitch and roll attitude. There is no explanation for this decoupling; but it is, nonetheless, quite significant in examining other problems associated with the LM, particularly in the analysis of landing gear performance.

Numerical Results of Attitude Control

The results of the statistical analysis performed on the test data previously mentioned are shown in table I. The limits x_1 and x_2 represent, with a confidence level of 0.94, the upper and lower bounds containing 95 percent of the expected variation in the variables shown. Since the variables are not normally distributed, the 95 percent limits, which represent very nearly the 2σ value for a normal distribution, are used instead of means and standard deviations from the mean because the 95 percent limits are more meaningful statistically. The means can be obtained by averaging the two values. The indicated limits are well within the present design envelope. The attitude rates are not shown, but they were essentially symmetrical about zero and generally less than ±1.0 deg/sec/axis.

TABLE I. - RESULTS OF STATISTICAL ANALYSIS

Limits	Velocity, fps			Attitude, deg		
	Vertical	Forward	Lateral	Pitch	Roll	Yaw
x_1	4.89	-1.38	-1.49	-2.34	-2.18	-2.86
x_2	6.88	1.41	1.54	2.40	3.39	1.57

Numerical Results of Velocity Control

The velocity control was also within the velocity envelope, as is shown in figure 18. Without radar errors being accounted for, the horizontal velocity vector was less than 2 fps, and the vertical velocity was between 5 and 7 fps about 99 percent of the time. Adding the expected 3σ radar velocity errors increased the 99 percent probability area, the horizontal velocity being slightly less than 3 fps and the vertical limits being between 8 and 4.5 fps. Hence, even in the presence of expected radar errors, the pilot is capable of controlling the LM within the design velocity limits. However, these results are conditioned by the test conditions and may, in the actual case, be slightly optimistic.

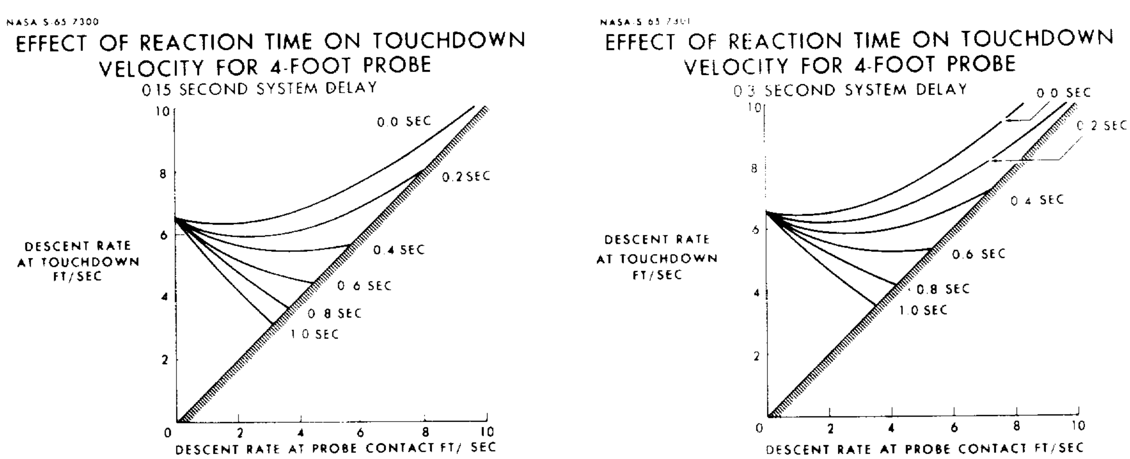
Summary of Touchdown Control

The test data indicate that pilot control of touchdown velocity and attitude is satisfactory with the test conditions used. These data show that the variables are expected to have a high probability of being within the design limits. The statistical analysis also indicates a low probability of two or more velocities being out of tolerance simultaneously. In fact, a review of the raw data of two simulation studies fails to reveal that more than any two variables (velocities, angles, angular rates) are out of tolerance at the same time. The simulations, however, used the primary control path, which incorporates the rate-of-descent control mode. Because of the direct thrust control, some degradation of performance, particularly in the vertical axis, can be expected when the landing is made with the backup control path. The degradation should not be large, but a wider spread in touchdown vertical velocity would be expected based on the previous simulation results where this control mode was utilized.

Descent Engine Shutdown

A shutdown of the descent engine before contact with the lunar surface is desirable because there is the possibility of damage to the descent engine nozzle if the engine is firing within 1 foot of the surface. In addition to the danger of the ascent stage being damaged by flying debris, there is also the chance that landing stability could deteriorate because of disturbance torques created by the nozzle collapsing prior to engine cutoff. Because the engine nozzle is slightly more than 1 foot off the surface when the landing gear touches down, it would normally be sufficient, from the viewpoint of the danger just cited, for the pilot to have the engine cutoff at the touchdown point. If sufficiently accurate altitude information (with the proper resolution) were available, the pilot could initiate engine cutoff just prior to touchdown and reduce the possibility of descent-engine and ascent-stage damage. However, because of radar inaccuracies and the computer processing inaccuracies after final update, the altitude information displayed to the pilot may be in error. This creates a hazardous situation in that engine damage or structural damage to the spacecraft could result from the engine being shut off either too near or too far from the lunar surface. To insure engine cutoff at a specific altitude, the feasibility of attaching four frangible probes to the LM landing gear pads is being investigated. The purpose of these probes is to sense a specific altitude above the surface and to provide an engine cutoff signal for display to the pilot.

An analysis of the manual engine shutoff problem was made by using the graphs of figures 19(a) and 19(b), which show how touchdown velocity is affected by pilot reaction time to the engine shutoff signal for 3- and 4-foot probe lengths. Because of the somewhat uncertain system delay times associated with shutting off the descent engine, delay times of 0.15 and 0.30 second, which represent the expected upper and lower limits of the present system, were assumed for the purpose of analysis. Figures 19(a) and 19(b) indicate quite clearly that maintaining the vertical velocity within the present design limits is not a problem for reasonable descent rates at probe contact. Even for the



(a) System delay time of 0.15 second. (b) System delay time of 0.30 second.

Figure 19. - Descent engine shutdown.

worst case considered (4-foot probe, 0.15-second system delay, zero pilot reaction time), the touchdown velocity does not exceed the present design limit of 10 fps if the descent rate at probe contact is less than 8 fps. From figure 19, it can be seen that all lesser descent rates at probe contact result in touchdown velocities of less than 10 fps.

The curves do show, however, that having the engine off at touchdown presents a problem if the descent rate at probe contact is too high or if the pilot reaction time to the probe contact signal is too long. Also, figures 19(a) and 19(b) indicate clearly that the touchdown velocity is more sensitive to pilot reaction time than to descent velocity at probe contact. The curves show that the descent rate at probe contact, the probe length, and the ability of the pilot to perform the control task must be matched. In addition, the velocity uncertainties associated with the landing radar, final update of the computer, and inertial platform drift must be considered in selecting probe length.

The analysis indicated that for the expected pilot reaction times, the probe length should be of the order of 3 to 4 feet. This was confirmed in a recent simulation of the lunar landing maneuver. The probe used was 3 feet long, and the system delay time was 0.15 second. In approximately 300 data runs, the simulation results revealed that touchdown velocity was 4.9 to 6.9 fps 95 percent of the time, and that 4 engine-on landings occurred. Examination of the test data showed that the average descent velocity at probe contact was about 4.6 fps and the average pilot reaction time to the probe signal was on the order of 0.3 second. As shown in figures 19(a) and 19(b), the pilots were operating very close to the maximum permissible descent rate at probe contact, and, therefore, some engine-on landings would be anticipated. Had the descent rate been of the order of 4 fps, it is doubtful that the engine-on landings would have occurred. However, a 4-foot probe length would accommodate the descent rate of 4.6 fps because the upper permissible reaction time is about 0.6 second. Based on this analysis and on the simulation results, it appears that the actual probe length can be selected analytically to accommodate the manual engine shutdown requirement and still afford safe touchdown velocities.

Control During Ascent

The ascent, in general, is a mirror image of the powered descent. The primary objectives of the ascent are: (1) to place the LM ascent stage in a coplanar orbit with the CSM and in an orbit which assures a clear pericynthion, (2) to phase the LM in such a way as to allow maximum information flow to the LM from both the CSM and the Manned Space Flight Network (MSFN) during terminal rendezvous, and (3) to bring the LM close enough to the CSM to allow the crew to rendezvous and dock with the CSM.

Powered Ascent

During the powered ascent phase of the LM mission, the vehicle characteristics change markedly from those encountered during landing (refs. 4 and 5). During lunar launch, the descent stage is released and remains on the moon. The gross weight of the spacecraft at the beginning of ascent firing is approximately 10 000 pounds (earth weight); and 7 minutes later, at the termination of ascent, the gross weight has decreased to 5000 pounds. The resulting increase in control powers during ascent, over those present during descent, causes marked changes in the vehicle as a navigation and control platform in either manual or automatic operation.

Basically, the powered ascent consists of a short, vertical-rise phase from the lunar surface, a rapid pitch over to 60° from local vertical, and a gradual pitch over, with the attitude at the end of the ascent firing being approximately 100° with reference to the landing site local vertical. During the normal ascent maneuver, the crew monitors the progress of the vehicle by comparing the outputs of the primary guidance system and the AGS with the nominally expected trajectory with verification by MSFN tracking and the rendezvous radar. Descent engine thrusting terminates at an altitude of approximately 60 000 feet with a positive flightpath angle.

Coelliptic Sequence Initiation

At an elapsed time after insertion selected by the crew, normally of the order of 30 minutes, a horizontal coelliptic-sequence-initiation (CSI) firing is made. This serves to absorb launch dispersions and to provide a launch window. The horizontal CSI firing is performed so that at the nominal terminal-phase-initiation (TPI) time, the LM will be on a specific inertial line through the CSM.

Constant Delta Height

The concentric sequence involves advancing in the elliptic trajectory to the predicted time of apocynthion, determined from pre-CSI computations, and performing a horizontal circularization firing termed the constant-delta-height (CDH) maneuver. After the CDH firing, the LM advances along the orbit until the elevation angle reaches the nominal TPI value, which varies only slightly with the normal concentric altitude differences of 15 to 30 nautical miles.

Terminal Phase Initiation

The time of TPI is a function of the landing site coordinates and spacecraft-to-spacecraft and MSFN tracking requirements. The general ground rule for lighting conditions is that there should be at least 40° between the LM-CSM and LM-sun lines of sight at TPI. There should be approximately 15 minutes of LM-CSM tracking between CDH and TPI with at least 5 minutes of tracking occurring just before TPI. Also, to allow sufficient MSFN backup for TPI, the TPI should be between 85° E. and 5° E. longitude. The terminal phase is initiated with a line-of-sight firing computed by Lambert's routine, and the terminal phase has a standard CSM travel angle of 140°.

Terminal Rendezvous

The terminal rendezvous and transition to the docking phase rely heavily on the line-of-sight relationships of the LM to the CSM. Consequently, the crew should intensify monitoring of the rendezvous at a range of approximately 20 miles. Within this range, any deviation from the desired rendezvous trajectory will become readily apparent by comparing the line-of-sight and range rates with those which should be encountered. Thus, the crew is allowed a maximum of continuity if manual takeover is required.

To complete the rendezvous under all likely conditions, it was necessary to develop a manual thrusting technique which would control the LM trajectory to within reasonable limits of that employed by the PNGCS. Thus, for a system deterioration at any point within the 20-mile range, the crew could successfully complete a rendezvous manually at a nominal increase in fuel expenditure.

A method of employing the LM rendezvous radar was developed to allow the crew to determine the velocity vector of the LM with respect to the CSM. Knowing this vector in terms of line-of-sight rate and range rate, the crew could control the relative trajectory by maintaining the range rate and line-of-sight rate on a predetermined schedule which approaches zero by thrusting along and perpendicular to the line of sight. The ability of the crew to perform this maneuver was verified by simulation at MSC. To afford the crew the necessary data to accomplish the required terminal rendezvous operations, a type of nomograph was provided which allowed graphical multiplication of the line-of-sight range and the line-of-sight angular rate to obtain the instantaneous orthogonal rates. With knowledge of the desired line-of-sight rate and range rate at any checkpoint during the terminal trajectory, the pilot determined and commanded the desired change in velocity vector in an efficient manner.

Results of the simulation showed that the manual backup rendezvous technique could be used to complete the maneuver within fuel constraints. The particular trajectories used for evaluating the piloting procedures were 180° transfers with various miss distances, a 210° transfer 0.5° out of plane with a 6-nautical-mile miss distance, and a 230° transfer from an aborted powered descent. The range-rate schedule originally intended for use was modified, when various problems arose, as the study progressed. The schedule finally evolved as being most acceptable for performing the maneuver is shown in table II.

TABLE II. - ACCEPTABLE RANGES AND RANGE RATES

Range, ft	Range rate, fps
120 000	100
60 000	80
30 000	60
15 000	40
5 000	18

If the range rate was below the specified value at the correction point in the initial runs, the pilots ignored the range rate and simply reduced line-of-sight rates. However, it was found that better control was effected by increasing the range rate to the specified value. The line-of-sight rates were generally corrected to 0.3 mrad/sec but this correction caused line-of-sight rate overcorrection, with some radar bias errors, which in turn resulted in ΔV penalties of the order of 15 fps. Digital program studies are being made to analyze this particular problem. As noted, the results are generally encouraging. A modified PNGCS program was run using the range-rate schedule evolved during the study. For the trajectories considered, the manual technique required slightly more ΔV (about 20 fps) than the modified PNGCS schedule.

Recent simulations have investigated the ability of the pilot to perform the terminal rendezvous using degraded guidance and navigation (no PNGCS guidance solutions). Three cases were examined: (1) The pilot had an MSFN update at a range of approximately 60 000 feet and controlled line of sight through a reticle. The pilot then used flight charts and elapsed time to control the maneuver. (2) The pilot had an MSFN update at a range of 60 000 feet, and subsequent range and range-rate data were available through the AGS computer with line-of-sight rate control through a reticle, and (3) The pilot had an operating rendezvous radar for range, range rate, line of sight, and line-of-sight rates.

An attitude-stabilized LM was also assumed. The numerical results of the study are shown in figure 20. Briefly, the simulation indicated that the pilot can satisfactorily complete the terminal rendezvous phase with a minimum of information and with extremely degraded guidance and navigation modes even in the presence of relatively large initial errors.

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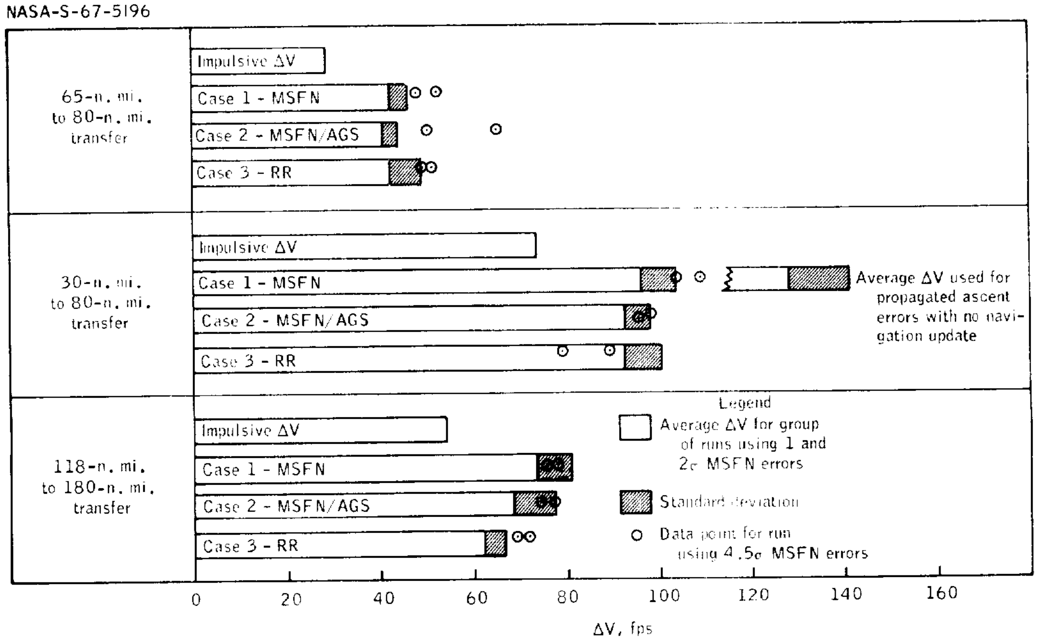

Figure 20. - Results of rendezvous simulation study.

Docking Maneuver

The mission phase that has received a large amount of attention is the docking maneuver. As the docking maneuver requires both precise control and visual coordination, it was decided that this phase would be best performed with the crew controlling the vehicle.

In the simplest configuration (head-on docking), it appeared that, because of the excellent visibility, there would be little to detract from the capability of the pilot to complete a successful docking (fig. 21). However, to perform this mission phase in such a manner required that a second docking hatch and tunnel be designed into the front of the LM. It was then suggested that the overhead window shown in figure 22 be included at the command-pilot crew station and that the pilot use the overhead window for the final docking maneuver.

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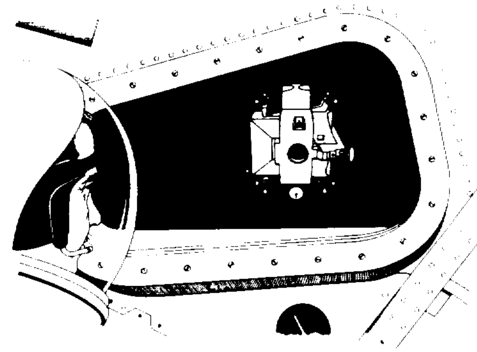

Figure 21. - CSM pilot view of LM at docking interface.

To verify the capability of the pilot to perform this task through the overhead window, the physical docking simulation at LRC (fig. 9) was utilized as a test bed. It was found that the pilot could complete the hard docking with the same degree of success as with the head-on configuration, using either the primary or the AGS control path. Docking using the primary control path was relatively easy because of the attitude-hold feature. The simulation also revealed that the small window size caused inertial reference to be lost because the CSM blocked out the background cues. Subsequently, several visual aids designed to augment available visual cues were evaluated. The results of this evaluation indicated that the standoff cross of figure 23 was the most useful aid for providing the pilots with cues.

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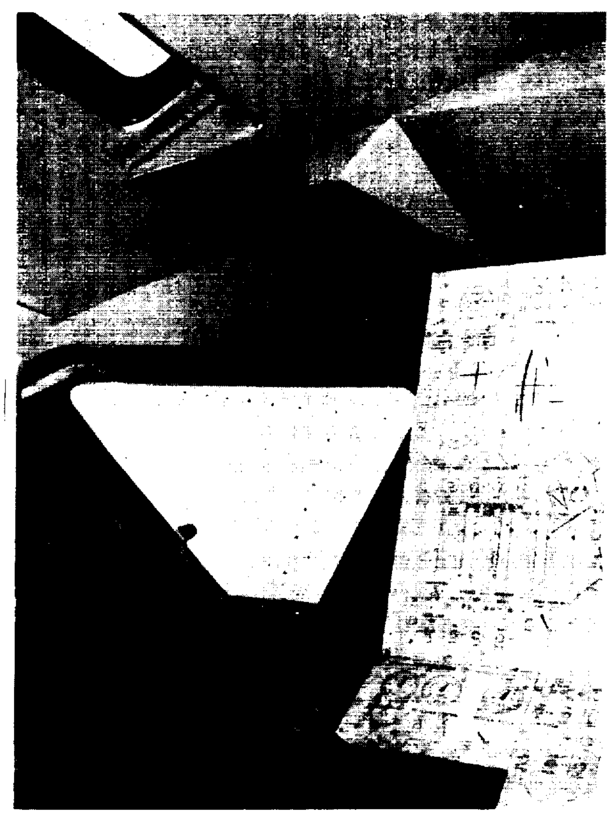

Figure 22. - LM overhead window.

The LM-active docking study at LRC provided a verification of the adequacy of the present docking design criteria. The study also showed that with the correct visual displays, the pilots could successfully dock the LM with the CSM from observations made through the overhead window without the use of flight instruments. Transfer of the control axis by 90°

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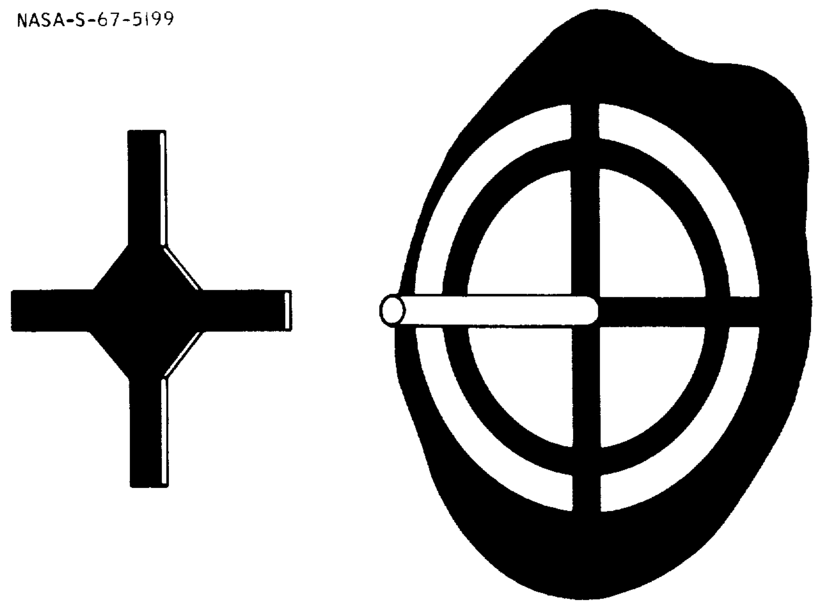


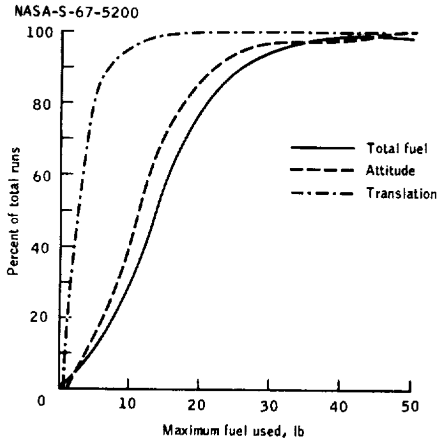
Figure 23. - Docking aid.

(necessary because of the overhead window) required training, but once the transfer was made, the test pilots had little difficulty in controlling the LM to the present design limits. Data from the study are contained in figures 24(a) to 24(f); the percentage of total runs is plotted as a function of the variables under examination. The data used are for pilot control through the primary control path (RCAH).

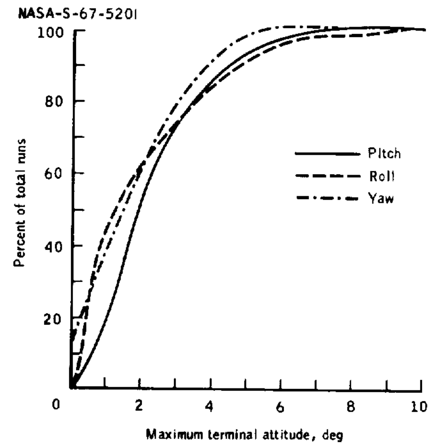
GENERAL LM CREW FUNCTIONS

The primary purpose of the LM spacecraft is to land men on the moon surface and provide for their safe return. To accomplish this mission successfully and efficiently requires a complete and total integration of the crew and the systems provided in the LM. In the course of this mission, the crew utilizes flying experience, judgment, and decision-making ability enhanced by flight experience, in connection with the data supplied by the guidance, navigation, and control systems. The crew controls the LM through the guidance system in all mission phases, exercising judgment in the execution of mission events. Because all situations that might arise cannot possibly be predicted, the crew will also use judgment (based on mission rules and assessment of how the design of the LM will meet these situations to operate and control the LM during unforeseen events.

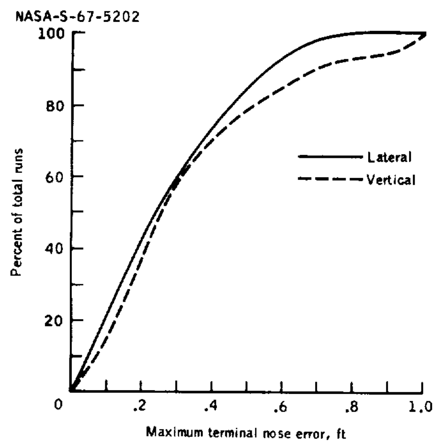
Systems aboard the LM must be managed in such a manner as to maximize the probabilities of mission success and crew safety. This management is a prime responsibility of the crew. The crew must also detect and isolate marginal or degraded systems and, if required, provide the information necessary to complete the mission or assure a safe abort.



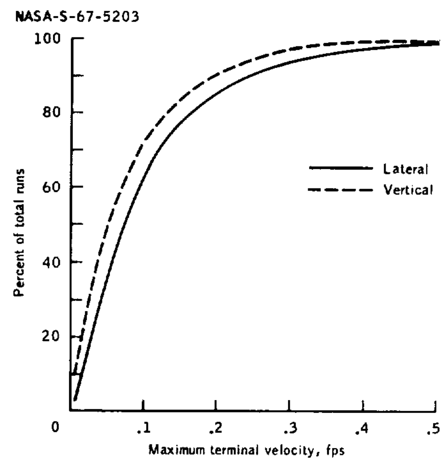
(a) Fuel.



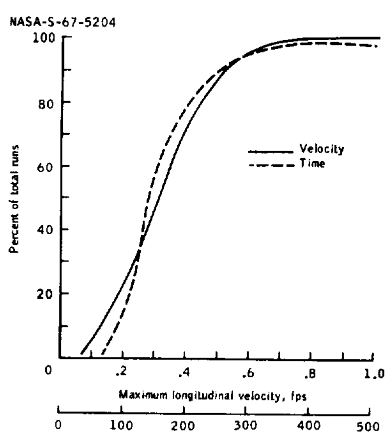
(b) Attitude.



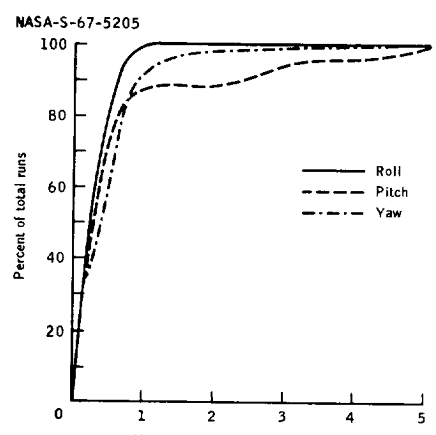
(c) Terminal nose error.



(d) Terminal velocity.



(e) Longitudinal velocity and time.



(f) Attitude rates.

Figure 24. - Results of LRC docking study.

Other directly related areas where the crew performs essential roles are in communications and evaluation, which consist of updating the computer, relating key events, and conducting experiments or other similar roles. As a final point, it should be noted that the LM will not have a complete test throughout its design range prior to the first lunar mission. In this respect, then, the pilots serve as a flight test crew for the LM missions.

CONCLUDING REMARKS

The lunar module mission and the automatic and manual control aspects of this spacecraft in landing on the lunar surface and returning to the orbiting command and service module have been discussed. The lunar module navigation, guidance, and control system has been designed to take maximum advantage of the capabilities of the crew in the areas of decision, judgment, management, and flying experience. The design has been based on an orderly sequence of analytical studies, simulation programs, flight tests, and operational and procedural requirements. There are areas in which more analyses and simulations must be conducted to resolve design and procedural problems. However, these do not appear to be of major concern in the lunar module development program. The program is proceeding toward the tests and operations phases, and at this time, the guidance and control system and the operational concepts appear to meet the majority of the requirements imposed by the lunar module mission and design constraints.

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APPENDIX

AGS	abort guidance system
AOT	alinement optical telescope
CDH	constant delta height
CSI	coelliptic sequence initiation
CSM	command and service module
DSKY	display and keyboard
LGC	LM guidance computer
LLRF	lunar landing research facility
LLRV	lunar landing research vehicle
LM	lunar module
LPD	landing point designator
LR	landing radar
LRC	Langley Research Center
MSC	Manned Spacecraft Center
MSFN	Manned Space Flight Network
NASA	National Aeronautics and Space Administration
PNGCS	primary navigation, guidance, and control system
RCAH	rate-command/attitude-hold
RCS	reaction control subsystem
ROD	rate of descent
RR	rendezvous radar
TPI	terminal phase initiation
T/W	thrust-to-weight ratio
ΔV	delta-V (characteristic velocity)

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