NASA CONTRACTOR REPORT

NASA CR-1619

LUNAR ESCAPE SYSTEMS (LESS) FEASIBILITY STUDY

Volume I - Summary Report

by J. O. Matzenauer

Prepared by
NORTH AMERICAN ROCKWELL CORPORATION
Downey, Calif.
for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JUNE 1970

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1. Lunar Escape Systems

Issued by Originator as Report No. SD 69-598-1

Prepared under Contract No. NAS 1-8923 by
SPACE DIVISION, NORTH AMERICAN ROCKWELL CORPORATION
Downey, Calif.
for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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FOREWORD

This summary report was prepared by the Space Division of North American Rockwell Corporation under Contract NAS1-8923 for NASA-Langley Research Center (LRC). A detailed technical volume, Contractor's Number SD 69-598, was also prepared. Both reports were prepared in the style required by NASA Publications Manual SP-7013, 1964.

The primary study team consisted of the following persons:

J.O. Matzenauer	- Program Manager
D.H. Hengeveld	- Project Engineer, Parametric Operational Information
D.A. Engels and G.C. McGee	- Project Engineers, Stability and Control
R.E. Oglevie	- Project Engineer, Guidance and Navigation
V.V. VanCamp	- Project Engineer, Design Integration
A.D. Kazanowski	- Consultant for Lunar Science and Visibility
D.F. Bender and M.R. Helton	- CSM Rendezvous Analysis

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LUNAR ESCAPE SYSTEMS (LESS) FEASIBILITY STUDY, SUMMARY REPORT

By J.O. Matzenauer
Space Division, North American Rockwell Corporation

INTRODUCTION

This report summarizes the results of a Phase A feasibility study of lunar emergency escape-to-orbit systems conducted by the Space Division of North American Rockwell (NR). Mr. A. W. Vogeley was contract technical monitor at NASA-LRC.

The mission of the lunar emergency escape-to-orbit system (LESS) is to provide a means for the crew of the lunar module (LM) or extended LM (ELM) to escape from the surface in the event that the LM/ELM ascent stage is unsafe or unable to take off into orbit. The LESS role is to carry the two astronauts to the CSM in orbit within three to four hours.

A determined effort has been made throughout the Apollo program to incorporate every reasonable means of assuring crew safety and mission success. Development of the LESS vehicle, however, will provide increased crew safety margins by covering possible failures of the critical single-engined LM/ELM ascent stage.

Both NASA and NR have carried on extensive study activities on missions and systems beyond early Apollo. These efforts have shown that crew safety largely paces the achievement of greater exploration. Thus, any system or procedure that promises to increase mission safety has potential for permitting a faster rate of achieving exploration goals.

Before this study, a preliminary feasibility analysis conducted at NASA-LRC had indicated that a simple flying platform concept might be adequate to carry the crew to a safe orbit. The intention was to obtain necessary safety and reliability through use of simple system concepts rather than through the more usual redundancy approach. Likewise, unsophisticated guidance and control techniques were desired for use with simple ascent profiles. In addition, potential availability of all the LM ascent stage propellants (5000 pounds) indicated that little emphasis need be placed on minimizing propellant requirements, although a low vehicle dry weight is necessary.

Study details and parametric data, which are summarized in this document, can be found in the main technical report volume SD 69-598. Also in the main report are more detailed conclusions and recommendations for further effort.

OBJECTIVES

The study objectives were to determine the feasibility of simple escape system concepts, to provide a spectrum of operational data on these concepts, and to identify techniques feasible and suitable for carrying out the emergency escape mission. This information, together with conceptual designs, surface preparation requirements, and long-range surface-to-surface flier application data, was to provide supporting material for system development decisions by NASA and for the simulation test program at NASA-LRC.

APPROACH

The overall objectives and the approach taken in the study are summarized in figure 1. Major inputs consisted of the most pertinent data from associated studies such as NASA-LRC initial system studies, the recent Phase B Lunar Flying Vehicle (LFV) Study for NASA-MSC, the large background of Apollo systems data, and the NASA-LRC flying lunar excursion experimental platform (FLEEP) proposal effort.

In the parametric data and system analysis effort, performance in terms of boost trajectories, CSM rendezvous and docking, and the subject of visibility conditions were treated parametrically to provide a background of operational information within which system and design iterations could be made. Guidance and stability concepts and techniques were also examined as broadly as possible as a basis for subsequent systems synthesis and integration.

In the systems integration and concept development activity, the guidance and control techniques that were previously treated as basic variables were integrated into practical design configurations. Realistic evaluation of weight and balance was used in the guidance and control analyses and the overall feasibility determination. Also at this stage in concept synthesis, iterations were made back through the performance loop. The results were feasible guidance and control combinations and conceptual configurations that reflect the features, constraints, and resulting characteristics for several classes of vehicles. The classes are established by the basic control mode and modified by the propulsion choices.

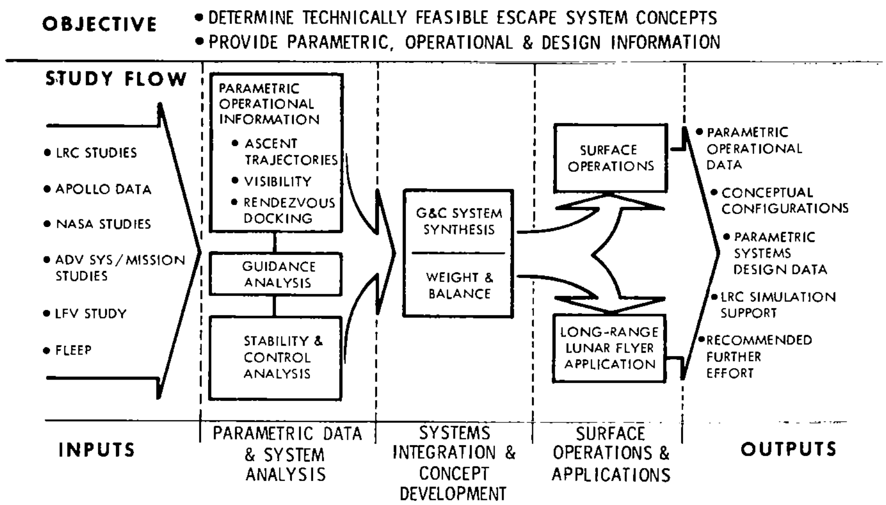

Figure 1. - Study Objectives and Approach

For the surface operations and applications effort, the problems of deploying and preparing the escape system for use were examined. The results were carried through design as appropriate to aid in establishing overall system feasibility. Utilizing the established Phase B LFV study ground rules and techniques, the application of LESS to long-range flyer (LRF) surface-to-surface missions was examined. Required changes were defined, and the resulting performance as a flyer was calculated. The effects of these flyer mission changes were then evaluated in terms of effect on the basic escape mission.

Outputs of the study spanned a spectrum of parametric operational information covering four basic ascent-to-orbit trajectories to various orbital altitudes. Also included were the effects of such system variables as thrust-to-weight, specific impulse, and trajectory sensitivity to major system errors. Visibility effects were determined for both the ascent and rendezvous portions of the mission. Energy and phasing requirements for rendezvous were treated extensively in a parametric manner for various conditions and relationships between CSM orbit and LESS final orbit. These energies were related to practical mission planning factors: timing, location of orbit nodes and apses, and plane changes. Equipment capabilities of the CSM were evaluated and performance estimated for the rendezvous tracking and intercept tasks. Five typical conceptual designs for kinesthetic, hardwire, and stability-augmented control modes were prepared to illustrate design features and interfaces between subsystems and elements. Variations in the designs were produced for different or alternative basic propulsion configurations. Deployment of the LESS from an LM/ELM was examined and the fueling and preparation for launch described. Long-range surface flyer adaptations of LESS were studied, and two conceptual designs were developed for concepts with basically different propulsion configurations.

A theory of handling qualities optimization was developed, and correlation was made with NASA-LRC simulation data obtained for kinesthetic control. Correlation between alternative pilot rating system was made to assist in making translations from one system to the other. Design curves were produced to show basic relationships between design variables for hardwire control. Fundamental guidance elements were examined and all possible visual and instrument reference systems identified. The best concepts were evaluated, and combinations of guidance and control elements were integrated to synthesize complete systems. Guidance error analyses were performed to show the estimated orbital injection errors expected with various mechanizations. Guidance and control equipment mechanization was studied to determine relative weight, volume, and power of candidate hardware as well as to assess the relative feasibility of those concepts.

PRINCIPAL ASSUMPTIONS AND GROUND RULES

The study was conducted with a minimum of restrictive ground rules. The principal ground rules followed were:

1. LM/ELM propellants are to be used from the ascent stage.
2. Minimum equipment and simplicity, rather than redundancy, are to be stressed.
3. The space-suit backpack is to be used for crew life support and environmental control and for communications.
4. Mission stay-times are to be up to 14 days.

SIGNIFICANT RESULTS AND DATA

Parametric Operational Information

Ascent Trajectories. - A series of ascent-to-orbit trajectory profiles was examined. Flight path shape is indicated in figure 2 with sketches of vehicle attitude during the various portions of the ascent profile. The calculus-of-variation optimum trajectory provides a constantly changing vehicle attitude to yield the minimum energy or ΔV required. (Not shown, but also considered, was a linear profile of vehicle attitude versus time which closely approximates the optimum profile energy requirement.) These profiles would be appropriate for a fairly highly mechanized guidance and control system concept. The three-step profile consists of a vertical rise portion followed by sequential pitchover to two other vehicle attitudes, the last one being near-horizontal (with the right value of thrust-to-weight). Three steps were found to be sufficient to provide a fairly close approximation to the minimum energy required (approximately 5 percent).

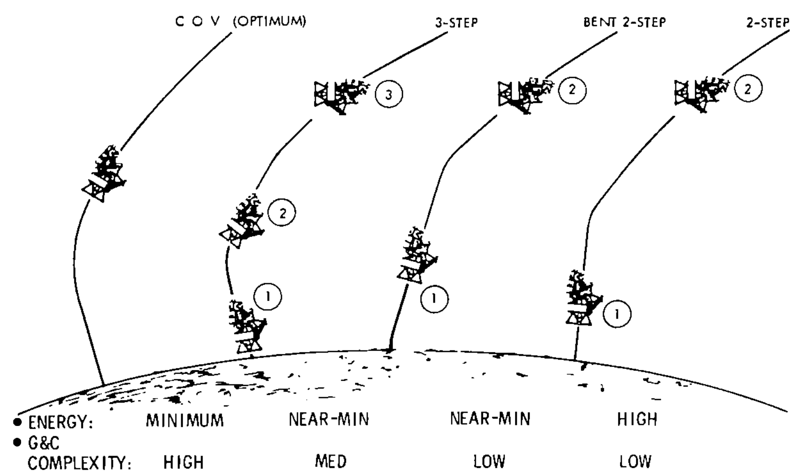

Figure 2. - Trajectory Profiles

The two-step profile on the right of figure 2 consists of a vertical ascent followed by a pitchover to near-horizontal thrust attitude. The profile has the advantage of being simple to mechanize in terms of guidance and control, but invokes a large penalty in the energy required, about 1000 fps compared to optimum profile (approximately 15 percent).

The bent two-step profile concept shown provides the combined advantages of low energy (equivalent to 3 step) and the simplicity of only one step change during ascent. The vehicle takes off with only a short vertical rise, then pitches over with the thrust axis about 30 degrees off vertical and begins to build up tangential velocity essentially from liftoff. This profile (with a 10-second vertical rise for orientation) has been utilized in the most recent simulation testing at NASA-LRC, for which trajectory data were informally furnished.

The trajectories were examined to determine the influence of many variables: initial vertical ascent time or altitude, initial thrust-weight ratio, attitude reference basis (inertial space or local lunar horizon), engine specific impulse, and step change timing. The variables were found to affect the various trajectory profiles in much the same way despite the basic profile differences.

Figure 3 shows the variation of ascent energy required (ΔV) for two typical attitude profiles as a function of initial thrust-to-weight (T/W) ratio for various target altitudes. It is noted that a T/W of about 0.3 is optimum for minimum boost energy for the higher orbits of most concern, 60 nautical miles. Turning losses associated with lower orbits causes optimum T/W to be shifted to higher values. The flight attitudes obtained in the various profiles are shown in figure 4, with the calculus-of-variation (COV) or optimum trajectory as a base. Pitch attitude is measured in degrees from local horizontal. The inset curve also shows how the number of trajectory profile steps affects the basic boost energy (ΔV) required.

Trajectory error sensitivity studies of the perturbations in the target orbit altitude revealed that the principal error sources were associated with pitch attitude and T/W. These errors result in variations in burnout conditions, of which the most critical is perilune altitude. T/W errors of the magnitude expected (±4 percent) could not be tolerated with engine cutoff controlled by a simple timer. Control by ΔV, utilizing output from an integrating accelerometer, was found to be required for both the attitude profile steps and engine thrust cutoff. Error sensitivities are shown in figure 5 for the original time basis of control and also for control of final cutoff with and without step change control by ΔV. Some combinations with high or low T/W would result in safe perilune but a very high apolune, which would make subsequent CSM rendezvous difficult. Cutoff on ΔV improves the perilune clearance for low T/W, but truly satisfactory orbits are only achieved with both step and cutoff by ΔV.

It was found that elliptical, rather than circular, target orbits desensitize the variation in perilune altitude with pitch attitude errors (fig. 6). For instance, a reasonable pitch error of plus one degree for a targeted 60-nm circular orbit would result in a perilune of about 20 nm; whereas, an

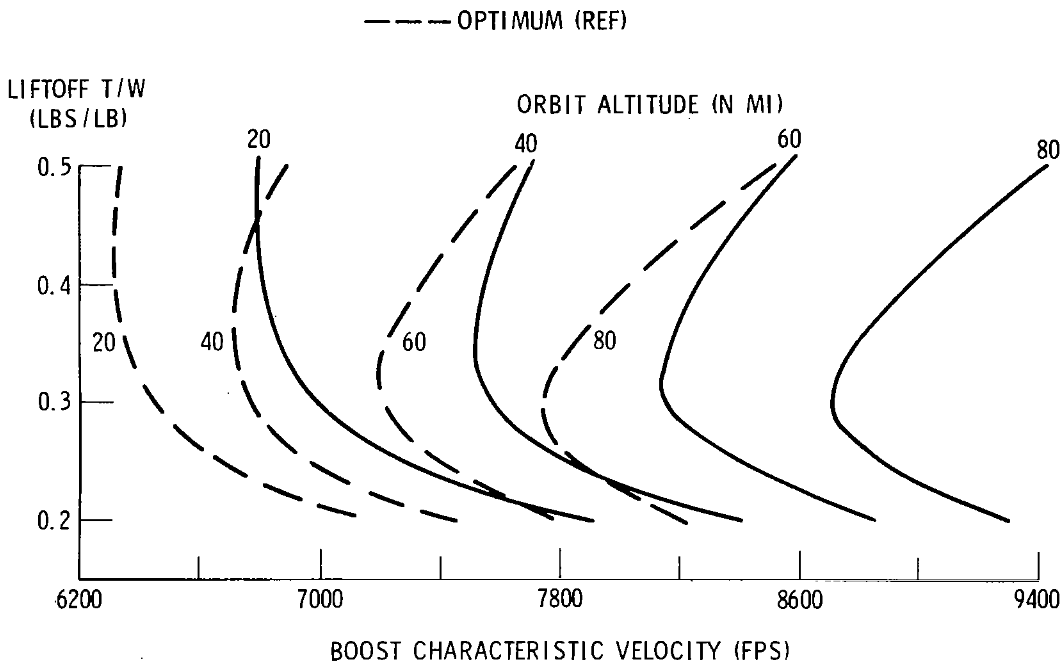

Figure 3. - Energy Requirements for Two-Step Steering Profile

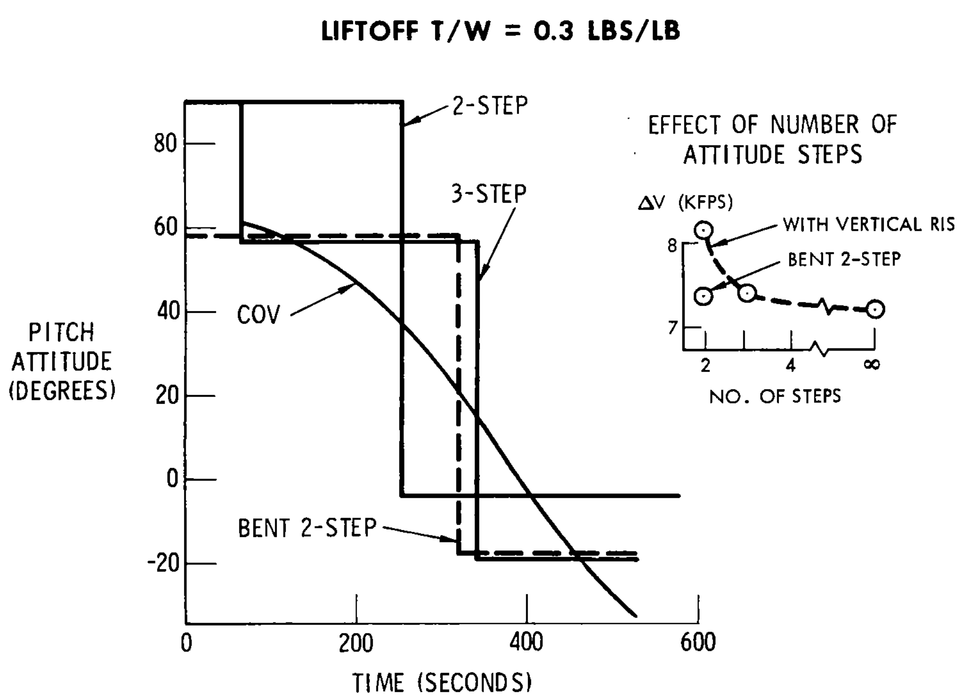

Figure 4. - Comparison of Steering Histories for Boost to 60-Nautical-Mile Orbit

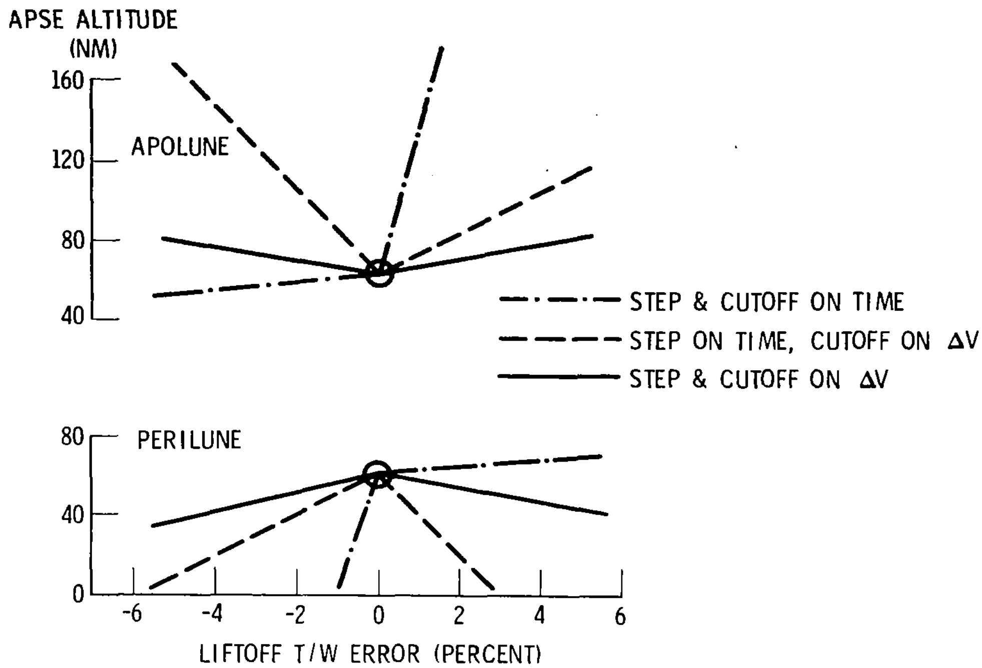

Figure 5. - Orbital Accuracies Versus Thrust-to-Weight Errors

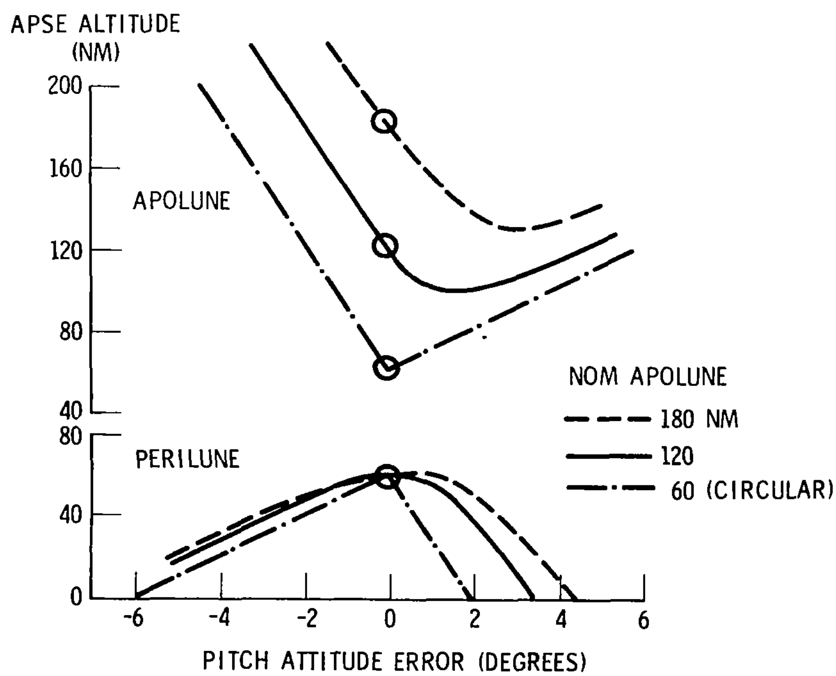

Figure 6. - Orbital Accuracies Versus Pitch Errors

error of 2-1/2 degrees would result in the same perilune for a 60- by 120-nm target orbit. These higher elliptical orbits, however, can cause subsequent rendezvous and transearth injection penalties because of non-optimum injection geometry; consequently, circular target orbits may provide the best overall compromise.

CSM rendezvous and docking. - An extensive computer analysis was undertaken for CSM transfers under various orbital conditions to establish parametrically the scope and character of the maneuvers involved in LESS rendezvous. Contour maps of energy required were developed for typical CSM and LESS initial conditions. The amount of energy required for most likely orbital conditions were found to be within the current CSM budget allowance of 790 fps for LM rescue maneuvers.

The CSM orbit determination and guidance capabilities currently aboard for backup LM rescue were found to be adequate for tracking the LESS and computing the rendezvous trajectory within one-fourth orbit from burnout (one-half hour). The LESS will require a VHF transponder and flashing-light beacon. The CSM can then perform the transfer to the LESS orbit within another one-half to three-fourths orbit (180- to 270-degree transfer). Typical rendezvous geometry is illustrated in figure 7.

Several methods were studied for docking the small LESS vehicle with the CSM. The preferred concept is a hard docking on the CSM nose with a special docking drogue on the LESS, as shown in figure 8. This scheme keeps the LESS firmly positioned while the crew transfers via hand holds and safety tethers to the CSM main hatch, reducing the possibility of damage to the heatshield by the LESS. Another consideration is the possible contamination or damage of the space suits from CSM reaction control system (RCS) jet impingement.

Visibility considerations. - Visibility was considered throughout the LESS mission study. Lunar conditions restrict the viewing of objects because of shadowing from blinding glare when sighting is near the sun and from solar glare or reflections from instruments. Reflected glare from the lunar surface also reduces sensitivity and contrast. The astronauts' visors must maintain filtering to preclude extremes of glare, yet allow perception of less well lighted objects.

Considering the wide spread of surface stay times to be considered, the range of possible sun angles becomes important. LESS abort could be shortly after LM/ELM landing at sun angles of 10 degrees behind or with sun angles up to 180 degrees ahead (on the horizon) with 14 days stay time, as seen in figure 9. During rendezvous, the LESS will be essentially in the sun at times, making visual tracking difficult. These problems of viewing tend to discourage use of simple visual guidance sights.

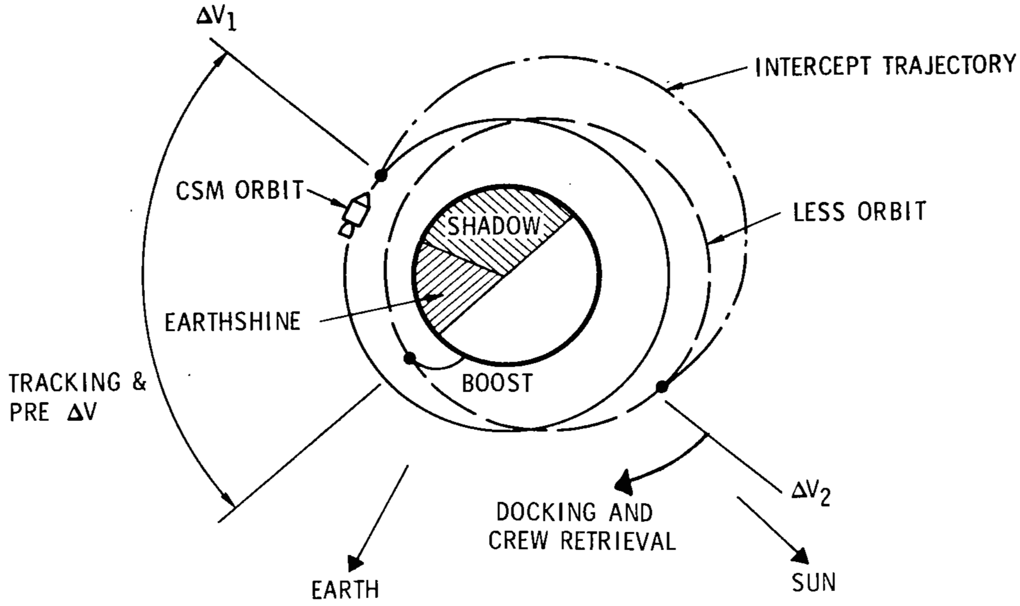

Figure 7. - Typical Geometry Picture

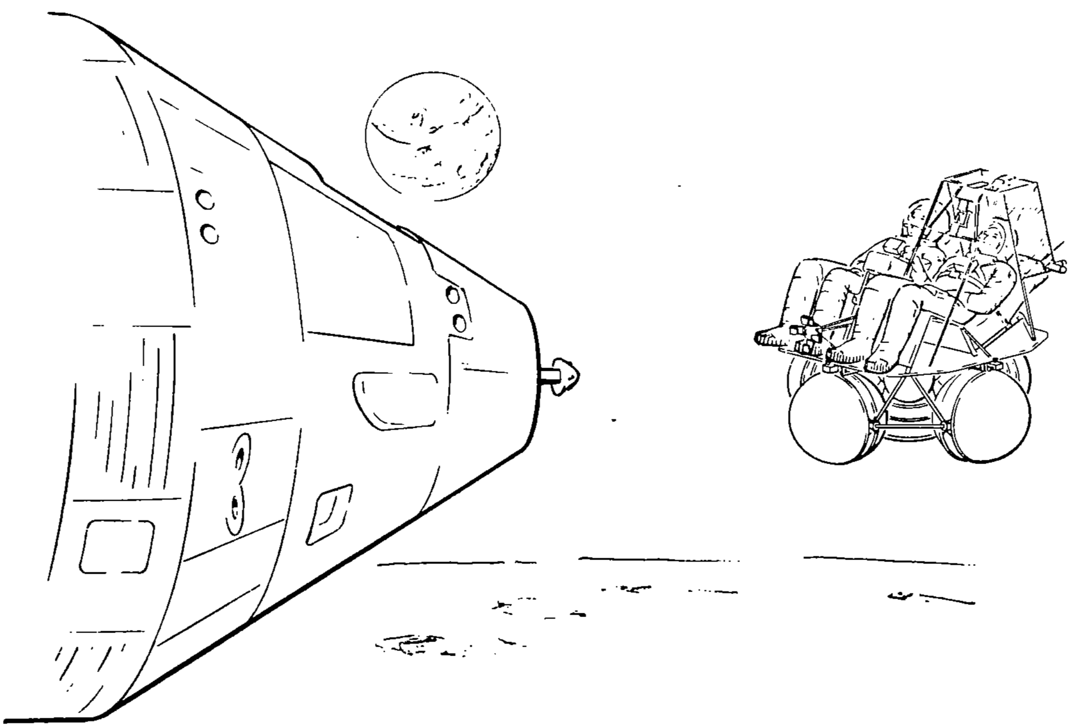

Figure 8. - Close Docking Maneuver

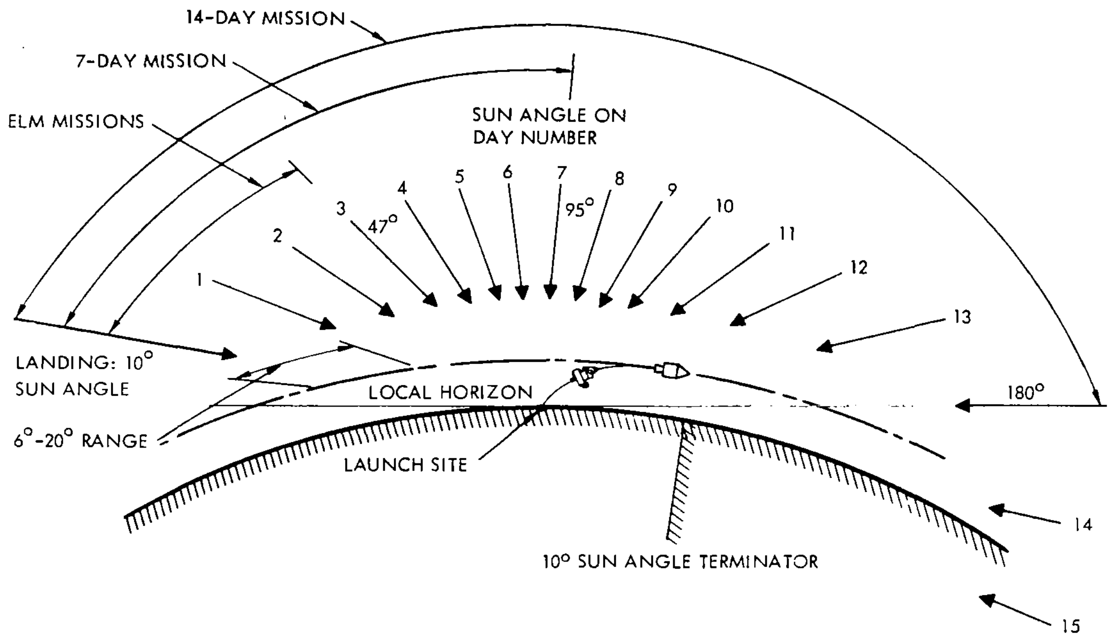

Figure 9. Sun Incidence as a Function of Stay Time

Visibility and acquistion of the target with the CSM optics was found to be a problem. It is currently under study at NASA-MSC in connection with LM rescue.

Guidance and Control Techniques

Stability and control. - Substantial quantities of data from contractor and other studies of the lunar flying vehicle were applied in this study, Tethered-flight-vehicle and fixed-base-simulator testing had indicated that the manual stability and control system (SCS) modes were not adequate where spot landings and small velocities at touchdown were required. The stability and control problem for the LESS is not so arduous, because there is no need to control translational velocities to a fine degree. The control task is reduced to maintaining the proper vehicle attitude for guidance rather than translational velocity control. System stability and handling qualities, however, were found to influence strongly the guidance accuracies achievable.

Considerable effort was expended in studying results of various other contractor and NASA simulations. Correlations between theoretical stability and pilot workload were determined. A handling qualities theory that was established permits prediction of the best handling qualities attainable as well as the system constants necessary for achievement of these handling qualities. It is expected that substantiating data will ultimately be an output of the NASA-LRC simulations.

Several possible vehicle configurations and stability and control modes were analyzed. It was concluded that kinesthetic control may be possible, though marginal, for the LESS (pending more simulator data) and that hardwire control appears promising. Hardwire theoretically permits more freedom in design layout and exhibits slightly better handling qualities (less pilot workload).

Figure 10 shows theoretical trends in handling qualities with the vehicle gain parameter changes during flight for both kinesthetic and hardwire manual control methods. To attain optimized kinesthetic control, there must be stringent constraints imposed on the thrust level and moment of inertia. Hardwire control is more easily optimized, since two additional parameters are available for adjustment: rotation controller sensitivity gear ratio, K_S, and the distance from the total center of gravity to the gimbal point. In the handling qualities optimization, the approach would be to center total parameter variation during flight near the bottom of the curve and thus reduce the total parameter variation from start of burn to end of burn.

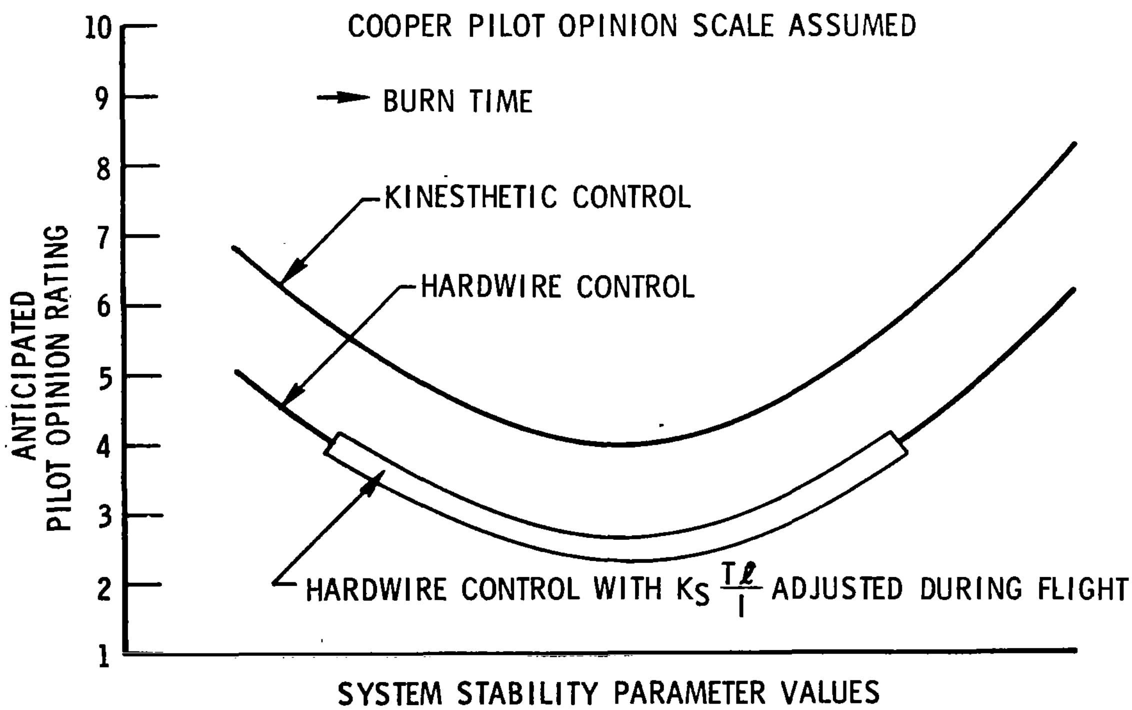

Figure 10. - Comparison of Kinesthetic and Hardwire Handling Qualities Optimization Capability

Guidance and Navigation. - A strong attempt was made to establish the feasibility of simple optical devices for attitude reference and/or for guidance. The problems associated with these displays include visibility limitations, keeping the visual reference in the pilot's field of view through the large pitch attitude change, cross coupling between visual and control axes, appreciable error because of roughness of the lunar horizon, and displays requiring the pilot's attention for interpretation and landmark identification. Azimuth references were narrowed to either the sun or surface landmarks. Neither, however, was found to be adequate throughout the 14-day staytime. These considerations resulted in a preference for a three-axis, gyro-driven attitude indicator display.

A system mechanization study was performed to establish the weight penalties associated with the various system concepts.

Guidance error analysis was conducted statistically, using error source magnitudes that are representative of simple system mechanizations without a high level of tolerance control. Such estimated error effects on LESS orbit uncertainties are illustrated in Table 1. Nominal conditions were 3-step boost profile to 60 nm orbit, $T/W_o = 0.3$, and constant thrust.

The manual steering error estimates are based on data from 27 runs recently made on the kinesthetic control simulation at NASA-LRC.

When statistically combined to provide three standard deviation (3σ) errors in resulting LESS orbits, the kinesthetic and hardwire modes were found to provide marginally acceptable orbital accuracies in terms of avoiding lunar impact. These results are believed to be slightly conservative in regard to the dominant error sources, pending results of further simulation testing at NASA-LRC. Figure 11 shows the effect on minimum altitude achieved as a result of steering errors encountered. The marginal condition shown with somewhat conservative error estimates may be improved with further data, as indicated. Uncertainty or deviation below the desired 60-nm altitude is plotted as a function of the main error source—manual control steering error. The minimum altitude for rendezvous allows for the CSM to descend even lower with safety for phasing maneuvers.

TABLE 1. - EFFECT OF INDIVIDUAL ERROR SOURCES

Error Source	Magnitude (3σ)	60-nm Injection Orbit Altitude Uncertainties (3σ)
Thrust/Weight	4.36%	21 nm
Thrust Vector Pointing Errors		
Thrust vector alignment versus vehicle (fixed gimbal) or effect of cg uncertainty (gimbaled)	0.4°	13 nm
Manual steering errors		
Kinesthetic	1.3°	41 nm
Hard-wire	1.1°	35 nm
Stability augmented	0.4°	13 nm
Autopilot	0.1°	3 nm
Step profile attitude maneuver rate errors		
Kinesthetic and hard-wire	±2.45°/sec	19 nm
Stability augmented	±0.54°/sec	7 nm
Thrust Ignition and Cutoff Errors		
Manual ignition and cutoff timing errors	1.0 sec	12.5 nm
ΔV meter	0.033%	5.5 nm
Engine tailoff impulse	Negligible	-

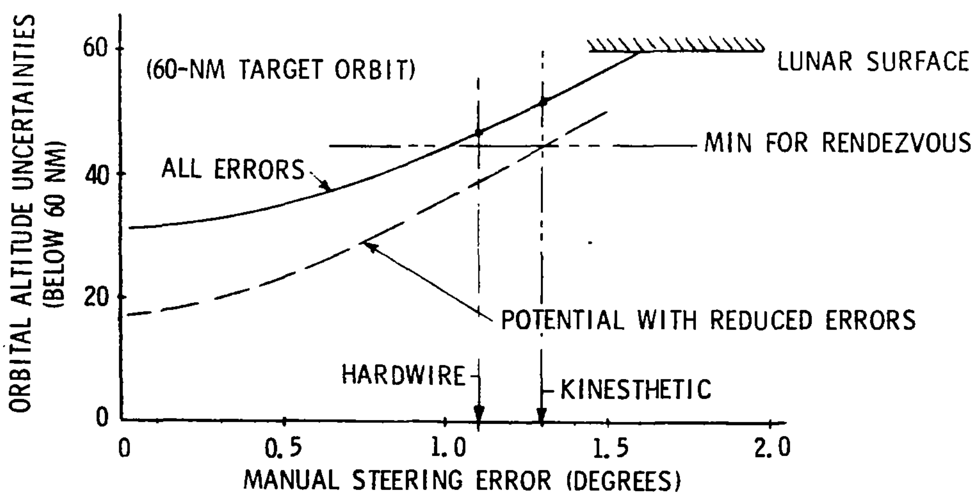

Figure 11. Effect of Three-Sigma Steering Errors on Minimum Altitude

Parametric Design Information

Five typical configurations were developed to illustrate interrelationships, to work out subsystem element interfaces, and to provide a basis for weight and balance analyses. Kinesthetic control configurations tended to be less compact because handling qualities studies indicated large inertias were desirable. Maneuvering response with large inertias, although poor, is not an adverse factor in the basic LESS mission. Lunar flying vehicle study results, on the other hand, showed that maneuvering response with kinesthetic control was of primary importance and, hence, required a small inertia. A kinesthetic concept is shown with variable configuration possibilities in figure 12.

The hardwire control configurations resulted in more compact arrangements because of decreased sensitivity to moments of inertia. Table 2 is a typical weight breakdown for a hardwire-controlled vehicle, using the bent two-step ascent profile. The vehicle itself is illustrated in figure 13 with a crewman boarding via a temporary ladder. The LESS protective cover used on the LM/ELM during transport is shown being utilized as both a sled and a launch pad.

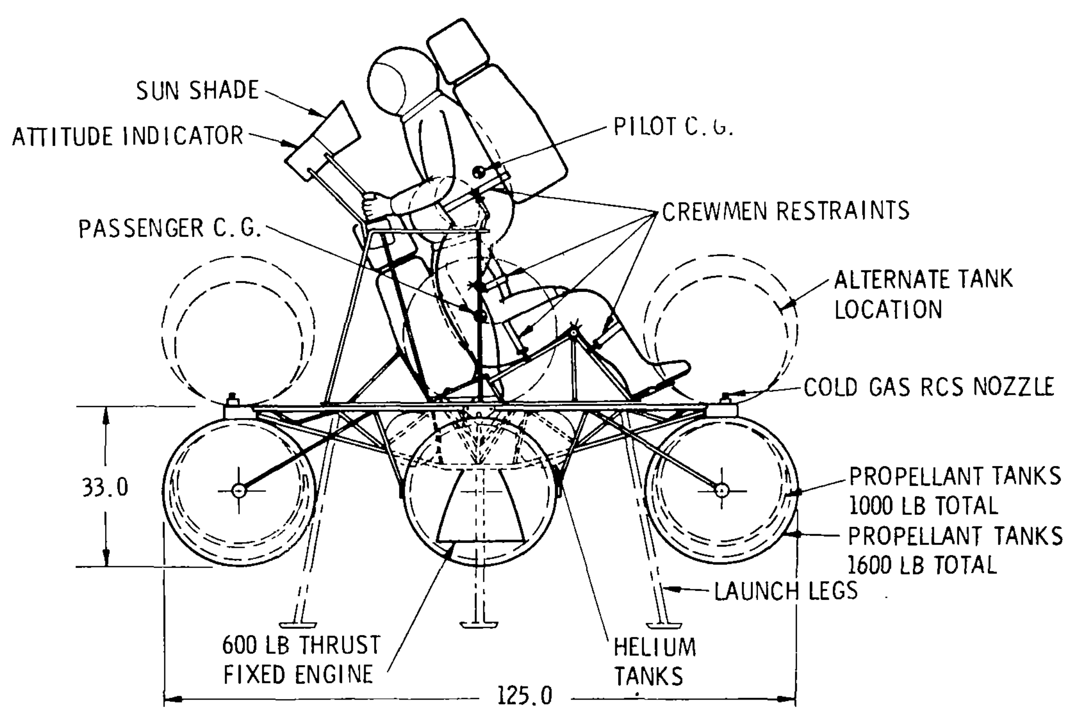

Figure 12. - Kinesthetic Control Configuration

TABLE 2. - WEIGHT BREAKDOWN - TYPICAL HARDWIRE CONTROL VEHICLE

Component	Weight (lb)
Structure	56.0
Guidance and control	57.5
Electrical system	30.0
Engine, gimbal, and mounting	40.0
Reaction control system	20.0
Propellant system	74.0
Pressurization system	41.0
Beacon and VHF transponder	25.0
Docking mechanism	20.0
Vehicle dry weight	364.5
Crew, PLSS, suits	750.0
Residuals and helium gas	13.5
Burnout weight	1128.0
Propellant	1160.0
Gross weight	2293.5

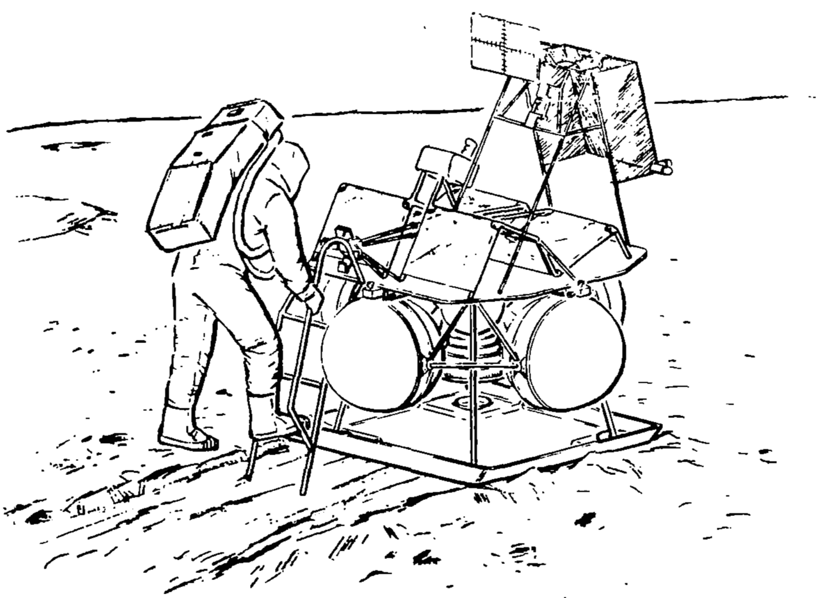

Figure 13. LESS Hardwire Control Configuration With Visual Sight Guidance

The inertias for this hardwire control vehicle concept range from 350 slug-ft^2 initially to approximately 125 slug-ft^2. Inertias varied from 400 to 800 slug-ft^2 initially to the 100 to 200 range at burnout in the study. Gross weights vary from about 2100 pounds to 2500 pounds for the LESS versions, depending upon the efficiency of the engines and ascent profiles employed. Corresponding propellant weights are 1000 pounds and 1600 pounds.

Surface operations. - Time-line analysis shows that a minimum of 45 minutes is required for one astronaut to unload, deploy, and make a preliminary checkout of the LESS. Figure 14 illustrates a possible unloading concept, assuming LESS storage on Quad I of the LM/ELM. Arms and cables assure astronaut safety. The protective cover can be used as a sled to move the LESS to the takeoff area, some 25 feet from the LM/ELM. The vehicle can be deployed after landing on a contingency basis or it can be left stowed on the LM/ELM until needed.

A two-hour preparation and checkout period is required before an abort. The LESS tanks are fueled, using special fittings on LM ascent tank drains (a minor change). Battery and gyro packages are loaded from the LM storage, guidance is aligned, systems are checked, and backpacks are recharged from the LM.

A concept utilizing a cluster of eight existing Apollo RCS pulse mode engines has configuration, control, and availability advantages. A view of such a vehicle is seen in figure 15.

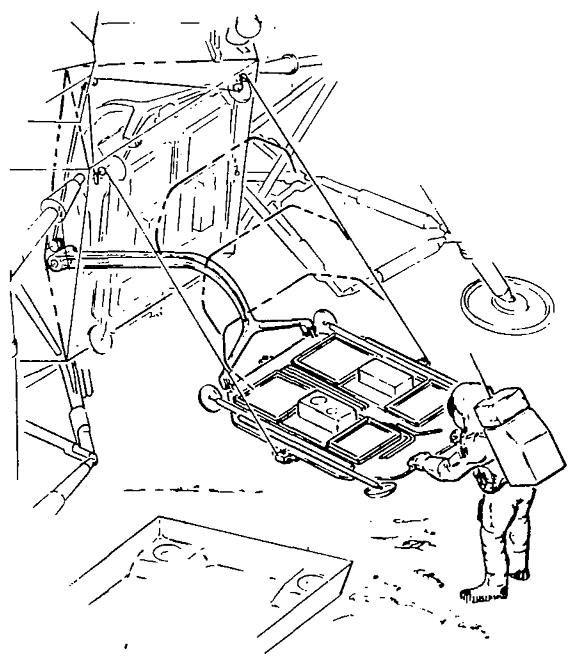

Figure 14. Lowering LESS to Lunar Surface

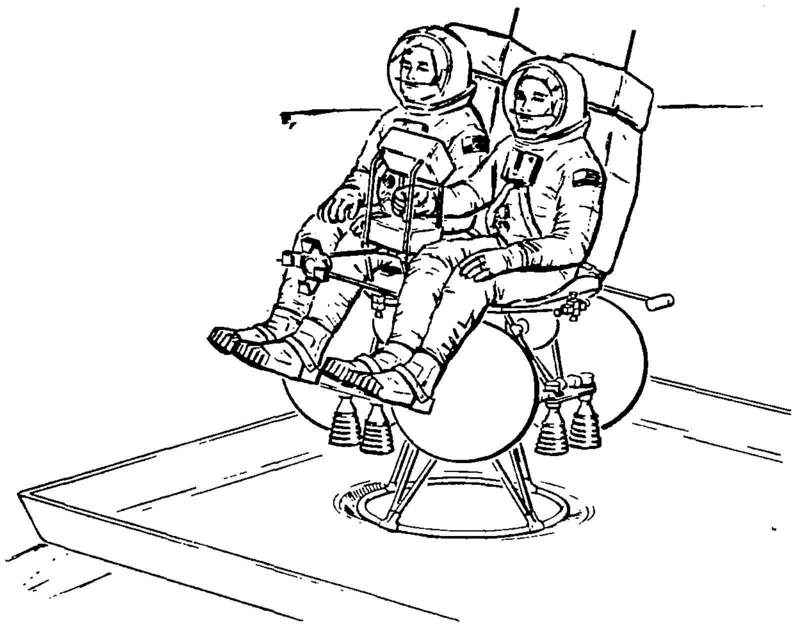

Figure 15. - LESS Flight Configuration

Lunar flying application - The LESS can be adapted to perform long-range, surface-to-surface, two-man, flying missions. Changes to the LESS for this operation include provisions for engine throttling, adding landing gear, strengthening the structure for landing loads, and adding a long-range telecommunication relay package. Design criteria were applied from the recent Phase B Lunar Flying Vehicle Study (NAS9-9045). Figure 16 is a typical configuration for such a vehicle using a single throttled engine. An attractive alternative concept could utilize a cluster of pulsed RCS engines (not shown).

The adapted LESS long-range flyer (LESS/LRF) is capable of a range radius of from 40 to 60 nm using 1200 to 1600 pounds of propellant (sized for escape missions). These order-of-magnitude increases in range, compared with that of the smaller lunar flying vehicles, should provide substantial exploration capability. It would combine relatively long range with the safety of short flight times. An attractive potential for improving mission safety could be achieved by using it as a rescue vehicle for a rover or another flyer, and as a reconnaissance vehicle for future landing sites.

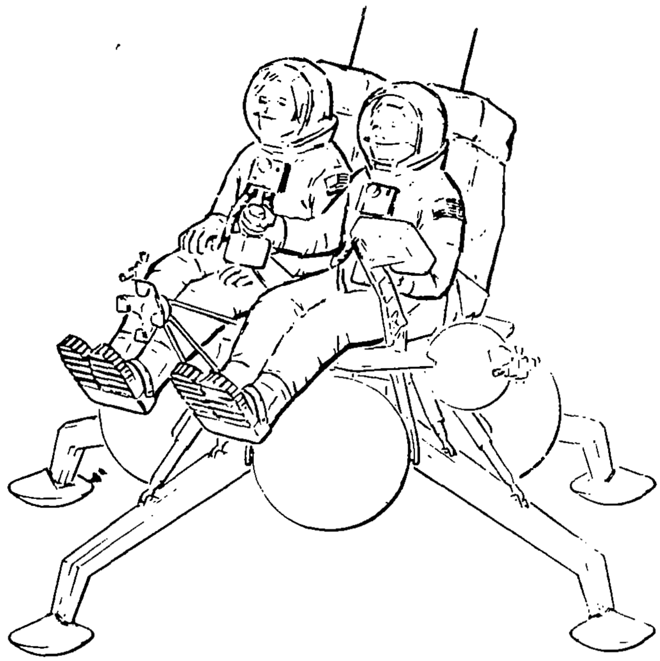

Figure 16. - Long-Range Flyer Version of LESS

CONCLUSIONS

The study results show that the basic LESS concept of a simple system for escape of two men to a safe orbit is feasible. Additional conclusions are as follows:

1. Simple manual control modes may suffice.

2. Simple boost profiles are acceptable.

3. Resulting orbital errors are acceptable for simple control concepts, but should be confirmed by further simulation testing.

4. Initial guidance data can be calculated for LESS by Mission Control Center and transmitted via LM/ELM updata link.

5. CSM-active rendezvous and docking requires no CSM changes.

6. Present CSM energy budget is adequate.

7. PLSS lifetime of 4 hours maximum is not exceeded.

8. One man can deploy and set up LESS.

9. Stowage of LESS on LM/ELM is possible.

10. LM/ELM changes for defueling are minimal.

11. LESS adapts well to alternate missions.

ADDITIONAL RESEARCH RECOMMENDATIONS

Aeronautics (Including Space Flight Systems)

1. Perform feasibility tradeoff analysis of LESS adapted to rescue missions, unmanned sample retrieval-to-orbit missions, orbital shuttle missions, logistics lander missions, experiment lander, and future landing site reconnaissance to ensure maximum system versatility and utility.

Biotechnology and Human Research

1. Additional simulation data are needed from flight-type and fixed-base-type simulators to establish the probability of successful missions with simple manual stability and control modes for the LESS. These data require statistical treatment to assure confidence.

2. Data are lacking on possible penetration of space suits by particles when crewmen are operating in the exhaust plume of CSM RCS jets. Also, propellant absorption by suits could cause toxic contamination after CSM entry. A vacuum testing program may be required if a rapid escape system development should become a reality.

3. The limits of visibility under lunar viewing conditions are not well established, particularly against the bright lunar surface background. Specific designated experiments may be necessary in early Apollo missions to provide definitive data.

Electronics and Control

1. Rendezvous Program 38 or its equivalent in the Apollo guidance computer could be deleted (no longer necessary) by NASA-MSC. An evaluation should be made to see if this program can be retained for possible use with the LESS.

Materials and Structures

1. Research on collapsible tanks is desirable to determine feasibility of such a concept for LESS and other applications wherein temporary empty storage must be tightly confined. For LESS this concept would ease the LM/ELM storage problem.

2. If a very rapid escape system development were to become a requirement, it may be desirable to perform dynamics analyses to determine tradeoffs and feasibility of possible locations for stowage of the LESS aboard LM or ELM. The locations are on Quad I or IV but within RCS jet impingement area, or on top of rear deck of descent stage.

Nuclear Systems (None)

Propulsion and Power Generation

1. Clustered Apollo RCS engines operating in the pulse mode appear attractive and have been considered for both lunar flying vehicle and LESS applications. While apparently complex, the concept promises distinct advantages in terms of package compactness, redundancy, guidance accuracy potential, early availability, and proved safety. The concept merits special consideration in future studies.

NASA-Langley, 1970 —— 31 CR-1619