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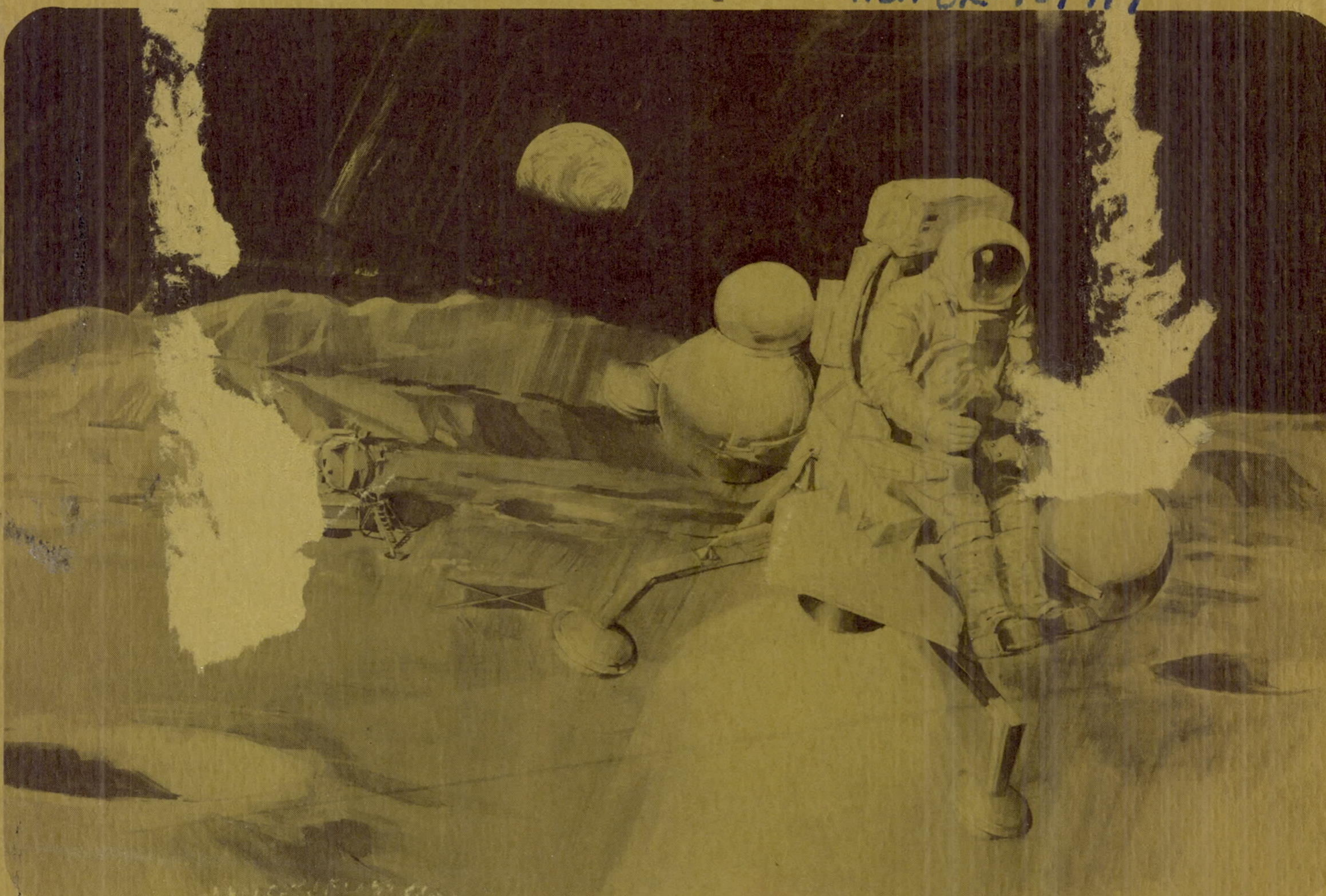

Study of ONE-MAN LUNAR FLYING VECHICLE
FINAL REPORT

Volume I
Summary

Space Division
North American Rockwell

SD 69-419-1

STUDY OF ONE-MAN LUNAR FLYING VEHICLE
FINAL REPORT

VOLUME I
SUMMARY

Contract NAS9-9045

31 August 1969

FOREWORD

This volume summarizes results of the One-Man Lunar Flying Vehicle Contract (NAS9-9045), conducted by the North American Rockwell Space Division for the National Aeronautics and Space Administration Manned Spacecraft Center, Houston, Texas. A detailed discussion of the study results is contained in Volumes 2 through 6:

Volume 2. Mission Analysis
Volume 3. Subsystem Studies
Volume 4. Configuration Design
Volume 5. Preliminary Design and Specifications
Volume 6. Training and Resources Plans

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ACCESSION NUMBER	DOCUMENT SECURITY CLASSIFICATION
	UNCLASSIFIED

TITLE OF DOCUMENT	LIBRARY USE ONLY
STUDY OF ONE-MAN LUNAR FLYING VEHICLE - FINAL REPORT: VOLUME 1-SUMMARY; VOLUME 2 - MISSION ANALYSIS; VOLUME 3 - SUBSYSTEM STUDIES; VOLUME 4 - CONFIGURATION DESIGN; VOLUME 5 - PRELIMINARY DESIGN AND SPECIFICATIONS; VOLUME 6 - TRAINING AND RESOURCES PLANS	
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*ONE-MAN LUNAR FLYING VEHICLE, *CONTROL SYSTEMS, *PROPULSION SYSTEMS, *TRAJECTORIES, *DESIGN, *TRAINING, *RESOURCES, *MISSION ANALYSIS

ABSTRACT

THE PRIMARY OBJECTIVES OF THIS STUDY WERE TO OPTIMIZE THE DESIGN AND TO DEVELOP SYSTEM SPECIFICATIONS OF THE LUNAR FLYING VEHICLE. THE SCOPE ENCOMPASSED PARAMETRIC INVESTIGATIONS, CONCEPT GENERATION, AND EVALUATION EFFORT FOR THE DEFINITION OF A RECOMMENDED CONCEPT; PRODUCTION OF A PRELIMINARY DESIGN AND DEVELOPMENT OF SYSTEMS SPECIFICATIONS OF THE RECOMMENDED CONCEPT; AND DEFINITION OF RESOURCES AND CREW TRAINING PLANS. IN ADDITION TO GENERATION OF THE LFV DESIGN, THE SCOPE OF THE STUDY INCLUDED LUNAR MODULE INTEGRATION, FLIGHT SUIT INTERFACE STUDIES, AND DEFINITION OF GROUND SUPPORT EQUIPMENT FOR EARTH AND LUNAR OPERATIONS.

AS A RESULT OF PARAMETRIC STUDIES CONDUCTED DURING THE FIRST PHASE OF THIS EFFORT, A CONCEPT WAS SELECTED WHICH HAS THE FOLLOWING CHARACTERISTICS: (1) STABILITY-AUGMENTED CONTROL, (2) FOUR GIMBALED ENGINES WHICH ARE CLUSTERED BENEATH THE VEHICLE, (3) A SEATED PILOT POSITION, AND (4) AN INTEGRAL X-FRAME LANDING GEAR WITH 6 HYDRAULIC ATTENUATORS. THIS VEHICLE IS CAPABLE OF CARRYING A 370-LB PAYLOAD IN ADDITION TO THE PILOT. THE DRY WEIGHT OF THE VEHICLE IS 304 LB. WHEN LOADED WITH 300 POUNDS OF LM DESCENT STAGE PROPELLANTS, THE VEHICLE CAN OPERATE WITH A 4.6 NAUTICAL MILE RADIUS WITH NO PAYLOAD.

INTRODUCTION

Exploration of the moon following the early Apollo landings presents a means of continuing a viable U.S. manned space program during the next decade. The national investment in the Apollo program has already created the necessary technology and prime hardware which allows continuation of a meaningful program concerned with exploration of the lunar surface.

Achievement of this objective requires increased mobility to allow transport of astronauts, scientific equipment, and samples between the lunar landing site and predetermined sites of scientific interest. Of the several mobility devices considered by NASA, a one-man, rocket-powered, lunar flying vehicle (LFV) provides unique mobility features which are significant to lunar exploration. With such a device, up to 370 pounds of scientific equipment and samples can be transported in a few minutes to sites which cannot be reached with surface devices because of adverse lunar surface features and limitations of life support systems. The speed of transport is also important in increasing the number of regions that can be visited during missions of limited duration. The rapid access to remote and, in some cases, otherwise inaccessible areas provided by the lunar flying vehicle permits a capability for a low incremental cost.

The speed of transport also provides a means of reducing mission hazards related to time-critical life support systems that must be used when the crew is away from the lunar module (LM). During a normal LFV mission, the LFV allows a rapid return in the event of a life support system failure. Furthermore, the LFV can provide a standby capability for timely surface rescue from walking and roving vehicle missions.

The primary objectives of this study were to optimize the design and to develop system specifications of the lunar flying vehicle. The scope encompassed parametric investigations, concept generation, and evaluation effort for the definition of a recommended concept; production of a preliminary design and development of systems specifications of the recommended concept; and definition of resources and crew training plans. In addition to generation of the LFV design, the scope of the study included lunar module integration, flight suit interface studies, and definition of ground support equipment for earth and lunar operations.

The study was conducted in accordance with the logic diagram shown in Figure 1. During the first phase of the study, which was of three months duration, conceptual tradeoff studies were conducted to provide parametric data for evaluation of a large matrix of design alternatives and the selection of a concept for the second phase preliminary design effort. During the second phase of the study, also of three months duration, more detailed tradeoff studies were conducted to develop a preliminary design and specification for the lunar flying vehicle system. During this second phase, a resources plan and a crew training plan were also developed.

The following sections of this report summarize the study results. A detailed description of the study results is contained in Volume 2 - Mission Analysis, Volume 3 - Subsystem Studies, Volume 4 - Configuration Design, Volume 5 - Preliminary Design and Specification, and Volume 6 - Training and Resources Plans.

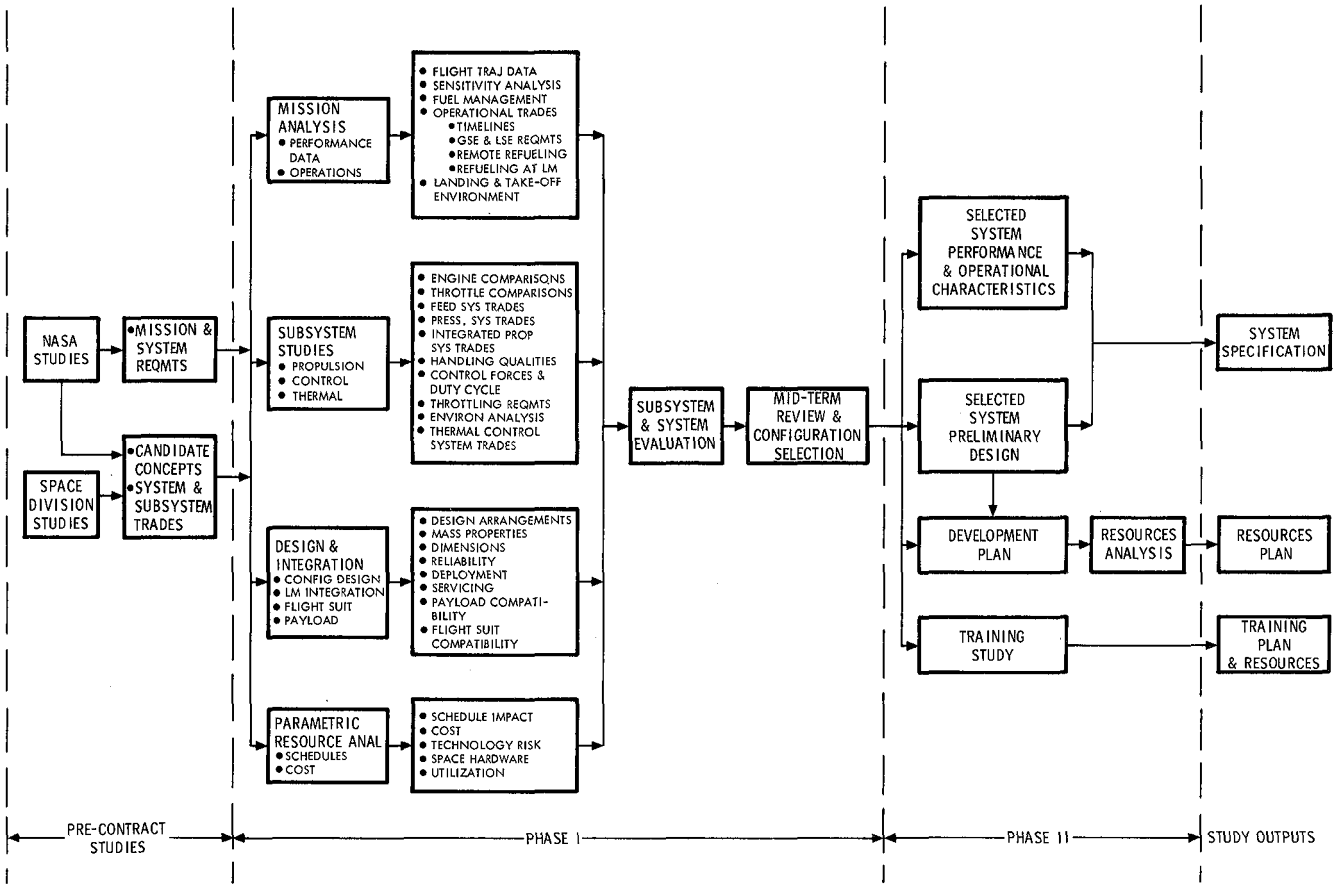

Figure 1. Study Approach

PRINCIPAL ASSUMPTIONS

At the beginning of the study, an assumption was established that two lunar flying vehicles were to be used on each mission; one of the vehicles was to be a surface rescue vehicle. During the second month of the study, the astronaut rescue requirement was eliminated, and a single LFV was then assumed to be employed in surface operations. This vehicle design was constrained by the following basic requirements: (1) use of LM descent stage residual propellants, (2) capability of carrying a fully suited astronaut and a maximum of 370 pounds of cargo, (3) capability of all LFV operations being performed by one man, (4) loaded propellant weight of approximately 300 pounds (including 10 percent fuel reserve), and (5) no allowance of engine shutdown during flight.

In addition to these basic ground rules, the Space Division applied important constraining criteria to the design relative to system crew safety and the characteristics of the lunar surface. Crew safety goals were assumed to be consistent with current Apollo system goals; therefore, single-point system failures were allowed only in those systems which could be judged to achieve high reliability within a reasonable design and development schedule. A Phase C and D schedule of 30 months, with delivery of the first operational flight article in April 1972, was assumed for this analysis. These crew-safety goals were also employed to establish a requirement for satisfactory handling qualities within a reasonable crew training period.

Because the lunar surface characteristics are poorly defined at this time, the landing system was designed to land consistently on a hard surface (which establishes attenuation requirements) or a soft surface (which establishes tip-over requirements). Application of these criteria results in a system with high lunar exploration utility, thus allowing exploration into those areas which cannot be reached by surface systems, such as crater walls and mountainous regions.

Since this system will be landed by a modified lunar module, constraints imposed by the LM system and the spacecraft-lunar module adaptor (SLA) interface were included in the study. Stowage in the LM descent stage corner compartments of quads I and IV was assumed for this study.

The baseline mission for this study was of three days duration, being initiated either at dawn or prior to sunset. The effects of longer stay times and earth-shine operations on design were also investigated.

LFV MISSION DESCRIPTION

A typical LFV mission profile is shown in Figure 2. The LFV, shown stowed in the corner compartment of quad IV of the LM descent stage, was designed to be stowed without removal of the landing gear and is covered during lunar transit to protect it from impingement of the reaction control system (RCS) rocket plume and from lunar surface debris upon landing.

After the lunar surface landing and post-landing LM operations, one of the astronauts leaves the LM and removes the LFV from the descent stage bay. The removal equipment is designed for one-man operation and does not allow the vehicle to swing pendulously as it is removed. One of the primary LFV design features is the intact LM installation. Only the payload racks and the control pedestal arms are folded during stowage. These are simply rotated into a locked position after the vehicle is removed from the LM.

The astronaut next prepares a landing and takeoff site located about forty feet directly in front of the LM exit. The site comprises a high-temperature ground cover which is staked to the ground. This cover prevents eroded soil from striking and damaging the LM ascent stage during takeoff and landing operations. The unfueled LFV is transported to the prepared site by the astronaut and is fueled by hoses attached to the fuel and oxidizer drain lines of the LM.

Following a brief checkout of the LFV system, a short flight within walk-back range is first accomplished to allow astronaut familiarization under actual lunar flight conditions. For landings at the remote site, damage from lunar soil erosion caused by rocket impingement is avoided by cutting thrust at a 58-inch level and dropping in. This also reduces the tendency to tip on landing. After deploying experiment modules and gathering samples, the astronaut deploys a small, expendable ground cloth beneath the LFV to eliminate soil erosion effects during takeoff and returns to the LM.

Following a sleep and rest cycle, replacement of the helium tanks, and refueling of the LFV, the astronaut accomplishes subsequent missions of longer range, depicted in Figure 2, to remote sites of scientific interest. Based on 1000 pounds of LM descent stage propellants, about two long-range and two short-range missions can be accomplished. Such operations appear to be consistent with a three-day surface-mission cycle.

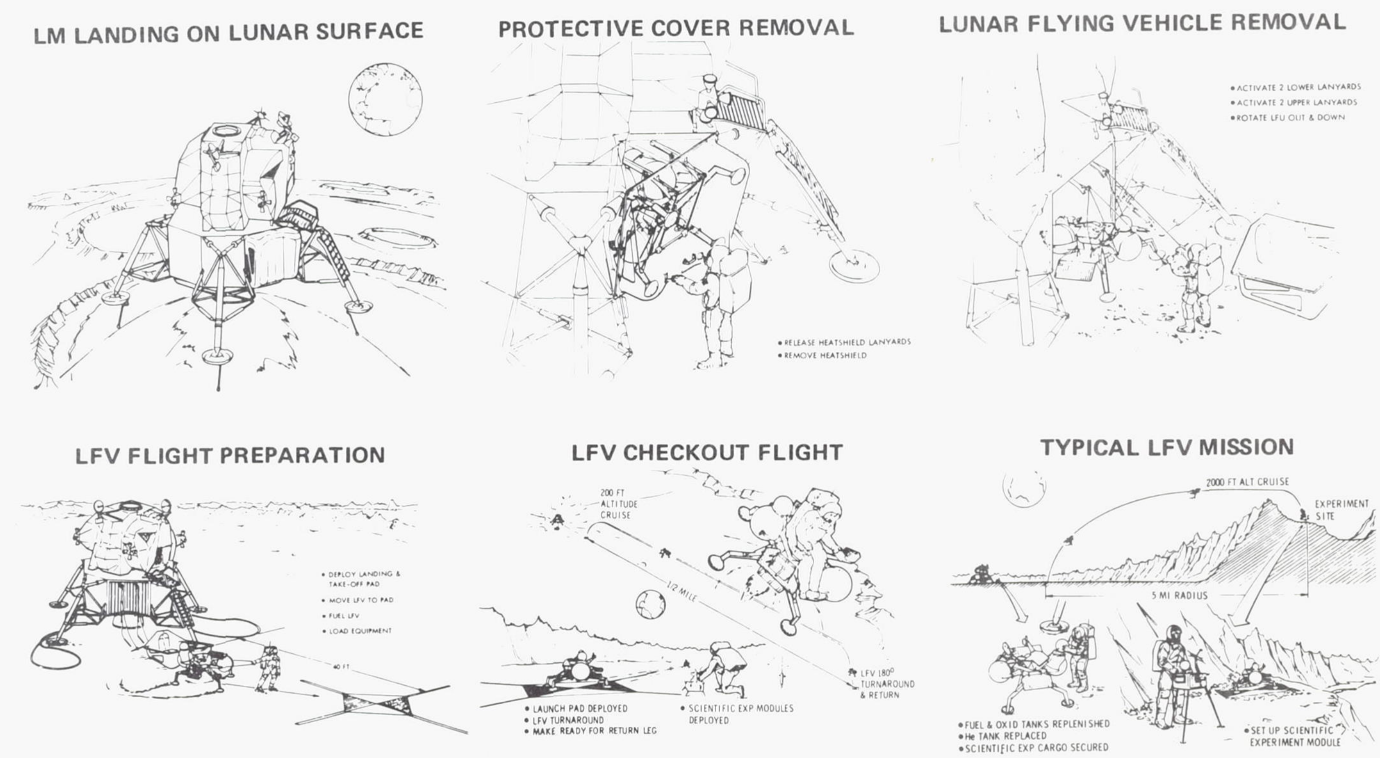

Figure 2. Lunar Flying Vehicle Operational Modes

SIGNIFICANT RESULTS

The first phase of the study concentrated on conceptual tradeoffs at the subsystem and system integration levels and resulted in the selection of a concept for preliminary design study during the second phase. The following sections describe the significant results of these two study phases.

TRADEOFF STUDIES

Of the many tradeoff studies conducted during the first phase, those of greatest significance were (1) control-system studies, (2) determination of the number and arrangement of engines, (3) studies of crew position and restraints, and (4) landing attenuation studies. The primary goal of the tradeoff studies was to determine the design characteristics that would produce a system in which high confidence could be placed. The following sections summarize the results of these studies, evaluation of the tradeoff data, and concept recommendations.

Control-System Studies

Three basic types of control systems were evaluated: kinesthetic (body motion), hardwire, and stability augmented. These are listed in the order of increasing complexity. Kinesthetic control allows a fixed main thruster. Thrust-vector control is obtained by platform tilting achieved in pitch and roll by movement of the body. Yaw control requires a reaction-control system or gimbaled thrusters. Hardwire and stability-augmented concepts were studied parametrically, with the thruster-pivot location varying from below the center of gravity to above the center of gravity. An option of the hardwire system, in which spring and damper networks were introduced into the system between a hand control and the engine, was evaluated to assess the potential for improving handling qualities by such an approach.

Several sources of data, illustrated in Figure 3, were utilized in assessing the handling qualities of the candidate control concepts. These included a visual simulator, a high-inertia tethered vehicle, and theoretical dynamics studies. The NR HOTRAN visual simulation facility provided a 6-degree-of-freedom, wide spherical screen (+100 degrees horizontal and +25 and -65 degree vertical view), and pitch and roll rotating base (±7 degrees). This provided an excellent peripheral view and the ability to change control modes and parameters quickly.

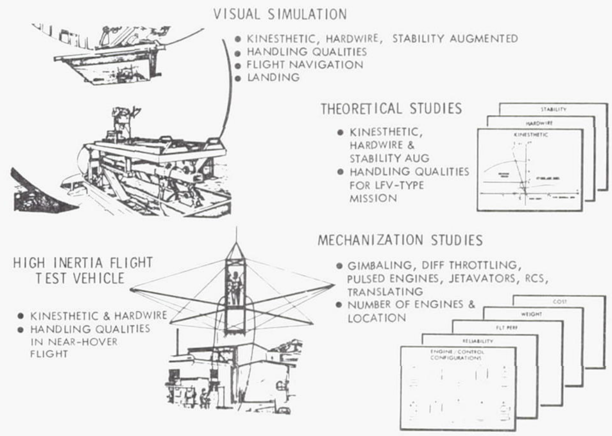

Figure 3. Control System Studies

The tethered flight vehicle was employed to assess handling qualities of both kinesthetic and below-c.g. gimbaled hardwire control systems in flight under 1-g conditions. This vehicle was designed to provide inertias up to 600 slug-feet^2 for simulating rotational dynamics at lunar conditions. High-pressure nitrogen gas, fed to the vehicle through hoses, provided rocket system thrust.

A sophisticated mathematical model, including man-in-the-loop dynamics, was also employed to correlate and explain the simulation results.

Figure 4 summarizes the visual and flight simulation results, presenting the Cooper ratings as a function of inertia (all conditions are related to lunar flight). Systems must have Cooper ratings between 1 and 3 to possess satisfactory handling qualities. The kinesthetic results indicate that satisfactory handling qualities did not occur across the entire range of inertias investigated. The best kinesthetic handling qualities occurred at an inertia of 50 slug-feet^2 under lunar flight conditions. In the region of interest for the LFV (I = 100 to 200 slug-feet^2), the handling qualities were rapidly degraded. At LFV inertias, 1-g tethered flights

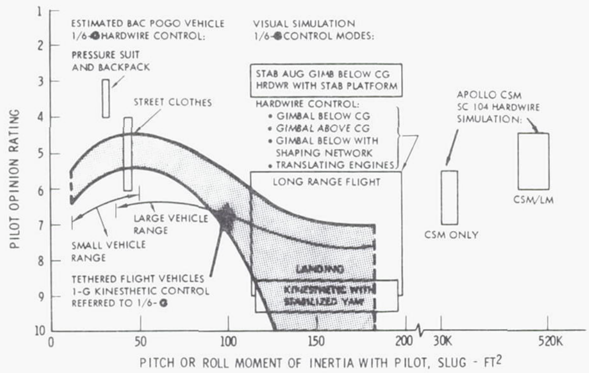

Figure 4. Handling Qualities Comparisons

were characterized by large, slowly damped, pitch and roll oscillations of relatively low frequency.

Hardwire control was also found to provide unsatisfactory handling qualities. Visual simulation resulted in Cooper ratings between 5 and 9 for long-range flight, but landing runs resulted in Cooper ratings of 9. Tethered-flight results indicated a mixing of kinesthetic and hardwire control during flight. Using this technique, the pilots could fly the vehicle at lunar flight inertias of 50 slug-feet^2, but had considerable difficulty at 100 slug-feet^2. Two other sources of hardwire data are also shown: (1) hardwire data obtained by Bell Aircraft Corporation (BAC) at the Langley Lunar Langley Research facility under tethered 1/6-g conditions and (2) visual simulation data for the CSM and LM/CSM utilizing hardwire control. The BAC data indicate Cooper ratings of between 3 and 6 at relatively low vehicle inertias. The Apollo data gave Cooper ratings for hardwire control between 4 and 7. It should be noted that the pilot task for the CSM and CSM/LM simulation involved primarily 3-axis attitude control and, thus, was a considerably simpler task than the LFV landing task. It was concluded that hardwire flight could be accomplished with considerable training, but the large percentage of time required for the control task takes too much time from navigation. If the pilot loses control concentration for even a short time, loss of vehicle control may result.

Stability-augmented systems, as expected, provided acceptable handling qualities. This system has an Apollo-type hand controller that is used to command pitch, roll, and yaw rotational rates and has attitude-hold when the control is returned to neutral.

Table 1 compares the Cooper ratings for the control systems investigated during this study. Good agreement among the methods of investigation is indicated. Only the stability-augmented system provides acceptable handling qualities.

Table 1. Handling Qualities Summary

Control Method	Pilot Opinion Ratings*		
	Theoretical Analysis	Visual Simulation	Flight Vehicle
Kinesthetic	7 to 8	9 to 10	5 to 8
Hardwire			
Above c.g.	5 to 6	6 to 9	---
Below c.g.	5 to 6	6 to 9	7 to 9
Stability augmented			
Below c.g.	2 to 3	3	---
At c.g. (level platform)	1 to 2	2	---
*Cooper rating scale			

Control System Mechanization

Control-system mechanization studies were significant in making the decision on the number of engines. An important portion of these studies was concentrated on the reliability aspects. Figure 5 presents a comparison of several combinations of outboard and high and clustered and low engine combinations, which were two primary alternatives in design. All of the outboard and high-mounted engine concepts were rejected, with the exception of the eight-engine, differentially throttled case

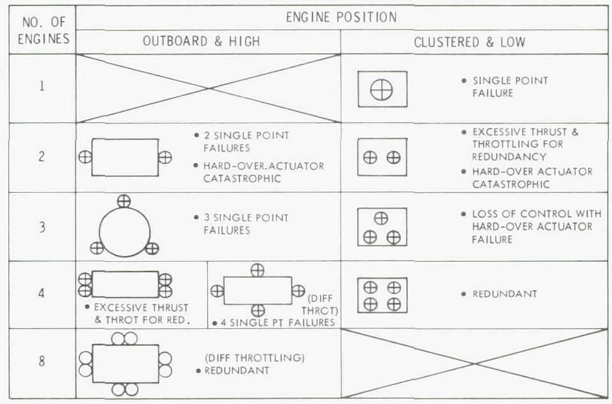

Figure 5. Comparison of Engine Location Options

which can be used with Apollo RCS pulse-modulated thrusters. This design case was further studied on a Company-sponsored program. All other outboard and high combinations resulted in sources of catastrophic multiple-engine failures or in excessive thrust and throttling requirements if redundancy were employed. Of the clustered and low-mounted engine concepts, only the single-engine and four-engine cases were retained. If an engine fails in a clustered and low case, engine gimbaling through the center of gravity can be achieved to provide control.

Several combinations of single- and four-engine configurations and one three-engine configuration (employing an RCS system for rotational control) were evaluated in arriving at the recommended concept. These included all-gimbaled concepts, all-RCS concepts, concepts which combined gimbaling and RCS, and translating plate concepts (Figure 6). A special case, the level-platform concept (number 15), has a hardwire main thruster that gimbals through the c.g. for translation control. The platform is stabilized by using an RCS system.

A summary of the data comparing the concepts is presented in Table 2. These data indicate that the single-engine concepts 2 and 3 provide the lowest dry weights and largest operational radii. However, crew safety for the four-engine concept is nearly ten times greater than for the single-engine concepts. These reliability estimates include estimates for the entire system, which is assumed in all cases to be stability augmented. The engine reliability was dominant for the single-engine case. All engine reliabilities were based on a 24-month engine development cycle.

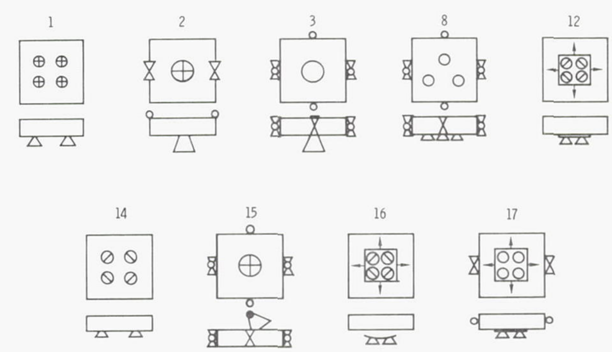

Figure 6. Engine and Control Configurations

Table 2. Engine and Control Evaluation Data Summary

Factor	Units	Configuration 1	2	3	8	12	14	15	16	17
Dry Weight	lb	301	284	279	290	302	296	302	302	302
Δ Radius	nm	0.00	0.76	0.70	-0.28	0.35	0.06	0.24	0.35	0.12
Reliability										
Crew Safety		0.9995	0.998	0.998	0.9995	0.9995	0.9996	0.998	0.9995	0.9995
Mission Success		0.984	0.988	0.974	0.970	0.988	0.988	0.974	0.988	0.982
Handling Qualities	C.R.	3.00	3.00	2.75	2.75	2.75	3.00	2.75	2.75	
Resources										
Cost	Ratio	1.02	1.12	1.11	1.14	1.02	1.02	1.15	1.02	1.15
Risk	Pen. Pts	2	3	2	3	4	2	4	4	6
Operational Character,										
Servicing Time	units	8	8	12	12	6	8	13	6	8

Despite the better performance and design simplicity of the single-engine concept, the four-engine concept was recommended because crew safety was considered undesirably low for the single-engine concept.

Landing-Attenuation and Crew-Position Studies

As a result of the unacceptable reliability associated with locating the engines outboard above the c.g., the potential hazards of lunar soil erosion upon landing related to low-engine mounting (12 inches from the ground at touchdown) had to be eliminated. Surveyor data, data from recent JPL tests at the NASA Langley Research Center, and theoretical results were analyzed to determine the altitude where soil erosion effects may damage the vehicle. This was conservatively estimated to be 58 inches from the rocket nozzle exit plane. These data, along with vehicle flight conditions, were then used to determine landing touchdown conditions, assuming engine cutoff at 58 inches above the surface. This altitude is determined by the use of a ground probe which activates a light; the pilot initiates the action for cutting thrust based on this information. Cutting off prior to landing is beneficial since resulting vertical velocity reduces the tendency to tip for a given horizontal velocity. Thrusting at touchdown significantly increases the tendency to tip over, particularly if the thrust is inadvertently increased by the pilot just as touchdown occurs.

The approach and considerations for resolving the landing attenuation scheme and the pilot position are illustrated in Figure 7. The initial LFV concept featured a standing and essentially unrestrained pilot. The landing gear concept led to difficult LM stowage and the requirement for 12 attenuators. Alternative attenutation studies included consideration of a Surveyor-type tripod attenuator and an integral-leg-type attenuator. In the integral-leg-type attenuator, the legs are rigidly tied together, and attenuation occurs between the leg frame and the body. These studies were combined with a study of the effects of pilot position and restraints on attenuation requirements. Stowage in the LM and ingress and egress were also considerations in selection.

Table 3 summarizes the characteristics of the alternatives. Based on NASA MSC experimental

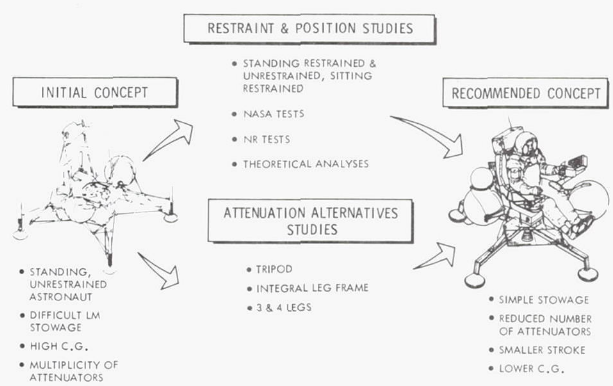

Figure 7. Landing-Attenuation and Crew-Position Studies

Table 3. Comparison of Crew Position and Attenuation Options

	Pilot	Standing, Unrestrained		Standing, Restrained		Seated, Restrained	
Characteristic	Attenuator	Tripod	Integral Leg Frame	Tripod	Integral Leg Frame	Tripod	Integral Leg Frame
Allowable g's							
Vertical		2	2	8	8	8	8
Horizontal		1	1	4	4	4	4
C.G. height (inches)		66	60	60	57.1	45	42
Landing gear spread (inches)		139	129	129	123	102	96
LM stowage		Remove legs	Fold legs	Remove legs	Fold legs	Remove legs	Intact
Weight (gear and restraint (pounds)		69.9	68.5	74.4	72.3	66.6	52.5
Attenuator Stroke (inches)		25.0	13.5	12.5	6.8	12.5	6.8

data and NR Space Division theoretical and experimental data, it was concluded that the standing, unrestrained pilot had to be attenuated to low horizontal and vertical g levels (one and two, respectively) to assure fully spacesuited stability during landing. Even under these low-g conditions, standing astronaut landing stability is questionable. The low-g level results in a large attenuation stroke and a large landing gear spread because of the resulting high c.g.

As a result of this study, the seated-and restrained-pilot position with the integral leg-frame gear was selected because of low weight, ability to stow in the LM without SLA interference, and the relative ease of providing positive restraints. Mock-up tests of ingress and egress showed that a fully pressurized astronaut with a portable life support system (PLSS) could easily ingress and egress from this position because of the resulting low seat (seat level about 36 inches from ground level.) A four-legged gear was selected over a three-legged gear primarily because it can be stowed in the LM intact.

Recommended Concept

In addition to the key tradeoff studies described above, several others were conducted at the subsystem and system integration level in arriving at a recommended concept. Included were thermal tradeoff studies, payload studies, and integrated propulsion system studies.

A three-view of the concept recommended at the midterm point of the study is shown in Figure 8. The characteristics of this concept and reasons for selection are described in Table 4.

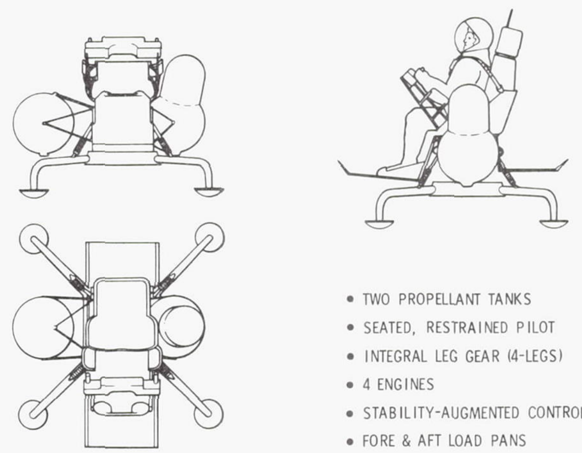

Figure 8. Recommended Concept at Midterm

Table 4. Midterm Recommendations

System Characteristics	Reason
Stability-augmented control	Others do not have acceptable handling qualities.
Four engines	Unable to achieve acceptable reliability for single-engine in 24-month delivery cycle.
Actuation system	Potential of alternative designs to be resolved.
Astronaut seated and restrained	Low c.g. reduces landing gear spread, restraints provide positive astronaut and PLSS landing stability.
Integral leg-frame landing gear	Simple, low-weight solution.
Four-leg gear	Leg folding or removal not necessary for LM stowage; lowest weight.
Two spherical propellant tanks	Lowest weight; stows in LM.

PRELIMINARY DESIGN STUDY

Several design issues were studied during the preliminary design phase. Key issues remaining after selection of the recommended concept included (1) selection of the control method for the four-engine design, (2) optimization of the landing-attenuation system, (3) optimization of the payload arrangement, and (4) definition of the ingress and egress method. As is usual in preliminary design studies, these issues were interrelated and involved several design layouts to resolve problems associated with clearances, visibility, payload integration, and ingress and egress. Some of the primary results are presented in the following sections.

Engine Actuation Study

Three methods of actuating the four-engine configuration to obtain thrust-vector control, shown in Figure 9, were studied to obtain detailed data on their characteristics. The requirement was imposed in all cases for redundancy in the event of an actuator failure. The eight-actuator case automatically provided redundancy with eight single actuators. Both the four-actuator and sliding-plate systems required "two-in-a-can" dual actuators for redundancy, and an automatic or pilot-initiated switchover to the redundant actuator was required. The sliding-plate concept achieved thrust-vector control by the translation in the vehicle horizontal plane of a plate containing the four engines for pitch and roll control. The engines are single-axis gimbaled for yaw control. Thrust-vector control for the four- and eight-actuator concepts is obtained by gimbaling (in both axes for eight actuators and in one axis for four actuators).

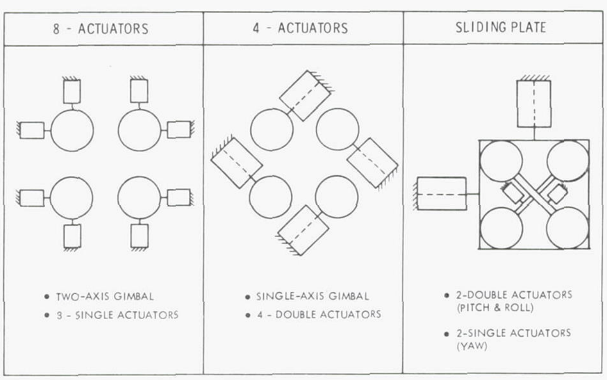

Figure 9. Candidate Four-Engine Actuation Configurations

A detailed analysis of these systems included determination of gimbal-angle or plate-translation requirements, actuation-hardware characteristics (including mechanism, weight, power, and reliability), and an analysis of the results of actuator failures. Because of the requirement for switchover from one set of actuators to the other for the four-actuator and sliding-plate concepts, both proved to possess time-critical failure characteristics. Since the "two-in-a-can" actuators are physically in the same environment, their true redundancy was questionable. The eight-actuator concept was fully redundant, without time-critical characteristics; had the lowest gimbal angle requirement; and could use relatively simple, modified, off-the-shelf hardware. Although it proved to be higher in weight than the four-actuator concept, the eight-actuator concept was selected.

The recommended control system concept provides complete redundancy in the event of system failures, including engine, actuator, control unit, power, or hand controller failures. Redundant control units, batteries, power distribution, and hand controller switches provide a completely redundant stability-augmented system. Hardwire control for backup instead of control system redundancy was considered, but the incremental training required to attain proficiency was considered excessive.

Landing-Attenuation System Optimization

Resolution of the landing-attenuation system design involved a study of two framing techniques shown in Figure 10 and a detailed landing dynamics study to resolve the number and orientation of attenuators.

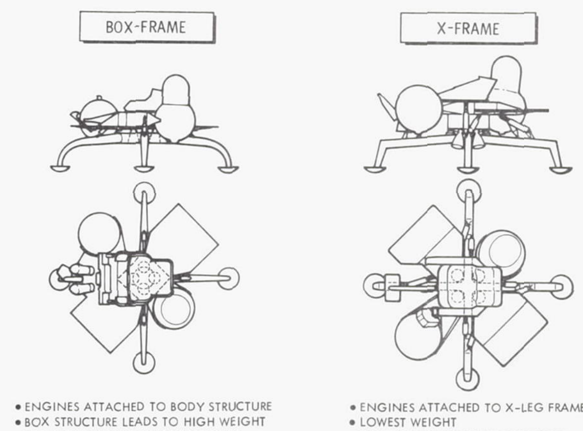

Figure 10. Optimization of Landing Attenuation System

The box-frame concept has the engines attached to the body frame with a box structure tying the four landing gear legs together. The X-frame concept has the engine attached to a landing gear X-frame. The X-frame concept led to the lowest weight and best clearance characteristics. Furthermore, location of the engines on the X-frame resulted in a larger c.g.-to-gimbal-axis distance, thus reducing engine-gimbal requirements. The landing dynamics analysis showed that the maximum g's are 20 at the engine location for a maximum velocity, hard-surface landing because of leg deflections. This level is well within the acceptable range for the engine and gimbal systems. For these reasons, the X-frame concept was selected.

Although it is possible to design a system without attenuators for soft lunar-surface landings, current lack of knowledge about soil conditions in regions other than the maria dictated the requirement to land on either a hard or soft surface. Relaxation of this assumption would lead to a simpler and lower-weight design, but must await further data on lunar surface conditions, particularly in the interesting selenological regions where the LFV would be most useful. The recent Apollo 11 flight provided data that tends to substantiate this conservative approach. The surface appeared to have many small craters, to be strewn with rocks of various sizes, and to have regions of both hard and soft soil.

A modified Surveyor-type hydraulic attenuator was employed in developing the attenuation system design. Other designs, including springs and reusable inelastic metal deformation attenuators, were also considered.

Four attenuators were initially studied (two in pitch and two in roll), but they proved to be unstable in providing attenuation in the yaw axis. Eight attenuators were found to provide stability and a certain degree of redundancy. Although the current design uses eight attenuators, further study may prove that six are sufficient. The preload on these attenuators is much higher than the forces and moments contributed by the engines. For this reason, location of the engines on the landing gear frame does not result in a dynamic interaction during flight.

Payload and Ingress and Egress Optimization

Figure 11 illustrates the process followed in optimizing the payload arrangement and the method of ingress and egress. The concept at midterm resulted in difficult ingress and egress because the payload racks were in the same plane as the pilot seat. This condition also resulted in poor landing visibility because of the forward payload location.

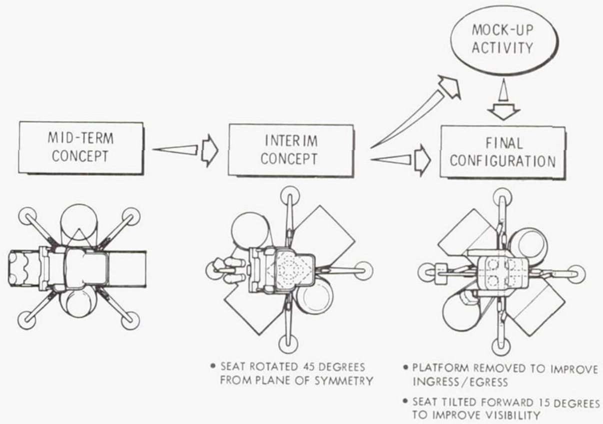

Figure 11. Optimization of Ingress and Egress

After the pilot's position was rotated 45 degrees from the plane of symmetry, the pilot no longer had to contend with payload interference in ingress, egress, and visibility. Full-scale mockup studies were conducted using a pilot in a fully-pressurized suit with a PLSS on his back (Figure 12). The pilot had considerable difficulty in stepping onto a platform and getting into a seated position, even with hand holds and steps. The final concept, which was derived in conjunction with the mockup activity, eliminated the

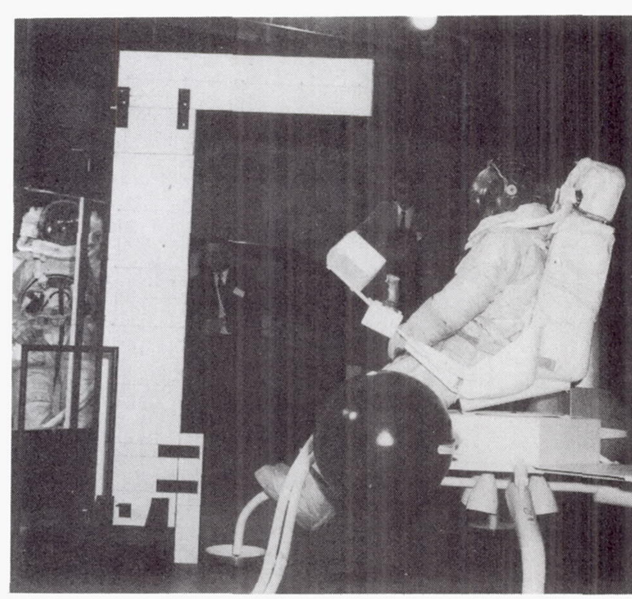

Figure 12. LFV Mockup With Pilot in PLSS

platform. Ingress was accomplished by backing up to the seat (straddling the forward landing leg) and simply pushing with the legs against bars located on the landing leg to get into the 36-inch-high seat. (This was accomplished in three seconds from the time the pilot reached the seat.) Although the current design has a footrest attached by a bar from the lower portion of the seat, the alternative of a footrest on the legs may prove practical when landing-dynamic tests are accomplished in future program phases. Egress is accomplished by taking the legs off the footrest and sliding off the seat onto the feet. Visibility in the seated position is excellent.

Preliminary Design

A perspective view of the preliminary design is shown in Figure 13. More detailed design drawings of the concept are presented in Volume 5 of this report, and a description of the subsystems is presented in Volume 3. A summary mass-properties statement is presented in Table 5.

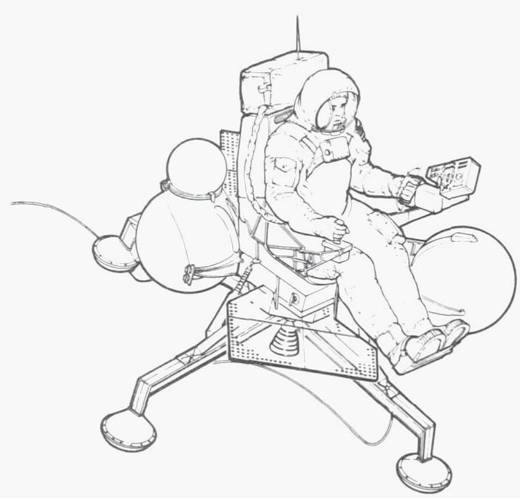

Figure 13. General Arrangement

Although the rescue mode was dropped from consideration during the study, the design can be modified to provide this capability by raising the payload pans a few inches and raising the seat enough to provide clearance between the bottom of the seat and an astronaut prone across the load pans. Design modifications required to utilize the LFV for two-man lunar escape were also considered. This requires removal of the load pans and

Table 5. Summary Weight Statement

System	Weight (pounds)
Body structure	36.9
Environmental protection	11.0
Landing	62.2
Propulsion	98.4
Power	18.3
Power conversion and distribution	8.8
Navigation and control	32.9
Personnel provisions	9.3
Crew station controls and panels	25.8
Dry weight*	303.6
Pilot	380.0
Residuals	9.0
Minimum burnout weight	692.6
Maximum payload	370.0
Propellant	300.0
Maximum gross weight	1362.6

*Includes 10-percent growth for all systems

INERTIAS (slug feet^2)

Axis	Burnout Weight			Gross Weight		
	Pilot Only	Pilot + 100-Lb Payload	Pilot + 370-Lb Payload	Pilot Only	100-Lb Payload	370-Lb Payload
Pitch	53	64	104	84	94	133
Roll	45	57	96	76	86	125
Yaw	37	57	130	87	106	180
Product	--	--	--	-1.2	-1.3	-1.4
Principal Axis (Deg)	--	--	--	-6.2	-3.6	-1.5

addition of two propellant tank and engine modules to provide sufficient propellants and thrust for the escape.

The flight performance capability of this concept was established by analysis of visual simulation results. Figure 14 summarizes propellant requirements (burned propellant to initial gross weight) as a function of one-way range assuming a constant-altitude flight mode. The effect of landing

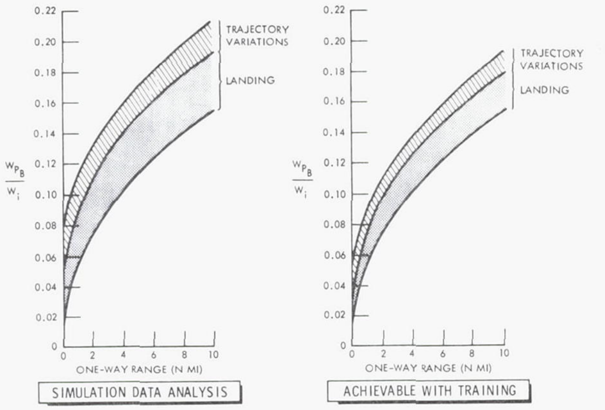

Figure 14. Pilot Influence on Propellant Requirements

and trajectory variation propellants on total propellant requirements is illustrated. The data on the left were taken directly from simulation, and the data on the right (achievable with training) illustrate anticipated requirements with additional training. Figure 15 presents weight and performance tradeoff data as a function of propellant weight. The current design point, using 300 pounds of propellant, has a dry weight of about 300 pounds, including a 10-percent growth factor. Since this weight is based on relatively conservative hardware estimates, future weight-saving programs could reduce the weight to about 260 pounds as a design goal. The ranging-capability curve uses the data from the "achievable with training" curve in Figure 14 and includes 3-percent residual and 10-percent reserve propellants in the propellant weight. This figure shows that the current concept has an operating radius of about 4.6 nautical miles with no payload. The maximum flight velocity for

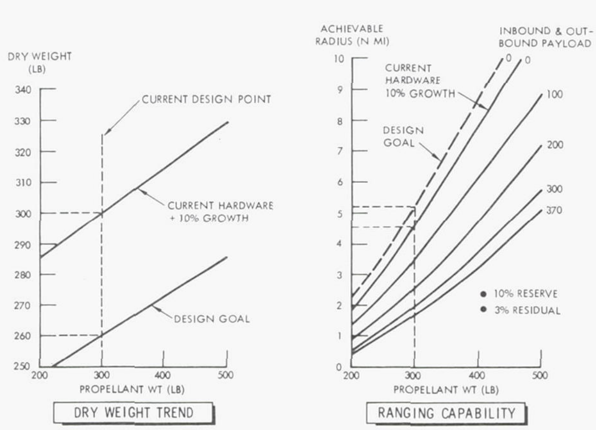

Figure 15. Achievable Ranging Capability and System Tradeoffs

this range is 280 feet per second. Increasing the propellant load to 400 pounds increases the radius to about 7.8 nautical miles at a dry-weight increase of 15 pounds. Future studies should consider the advisability of increasing the propellant load based upon mission considerations.

DEVELOPMENT SCHEDULE

The development schedule for the LFV is primarily influenced by development of the engines, which requires from 17 to 20 months. The simplified schedule, shown in Figure 16, was derived after an analysis of data received from several subcontractor sources. This schedule assumes combined Phases C and D activities initiated in October 1969. Delivery of the first operational vehicle is scheduled by April 1972.

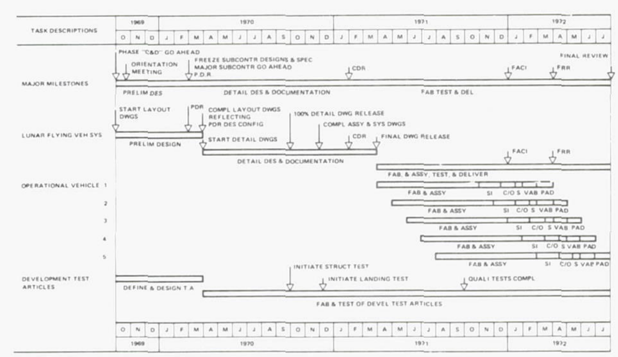

Figure 16. Development Schedule

CREW TRAINING

Crew training for the LFV mission would be similar to that currently used for the Apollo program. Training would sequentially involve (1) mission and system familiarization classes, (2) detailed system and mission classes along with full-scale mockup and visual simulation activities, and (3) flight training. The flight-training program would require both 1/6-g tethered training at a facility like the NASA Langley Lunar Landing Research facility and free-flight training. A modified FLEEP or 1/6-g LM training vehicle could be used for the 1/6-g training. The 1-g LM training vehicle could be modified for free-flight training. Because of the similarity between the LFV and LM control characteristics, it is anticipated that only a brief training period would be required.

Navigation training would be accomplished with a visual scene simulator with a lunar surface model closely resembling the actual mission site. Since the 1-g LM training vehicle can not fly at the high velocities related to the LFV peak cruise velocity, it is recommended that a helicopter be employed for navigational training at high velocity over regions of the earth with topography similar to the moon.

RELATIONSHIP TO OTHER NASA PROGRAMS

This effort is closely related to all other activity associated with advanced lunar missions and with certain research work currently being conducted within NASA. Because this system represents a potential payload on an extended lunar module (ELM), a strong interface exists that implies potential modifications to the lunar module to provide translunar LFV stowage and propellant transfer on the lunar surface.

In the planning of future lunar mission objectives, the potential of the flying-vehicle concept when used alone or in combination with other lunar transport vehicle (such as the Rover) will be important in defining mission objectives. The technology and potential mission uses associated with the LFV and the work currently in progress on the Lunar Emergency Escape to Orbit Vehicle Study (NAS1-8923) are also closely related.

Planned fabrication and flight testing at the NASA Langley Lunar Landing Research facility of a FLEEP vehicle (for kinesthetic and hardwire control system research) will provide additional valuable data related to these simple control system approaches. This vehicle could also provide, when appropriately modified, a system for astronaut training.

SUGGESTED ADDITIONAL EFFORT

Results of the present effort have produced a design that provides a safe and useful technique for lunar exploration. Additional effort falls into two categories: Phases C and D effort, and supplemental preliminary design and simulation effort to permit further simplification of the system or to establish firmly that further simplification is undesirable due to impact on safety.

PHASES C AND D EFFORT

For timely development of the LFV system, Phases C and D should be combined. Initiation of these phases should, if possible, follow completion of the present effort almost immediately. Combining of Phases C and D will allow the early procurement of the long lead time engines and the orderly design, procurement, fabrication, and testing of other subsystem and system elements. The Training and Resources Plans, Volume 6, contains the schedules related to these activities.

IMMEDIATE FOLLOW-ON EFFORT

Immediate follow-on activities have been identified in two areas: simulation and design.

Simulation

The objective of devising a simple control system with acceptable handling qualities for application to the LFV should continue to be pursued. During the current program, none of these systems has been found to possess acceptable characteristics. Flight simulation research was conducted with the pilot in a standing and unrestrained position employing a relatively short control handle (5 inches) for hardwire gimbal control of the engines. Although visual simulation work included the introduction of spring and damper networks between the hand control and the engines, this has not been attempted in tethered flight. Additional visual simulation work with spring and damper networks should be accomplished to probe further the parametric characteristics of these systems. Variations in the length and configuration of the hand control may also prove to be of parametric interest. A combination of flight and visual simulation would be required to allow the regions of interest to be effectively analyzed.

The tethered-flight vehicle could be modified to restrain the pilot and to include spring and damper networks in the region that is determined optimum from visual simulation and to include hand-control configuration variations.

Design

A brief study was conducted of concepts using the LM RCS engines in the pulsed mode. It was found that a feasible concept could be realized which had a radius of operation about one nautical mile less than the current design concept. This concept used the pulsed thrusters for main propulsion as well as for platform rotation. (Rotational control was obtained by differential pulsing.) Since use of these engines may reduce program cost and shorten the development cycle, it is recommended that a preliminary design of this concept be produced.

Additional effort related to the current concept would be desirable to optimize the design. The mockup work, conducted during the later portion of the program, resulted in improvements in the astronaut and vehicle interface areas. Although these changes have been incorporated into the preliminary design, they further impact total system integration and may lead to a more compact arrangement. For this reason, a final iteration of the design would be of value.