STUDY OF

ONE MAN LUNAR FLYING VEHICLE

FINAL REPORT

REPORT NO. 7335-950010

JULY 1969

Prepared under Contract NAS9-9044

by Bell Aerosystems Co.

For

National Aeronautics and Space Administration
Manned Spacecraft Center

Houston, Texas

BELL AEROSYSTEMS - A textron COMPANY

NASA CR-101944

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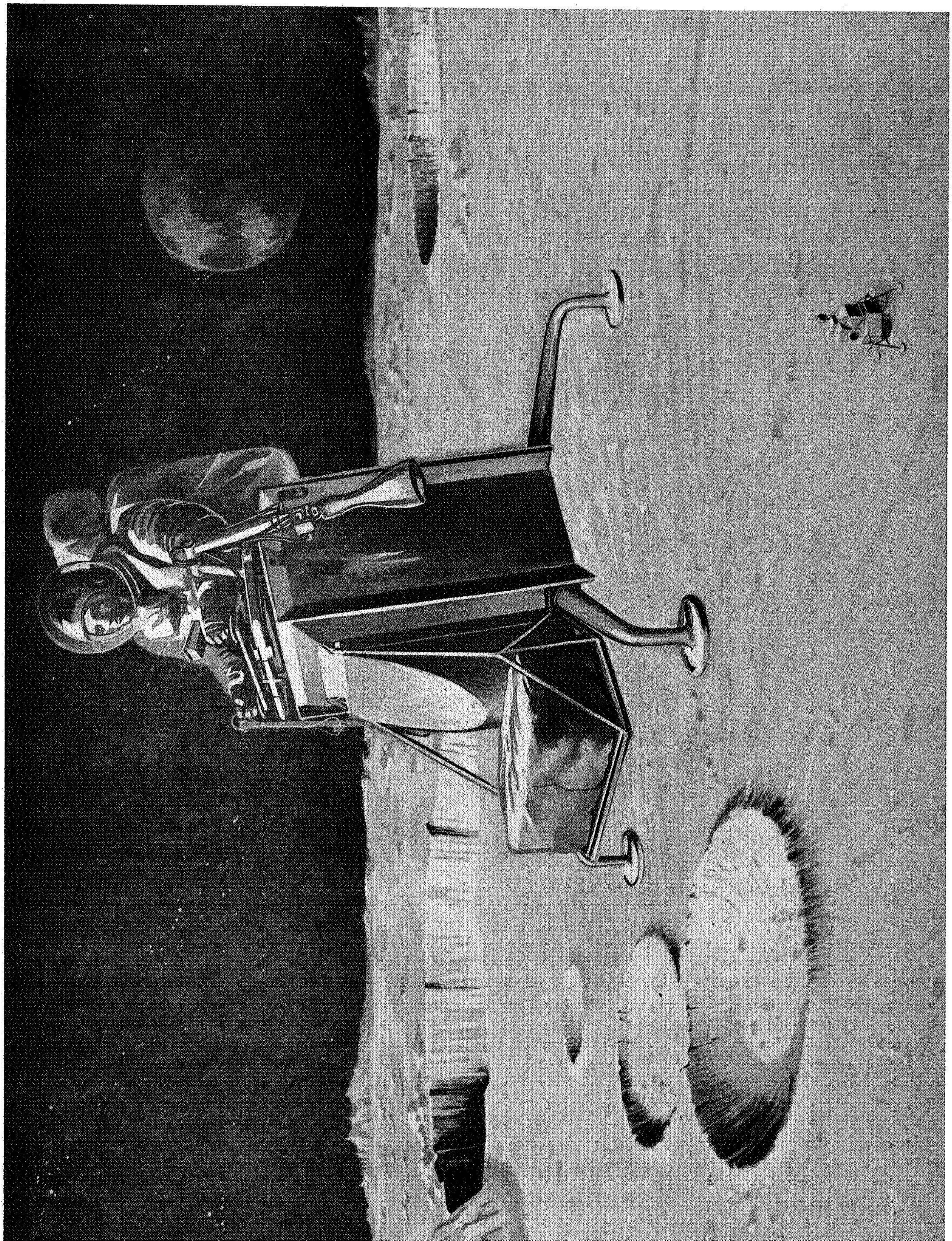

FOREWORD

This report presents the results of a study of a One Man Lunar Flying Vehicle, conducted for the National Aeronautics and Space Administration, Manned Spacecraft Center, Houston, Texas, under Contract NAS 9-9044. The study was conducted by personnel of the Bell Aerosystems Company under the direction of Mr. K. Levin, Program Manager and Mr. R. Nelson, Technical Director.

The NASA Technical Supervisor for the study was Mr. William R. Humphrey, Lunar Missions Office, Manned Spacecraft Center. Because this report makes frequent reference to Manned Flying System Report No. 7243-950002, Volumes I and II, produced by Bell Aerosystems under a prior NASA contract, copies are included with the distribution of this report.

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1.0 INTRODUCTION

1.1 BACKGROUND

Prior studies of lunar surface missions established the desirability of providing the lunar explorer with mobility aids to enhance his exploration capability and his safety. It was shown that flying mobility could increase scientific time at remote sites and permit exploration of many features of interest to scientists but inaccessible by surface travel. Scientific return can be improved by making observations and measurements from the vantage point of altitude. The use of flyers enhances overall mission safety because (1) rapid return to the LM can be accomplished in the event of developing contingencies in other systems or natural events; e.g. the suit, PLSS, LM, solar flares, etc. and (2) for a given amount of scientific time at a remote site(s), total EVA time is minimized. Furthermore, the flying vehicle is ideal as a rescue vehicle because of its speed. By addition of auxiliary propellant tanks and a simple guidance system, the exploration flyer might provide a capability for emergency ascent to lunar orbit.

The prior Lunar Flying Vehicle (LFV) and Manned Flying System (MFS) contracts, conducted for NASA by Bell Aerosystems established the feasibilty of providing a small lunar flyer using present state-of-the-art technology and components from the Apollo and other space programs. The vehicle investigated in the MFS study was required to carry two astronauts to a round trip radius of 15 miles, employ dual LM type augmented controls, and meet LM landing criteria. The resulting vehicle carried the two astronauts seated side by side, and incorporated four LM RCS type rockets modified for throttling, a sophisticated electronic guidance and control system, and a modified LM type landing gear.

This vehicle weighed over 400 pounds (earth weight) empty, and carried 600 pounds of LM propellants. The weight of the vehicle and its propellants dictated that this mobility aid could be made avialable only on those lunar missions which employed a dual Saturn V launch. It became apparent that a gap would exist between the initial landings on which no mobility aids would be used, and the later dual Saturn V missions. This gap could be filled only if a mobility aid could be developed whose system weight was compatible with the modified LM payload capability of approximately 1000 pounds. Mission application and vehicle design studies were therefore initiated on a smaller, shorter range, less sophisticated one man flyer. These studies showed that many of the early mission requirements could be met with a small one man, stand-up, vehicle employing a mechanical thrust vector control system, and a helicopter type landing gear.

Prototype flight tests in free earth flight, and also in gimbal and tether type simulated lunar flight, with both shirt sleeved, and Apollo pressure suited pilots, demonstrated the feasibility of the standup configuration with the simple all mechanical flight controls. Lunar vehicle design studies showed that this vehicle would weigh about 200 pounds empty, and should carry about 300 pounds of LM propellant. Furthermore, Grumman studies of LM utilization showed that between 300 and 1500 pounds of propellant would be available in the descent stage tanks after landing, and

that this propellant could be withdrawn for use in the lunar flyer. Thus, only the flyer dry weight plus its support equipment would come from the LM payload allotment. In view of these developments, NASA instituted a new study of lunar flyers to optimize the design and develop system specifications for a simple lightweight one man vehicle. This report presents the results of the study conducted by Bell Aerosystems under NASA Contract NAS 9-9044.

The organization and task sequence of the study is presented in Figure 1.1. Each of these tasks is discussed and results presented in subsequent sections of this report.

1.2 MISSION AND SYSTEM REQUIREMENTS

Mission requirements for the lunar flyer are to provide for:

1. horizontal and vertical mobility
2. reconnaissance and exploration over rough terrain.
3. up to three month lunar storage prior to use
4. up to 30 sorties during one lunar mission
5. total range of 10 to 15 miles
6. one man deployment
7. flyer payloads from 0 to 370 pounds (alternate - 100 pounds maximum payload)
8. rescue of disabled astronaut (requirement deleted during study)

In explanation of item 7, it was required to determine the vehicle dry weight reduction if the maximum payload capability is reduced from 370 pounds to 100 pounds. In regard to item 8, it was required to determine the effect of deleting the requirement to carry a disabled astronaut, while retaining a capability to carry a 370 pound payload.

System and subsystem requirements were provided in the contractual statement of work, were derived from mission requirements, and were developed during the study. These are:

1. vehicle dry weight target - 180 pounds
2. system dry weight target - 450 pounds (2 vehicles and LSE)
3. vehicle propellant load - 300 pounds of LM propellants
4. minimum complexity - only mandatory flight equipment
5. communications by PLSS radio
6. maximum use of available components
7. design to Saturn V/Apollo environmental requirements
8. maintain propellants temperature within 40°F to 100°F

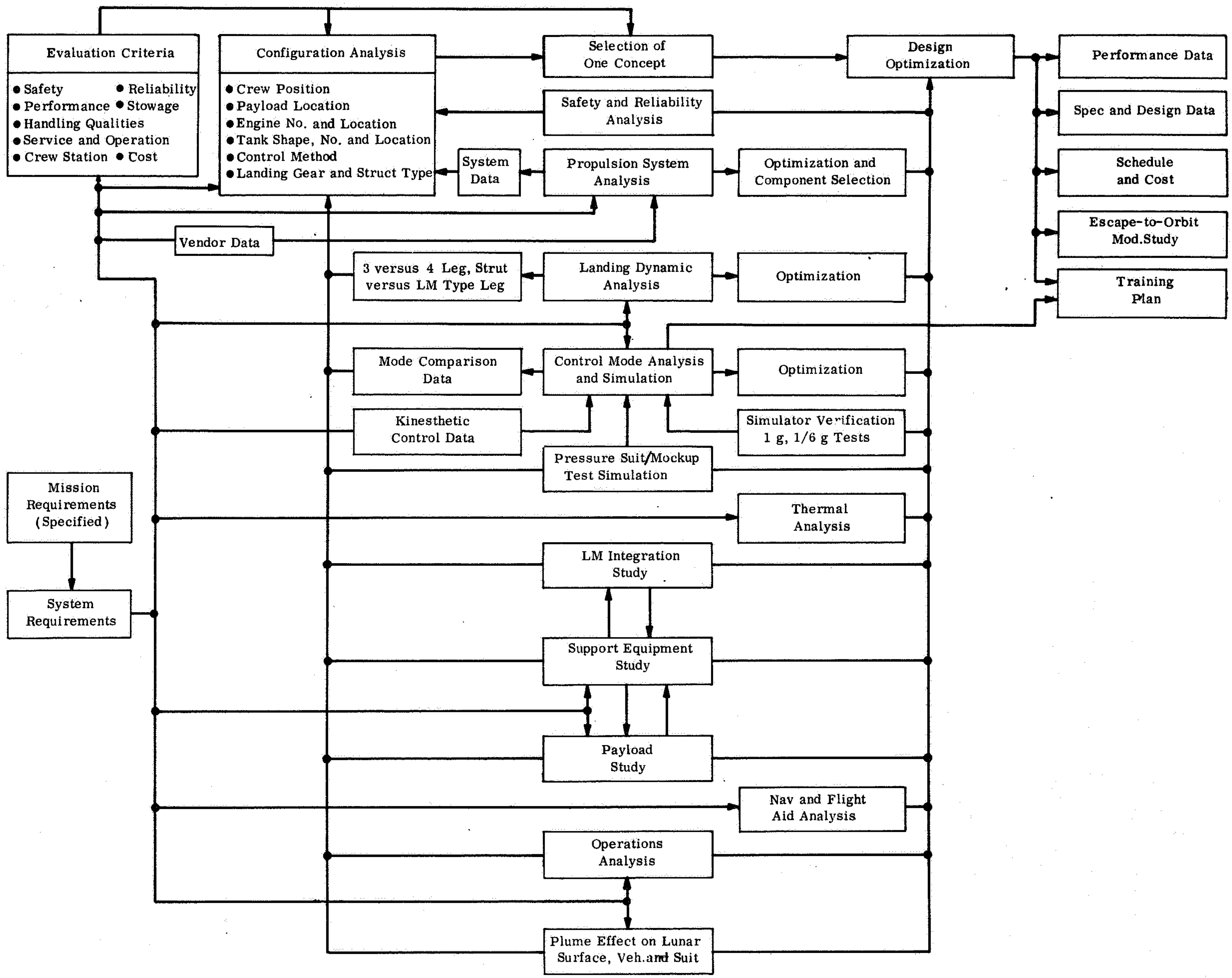


Figure 1.1. OMLFV Study Program

9. design engine, structure, and landing gear for a future 25% vehicle weight growth.

10. total engine thrust range - 50 to 300 pounds

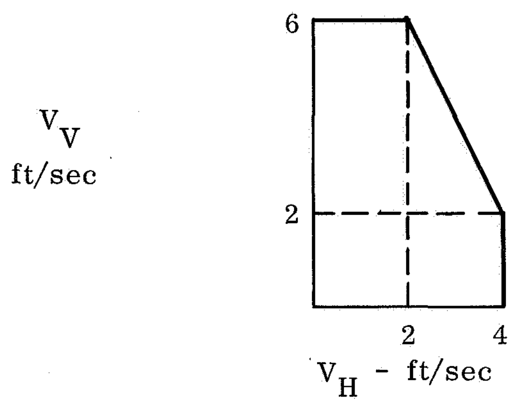
11. regulated pressure-fed propulsion

12. two flyers stowed on LM descent stage

13. refuelable from LM descent stage

14. takeoff and landing within refueling distance from LM

15. landing envelope:



$\theta = \pm 10^\circ$

$\dot{\theta} = \pm 6^\circ/\text{sec}$

1.3 VEHICLE SUMMARY DESCRIPTION

The vehicle recommended to best meet these requirements is shown in the frontispiece. This vehicle was developed through a systematic analysis and comparison of various crew positions, engine arrangement, flight control concepts, tankage arrangements, and landing gear types. In each case, selection was made on the basis of optimizing safety and reliability, followed by consideration of weight, performance, cost, development risk, LM stowage volume, and convenience of handling and service operations. Reliability was maximized by simplicity of design and operation, and by use of conservative design margins.

The vehicle is supported and controlled by two engines, similar to any one of several available, including the LM RCS engines, designed for throttling from 150 to 25 pounds thrust each, and for operation at a low temperature to provide high reliability. Engines are pivoted in pitch and mechanically linked to handlebar type controllers. Yaw control is by differential pitch motion. Roll control is by differential throttling. Throttle and on-off valves are mechanically linked to the astronaut operated controls. Propellant is contained in two tanks similar to the LM RCS tanks but without bladders. The landing gear consists of four cantilevered legs and pads. The legs fold for stowage on LM, but once unfolded, no moving or sliding joints are required. The engines are mounted high enough on the vehicle to minimize the effect of the rocket exhaust plume on the lunar surface and permit safe takeoff and landing close to LM. The vehicle and astronaut are protected from the rocket and plume heat

by a multi-layer thermal shield. Payloads, up to and including a second astronaut, are carried on a specific mission designed pallet attached to the front of the vehicle.

This vehicle has a calculated dry weight of 235 pounds, however, engine thrust and the vehicle structure and landing gear have been designed for a future dry weight growth of 25% to 294 pounds. An analysis of the effect of reducing the payload capability from 370 pounds to 100 pounds, shows that vehicle present dry weight will be reduced by 9.7 pounds. A more detailed vehicle description and mass properties will be found in Section 3.0.

Tradeoffs which resulted in this configuration concept are presented in Section 2.0. Subsystems are discussed in Sections 4.0 through 8.0. Operations and performance are discussed in Sections 9.0 through 13.0. A resources plan is presented in Section 14.0 and the modifications required to provide an escape-to-orbit capability presented in Section 15.0. A description of mathematical models and simulation equipment used in landing dynamics, flight control, and vehicle performance analyses is included in Appendices A, B, and C. The hardware on which the resources plan is based is defined in Appendices D, E, and F.

Appendix G discusses additional tasks recommended as a follow-on to the present study.

This report was prepared just after the completion of the Apollo 11 manned lunar landing mission. A cursory review of the results of this mission indicates no unforeseen conditions and increases confidence in the design and conclusions reached in this study.

2.0 CONCEPTUAL DESIGN AND COMPARISON

The objective of this task is to determine the best overall vehicle configuration for the One Man Lunar Flying Vehicle. In order to meet this objective, it is necessary to consider a wide variety of configuration parameters and combinations to insure that no good combination of parameters has been overlooked. The overall approach used to insure complete coverage of configuration possibilities is illustrated in Figure 2.1. A matrix of configuration variables was established which indicated the wide variety of possible combinations which could be employed to meet the requirements of the One Man Lunar Flying Vehicle. From this basic matrix, a number of integrated configurations were conceived. Concurrently, suitable evaluation criteria were established to compare the relative merits of these integrated concepts. An evaluation procedure was employed to select the best features and most promising integrated configurations and, finally, the configuration for the preliminary design phase of the study. These steps of the configuration selection approach are discussed in the following paragraphs.

2.1 EVALUATION CRITERIA AND DESIGN PHILOSOPHY

The characteristics which have been used to evaluate the suitability of each configuration are shown in Table 2.1. Also shown is the measure or figure of merit which was used to compare the alternate configurations and the relative importance or weighting for each of the characteristics. An overall score for each configuration evaluated was obtained by summing the scores on each characteristic. This total score was used as a guide in the selection of the best overall concept for preliminary design. It can be seen that vehicle reliability recieves a high ranking relative to other characteristics chosen for comparison. The high importance associated with these characteristics greatly influenced the design philosophy during the study. The approach to high reliability and maximum safety was to minimize the number of components in the system and to apply large design margins. In addition, performance was traded for reliability. A further discussion of these reliability and safety aspects is included in Section 5.0, Propulsion Analysis and Design. Comparative failure rate data for nine final vehicle candidates is presented in the next subsection.

2.2 SELECTION PROCESS AND CANDIDATE CONFIGURATIONS

The matrix of configuration variables which was utilized in the study is illustrated in Figure 2.2. Some combinations of these variables are impossible, others were found impractical. Still others had been investigated in previous lunar flying vehicle studies and were determined to offer no particular advantage (see, for example, References 1 and 2).

Based on this matrix, 20 integrated configuration concepts were developed to determine the effect of these variables upon the overall configuration. The crew position, number of engines, and control moment mode were found to be the most important configuration-influencing parameters and formed the basis for the

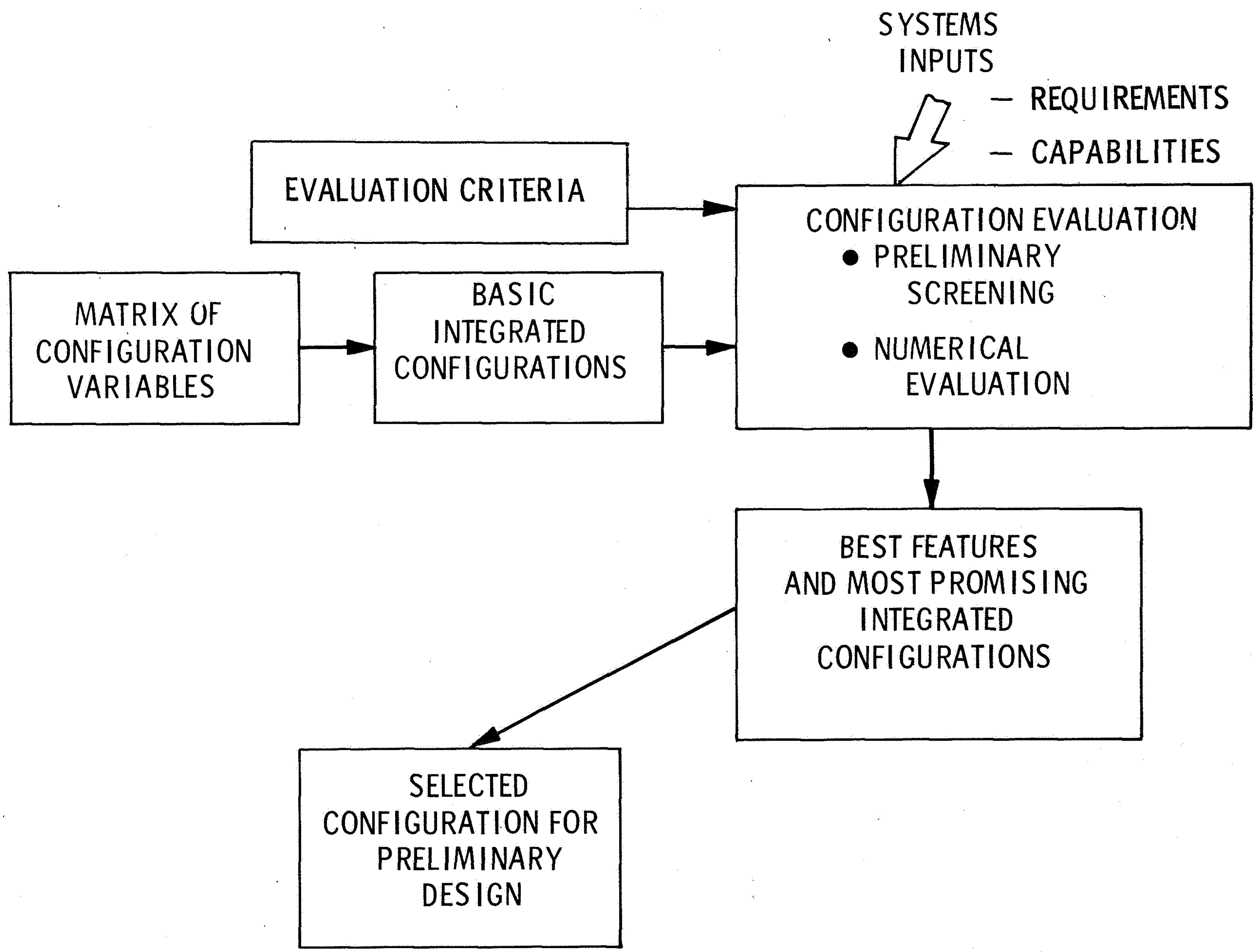


Figure 2.1. One Man Lunar Flying Vehicle Configuration Selection Approach

TABLE 2.1

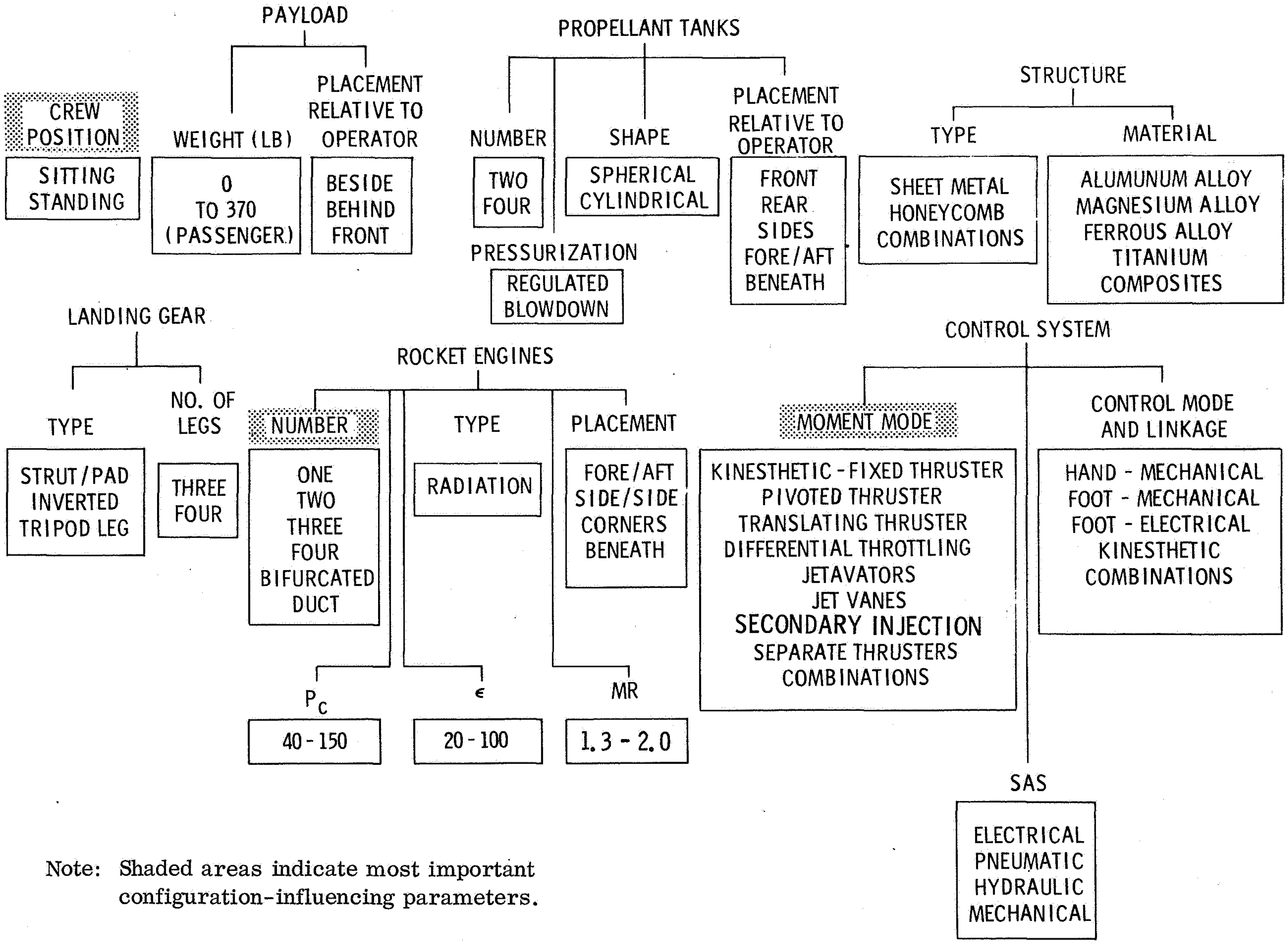
CONFIGURATION EVALUATION CRITERIA

Characteristic	Relative Inportance (R)	Measure (M)	Score Calculation
Reliability: Safety	25	Number of Unsafe Failures per Million	$\frac{M_B}{M_C} \times R$
Reliability: Abortive	25	Number of Failures Causing Mission Abort per Million	$\frac{M_B}{M_C} \times R$
Performance	12	Configuration Dry Weight	$\frac{M_B - 200}{M_C - 200} \times R$
Stowage	12	Square Feet of Thermal Shielding Required to Protect Flyer when Stowed on LM	$\frac{M_B}{M_C} \times R$
Handling Qualities	12	Relative Rating from 0 to 1.0*	$M_C \times R$
Service and Operations	6	Relative Rating from 0 to 1.0* Considering: 1 Ease of Deployment 2 Ease of Fueling 3 Ease of Loading and Unloading Payload 4 Ease of Trim Adjustment	$M_C \times R$
Crew Station	6	Relative Rating from 0 to 1.0* Considering: 1 Ease of Ingress/Egress 2 Ease of Employing Restraint 3 Operator Comfort (Standing/ Seated) 4 Visibility	$M_C \times R$

M_B = Measure of Best Configuration for this characteristic.

M_C = Measure of Configuration being Evaluated

* 1.0 is Best



Note: Shaded areas indicate most important configuration-influencing parameters.

Figure 2.2. One Man Lunar Flying Vehicle Matrix of Configuration Variables

configuration development. Other parameters (i.e., number of tanks, tank placement, engine placement, etc.) were selected to make the best overall configuration. Other special case configurations were included to determine the effect of a particular variable. A summary of the 20 configurations established in this manner is illustrated in Table 2.2. These configurations were analyzed and subjected to a preliminary screening and evaluation. Nine configurations were selected for more detailed analysis and numerical evaluation. These are discussed in the following paragraphs.

2.2.1 Candidate Configurations

Seated Configurations (5.3 and 6.2)

In seated configurations, four closely coupled tanks were preferred because of: (1) minimum volume and (2) structural efficiency due to compactness and a convenient seating arrangement for the astronaut above the tanks. Additionally, vehicle trimming (due to variable payload requirement) by thruster adjustment was found to be mechanically and operationally simpler than payload and astronaut/controller repositioning. These findings are reflected in the seated configurations shown in Figures 2.3 and 2.4. Configuration 5.3 employs two engines mounted outboard of the astronaut and tanks. Hard point provisions are made for carrying payload on a pallet behind the astronaut. This payload pallet concept has been employed on all of the flying vehicle designs and is described more fully in Section 8, Payload Studies. Vehicle trim is achieved by a preflight adjustment of the thruster fore or aft depending on the amount of payload carried. Pitch control is achieved by pivoting of the main engines about a high pivot point, roll control is achieved by differential throttling, and yaw control is achieved by differential pivoting. Configuration 6.2 is a seated configuration employing a single central engine. In this configuration pitch control is achieved by translation of the main engine fore and aft, roll control is achieved by pivoting of the main engine and yaw control is achieved by separate small thrusters. An analysis which is reported in Section 9.0 Operations Study, showed that the minimum allowable height from engine exit plane to the surface for a single 300 pound thrust engine is 20 inches due to plume/surface interaction effects. This results in a high configuration center of gravity and a large landing gear. In order to reduce vehicle height, the engine length was reduced to the minimum practical by employing a high chamber pressure (P_c = 125 psia) and a low area ratio (ϵ = 20) nozzle. Further reduction in vehicle height (to reduce landing gear spread and leg length) is possible by burying or partially burying the engine-moving the tanks down around the engine. Such configurations require more stowage volume because the configuration is wider. Furthermore a buried engine will operate at a higher temperature, reducing engine reliable life.

Disadvantages of the seated configurations are the relative discomfort of the seated position in a pressurized suit and a rather complicated ingress/egress procedure to arrive at the seated position. Mockup/pressure suited subject tests conducted during the Manned Flying System Study (References 2 and 3) showed that the best procedure for achieving a seated position involved stepping onto a platform, making a turnaround and lowering oneself (using a suitable hand hold) into the seated position.

TABLE 2.2

INITIAL CONFIGURATION DEVELOPMENT

Configuration Designation	Crew Position	Main Engines	Pitch Control Mode	Remarks
-	Seated	1	High Pivot (HI)	Complex Structure
6.1			Low Pivot (LO)	
6.2			Translate (TR)	
-			Diff Throttle (DT)	Impossible
5.1, 5.2, 5.3		2	HI	
7.1			LO	2 Eng Translation Investigated in 4.4
-			TR	Ingress/Egress Difficult
-			DT	
-		4	HI	Rejected due to Reliability Considerations
-			LO	
-			TR	
-			DT	
-	Standing	1	HI	Complex Storage
3.1			LO	
3.2			TR	
-			DT	Impossible
8.1, 8.2		2	HI	
4.2, 4.3			LO	
4.4			TR	
4.5			DT	
-		4	HI	Rejected due to Reliability Considerations
-			LO	
-			TR	
12.1			DT	Investigated to Determine Advantages
13.1	Standing	2	HI	To Investigate Foot Control Mode
2.1	Standing	2	Kinesthetic	To Investigate Shifting Astronaut and Payload
10.1	Standing	2	HI	To Investigate SAS and LM Hand Controller
9.2	Standing	4	HI	To Investigate Redundancy
3.2	Standing	2	TR	To Investigate Redundancy

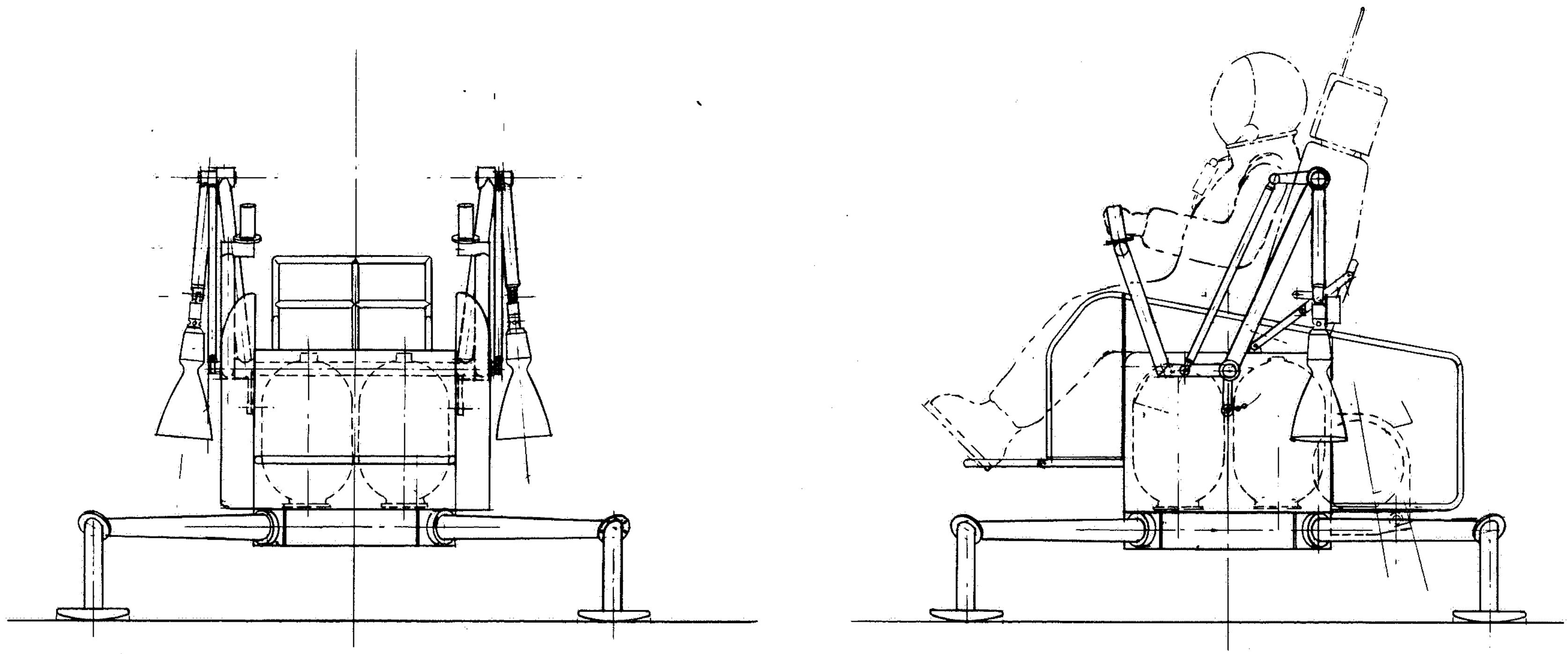


Figure 2.3. Lunar Vehicle Configuration 5.3

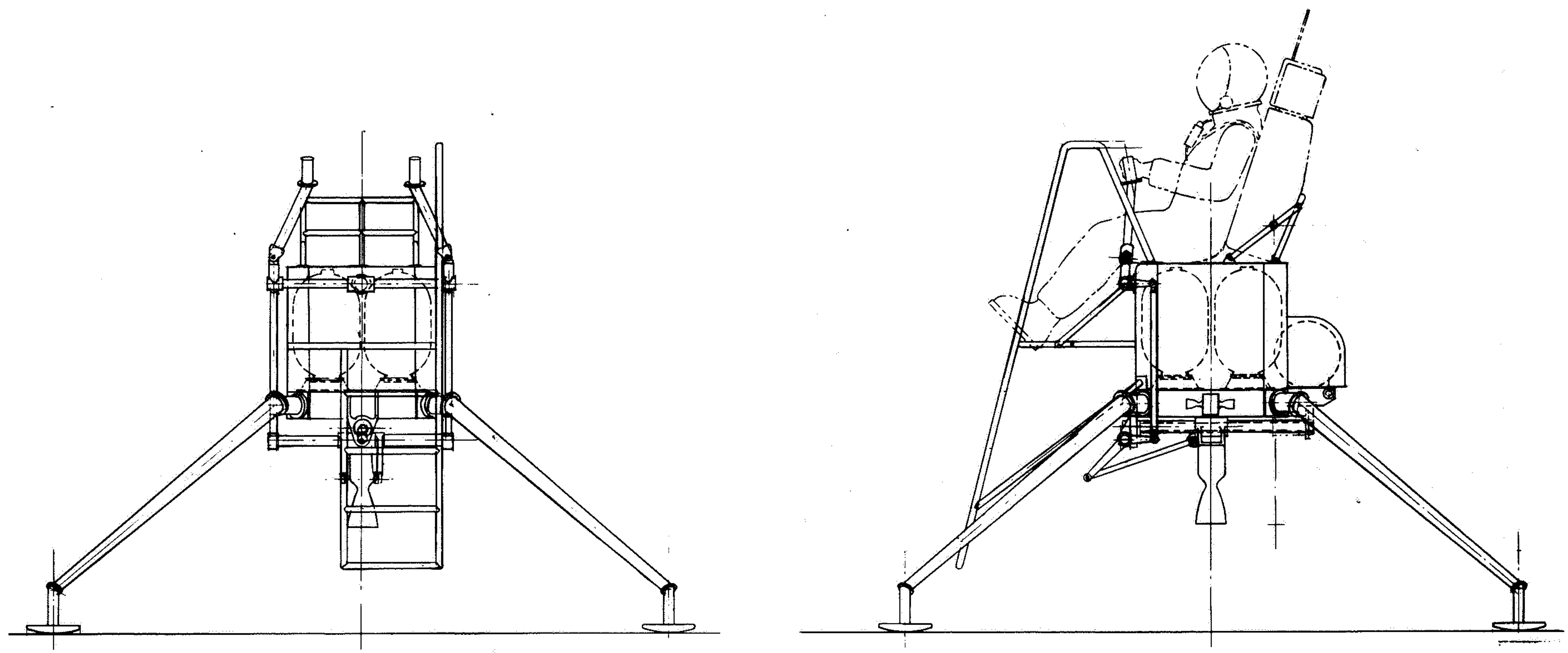


Figure 2.4. Lunar Vehicle Configuration 6.2

In order to circumvent these disadvantages, a number of configurations employing a stand-up operating position were investigated.

Stand-up - Two Engine Configurations (4.3, 4.4, 8.2, and 8.1)

A factor favoring the use of a stand-up configuration is the attenuation of landing g loads experienced by the upper torso and PLSS with modest amounts of knee bending. Figure 2.5 shows the upper Torso/PLSS load factor as a function of the vehicle load factor for a vertical velocity landing impact of 6 fps. At a vehicle load factor of 3 g's, three inches of leg deflection results in a load factor of 1.4 g's on the PLSS and upper torso. Landing gear studies show that the most severe landing condition produces a vehicle load factor ranging from 2 to 4 g's depending on the vehicle loading condition. Drop tests were conducted at MSC (see Ref. 4) with a suited subject, up to a peak vehicle acceleration of 3 g's. These tests indicated that the stand-up position is acceptable if the astronaut is suitably restrained to prevent toppling out of the vehicle on a severe landing. Additional tests conducted at MSC on another program provides evidence that an astronaut can support landing loads while on his feet (Ref. 5). A pressure-suited subject, wearing a backpack PLSS, supported by a gimbal and lunar gravity tether, jumped from an eight foot step ladder to the ground with no difficulty. His velocity at contact with the ground was 9.2 ft/sec., which is greater than the maximum touchdown velocity for the lunar flyer. Accordingly, a horizontal restraint system has been incorporated in the stand-up configurations, but support of astronaut or PLSS for vertical loads is not required.

Stand-up configurations employing two propellant tanks and two outboard engines are illustrated in Figures 2.6 through 2.9. Configuration 4.3 illustrates one of these configurations with a low pivot location for pitch control. Roll control is achieved by differential throttling and yaw control is achieved by differential pivoting of the main engines. In this configuration, because the pivot is located close to the vehicle center of gravity, large trim angles on the order of 5 to 10 degrees are experienced due to propellant burn-off at certain payload loading conditions. In addition, simulation studies of low pivot types of control systems indicated that the handling qualities of this type of vehicle were inferior at low sensitivities. These effects can be minimized by making the pivot location closer to the ground, that is further away from the center of gravity at some increase in structural complexity and weight.

These effects are overcome in a configuration such as that shown in Figure 2.7. In this configuration pitch control is achieved by translation of the engines fore and aft, and roll control is achieved by differential throttling, and yaw control is achieved by differential pivoting of the main engines. Zero trim angle is maintained because the thruster is positioned in the neutral position directly beneath the instantaneous center of gravity. One of the disadvantages of this configuration is the relatively large weight penalty paid for mounting the engines in the fashion shown.

Configuration 8.2, illustrated in Figure 2.8 shows a configuration with the pitch pivot located high. This configuration also employs roll control by differential throttling and yaw control by differential pivoting of the main engines. Since the pivot is located further away from the center of gravity, the trim angles range from 0 to approximately 6 degrees depending upon the magnitude of the payload on board.

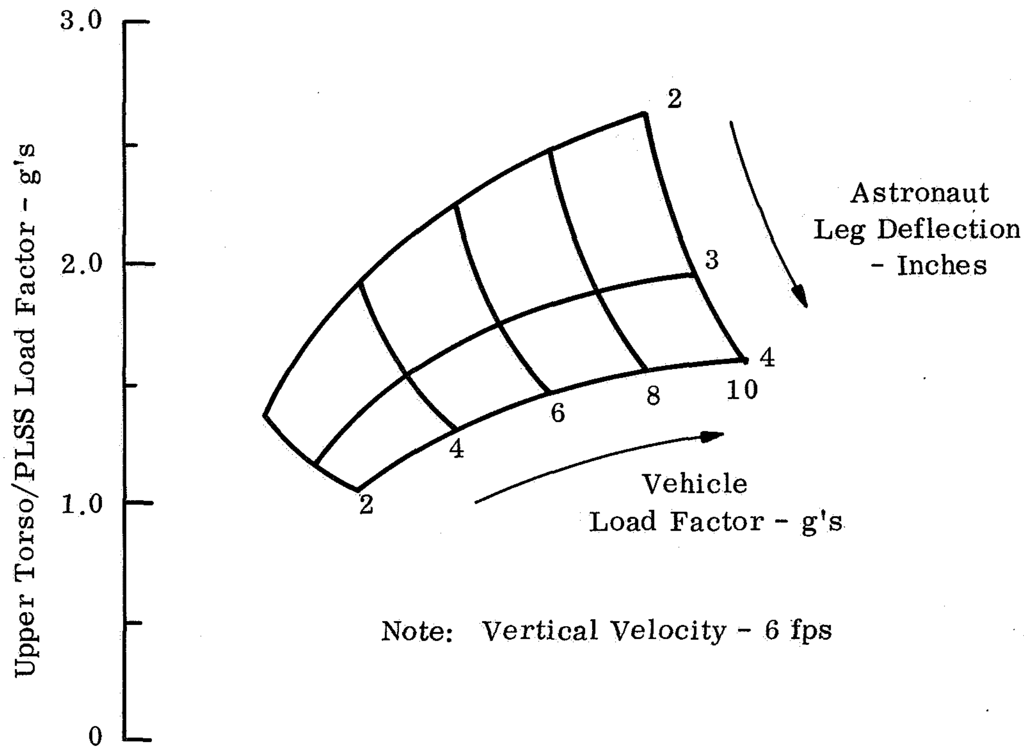


Figure 2.5. Attenuation of g Loads due to Leg Deflection

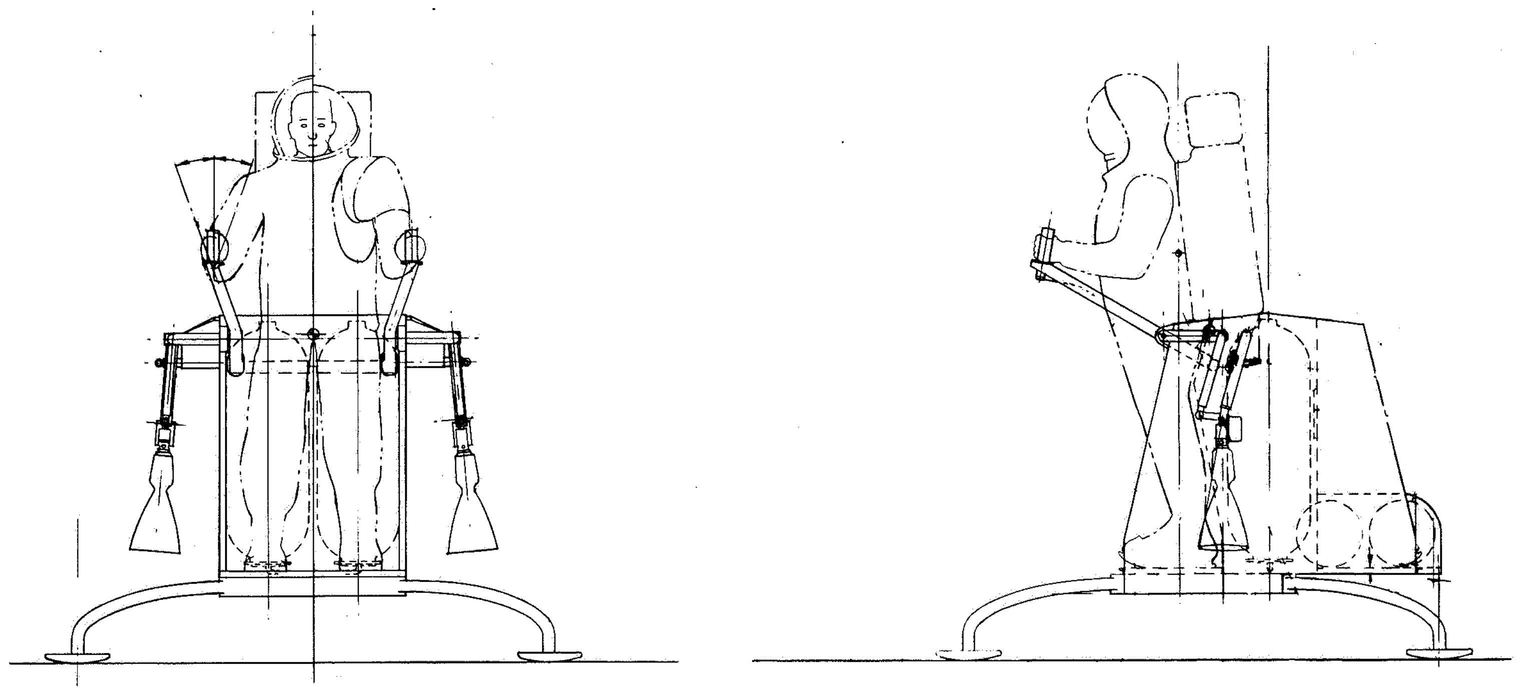


Figure 2.6. Lunar Vehicle Configuration 4.3

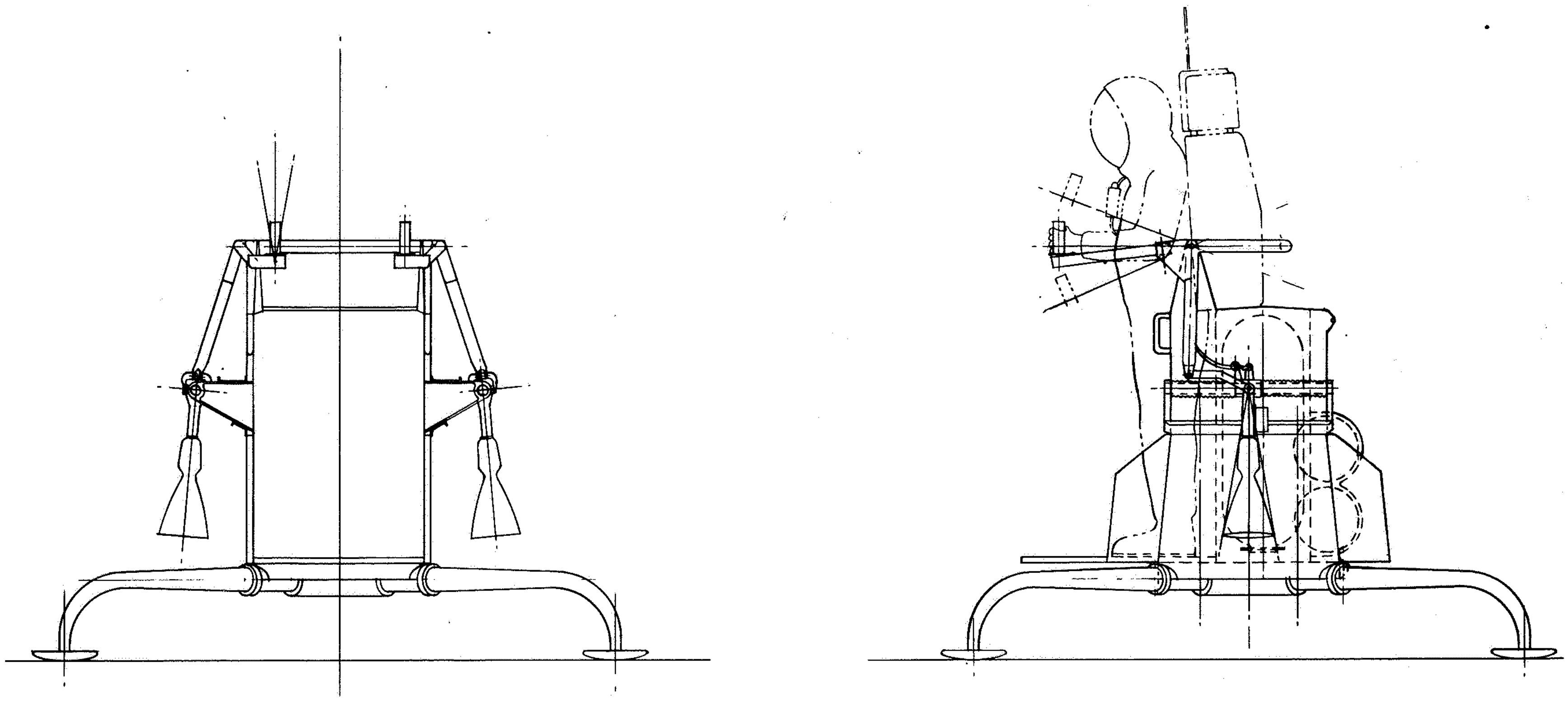


Figure 2.7. Lunar Vehicle Configuration 4.4

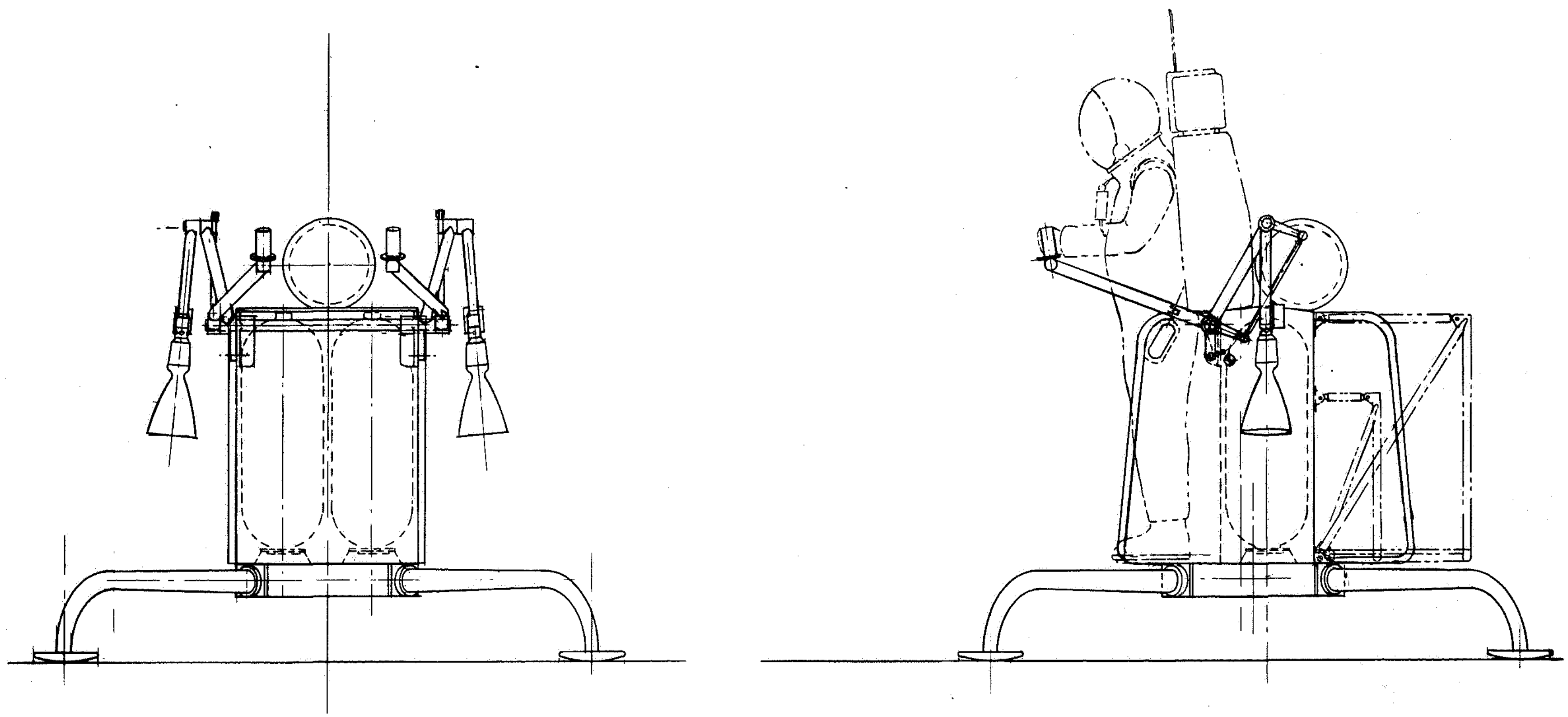


Figure 2.8. Lunar Vehicle Configuration 8.2

The simulation studies reported in Section 6.0, Control System Analysis and Simulation, showed that this configuration also provided good handling qualities.

The three configurations just described all employ a crew station which is in front of the payload and propellant tanks. Configuration 8.1, which is illustrated in Figure 2.9, shows a crew station to the rear of the payload and tanks. Two tradeoff factors are involved in making a selection between a front and rear located crew station; these are visibility and ease of ingress. Figure 2.10 illustrates the visbility comparison of the two crew station locations. Payload size and arrangement affects the downward visibility directly in front of the operator for a configuration employing a crew station to the rear. During the approach to the destination, the destination will be at a line of sight angle ranging between 36 and 51 degrees depending upon a type of flight profile used and the distance to the destination. Acceptable visibility in this region can be provided by proper arrangement of the payload on the vehicle. This acceptability is illustrated by comparison with helicopter visibility requirements as extracted from helicopter visibility specifications for tandem and side-by-side seating arrangements. When comparing ingress characteristics of the two configurations, a clear preference for the crew station in the rear exists. For this crew station, the operator simply steps up to the operating position for the vehicle. With the crew station in the front the operator is required to step up, make a turnaround and back into the operating position. He must then reposition the controllers which had to be swung out of the way in order to perform the ingress. This hinging for swing-out requires a more complex, less reliable, and heavier mechanical control system. For these reasons, a crew station to the rear was selected.

Stand-up - Single Main Engine Configuration (3.2)

All of the previous stand-up configurations employed two outboard engines. It was of interest to determine the characteristics of a vehicle which employs a single centrally located engine for main propulsion. Such a configuration is illustrated in Figure 2.11. In this configuration, pitch control is achieved by translation of the engines fore and aft while roll control is achieved by pivoting the main engine. Yaw control is achieved by separate attitude control thrusters which utilize propellants from the main tanks. This configuration has the same deficiencies as were noted for the seated single engine configuration.

Redundant Engine Configurations (9.2 and 3.3)

The effect of engine redundancy was investigated and is illustrated in Figures 2.12 and 2.13. Configuration 9.2 is a redundant version of configuration 8.2. Here, two outboard engines are placed on each side so that, in the event of a single engine failure, its mate on the other side is shutdown to preserve vehicle balance. Sufficient thrust is provided by the remaining two engines to make a safe landing. In all other respects the vehicle is similar to configuration 8.2. The redundant version of configuration 3.2 is illustrated in Figure 2.13. Here again, sufficient thrust is provided with one engine to make a safe landing, if an engine failure occurs.

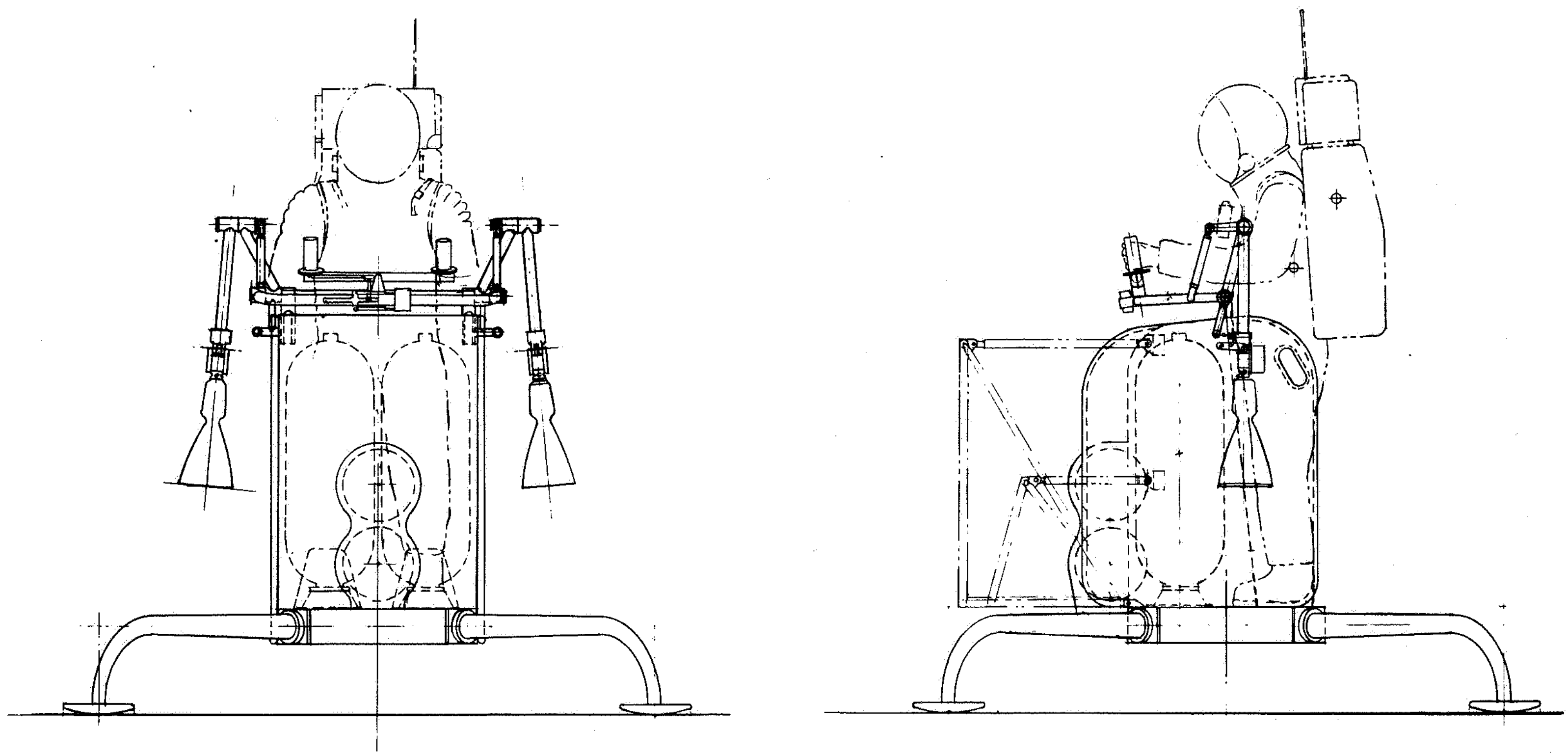


Figure 2.9. Lunar Vehicle Configuration 8.1

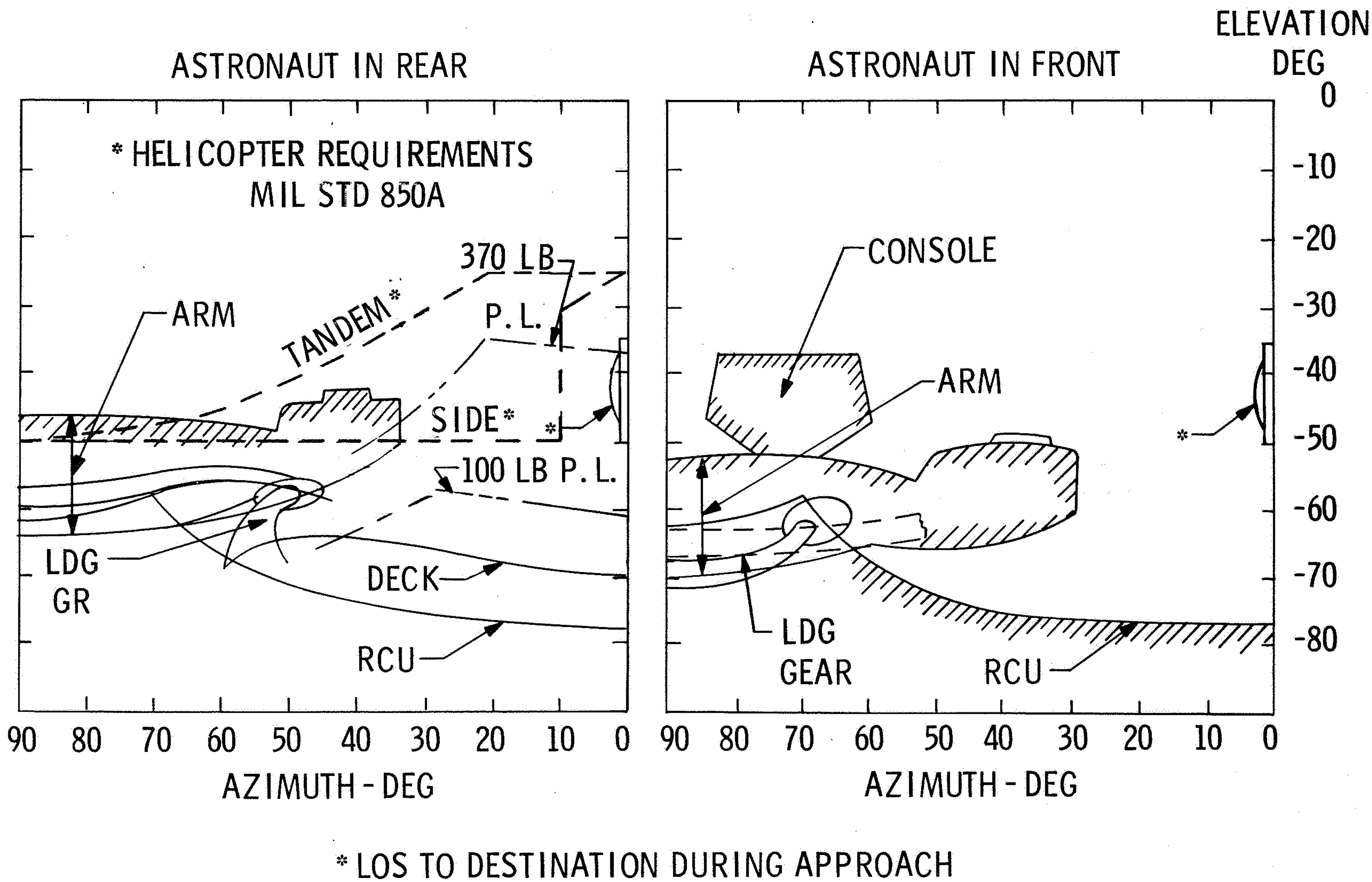


Figure 2.10. Visibility Comparison

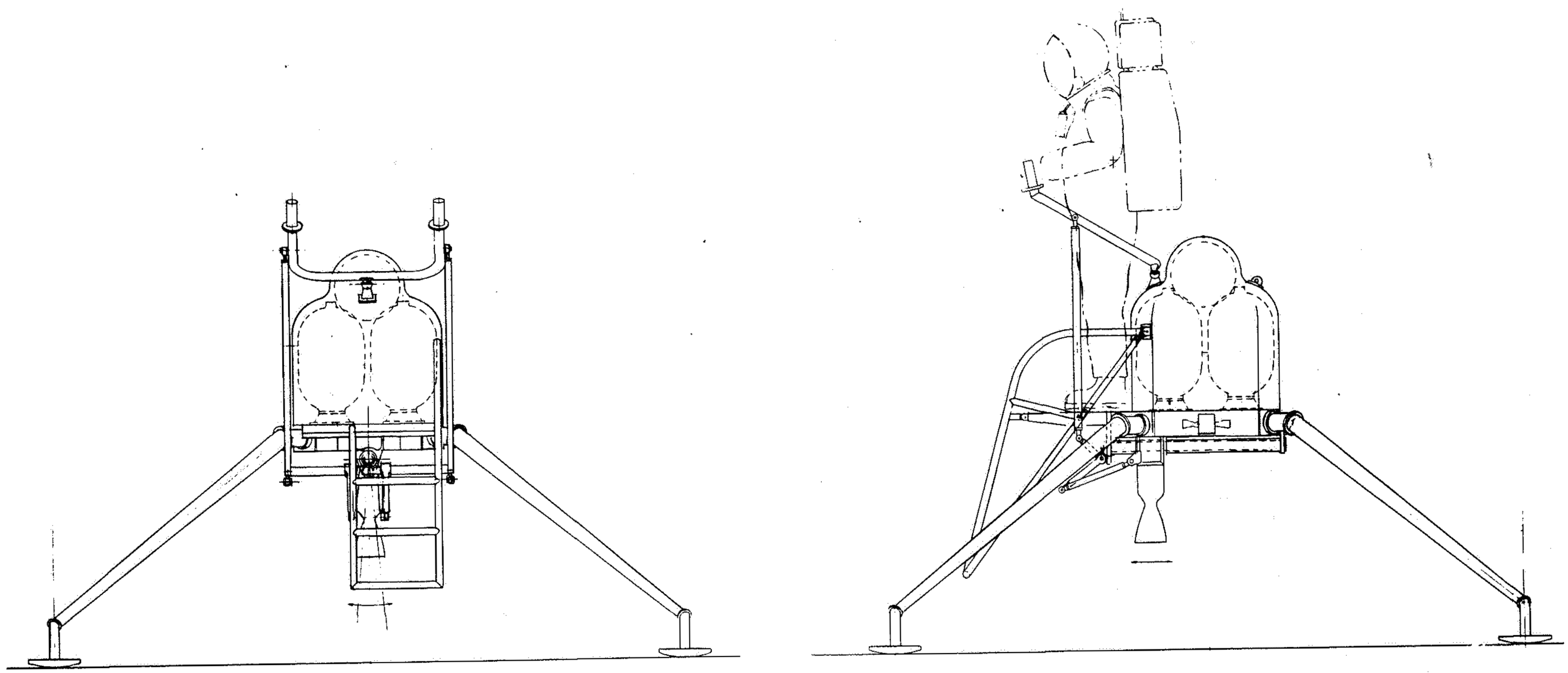


Figure 2.11. Lunar Vehicle Configuration 3.2

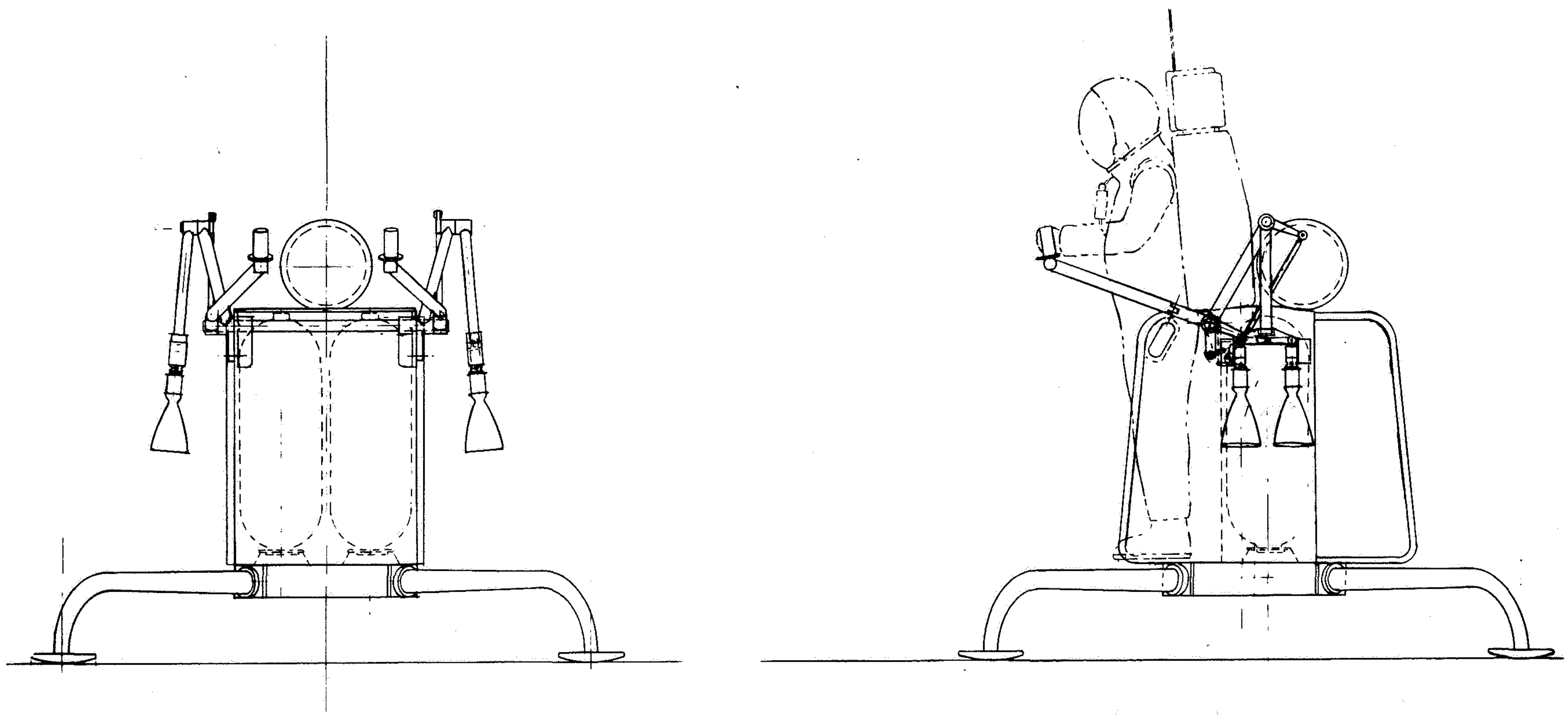


Figure 2.12. Lunar Vehicle Configuration 9.2

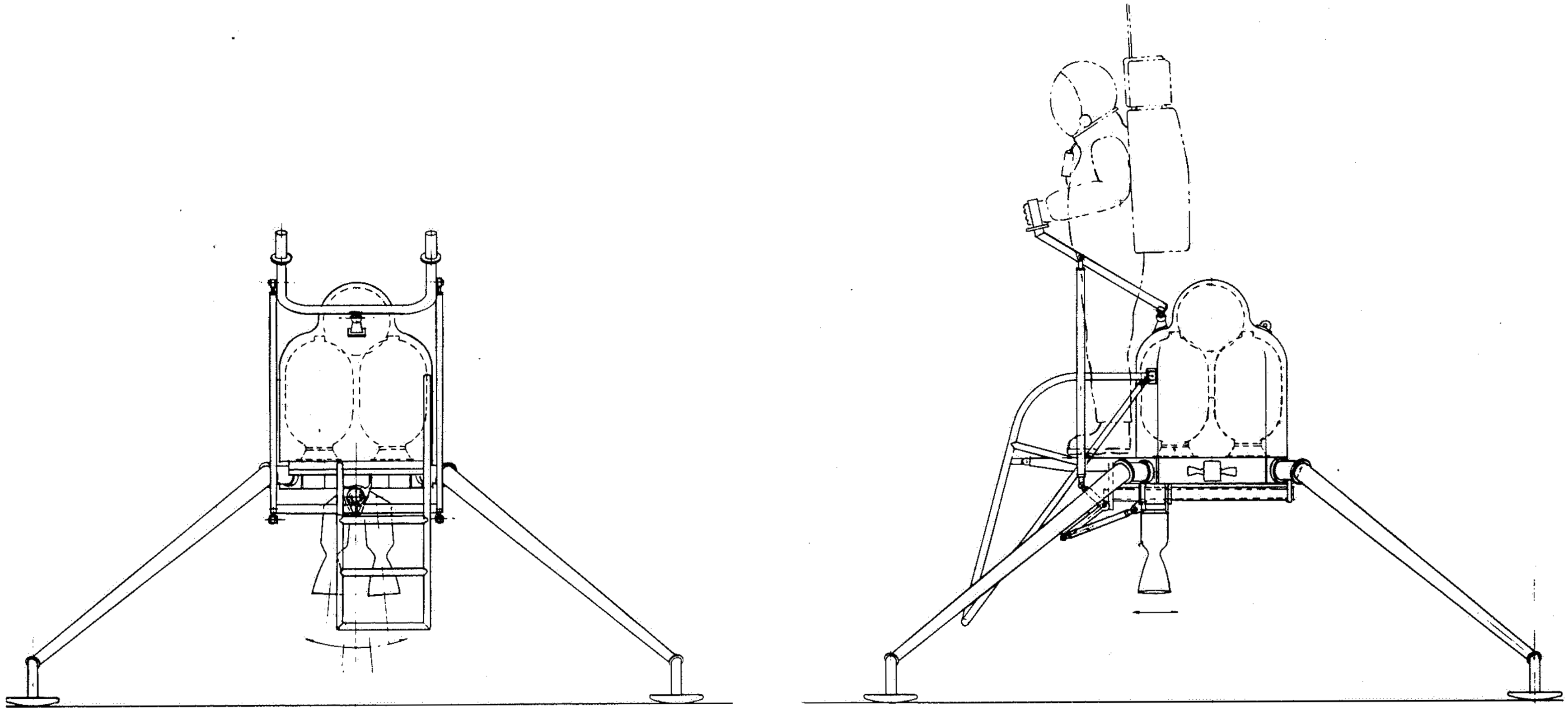


Figure 2.13. Lunar Vehicle Configuration 3.3

2.2.2. Evaluation

The discussion of the nine candidate configurations has indicated in a qualitative way the relative advantages and disadvantages of these configurations. A quantitative comparison is provided in Table 2.3. This shows how the vehicles compare based on the figure of merit established for each of the characteristics. The reliability characteristic was separated into two factors, one dealing with astronaut safety and the other dealing with the system failures which would cause sortie abort but which would not compromise astronaut safety. These reliability estimates were based upon the component failure rates for propulsion system components only (shown in Table 3.4). Other system components were minor contributors to the overall vehicle reliability and were the same for all of the configurations investigated.

The figure of merit data was converted into point scores (Table 2.1) by characteristic and by configuration to get an indication of what the most desirable configurations were and their characteristics. The results of this analysis are presented in Table 2.4. On the basis of the comparisons made in these two tables and the eariler discussion of factors, configuration 8.1 was selected for the preliminary design phase of this study. The redundant version, configuration 9.2, scored slightly higher on the basis of fewer unsafe failures. However, this theoretical advantage is questionable when it is considered that two additional engine systems and some kind of switching technique or hardware is required. This additional vehicle complexity increases the probability of an abortive type failure as indicated in Table 2.2. A more thorough discussion of the reasons for selection of a two-engine system, with large design margins, over a redundant four-engine system is presented in Section 5.0.

None of the configurations presented showed a stability augmentation type of control system since this conceptual design effort was primarily oriented toward defining the best general arrangement of tanks, crew, payload and engines. Stability augmentation can be added to all configurations but will not affect the relative rating of the vehicles. The benefits and disadvantages of stability augmentation, as applied to the selected configuration, are discussed in Section 6.0.

TABLE 2.3

CONFIGURATION COMPARISON DATA SUMMARY

CONCEPT DESIGNATION AND DESCRIPTION	RELIABILITY		PERFORMANCE (DRY WEIGHT)	STOWAGE (SQ. FT. OF THERMAL PANEL)	HANDLING QUALITIES (RELATIVE RATING)	SERVICE AND OPERATIONS (RELATIVE RATING)	CREW STATION (RELATIVE RATING)
	SAFETY (UNSAFE FAILURES PER MILLION)	ABORTIVE (ABORT FAILURES PER MILLION)					
RELATIVE IMPORTANCE →	25	25	12	12	12	6	6
5.3 SEATED 2 ENG	13	238	219.0	26.3	1.0	0.5	0.3
6.2 SEATED 1 ENG	9	193	235.6	16.1	0.75	0.5	0.3
4.3 LOW PIVOT 2 ENG	13	230	209.3	17.9	0.75	1.0	0.9
4.4 TRANSLATE 2 ENG	13	230	231.4	26.3	0.9	1.0	0.9
8.2 HIGH PIVOT 2 ENG	13	230	217.8	9.7	1.0	1.0	0.9
8.1 HIGH PIVOT 2 ENG	13	230	205.3	13.5	1.0	1.0	1.0
9.2 HIGH PIVOT REDUNDANT 4 ENG	4	351	228.1	9.7	1.0	1.0	0.9
3.2 TRANSLATE 1 ENG	9	193	235.6	12.9	0.5	0.75	0.75
3.3 TRANSLATE REDUNDANT 2 ENG	4	257	241.6	17.9	0.75	0.5	0.75

TABLE 2.4

SUMMARY SCORING OF CONFIGURATIONS

Concept Designation and Description			Reliability		Performance	Stowage	Handling Qualities	Service and Operations	Crew Station	Total
			Safety	Abort						
5.3	Seated	2 Eng	7.7	20.3	3.3	4.4	12	3	1.8	52.5
6.2	Seated	1 Eng	11.1	25	1.8	7.2	9	3	1.8	58.9
4.3	Low Pivot	2 Eng	7.7	21	6.8	6.5	9	6	5.4	62.4
4.4	Translate	2 Eng	7.7	21	2.0	4.4	10.8	6	5.4	57.3
5.2	High Pivot	2 Eng	7.7	21	3.6	12.0	12	6	5.4	67.7
8.1	High Pivot	2 Eng	7.7	21	12.0	8.6	12	6	6	73.3
9.2	High Pivot	Redundant 4 Eng	25	13.8	2.3	12.0	12	6	5.4	76.5
3.2	Translate	1 Eng	11.1	25	1.8	9.0	9	3	4.5	63.4
3.3	Translate	Redundant 2 Eng	25	18.8	1.5	6.5	9	3	4.5	68.3

3.0 PRELIMINARY DESIGN

3.1 INTRODUCTION

Section 2.0 of this report presented conceptual designs developed under the ground rules of this study and explained the process used to select one of these designs for more intensive study.

This section discusses and presents data on:

(1) A preliminary design study of the selected configuration and its subsystems including mass properties data.

(2) The contribution of the OMLFV system to the Apollo launch weight.

(3) The ground rules established for, and the estimate of, the reliability of the vehicle.

(4) Some modifications to the baseline vehicle to provide additional capability.

3.2 BASELINE CONFIGURATION - GENERAL ARRANGEMENT AND OPERATION

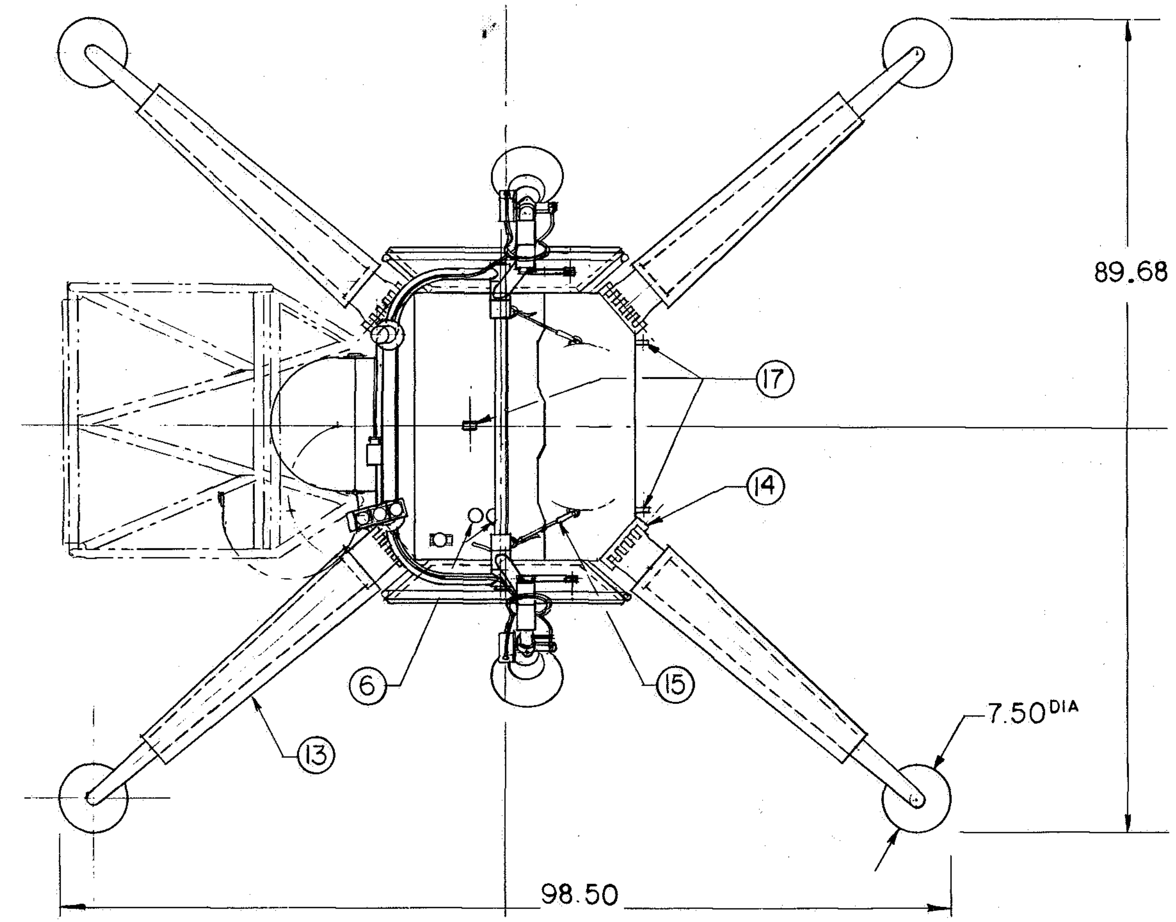
A three-view drawing of the selected configuration, with principal dimensions and callouts is shown in Figure 3.1. The vehicle employs two engines with a high pivot location. The crew station is to the rear and provisions are made for a mission tailored payload pallet at the front. The vehicle is manually controlled and employs a simple strut/pad type landing gear.

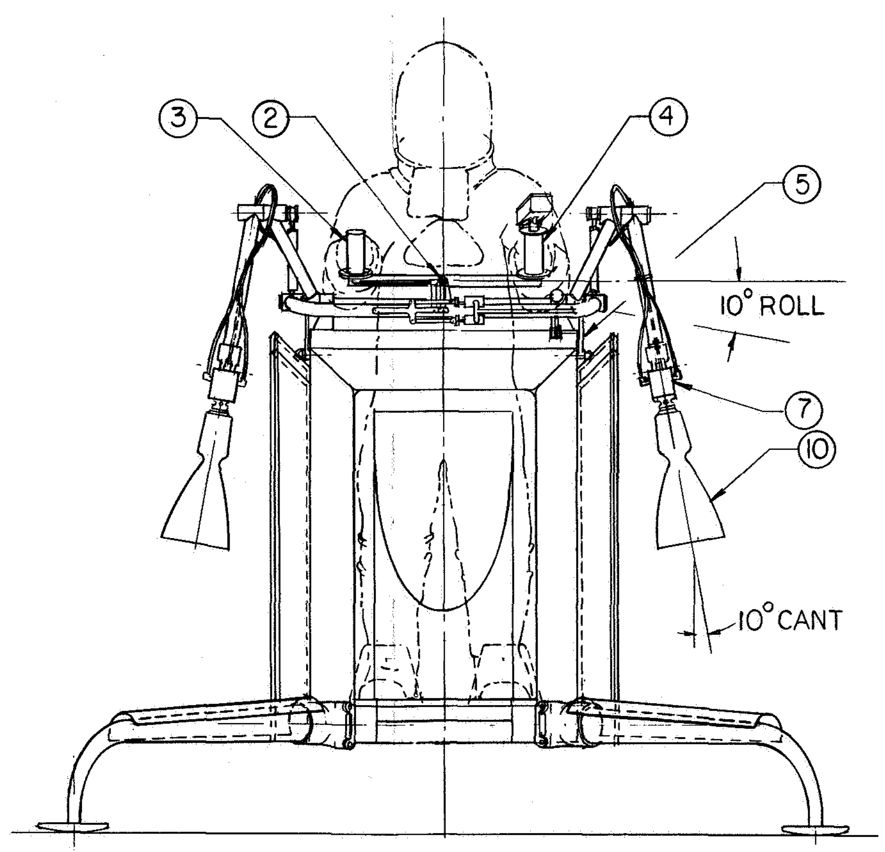
After the vehicle has been deployed from the LM, the propellant fueling lines are brought to the vehicle, the service door located at the front of the vehicle is unlatched and swung open and the vehicle is fueled. This operation is monitored by the fuel quantity gages and pressure gages on the instrument panel; the instrument panel having been rotated and swung down for this purpose. After the fueling lines have been disconnected the helium tank isolation valve is opened and verification of the propellant tank pressures is read from the gage on the instrument panel. The service door is then secured, the instrument panel returned to its flight position, and the fueling hoses returned to LM. The payload pallet and the payload are then installed and the engine trim positioner is unlocked, moved to the proper position for the payload onboard and relocked. The astronaut then steps onto the platform at the rear of the vehicle and secures the two restraint straps which clip into the "D" rings at his waist. These are the same "D" rings that secured him to LM.

In front of him, the right hand controller is the throttle, the left hand controller is for yaw. Vertical motion of the control assembly provides pitch control, rotation of the hand controller assembly about its x axis provides roll control.

LEGEND

1 PITCH CONTROLLER PIVOT AXIS
2 ROLL CONTROLLER PIVOT AXIS
3 THRUST CONTROLLER
4 YAW CONTROLLER
5 THRUST POSITIONER ASSY
6 PROPELLANT ISOLATION VALVES (2)
7 PROPELLANT SHUTOFF VALVE (2)
8 PROPELLANT SHUTOFF VALVES CONTROLLER
9 PROPELLANT THROTTLE VALVE
10 THRUST CHAMBER (2) 150LB MAX. THRUST, THROTTLEABLE
11 INSTRUMENT PANEL
12 MULTI-LAYER HIGH TEMPERATURE INSULATION
13 LANDING GEAR PLUME SHIELD
14 LANDING GEAR FOLDING HINGE
15 ASTRONAUT RESTRAINT HARNESS (2)
16 PAYLOAD PALLET
17 LM/OMLFV INTERFACE FITTINGS (3)





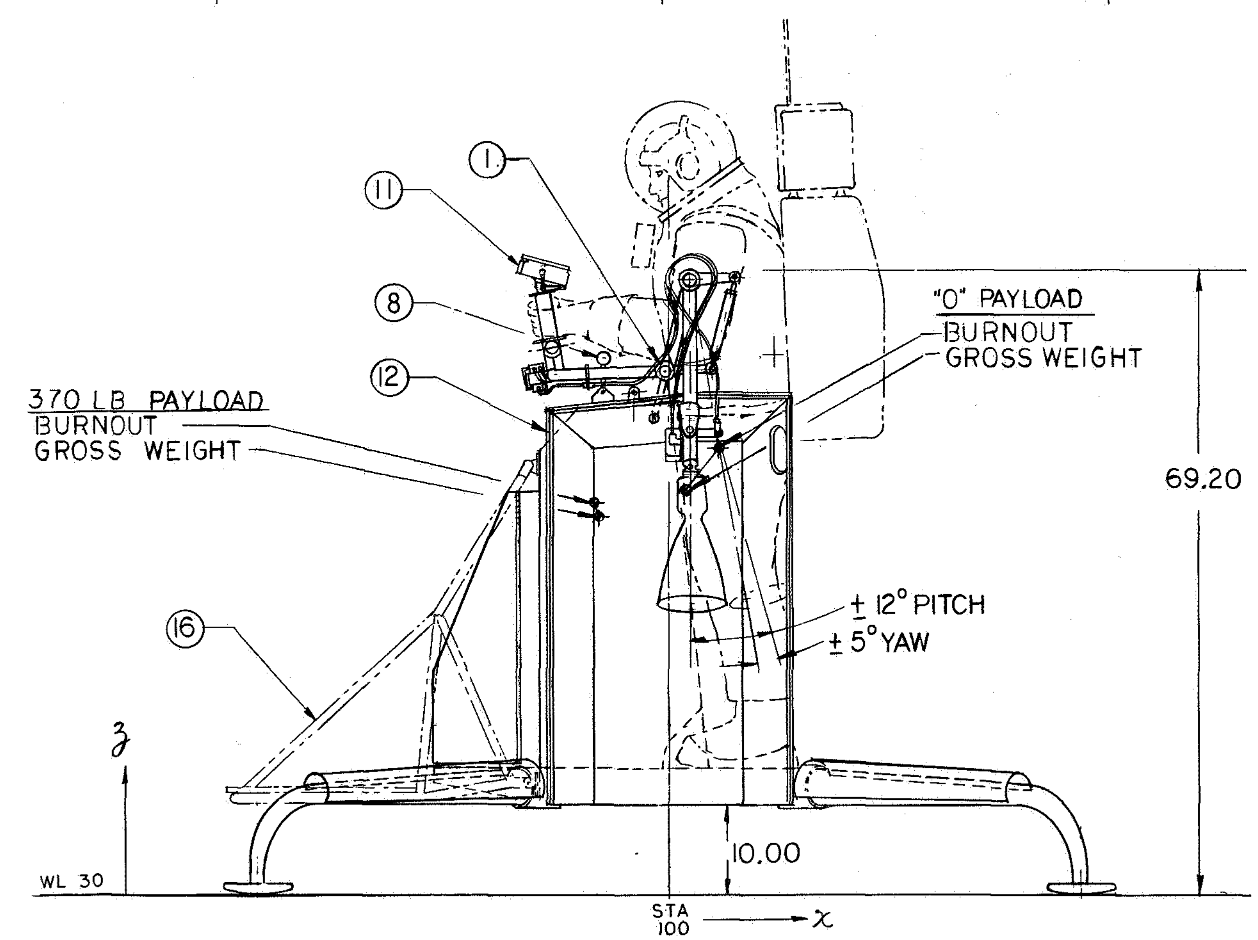


Figure 3.1. General Arrangement

The start sequence is initiated by moving the start control handle from the NEUTRAL to the OFF position. The propellant isolation valves are then opened, sending propellants from the tanks through the throttle valves to the start valves at the thrusters. All controls are then "rocked" to verify smoothness of operation, the throttle controller is rotated to the minimum thrust position, and the start valve control handle is moved forward to the ON position to initiate thrust. The throttle is then advanced for takeoff.

3.3 BASELINE CONFIGURATION - INBOARD PROFILE

A drawing of the inboard profile is shown in Figure 3.2. The vehicle subsystems are identified as: landing gear, body structure, propulsion, control system, thermal protection, instrumentation and electrical power, and astronaut restraint.

3.3.1 Landing Gear

The geometry of the landing gear shown is similar to that of a helicopter gear. Having no moving parts, it is both simple and reliable. Landing energy is absorbed by a combination of lunar soil compression/penetration and soil/pad friction and displacement. Deflection of the gear legs reduces landing forces on the vehicle and astronaut.

The legs of the gear are tapered tubes made from titanium alloy. The landing pads have been optimized at 7.5 inches diameter. The pads are made of titanium honeycomb sandwich and are brazed to the landing gear legs. Titanium fittings at the root section of the legs mate with fittings which are brazed to the vehicle honeycomb platform. This attachment consists of a hinge pin at the top of the fitting about which the leg can be pivoted for stowage and two latching bolts that automatically engage when the leg is brought down into position as part of the deployment sequence. Criteria and design of the landing gear described in detail in Section 4.0.

3.3.2 Body Structure

Body structure design is based upon 2 g operational landing loads (based on vehicle operational gross weight when carrying a 370 lb payload and full of propellant and including a 25% growth factor on estimated vehicle dry weight) and 8 g Saturn V delivery loads (based on vehicle dry weight).

The body structure consists of a honeycomb sandwich platform and a box structure mounted to it. The platform is the key assembly of the structural system. The landing gear, propellant tanks and box are attached to it and it serves as the platform for the astronaut. The lower LM attaching fitting and the payload pallet fittings are also part of this assembly. The platform is a four-inch thick brazed titanium honeycomb sandwich. The sandwich has an upper and lower face of 0.015 sheet and a core

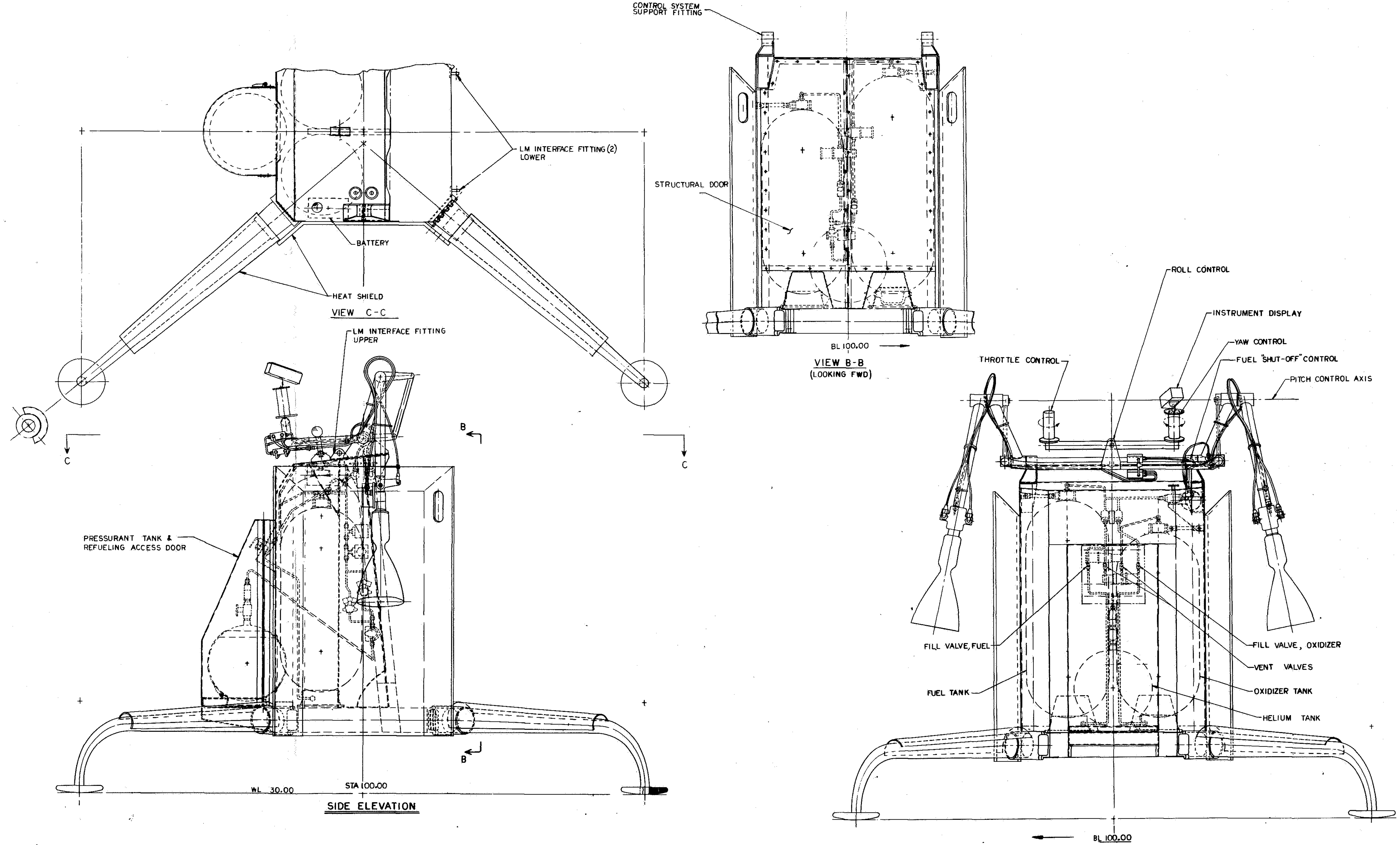


Figure 3.2. OMLFV Inboard Profile

of 1/4 inch cells of 0.0015 honeycomb. The outboard edges between the corner fittings are titanium channels. The upper and lower faces, corner landing gear attaching fittings and edge channels are brazed directly to the honeycomb core in one brazing operation. Titanium was chosen because of its excellent strength to weight ratio at the temperatures experienced in lunar storage and operation.

The box structure attached to the platform contains, and provides mounting attachments for, the propulsion and electrical system components. Mounted to the tope of this structure are the control system, engine supports and upper LM attachment fitting.

The front of the box has a reinforced frame around the helium tank and refueling access door. This door has a continuous hinge on one side and latches on the other. The aft side of the box has a large structural access door to facilitate the installation of propulsion components. The box is made of reinforced titanium sheet.

3.3.3 Propulsion

This system is composed of the pressurization, propellant storage/delivery, rocket engine and instrumentation subsystems. The following paragraphs discuss the installation of these systems in the vehicle. Section 5.0 presents the analysis of the propulsion system in detail.

The pressurization system consists of a helium tank assembly, regulator, filter, quad-redundant check valves, pressure relief valves, vent valves for refueling, and lines and fittings. The helium tank assembly is replaced with a precharged tank between sorties. The helium tank assembly consists of the tank, an isolation valve and a quick disconnect fitting. The tank is installed in a strapdown mount located at the front of the vehicle. A hinged access door is provided for access to the tank.

The propellant storage/delivery system consists of fuel and oxidizer tanks, propellant filters, isolation valves, refueling valves, and lines and fittings. The propellant tanks are sized for 300 pounds of usable propellant at a mixture ratio of 1.3. During operation and storage on the moon the propellant temperature is maintained at $70^\circ \pm 30^\circ$F by insulating the inside of the structure surrounding the tank and by thermal coatings on the outside of that structure. The tank working pressure is 235 psi. Antislosh baffles and antivortex vanes are installed in each tank. The tanks are made of titanium alloy. The bottom tank mounts, also made of titanium, react vertical and lateral loads. The upper mounts, also titanium, react lateral loads. The fueling system consists of two quick disconnect fittings, one for each propellant mounted on a panel located under the front access door.

The rocket engine system consists of the engines, start/stop valves, throttle valves and lines and fittings. Each 150 pound engine with its 40 to 1 expansion nozzle is yaw pivot mounted to a tubular strut whose upper pivot is the engine pitch axis. The engines are mounted high off the surface to minimize plume/surface

impingement effects (see Section 9.0). A start/stop valve is mounted to each engine. The throttle valves are mounted to the pitch control structure at the front of the vehicle. The operation of the throttle valve is described in paragraph 3.3.4. Flexible propellant delivery lines are used to accommodate the pitch, yaw and trim control motion of the thruster assemblies. These lines are thermally protected with multi-layer insulation.

The propulsion instrumentation system is described in paragraph 3.3.6.

3.3.4 Control System

The manually operated control system is shown in Figure 3.3. The system provides pitch, yaw, roll and thrust control and preflight trim adjustment.

A proportional acceleration pitch command is obtained by an up/down motion of both hands through a $\pm 12^\circ$ angle. The pivot axis for this input motion is adjacent to the astronaut's elbows. The controller motion is transmitted to the thruster assemblies through a four bar linkage; the thruster pitch pivots are located adjacent to the astronaut's shoulders, above the vehicle cg. Moving the hands downward produces a nose down pitch command.

A proportional acceleration roll command is produced by rotation of the handle bar assembly (to which the yaw and throttle controllers are attached) around its "x" axis pivot; clockwise rotation produces clockwise roll. At the 10 degree maximum deflection of the roll control bar the thrust of one engine is increased by 8.75 pounds, the thrust of the other is simultaneously reduced 8.75 pounds. The roll force couple is proportional to the angular rotation of the bar; it is not dependent upon the throttle setting. The linkage shown is one of several possible arrangement which meet the requirement for a minimum of undesirable roll/throttle coupling.

The yaw controller is operated by the left hand; rotating clockwise produces a clockwise yaw command. The maximum controller motion is set at $\pm 20^\circ$ which, through a push/pull cable and linkage system, rotates the thrusters about their yaw pivots $\pm 5^\circ$ differentially.

The thrust controller is operated by the right hand. The linkage shown in Figure 3.3 is arranged to operate the two bipropellant throttle valves collectively. The throttle handle moves through an arc of 30° for a throttle stroke of 0.5 inches. The unbalanced internal hydraulic force in the valves is balanced externally by a spring. A breakout force of about two pounds is required to overcome the static friction of the "O" ring seals in the valve. This force effectively prevents throttle creep for hands off operations.

Vehicle Pitch and Roll Trim Control

The vehicle system center of gravity can shift forward almost 12 inches as payload mounted on the vehicle is increased from 0 to 370 pounds. In addition, the

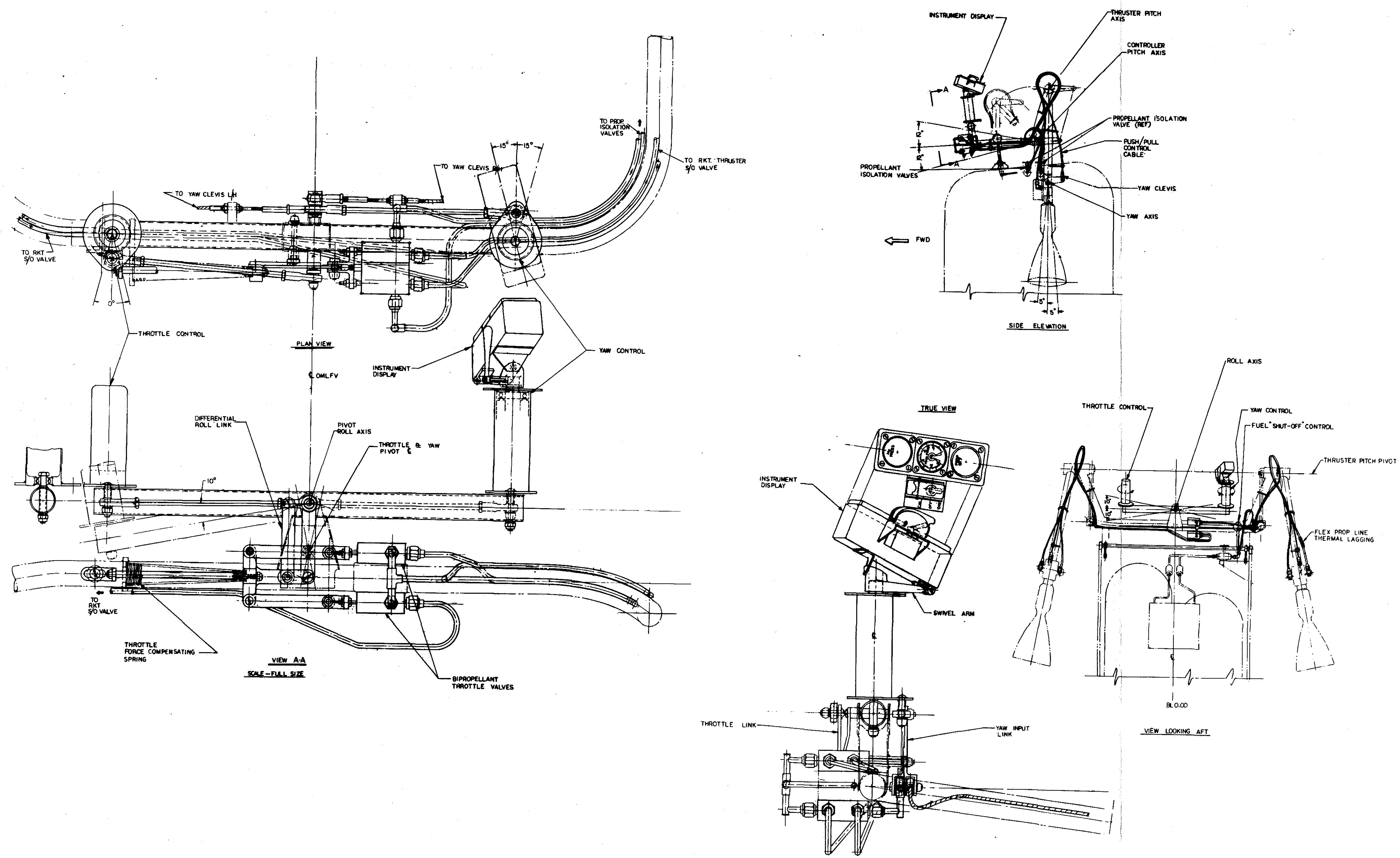


Figure 3.3. OMLFV Control Systems

center of gravity will shift up to 4.0 inches with propellant consumption (Ref. Paragraph 3.4, Table 3.2).

Pitch trim to compensate for variable payload is accomplished by a pre-takeoff adjustment of engine position. Each engine pitch pivot is mounted on an adjustable arm, so it can be moved in steps from station 91.9 to 103.4 by removal and reinsertion of a locking pin. This adjustment is shown schematically in Figure 3.4. Note that the hand controller location is not affected by this adjustment.

The 4.0 inch shift with propellant consumption (at zero payload) is trimmed by the pitch attitude controller. This uses up $\pm 5^\circ$ of the total $\pm 12^\circ$ control range available, leaving sufficient range for attitude control.

Since the payload and propellants are located symetrically about the roll axis, roll trim can be easily accomplished with the roll attitude control.

3.3.5 Thermal Protection

The thermal protection system has been developed from the data presented in Section 7.0. The system is entirely passive, consisting of thermal coatings, internal insulation and external engine radiation/plume protective shields.

The propellant tanks, propulsion components and battery pack are used as a common heat sink by insulating the inside of the body structure with 18 layers of aluminized mylar. Where feasible all components are mounted to the internal aft center frame so as to minimize the number of heat conducting paths. Where internal/ external attachments cannot be avoided, as is the case for some fittings, an insulating gasket between the fitting pad and the body structure is installed to reduce heat flow.

The shields along the sides of the body structure protect the vehicle from the rocket engine plume and its radiation. These shields are of multilayer construction similar to that used on the LM descent stage, see Figure 3.5. Titanium shields over the landing gear legs in the area of plume impingement are utilized to protect the gear from excessive temperature due to plume heating.

3.3.6 Instrumentation and Electrical Power

The vehicle instrumentation consists of a propellant quantity gage for each tank, propellant tank pressure sensors, a helium tank pressure sensor, a battery power source and a panel display. The display is mounted on the stationary post of the yaw control handle. This location was selected because it is within the astronaut's view and offers minimum restriction to his landing visibility. The face of the panel is shown in Figure 3.6; the field of view is shown in Figure 3.7.

A battery power source is provided for the instrumentation system. The transonic fuel tank quantity system is the principal power load, requiring 10 watts. Total flight and refueling time is approximately one hour so the power requirement is slightly more than 10 watt hours. Two 10 watt hour batteries are provided for redundancy and to provide 100% power reserve power capacity. Twenty Yardney

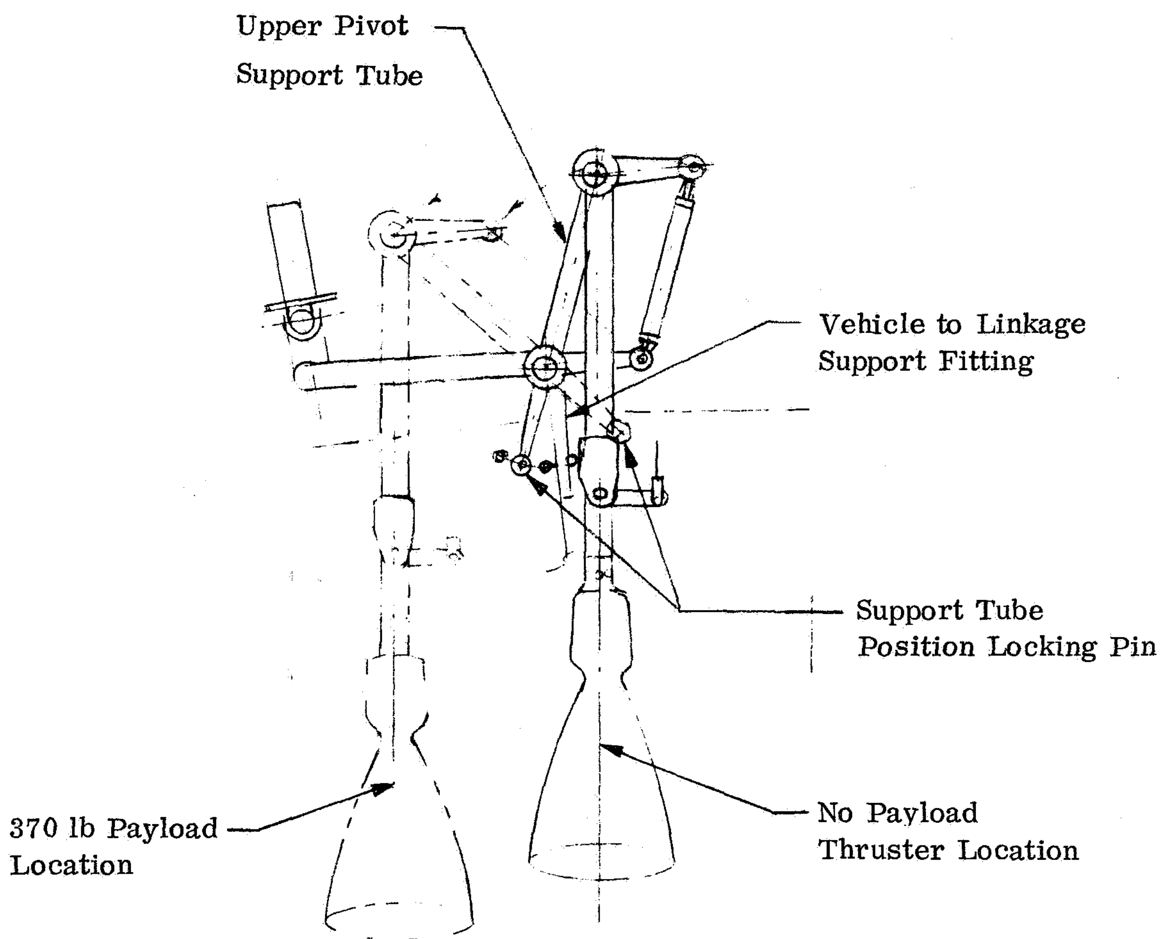


Figure 3.4. Trim Adjustment Schematic

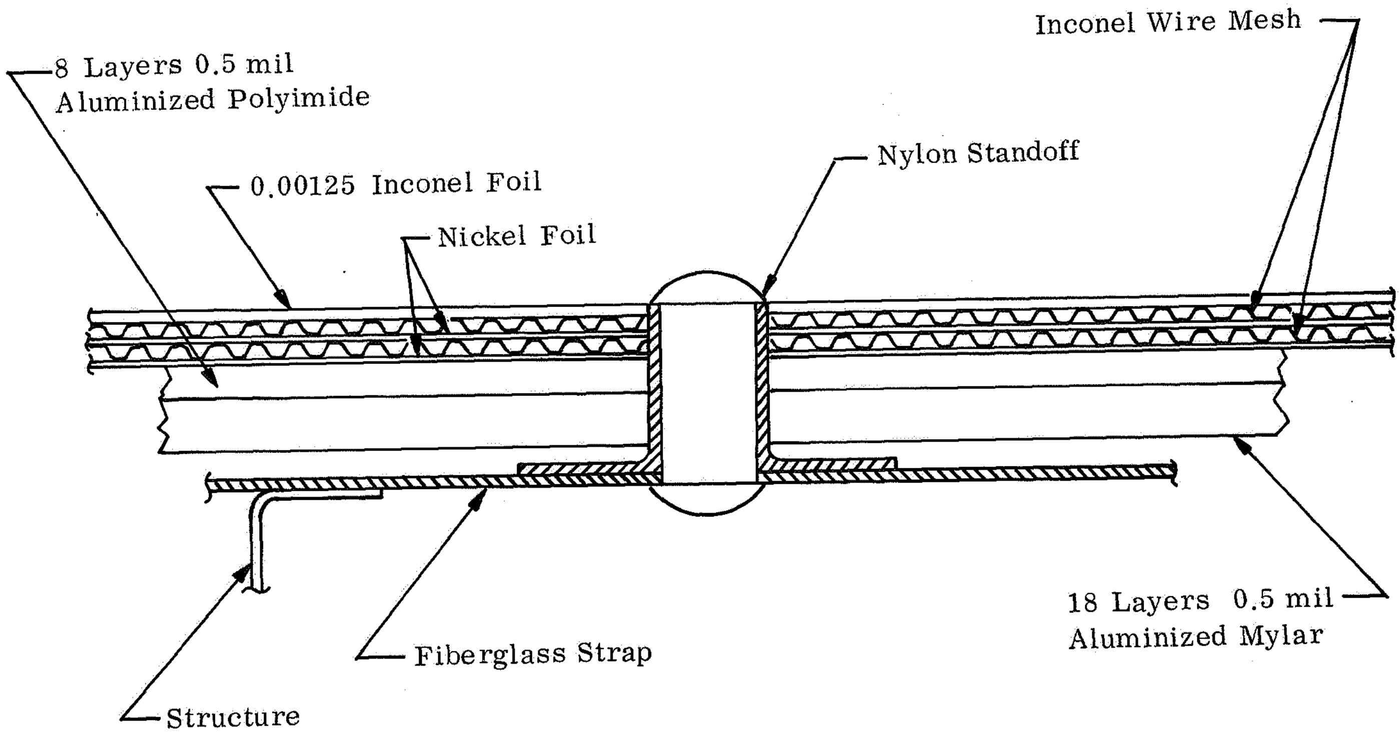


Figure 3.5. LM Thermal/Micrometeoroid Shield

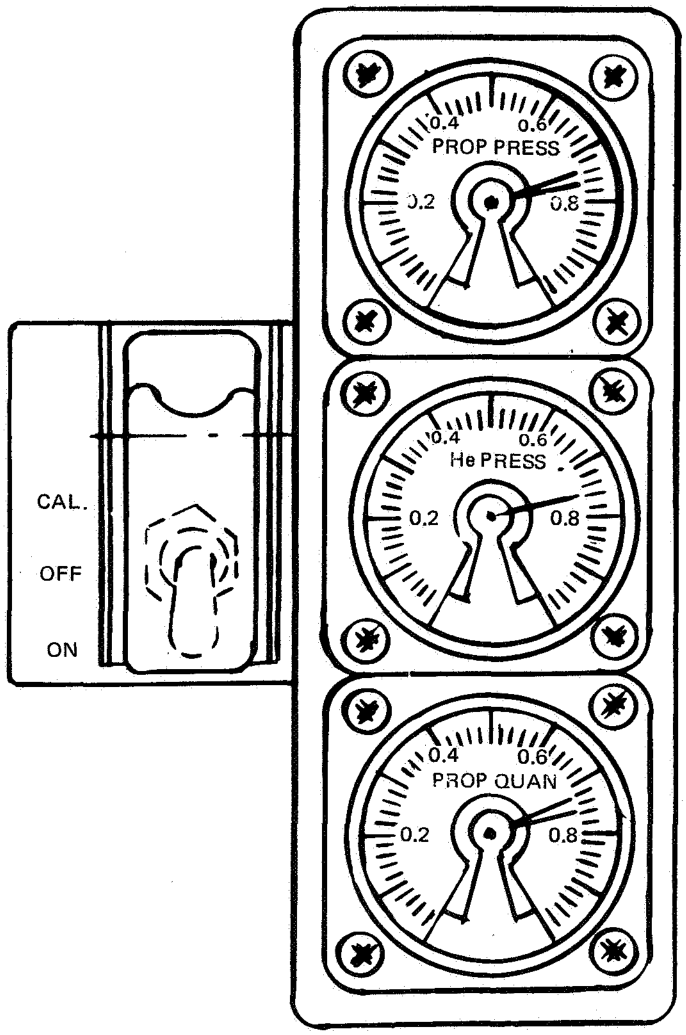


Figure 3.6. Display Panel

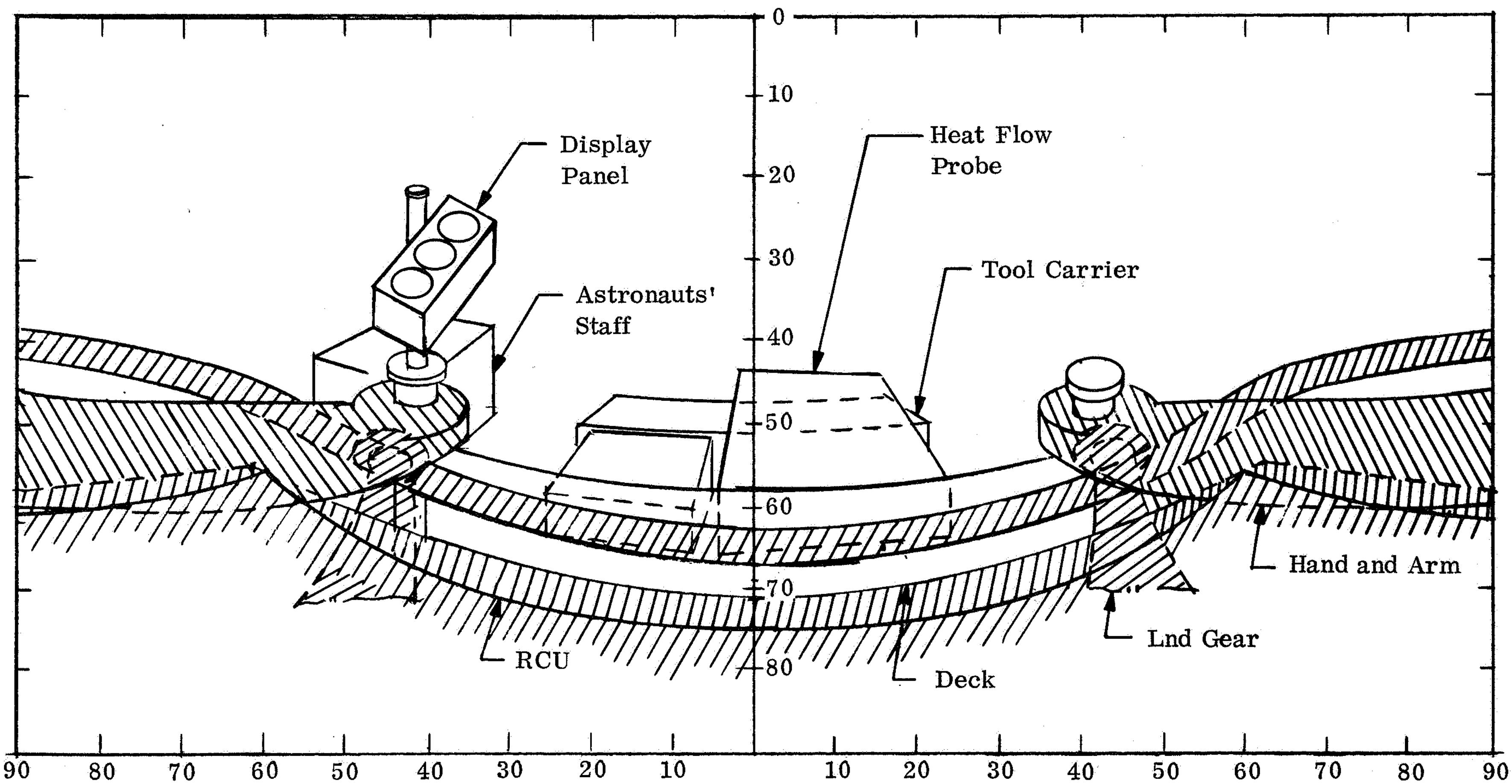


Figure 3.7. View Angles - Selected Configuration

Electric Corporation LR-05 silver zinc battery cells, each providing a nominal 1.5 volts are used in each battery. The electrical diagram for the system is shown in Figure 3.8. The two batteries are installed in one pressure tight container 3.2 inches in diameter and 6.5 inches long. A pressure relief valve vents the container from earth atmospheric to approximately 5 psia for lunar operation.

A three position power switch is mounted on the left side of the display panel. When the switch is in the test position each indicator should register a preset value, serving as a calibration check for each indicator.

3.3.7 Astronaut Restraint

Two straps are provided to restrain the astronaut in the vehicle as shown in Figure 3.9. These straps, one at each side, are anchored to the upper aft corner fitting of the vehicle. They are snapped into the "D" rings on the suit at the waist (these "D" rings are used for restraint in the LM) upon entry into the flyer crew station.

3.4 MASS PROPERTIES DATA

A summary weight statement for the flying vehicle is presented in Table 3.1. The vehicle dry weight of 235 pounds was increased by a factor of 25% for design of the engine (thrust requirements), landing gear, and structural weight. In addition the vehicle performance data presented in Section 10.0 accounted for this potential growth factor.

Summary mass properties, including weight, center of gravity, mass moment of inertia, and cross products of inertia for five payload conditions are shown in Table 3.2.

Section 6.0 shows that vehicle handling qualities are acceptable at sensitivities corresponding to the complete spectrum of cg and inertia characteristics indicated in Table 3.2.

Paragraph 3.3.4 discusses the methods of trimming the vehicle for level flight in spite of cg shift due to variable payload and propellant weights.

The total weight chargeable to a two flyer system is summarized in Table 3.3. Two fully charged helium bottles are on the flying vehicles and two additional charged bottles are stowed on the LM descent stage to provide sufficient pressurant for four sorties. Additional thermal shielding and structural modifications are required on the LM to accommodate the flyer system. The weight of these items has been estimated as 68 pounds per flyer (see Section 11.0).

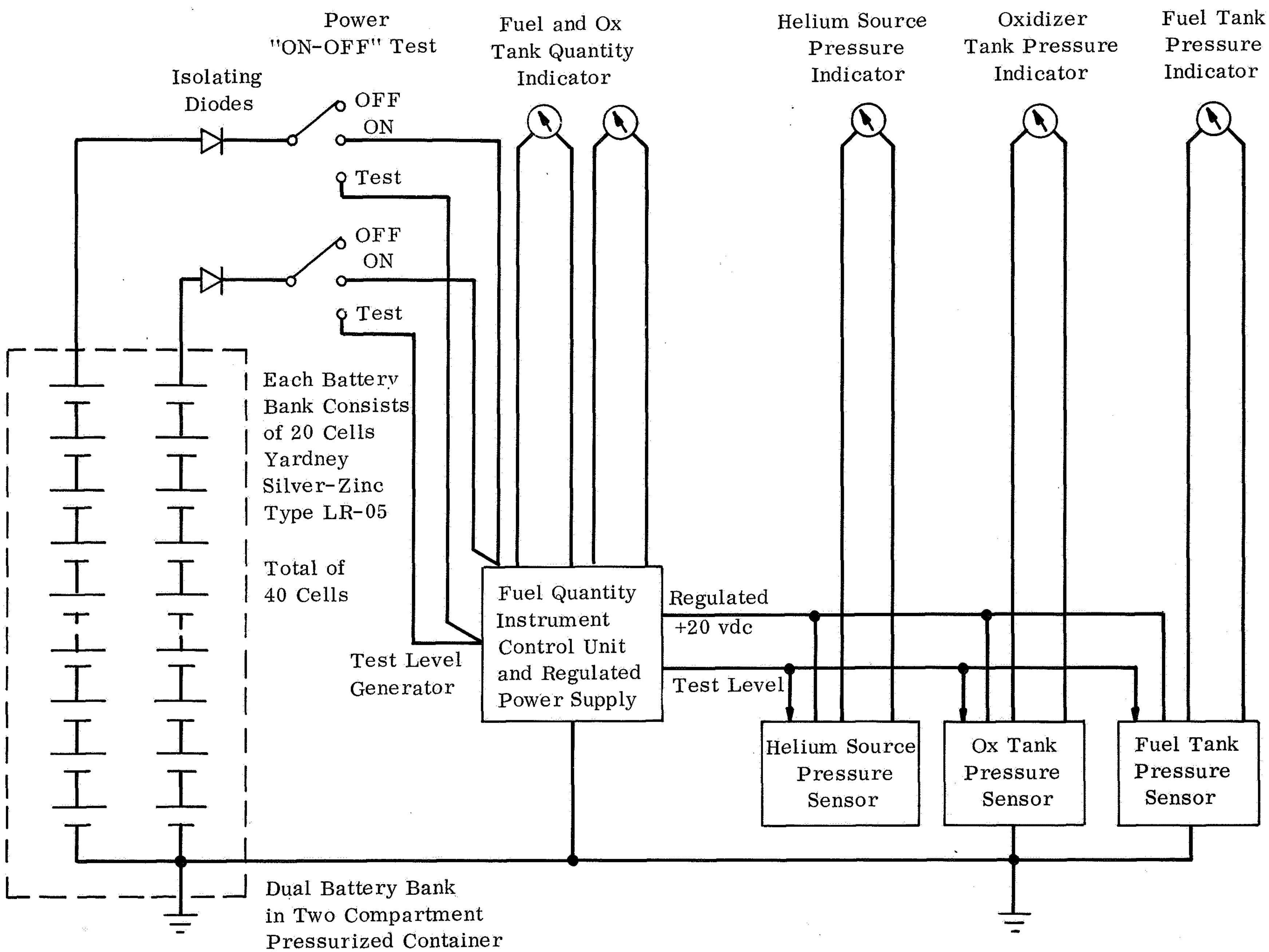


Figure 3.8. Power Supply Schematic

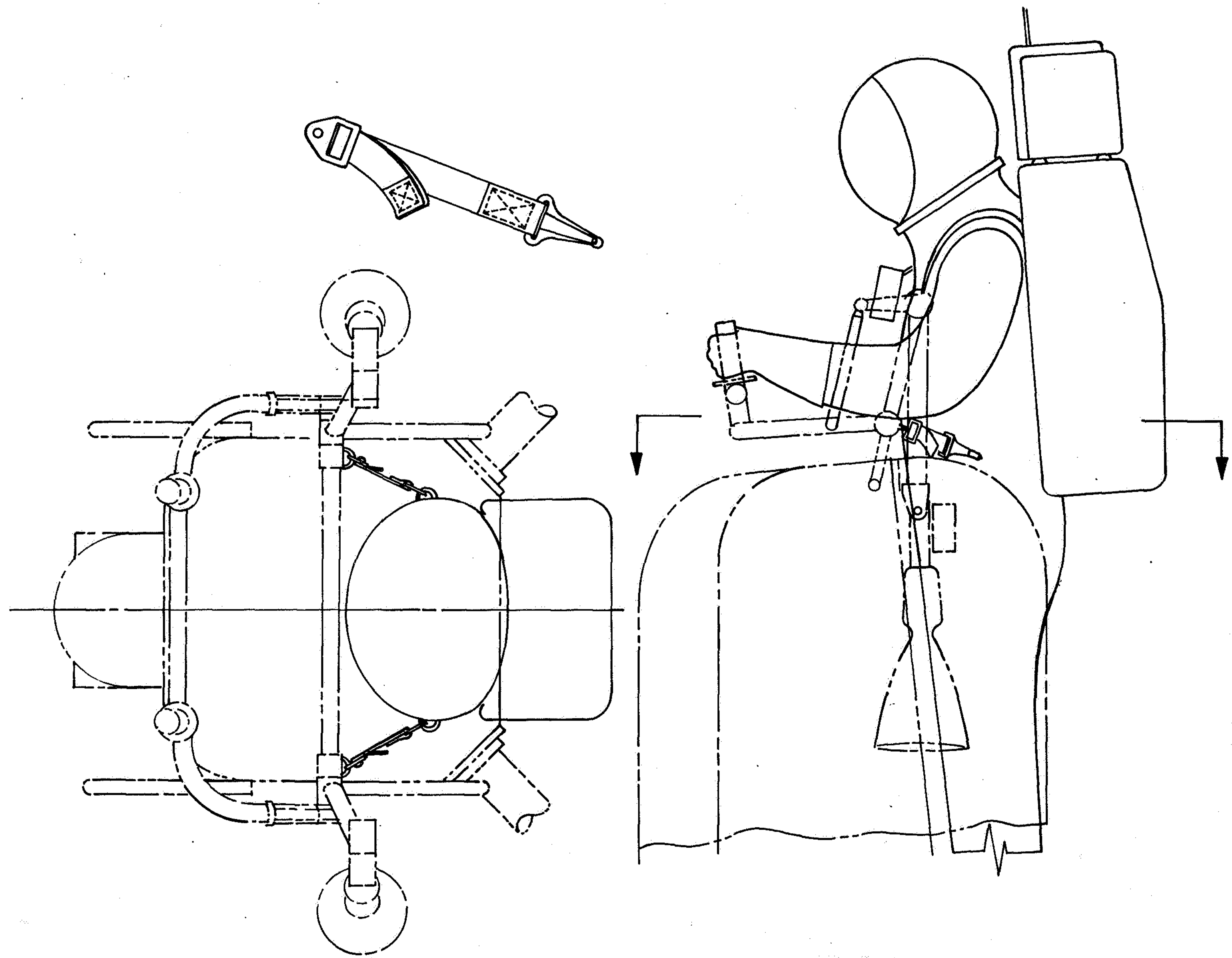


Figure 3.9. Astronaut Restraint System

TABLE 3.1

SUMMARY WEIGHT STATEMENT FOR OMLFV

Item	Weight (lbm)		
Structure			71.5
Honeycomb Floor		20.1	
Floor Edge Channel		4.4	
Propulsion Tanks Housing		33.0	
Handholds and Grips		1.0	
Pressure Tank Surround		3.8	
Pressure Tank Mount Reinforcement		0.5	
Propulsion Tank Mount Reinforcements		1.0	
Control System Attachment Reinforcements		1.6	
Payload Hardpoints		0.6	
Miscellaneous Reinf., Gussets, and Standards		2.5	
LM Attachment Fittings and Reinforcements		3.0	
Induced Environmental Protection			15.6
Radiation Shield		8.4	
Propulsion Tanks Insulation		3.0	
Pressure Tank Insulation		1.0	
Landing Gear Strut Shields		3.2	
Launch, Recovery and Docking (Landing Gear)			37.2
Landing Gear Struts		15.6	
Landing Gear Pads		3.6	
Landing Gear Fitting (Floor Attach.)		8.4	
Landing Gear Fitting (Strut)		8.0	
Landing Gear Pins (Fittings)		1.6	
Main Propulsion			71.6
Propulsion System		34.5	
Oxidizer Tank	9.4		
Fuel Tank	10.3		
Valves, Connectors, Filters	5.6		
Lines and Fittings	5.2		
Propulsion Supports	4.0		
Pressurization System		22.5	
Pressure Sphere	12.9		
Valves, Connectors,	3.9		
Regulator and Filter	1.2		
Pressurization Supports	2.3		
Lines and Fittings	2.2		

Item	Weight (lbm)		
Engine		14.6	
Thrust Chambers	9.6		
Engine Mounts	0.6		
Throttle Valves	2.4		
Engine Support Tubes	2.0		
Orientation Controls			21.9
Yaw Controller		1.0	
Throttle Controller		1.0	
Main Control Tube		1.4	
Main Cross Tube		0.8	
Attachment Main Cross Tube to Structure		1.2	
Thrust Vector Position Controller		1.0	
Tube - Cross Tube to Eng. Sup't. Top Sleeve		2.0	
Misc. Tube, Links, Rods, Bearings etc.		8.3	
Yaw Control Cable System		3.2	
Misc. Stds., etc.		2.0	
Prime Power Source			3.0
Battery		2.5	
Wiring		0.5	
Instrumentation			9.6
Quantity Sensing Probes		6.0	
Quantity Sensing System Controller		3.0	
Gas Pressure Transducer		0.2	
Propulsion Tank Press. Transducers		0.4	
Personnel Provisions			4.7
Instrument Panel		2.0	
Astronaut Restraint		1.5	
Payload Attach. Fittings		1.2	
Subtotal			235.1
Personnel		370.0	
Subtotal			605.1
Residual Propellant and Service Items		6.6	
Subtotal			611.7
Full Thrust Propellant		300.0	
Total			911.7

TABLE 3.2

MASS PROPERTIES SUMMARY

	*** Weight	** X	** Y	** Z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
	C.G. Location									
No Payload	(lbm)	(in.)	(in.)	(in.)	(Slug-ft^2)					
Burnout Weight	611.7	105.4	99.9	78.6	80.5	88.5	34.1	0.2	14.7	0.2
Gross Weight	911.7	101.4	99.4	74.7	98.1	109.6	44.9	1.0	21.1	3.8
100 Pound Payload										
Burnout Weight	717.8*	100.5	99.9	75.2	93.6	124.1	59.1	0.2	30.3	0.1
Gross Weight	1017.8*	98.3	99.4	72.7	108.4	138.3	66.0	0.7	33.2	3.6
138.3 Pound Payload (Group I)										
Burnout Weight	756.1*	98.9	99.7	73.8	102.3	140.6	69.1	0.8	37.2	0.6
Gross Weight	1056.1*	97.3	99.4	71.7	116.0	152.8	75.0	1.2	39.1	3.9
255 Pound Payload (Group II)										
Burnout Weight	872.8*	91.8	99.9	74.7	100.7	191.7	124.0	0	39.4	0.1
Gross Weight	1172.8*	92.2	99.5	72.6	115.3	203.2	128.5	-0.1	38.9	3.6
370 Pound Payload										
Burnout Weight	987.8*	91.7	99.9	73.5	104.8	177.9	110.7	0	39.4	0.1
Gross Weight	1287.8*	92.0	99.6	71.9	118.6	188.7	115.1	-0.1	38.9	3.5

* Includes payload and payload pallet (6.1 pounds).
** Refer to Figure 3.1 for reference lines and axes.
*** All weights include pilot (370 pounds).

TABLE 3.3

TOTAL SYSTEM WEIGHT

Item	Single Flyer	Two Flyers
Vehicle Dry Weight	235.1	470.2
Helium Gas (Not Included in Dry Weight)	1.6	3.2
Payload Pallet	6.1	12.2
Helium Tanks and Gas for 2 Sorties	27.4	27.4
Propellant Resupply Equipment	30.0	30.0
Deployment Equipment	27.5	55.0
LM Structural and Thermal Modifications	68.0	136.0
	395.7	734.0

Table 3.3 indicates a total system weight of 734 pounds to carry two flyers on LM, an increase of 63% over the 450 pound weight goal established at the initial orientation meeting. Several possibilities exist for weight reduction; on the other hand, no weight growth is included in the 734 pound total. Thus, it is unlikely that two of the present flyers with their support equipment can meet a weight goal of 450 pounds. Table 3.3 indicates that the goal can be met with a single flyer, at a total weight of 395.7 pounds.

The following are weight reduction possibilities for the one or two flyer systems.

Vehicle dry weight can be reduced about 10 pounds per flyer by reducing flyer payload capability. The use of advanced composite structural materials in place of aluminum and titanium has reduced weight as much as 40% in other applications, and might reduce flyer weight by 30 pounds per flyer.

Flyer deployment equipment weight might be reduced by additional design and mockup/pressure suited subject tests. The system used in this study is based on the technique used for unloading ALSEP, already tested. Also two sets of unloading gear are provided, one for each flyer. It may be possible to develop other techniques in which one lower weight deployment system might be employed to unload both flyers, for a weight reduction of 30 pounds.

LM modification weight includes a conservative allowance of 57 pounds of structure per flyer for support of thermal insulation, vehicle, and unloader. This

represents 21% of the mass being supported. Additional detail design should be able to reduce this factor to 15% for a saving of 17 pounds per vehicle.

If all methods suggested are employed, the 734 pound two flyer system weight will be reduced to 590 pounds, not including a weight growth factor. A 25% growth factor applied to the 590 pounds, results in a system weight of 738 pounds. It is therefore recommended that the 734 pound estimate be accepted as a prediction of the final two flyer system weight including growth.

3.5 RELIABILITY ESTIMATE

The approach taken to achieve high reliability in the lunar vehicle is to employ a minimum number of components and to apply large design margins, trading performance for high reliability if necessary. The design has been defined in sufficient detail to permit accomplishment of a failure modes and effects analysis (FMEA) and it is recommended that such an analysis be undertaken at an early date to indicate whether additional design modifications can be made to improve vehicle reliability and safety. It is recognized that, until such an analysis is accomplished, it is not very meaningful to make a quantitative reliability prediction. This is especially true if the reliability figure is to be compared with estimates made for other lunar vehicle systems or with estimates made for existing systems where a large test experience data base exists. The mission, assumptions, and loss being analyzed for must be clearly identified. However, for this study as an aid to concept selection, a consistent set of failure rate data was established and used for the purpose of comparing one concept with another (see Section 2.0). This data is presented in Table 3.4 along with notes regarding the original source of information and the rationale used in modifying the source data for flyer mission and duty cycle usage. Using this data, an overall vehicle reliability estimate was made for the preliminary design vehicle and is presented in Table 3.5. This can be considered as an estimate of the probability of successful sortie completion.

3.6 ADDITIONAL CAPABILITIES

The basic vehicle can meet the mission requirements. However, the incorporation of additional hardware can enhance the vehicle capabilities for specific missions. Four systems are presented, which can be applied without major change to the basic vehicle.

3.6.1 Control Augmentation

Section 6.0 presents data to show that more pleasant handling qualities can be achieved by adding control or stability augmentation. A fail-safe spring damper system can be introduced into the control linkages which will provide an approximation of rate command without compromising the high reliability of the basic mechanical system. This requires the addition of caged compression springs and a damper, at a weight estimated at 2.0 pounds per axis.

TABLE 3.4

COMPONENT FAILURE RATES BASED ON OMLFV MISSION

Component	Overall Failure Rate per 10^6 Sorties (10 Minute Sortie with 3 Stops)	Source and Rationale
Pressure Tank	6.3	Reference 1, Apportionment = 5.0 x 10^{-6}; Increased by 25% due to Four Landing Cycles.
Gas Isolation Valve (Manual)	1.0	Reference 2, Manual Shut Off Valve Industry Predicted Failure Rate = 0.1 x 10^{-6}; Increased by Factor of 10 due to Environmental Conditions.
Quick Disconnect	0.6	Reference 3, pg 2.481 Industry and Military Aircraft Experience Failure Rate = 4.6 x 10^{-6}; Decreased due to Mechanical Arrangement and Checking by Personnel.
Gas Filter	1.9	Reference 3, pg 2.380 Aircraft Experience Rate.
Regulator	58.0	Reference 1, Apportionment = 469 x 10^{-6}; Decreased due to less Cycles (by a Factor of 10), Further Modified because of Landing Cycles.
Quad Check Valves	0.2	Reference 3, pg 2.486 Laboratory and Missile Experience Rate = 151 x 10^{-6} per Cycle and Individual Valve - Redundancy 0.005 x 10^{-6} per Cycle; 4 Cycles and Factor of 10 for OMLFV Mission.
Relief Valve	12.6	Reference 1, Apportionment = 10 x 10^{-6}; Increased by 26% due to Four Landing Cycles.
Tank Vent Valve (Manual)	0.3	Reference 2, Manual Shut Off Valve Industry Predicted Failure Rate = 0.1 x 10^{-6}; Increased by Factor of 3 due to No. of Cycles and Differences in Environment.
Fuel Tank	6.5	Apollo Tank Shell Apportionment 0.6 x 10^{-6}; Increased by Factor of $\approx$ 10 due to Landing Cycles.
Oxidizer Tank	8.5	Apollo Tank Shell Apportionment 0.8 x 10^{-6}, Increased by Factor of $\approx$ 10 due to Landing Cycles.
Fill and Drain Valve (Manual)	0.6	Reference 1, Apportionment = 0.5 x 10^{-6}; Increased by 20% due to Environmental Differences.
Propellant Filter	1.9	Reference 3, pg 2.380 Aircraft Experience.
Isolation Valve (Manual)	1.0	Reference 2, Manual Shut Off Valve Industry Predicted Failure Rate = 0.1 x 10^{-6}; Increased by Factor of 10 due to Environmental Conditions.
Bipropellant Throttle Valve (Manual)	27.0	Reference 3, pg 2.486, Aircraft Experience Rate = 2.08 x 10^{-6} per Cycle; Increased by Factor of $\approx$ 10 Assuming $\approx$ 10 Extended Motions of Valve
Bipropellant Shut Off Valve (Manual)	5.5	Marquardt Data on LM RCS Electrically Actuated Bipropellant Shut Off Valve = 1.5 x 10^{-6} per Cycle; Four Cycles per Sortie - Reduced due to Manual Rather than Electrical Actuation.
Thrust Chamber	20	LM Ascent Engine (with Redundant Valves) Apportionment.
Lines and Fittings	9.2	NRA Failure Rate Manual - Industry Predicted = 4.3 x 10^{-6}; Increased by Factor of $\approx$ 2 based on Environment and Landings.
Propellant Quantity Sensor	20	Estimate
Pressure Transducer	6	Estimate
Meter	3	Estimate
Throttle/Roll Controls	20	Estimate
Pitch Control	8	Estimate
Yaw Control	2	Estimate

Reference 1 - Minuteman III PBPS
2 - Autonetics Standard Failure Rate Manual
3 - Farada 9/1/68

TABLE 3.5

OVERALL VEHICLE RELIABILITY ESTIMATE

Failures per Million Sorties
(10 Minute Sortie with 3 Stops)

Pressurization System	98.6
Propellant Feed System	26.6
Engine System	105.0
Control Linkage and Gimbals	30.0
Instruments/Displays	47.0
Structure and Landing Gear	10.0
Total	317.2

Reliability 0.999683

3.6.2 LM Controller and Autopilot

If autopilot type flight control is required to accomplish certain scientific objectives such as in-flight stabilization of remote sensor payloads, LM side-arm controllers and an electronic control system can be accommodated by mounting the hand controllers to the tope of the structural box enclosing the propellant tanks. A small extension of the vehicle envelope above the helium tank will permit installation of electronic components and power supply.

3.6.3 Navigation Aids

For longer range flights, more efficient propellant utilization will result if the vehicle is flown at higher velocity, and visual navigation will be easier at higher flight altitudes. However, this may require additional instrumentation.

Flight aids, such as attitude gyros, optical sight, timer, velocity meter, radar altimeter, and radio direction finder equipment can be accommodated. An optical sight would be removable or foldable for stowage onboard the LM. Electronic and power supply equipment would be installed inside the vehicle, closely coupled to the propellant tanks for good temperature control, with modest extensions to the vehicle envelope. Two single slot waveguide radar antennas (1/2 x 1 x 4 inches each) are adequate at the flight altitudes projected for the flyer. These would be attached to the bottom of the structural platform. The additional displays associated with such aids would be mounted to the stationary post of the throttle controller.

3.6.4 Redundant Engines

Since the engines are mounted outboard, space is readily available for a redundant four engine system. Engine redundancy would be provided by modifying each engine mount structure to accommodate two engines. The throttle and roll control assembly would be modified for two pairs of throttle valves.

Sensing and switching equipment would be added to sense which engine has failed and shut down the failed engine and its opposite engine on the other side. Descent to the surface is accomplished on the remaining two engines.

3.7 VEHICLE FOR 100 LB MAXIMUM PAYLOAD

The payload requirement of 370 pounds was established so that a second astronaut could be carried. However, many scientific missions can be accomplished with a payload of 100 pounds or less. A design study was conducted to determine the extent to which the vehicle and mission was being compromised to accommodate the rare occasion when a second astronaut might be carried. All vehicle concepts generated earlier in the study were re-examined as 100 lb payload vehicles. It was established that the same vehicle concept selected for the 370 pound payload was also best for the 100 lb payload. This vehicle was therefore examined to determine the effect of reducing capability to 100 pounds maximum payload. It was found that the

pitch trim system was not necessary, and could be deleted, and a structural weight reduction could be made.

The elimination of pitch trim deletes the operational pretakeoff trim adjustment, and allows replacement of the movable pitch pivot mounts with fixed mounts, at a 1 lb weight saving.

Because of minimum gage restrictions, the structural weight reduction is not proportional to gross weight reduction. Weight reduction is summarized in Table 3.6. The engine weight reduction is possible due to the reduction in thrust (maintaining the same maximum design thrust to weight ratio).

It is recommended that the 370 pound payload capability be retained until such time as a need for carrying a second astronaut can be more firmly established, or eliminated.

TABLE 3.6

REDUCTION IN VEHICLE DRY WEIGHT FOR PAYLOAD CAPABILITY REDUCTION FROM 370 LB TO 100 LB

Item		Pounds Saved
Engines		2.5
Chambers	1.5	
Throttle Valves	0.8	
Shut-off Valves	0.2	
Structure		6.2
Top Payload Attachments	0.4	
Top Payload Reinforcement	0.3	
Vertical Angles	0.4	
Vehicle Base	1.6	
Landing Gear Corner Fitting	0.4	
Landing Gear Struts	1.4	
Landing Gear Pads	0.2	
Payload Shield	0.5	
Payload Pallet	1.0	
Trim Adjustment		1.0
Total		9.7

4.0 LANDING GEAR STUDY

In order to establish the detailed design requirements for a stowable, re-useable landing gear for the OMLFV, parametric studies were conducted to accomplish the following:

1. Establish landing criteria
2. Establish worst case combinations for stability and for loads
3. Compare compability of inverted tripod and cantilever strut legs
4. Compare three-leg and four-leg gears
5. Establish effect of leg stiffness
6. Investigate materials
7. Investigate temperature effects
8. Establish footpad diameter
9. Investigate capability for landing on a hard unyielding surface
10. Design a gear for the selected OMLFV.

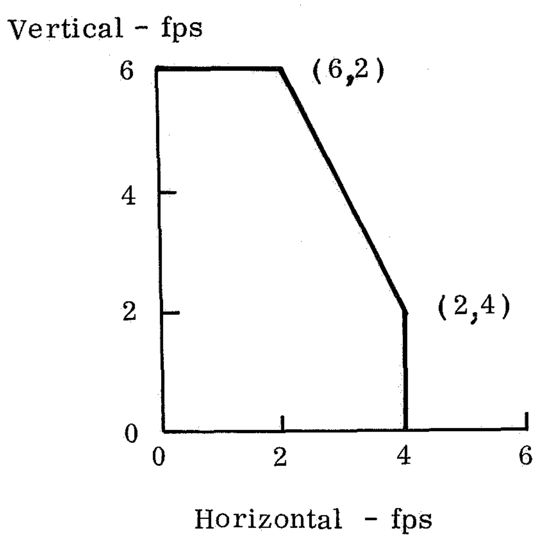
4.1 CRITERIA

The criteria chosen for the study of the vehicle landing gear is summarized in Figure 4.1. This criteria consists of a vertical and horizontal landing velocity envelope, pitch attitude, pitch rate, ground slope, loading conditions, and landing surface definition. The velocity envelope is based on the data shown in Figure 4.2. The figure depicts LM design criteria, LLRV experience, LM-LLRF experience, and Bell Aerosystems experience with a tethered one-sixth g simulator of both pivoted thruster and kinesthetically controlled vehicles. Also shown is a shaded area bounded by six foot-per-second vertical and two foot-per-second horizontal. These values were suggested in the RFP work statement. The selected criteria includes both the shaded area and the Bell simulator experience. It is less severe than the LM criteria, but this is consistent with the type of vehicle.

The OMLFV is much smaller than LM, the pilot is closer to the ground, has a better view of the ground, and can maintain lift and control thrust until the instant of ground contact.

The criteria values for attitude, attitude rate, and ground slope were suggested by the RFP work statement. These are all considered to be conservative for this vehicle since the pilot/control capability is sufficient to keep all these parameters close to zero in the hovering mode. The ground slope of 10° represents landing with two adjacent footpads on a 15 inches high mound or ridge which is easily discernable and could be avoided. However these criteria values do not impose a severe

Velocity



Planar Motion (2 -2 and 1-1 Landings)

Attitude

θ = ±10 Degrees

Attitude Rate

$\dot{\theta}$ = ±6 Degrees/Second

Ground Slope

γ= ±10 Degrees

Vehicle Loading Condition

Stability - Highest cg

No Payload

Tanks Empty

Structural Loads

370 lb Payload

Tanks Full

Landing Surface

Soil

Figure 4.1. Landing Gear Criteria and Design Conditions

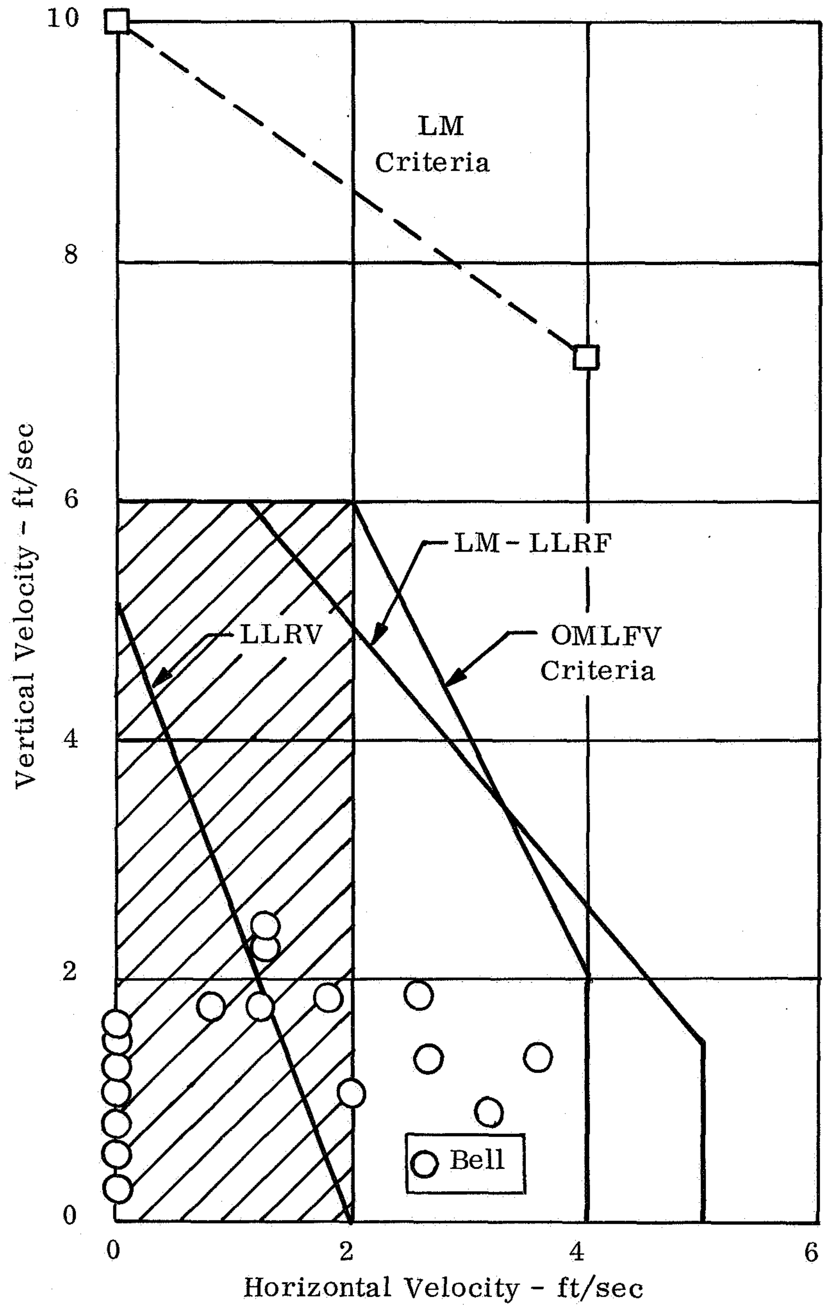


Figure 4.2. Lunar Flight Simulation Landing Data

requirement on the landing gear and they do give the vehicle the capability to touch down on irregular terrain before a perfect hover is fully achieved.

Two vehicle loading conditions were found to be critical and are shown in Figure 4.1. They are minimum weight condition which has the highest center of gravity and is therefore critical for stability, and the maximum gross weight condition which produces the highest loads in the gear and is therefore critical for strength.

The landing surface for which the gear is primarily designed is soil rather than hard rock. Although most previous lunar landing studies have assumed a hard surface for all landings, the incorporation of soil in the landing analysis is based on actual data from the Surveyor Program. (Refs 33-37). A review was made of the data from all the Surveyor flights and this data relative to the soil characteristics is summarized in Table 4.1. It can be seen that the lunar soil at the point of impact of each Surveyor behaves in a similar fashion. Since the Surveyors landed in a variety of locations, including maria and highland areas, and all the locations showed similar soil characteristics, it can be assumed that the lunar soil exists all over the moon surface and it is of sufficiently uniform consistency as to provide a predictable shock absorbing medium for a piloted vehicle. This conclusion is further reinforced by the observations of the surface obtained in the Apollo 11 mission. Surveyor and Apollo photographs did show scattered boulders of various size and concentrations but these can be avoided by the pilot as was done in the Apollo 11 landing.

The parameters of soil characteristics in Table 4.1 are based on a JPL mathematical soil model (Ref 35) and are presented for the purpose of indicating the uniform nature of the lunar surface. The work done in this study utilizes the Bendix mathematical soil model, (Ref. 25) using soil parameters determined by controlled drop test measurements.

The tool used in this landing gear study is a Bell Aerosystems landing dynamic analysis electronic computer program which is described in Appendix A. This program is based on Bendix studies of landing on lunar soil which include correlation with Surveyor landing data as shown in Appendix A.

4.2 DESIGN LANDING CONDITIONS

In order to compare various landing gear designs quickly and efficiently, a preliminary study was performed with a typical vehicle to determine which combinations of the criteria parameters were most critical. This resulted in the six combinations shown in Figure 4.3 which were established as design conditions for the remainder of the study. Conditions A and B are critical for stability and are associated with the high center of gravity low weight configuration. Conditions C through E are load conditions and are associated with the maximum weight configuration. Condition F is a four leg vertical landing and is not critical for stability or loads but is included for comparative purposes. The computer program is limited to planar three-degree-of-freedom landings in order to achieve the most information in minimum time. Conditions A, B and D are conventional 2-2 landings, but conditions C and E are 1-1 landings rather than the conventional 1-2-1. This is an additional simplification which is more realistic when landing on irregular terrain and also results in more critical loads on the vehicle. (See Appendix A for landing definitions.)

TABLE 4.1

SURVEYOR DATA SUMMARY

Surveyor Number	Site	Velocity (m/sec)	Shock Absorber Load (N)	Penetration (CM)	Ave. Static Bearing Strength (N/CM^2)
I	Mare	3.6	6000	3-8	3.4
III	Oceanus Procellarum (Mare)	1.8	4500	3-5	3.4
V	Mare Tranquillitatus	4.2	7300	12	2.7
VI	Sinus Medii (Mare)	3.4	8000	5-7	3.4
VII	Crater Tycho (Highlands)	3.8	7800	4-4.5	3.4

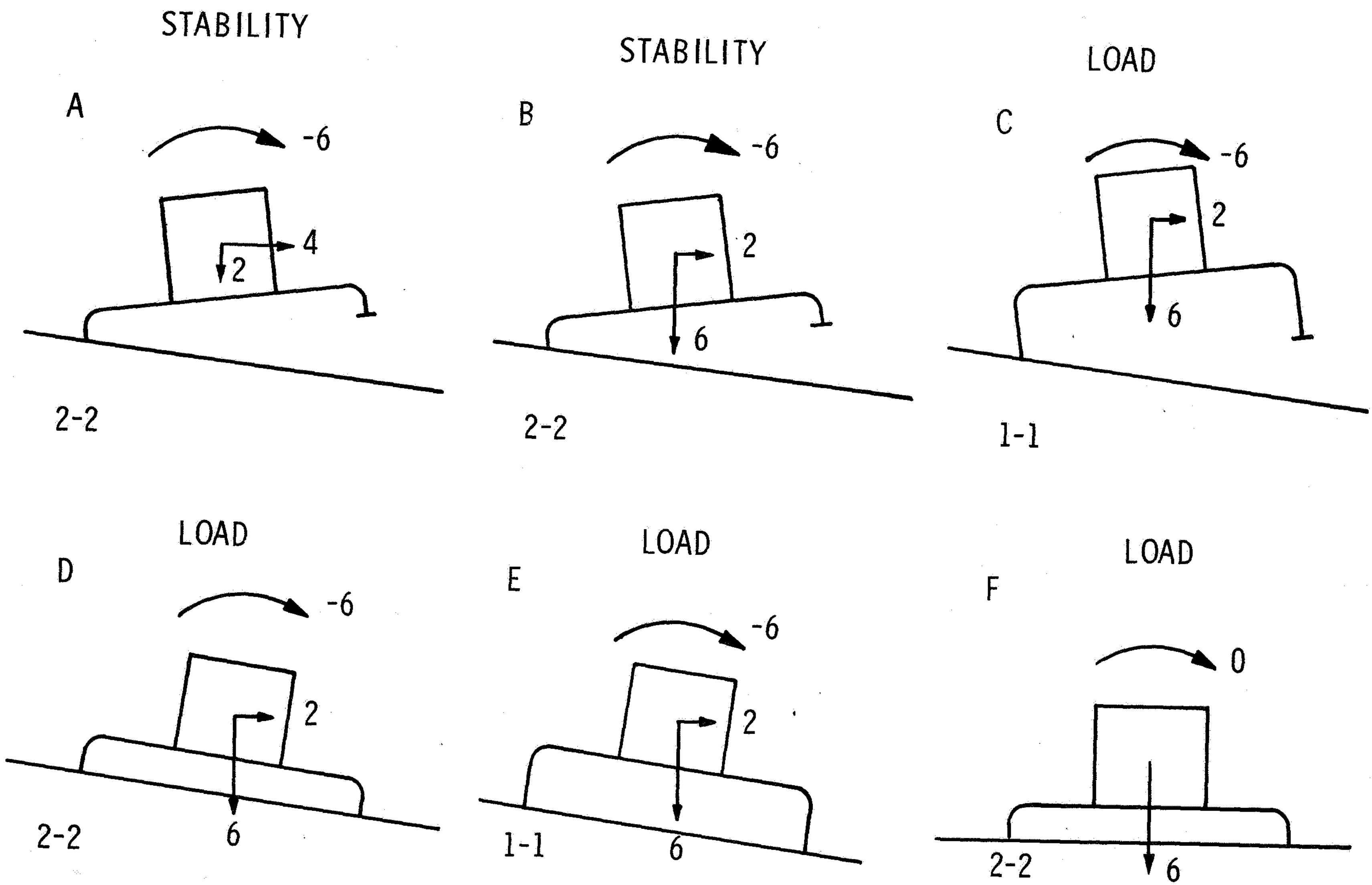


Figure 4.3. Design Combinations

Although the above combinations probably cover all the maximum loadings, it is known that some out-of-plane yawed heading with cross slope velocity combinations are more critical for stability. Therefore a gear designed to accommodate the selected combinations will have a reduced velocity and/or slope capability for some out-of-plane situations. An expanded three-dimensional computer program and study would be required to determine this capability.

4.3 COMPARISON OF INVERTED TRIPOD AND CANTILEVER STRUT LEG

Using the computer program, the gear geometry required to meet the established criteria was determined for both an inverted tripod type gear and a cantilever strut gear for a representative vehicle. Stability performance is compared in Table 4.2 and significant design considerations are compared in Table 4.3. Stability and weight are both nearly identical for the two designs. However the cantilever strut is superior with regards to stowability, lunar erection, and reliability. The tripod design requires the stowage of twelve members but the cantilever design involves only four. Unstowing and erecting the tripod gear is more difficult, and it also requires four energy absorbing mechanisms as compared to none for the cantilever design. On the basis of these comparisons the cantilever leg gear was selected for the OMLFV.

4.4 COMPARISON OF THREE-LEG AND FOUR-LEG GEARS

A review was made of previous studys of three legs versus four legs (Refs 39-43). These studies indicated that the gear radius from the center of the vehicle to the footpad for the three leg design should be 20 to 25% greater than that for the four leg system, based on equal probability of stable landings. A review of the results of the computer runs performed in the search for the design conditions in the present study indicates that the maximum ground reaction for a three leg gear is about the same as for a four leg design. Using a 20% greater leg length for the three leg gear and the same maximum load per leg as for the four leg gear, a typical gear was analyzed for each arrangement with the results shown in Table 4.4. The leg systems are equal in weight, but the three leg gear is less convenient to stow due to the longer leg length. The four leg gear was therefore selected for the OMLFV.

4.5 EFFECT OF LEG STIFFNESS

The effect of leg spring rate on the landing gear radius required for stability is shown in Figure 4.4. The spring rate is the deflection of one leg when a 1000 pound vertical load is applied at the top of the leg, i.e., at the footpad. A vertical load will also cause horizontal deflection, and horizontal loads will likewise cause vertical as well as horizontal deflections. The spring rate plotted in Figure 4.4 is the vertical deflection due to vertical load, to illustrate the effect of stiffness and is not actually a design tool. Relative values of the other spring rates were used in determining the required landing gear radius. This investigation was made for a typical vehicle with a center of gravity 55 inches above the footpad, but the data is plotted in terms of R/H which is the horizontal distance from the center of the vehicle to the center of the

TABLE 4.2

COMPARISON OF INVERTED TRIPOD AND STRUT/PAD PERFORMANCE

Stroke Load	V_v	V_h	R/H	Maximum Pitch Angle
232	6	2	0.95	-26.5
290	6	2	0.95	-28.0
348	6	2	0.95	-29.0
Strut/Pad	6	2	1.03	Marginally Unstable
148	2	4	1.07	-31.0
290	2	4	1.07	-35.0
435	2	4	1.07	-37.0
Strut/Pad	2	4	1.07	-20.5

TABLE 4.3

COMPARISON OF INVERTED TRIPOD LEG AND STRUT/PAD TYPE LANDING GEAR

Comparison Factor	Strut/Pad	Inverted Tripod
Stability[1]	-21°	-31°
Weight[1]	28 lb	32.4 lb
Stowability	4 Struts Maximum Length 59 in.	12 Struts Maximum Length 56 in.
Operational Considerations	4 Connections to Main Frame	12 Connections to Main Frame (3 Per Leg)
Complexity	No Moving Parts - Simple Strut	Four Energy Absorbing Mechanisms

1 Maximum Vehicle Pitch Angle during Landing - Vehicle Topples at -38°, Landing Condition A

2 Does Not Include Four Main Structure Fittings for the Strut/Pad Gear or Eight Main Structure Fittings for the Inverted Tripod Gear.

TABLE 4.4

COMPARISON OF THREE-LEG AND FOUR-LEG SYSTEMS

Comparison Factor	Three-Leg	Four-Leg
Leg Length (Stowability)	71 in.	59 in.
Total Weight of Legs	15.6 lb	15.8 lb

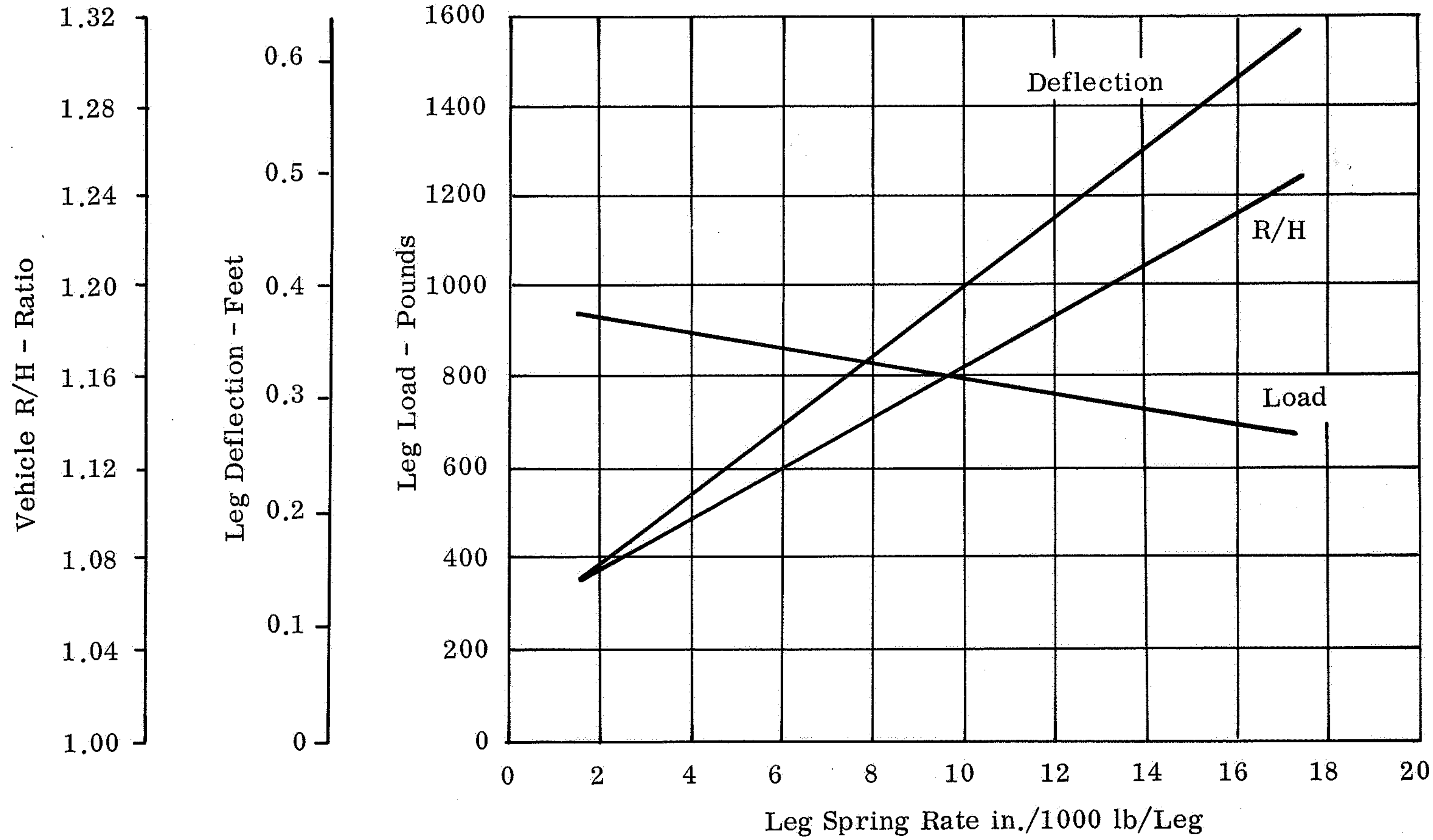


Figure 4.4. Vehicle R/H, Leg Deflection, and Leg Load versus Leg Spring Rate

footpad divided by the vertical distance from the center of gravity of the vehicle to the bottom of the footpad. The load and deflection curves shown are the maximum instantaneous vertical ground reaction on one leg and the resulting leg deflection at the tip relative to the vehicle body. A decrease in leg stiffness, i.e., an increase in spring rate, requires only a modest increase in leg length, and actually results in a decrease in leg load. However, the leg deflection rapidly becomes intolerable. It is concluded from this investigation that the leg spring rate should be in the 2 to 8 inch/1000 pound region.

4.6 EFFECT OF MATERIALS

Results of an investigation of possible strut materials are shown in Table 4.5. The data presented are for a leg radius of 62 inches and a limit leg load of 1000 pounds. The four legs are joined to each other at the center of the vehicle, and are attached to the vehicle only at the four corners of the vehicle at a radius of 17 inches. The weight shown is for four struts including the carry-through portions inboard of the corner support points. On the basis of this comparison it was concluded that titanium offered the best compromise of root diameter, weight, and spring rate. Carbon fiber composite is considered to be a possible future state-of-the-art improvement.

4.7 EFFECT OF STRUT TEMPERATURE

An investigation was made on the effect of strut temperature for a titanium strut with a 42.5 in. outboard length and a 7 in. inboard stub. This represents a strut which plugs into a body center section. Results are presented in Figure 4.5. The weight shown is for the four struts only with no weight included for the body center section. For a body with the corners at a 17 inch radius from the center the leg radius would be 59.5 inches. The data shown are for a design limit load of 1200 pounds with a factor on ultimate load of 1.5. The analysis includes the effect of temperature on the modulus of elasticity, and the effect of combined bending and torsion buckling at ultimate loading due to a combination of a horizontal force with 600 pound limit lateral and axial components acting simultaneously with the 1200 pound vertical force. It should be noted that the data in Figure 4.5 are based on strength, not stiffness. From a heat transfer study of the heating effect of the engine plume on the legs it was determined that even with a leg designed for 800°F a shield would be necessary for protection from the plume. The weight of shield or insulation required to bring the maximum leg temperature down from 800°F to 400°F is less than the weight increase to design the leg for 800° use.

Therefore, a design temperature of 400°F was selected for the leg, and a configuration with a root diameter of four inches and a wall thickness of 0.060 was selected for the final leg geometry.

TABLE 4.5

COMPARISON OF LEG MATERIALS

Material	E	Density	Thickness	Limit Stress	Root Diameter	Weight	Spring Rate
	$x10^{-6}$	lb/in^3	in.	psi	in	lb	in/1000 lb
Fiberglass	3.3	0.070	0.100	45,000	3.43	13.9	19.2
Steel	29.0	0.290	0.030	200,000	2.97	15.0	11.2
Titanium	16.5	0.165	0.040	96,700	3.70	14.2	7.7
Aluminum	10.5	0.100	0.050	52,000	4.51	13.1	5.3
Lockalloy	27.0	0.076	0.060	31,300	5.31	14.1	1.05
Carbon Fiber	15.0	0.050	0.080	40,000	4.07	9.5	3.1

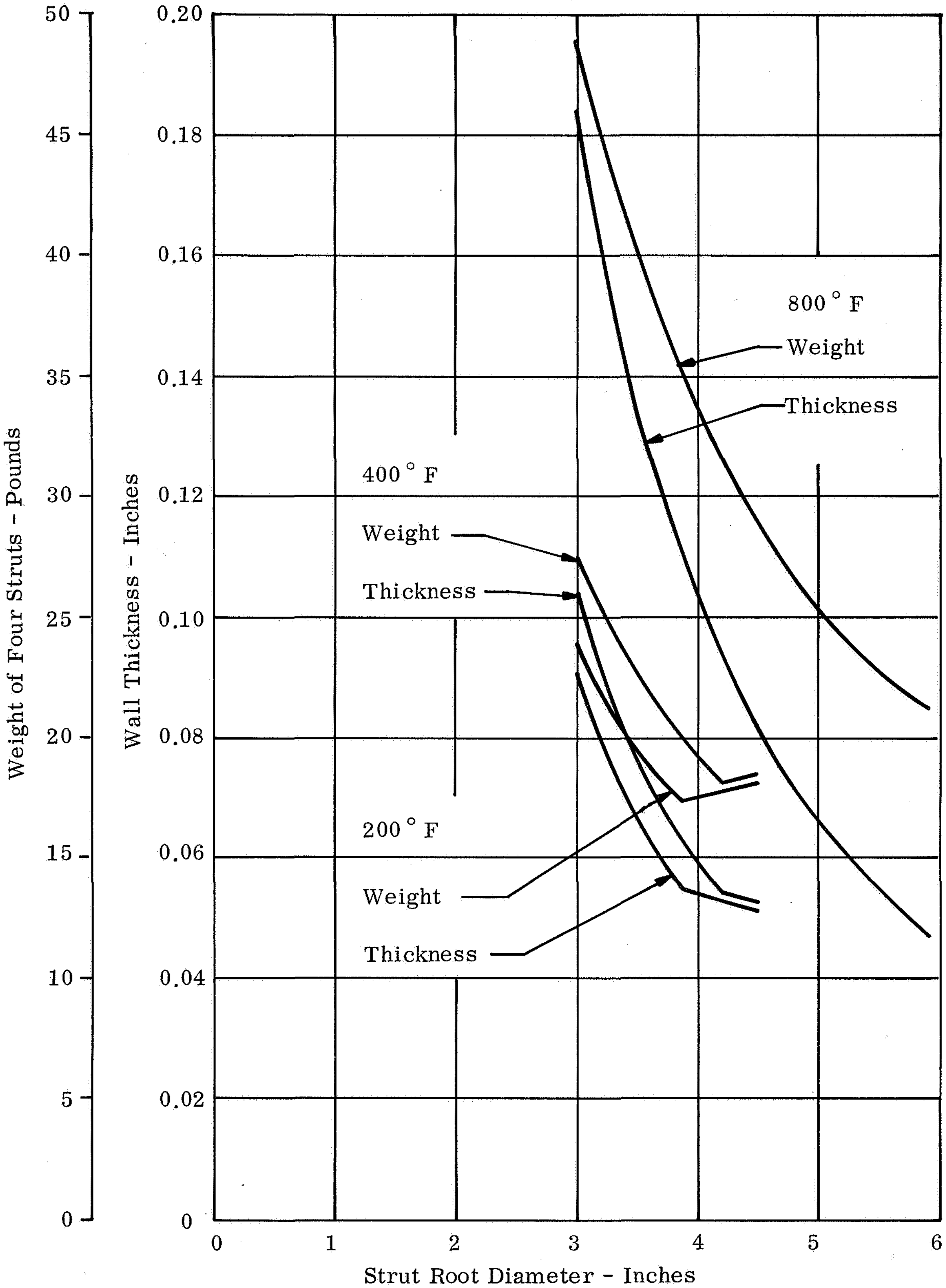


Figure 4.5. Titanium Strut 42.5 Inches Long with Taper to 1.5 Diameter at Tip - P_V = 1200 Pounds (Limit)

4.8 EFFECT OF FOOTPAD SIZE AND SOIL DENSITY

An investigation was made into the trade-off between lunar soil stiffness and footpad size. As explained in Appendix A the Bell soil model is based on the Bendix soil model (Ref 25) which was correlated with Surveyor landing data. A range of soil stiffness was determined from the Surveyor data. (Ref 38)

It was observed in this present study that changing the soil stiffness within the predicted range had a small effect on vehicle stability but a noticeable effect on loads. However it is possible to adjust the maximum ground reaction for any specific set of soil parameters by picking the appropriate footpad size, as shown in Figure 4.6. The stiff Surveyor soil has a density of 5 slugs per cu ft, a relative density of 1.0, and a friction angle of 45°. The soft soil has a density of 3.11 slugs per cu. ft, a relative density of 1.0, and a friction angle of 45°. Based on the stiff lunar soil a footpad of 3.75 inch radius was chosen for the final configuration. This gives a maximum limit ground reaction of 1200 pounds with a corresponding penetration of about 9 inches.

4.9 EFFECT OF CRUSHABLE FOOTPAD

Using the final vehicle configuration, an investigation was made of the capability of landing on a hard unyielding surface (rock) with energy absorption in the footpads. Safe landings can be made on rock at low touchdown velocities, but the energy absorption in the footpads will expand the operating envelope. Assuming an energy absorbing footpad with honeycomb material designed to crush at a constant load of 1200 pounds through a stroke of four inches, the final vehicle design was checked for the design conditions. Load conditions C through F resulted in strokes of 3.77, 1.60, 4.0, and 0.79 inches respectively. Stability conditions A and B, which are identical except for the initial velocities, could not be accomplished. At reduced velocities stability was achieved as shown in Figure 4.7. Stability is measured in terms of maximum tip up angle relative to the 10° ground slope. In the stable landing combinations plotted in Figure 4.7 no stroking occurred. The dashed line on Figure 4.7 is a reduced velocity envelope for a hard surface.

4.10 FINAL LANDING GEAR DESIGN

The landing gear leg design arrived at as a result of this study is a tapered curved tube,as shown in Figure 3.1,with a diameter of 4.0 inches at the root and 1.5 inches at the tip. A constant wall thickness of 0.060 inches is used, and the material is titanium alloy. Each leg is attached to a corner of a four-inch deep titanium honeycomb centersection with a fitting as shown in Figure 4.8. This design permits each leg to be folded upward for stowage during transport to the moon, and erected easily by folding downward until the locking pins snap into position. The stability capability with a 7.5 inch diameter footpad on stiff lunar soil is shown in Figure 4.9. Also shown is the reduced envelope capability for landing on a hard surface.

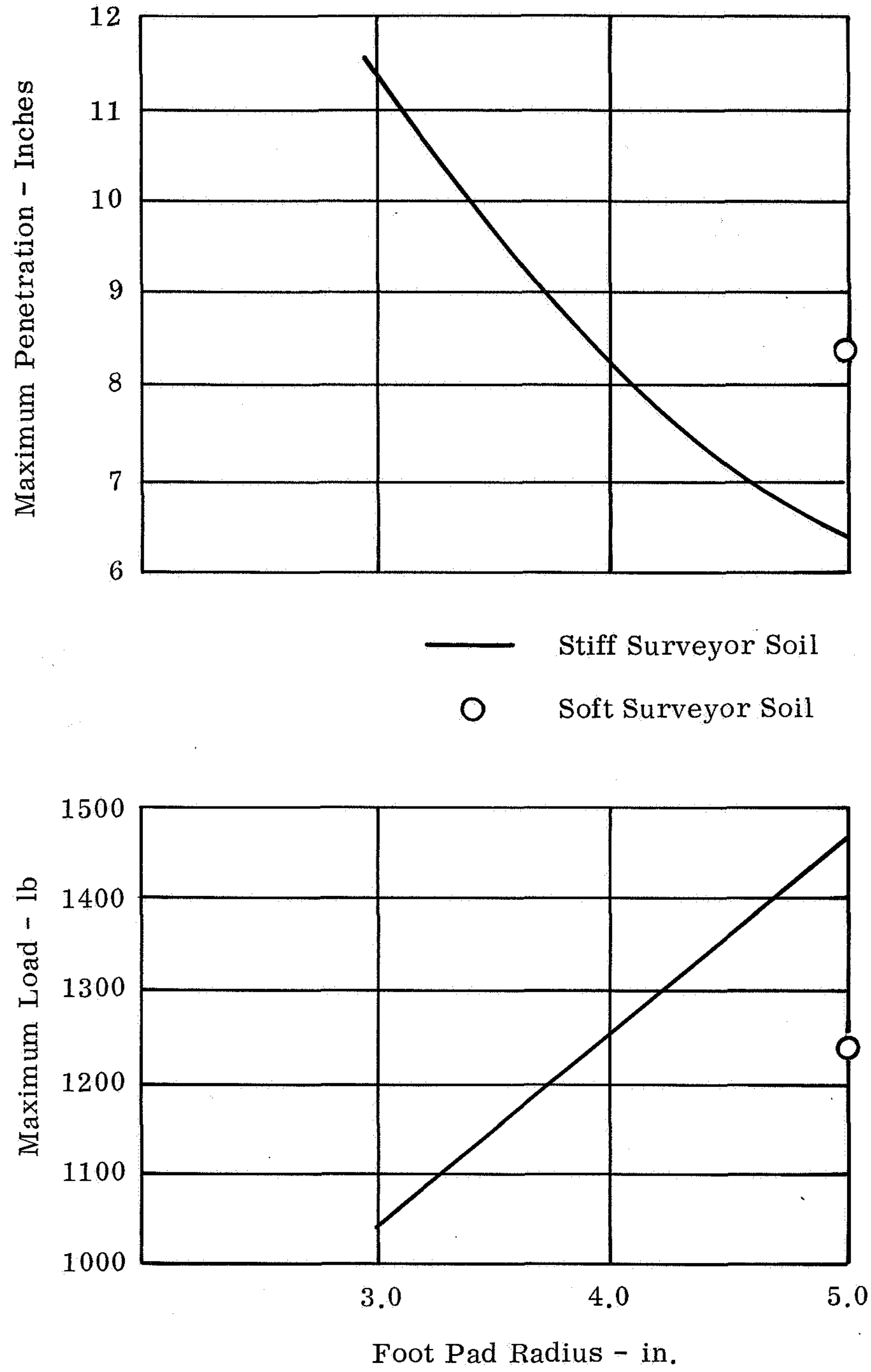
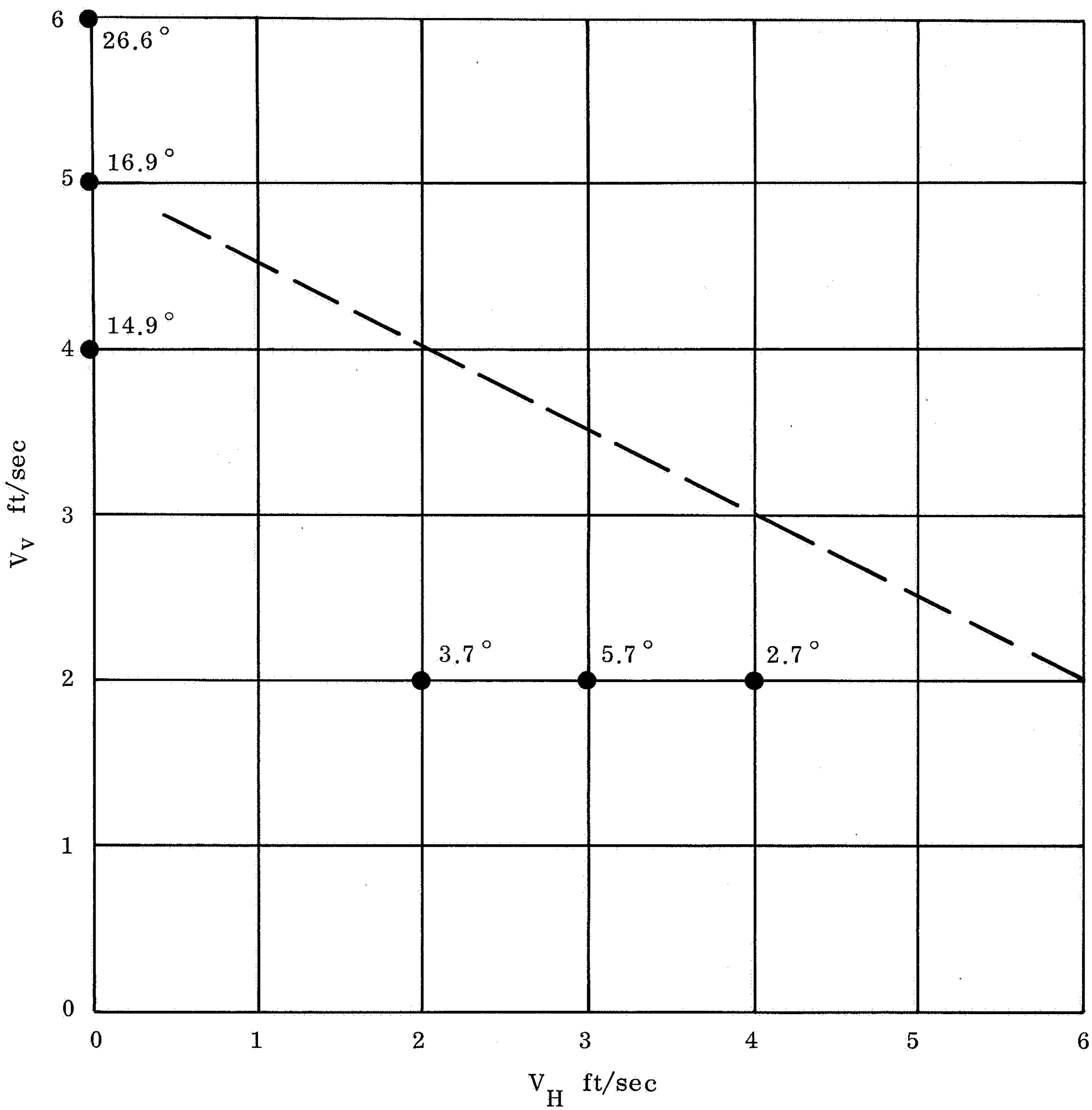


Figure 4.6. Soil/Footpad Radius Tradeoff



Degrees shown are maximum tip up relative to 10° slope.

Figure 4.7. Hard Surface Stability

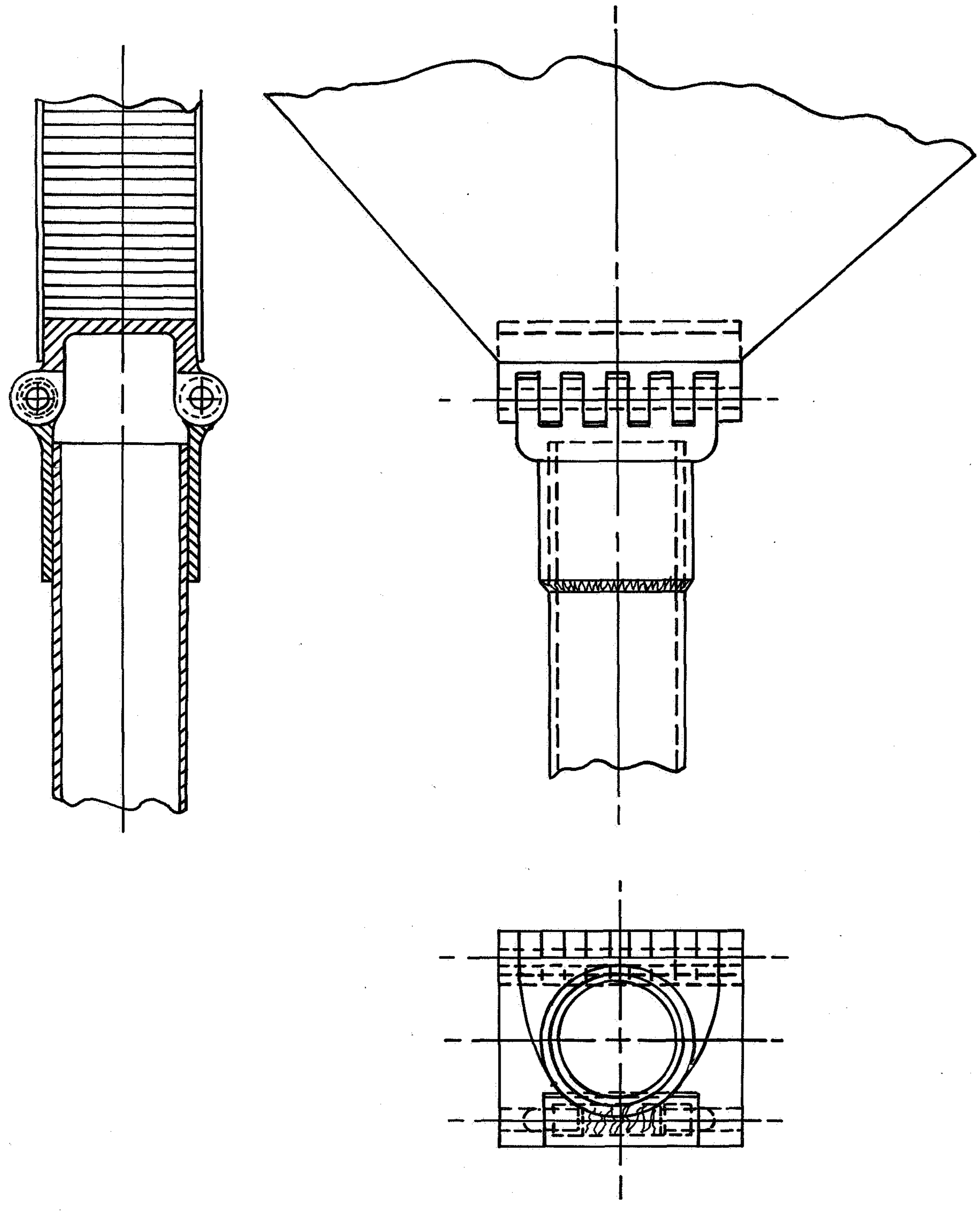


Figure 4.8. Landing Gear Leg Attachment

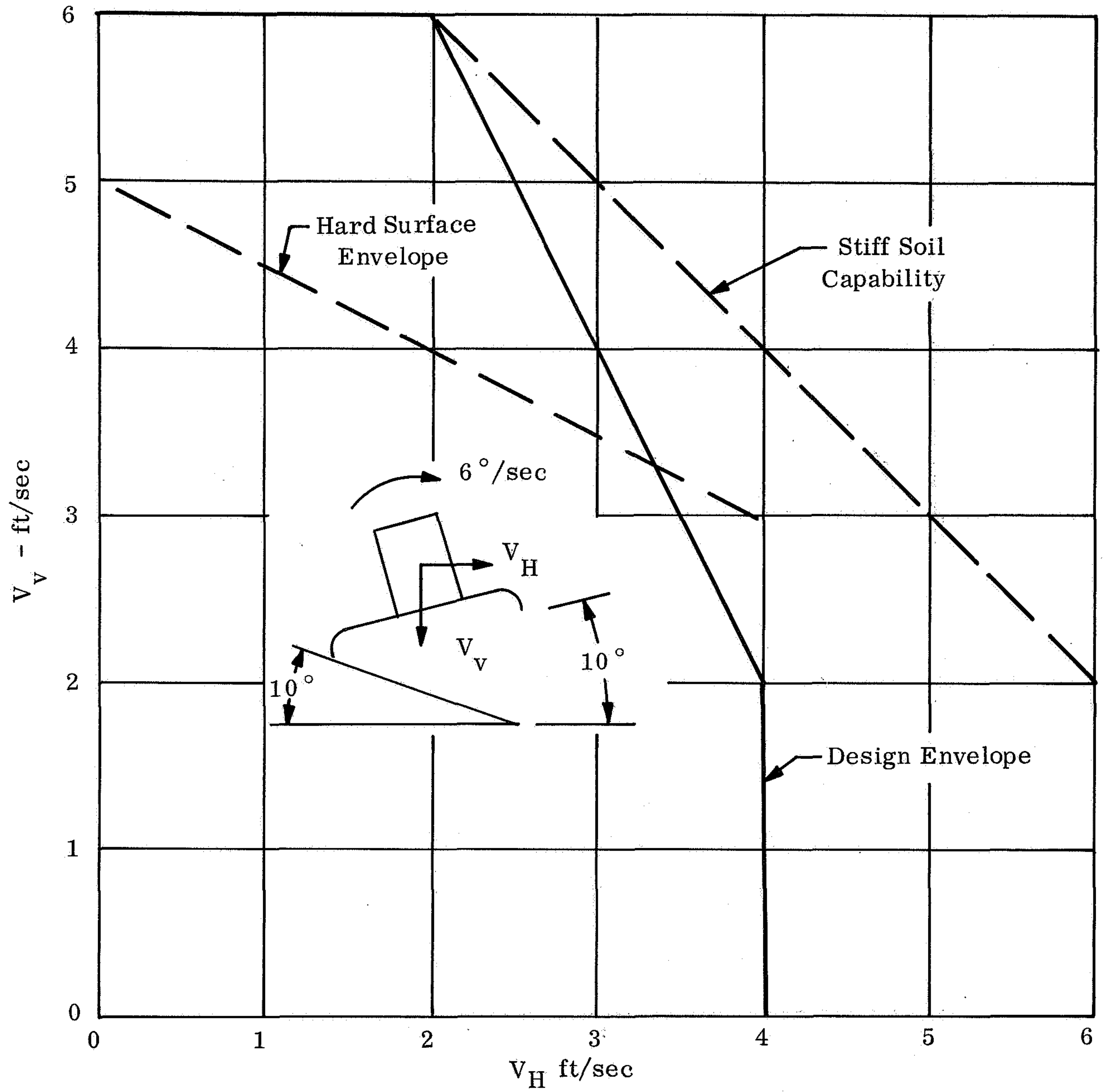


Figure 4.9. Final Design Gear Stability

5.0 PROPULSION ANALYSIS AND DESIGN

5.1 REQUIREMENTS

The propulsion studies were bounded by four primary requirements established by previous NASA and Bell studies of Lunar Flying Vehicles. The four primary requirements listed below are, in effect, a selection of major features of the OMLFV propulsion system.

1. Continuous thrust flight - obviates requirement for positive expulsion propellant acquisition.
2. LM descent propellants - $N_2O_4/.5N_2H_4$ + .5 UDMH.
3. Regulated pressure fed type propulsion system.
4. Current technology and developed components.

These primary requirements, together with the additional requirements generated by vehicle design and mission studies, result in the overall propulsion system requirements for the One Man Lunar Flying Vehicle given in Table 5.1.

TABLE 5.1

OMLFV PROPULSION SYSTEM REQUIREMENTS

Type	Regulated Pressure Fed
Thrust	50-300 lb (continuous)
Throttle Ratio	6:1
Propellants	N_2O_4 - 0.5 N_2H_4 + 0.5 UDMH
Propellant Quantity	300 lb
Mixture Ratio	1.3 to 2.0 (design range)
TVC Angle	$\pm 12^\circ$ Maximum
Mission Firings	60 to 150 (one firing per flight)
Engine Life	3.5 Hours (total)
Technology	Current operational or demonstrated
Operational Date	Mid-1972

Vehicle thrust and throttle ratio requirements were established during the MFS study (Ref. 2). Figure 5.1, extracted from Reference 2 , shows the effect of the maximum and minimum thrust to weight ratio upon ΔV required for a five mile flight. It can be seen that the knee of the curve on T/W_{min} is between 0.4 and 0.6. The best T/W_{max} is approximately 1.5 but the curve is very flat from 1.4 to 2.0. Table 5.2

TABLE 5.2

VEHICLE THRUST REQUIREMENTS

Vehicle Thrust			Vehicle Load Condition		
T_{MAX}	T_{MIN}	Throttle Ratio	Highest Weight (Payload 370 lb), earth lunar 1347* 222*	Typical Weight (Payload 100 lb) earth lunar 1077 178	Minimum Weight (No Payload - (Empty) earth lunar 671 112
250	50	5	T/W = 1.13 Lunar	1.41	0.45
300	50	6	1.35	1.69	0.45
400	50	8	1.8	2.25	0.45

*Weights include 25% growth factor on estimated vehicle dry weight.

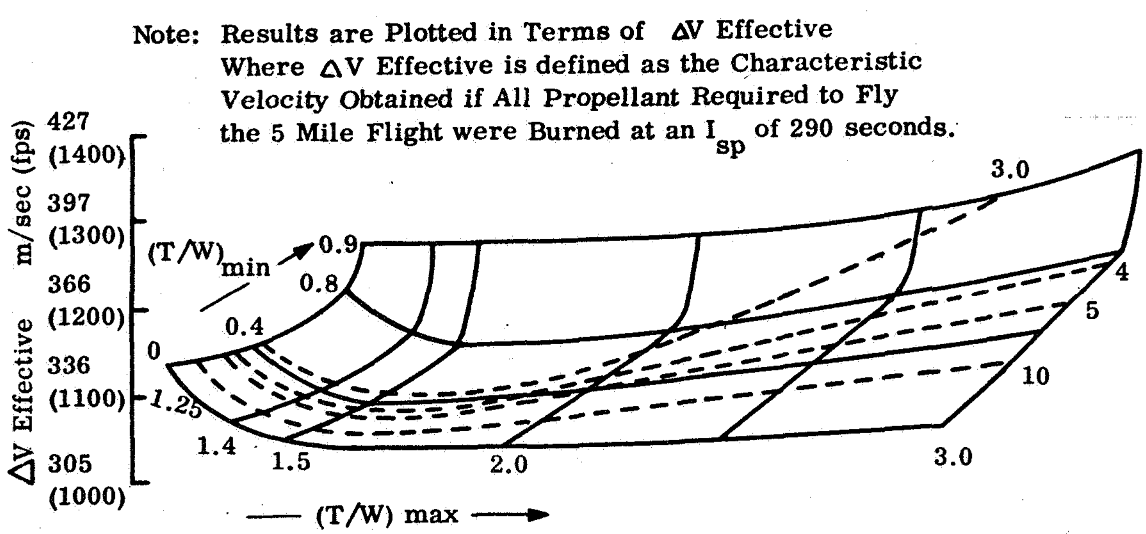


Figure 5.1. Summary of Simulation Test Results for 5 Mile Flights Accounting for I_{sp} Degradation

shows the thrust to weight ratio conditions which occur as a function of vehicle loading condition for various levels of T_{max} and a fixed T_{min} of 50 pounds. This T_{min} provides an acceptable thrust to weight ratio of 0.45 as an upper limit to the minimum thrust to weight ratio which would occur during normal operations. For all other vehicle loading conditions the minimum thrust to weight ratio would be lower than the values shown. On the basis of this comparison a thrust T_{max} of 300 and T_{min} of 50 with a throttle ratio of six to one is optimum for the one man lunar flying vehicle. A T_{max} of 250 is pounds marginally low for the maximum payload case.

The great diversity of possible mission duty cycles (MDC) for the OMLFV precludes design for a typical MDC. The limits of the possible mission duty cycles are shown in Figure 5.2. However, the thrust level sequence characteristics of all MDC's is relatively common because the major portion of the thrusting for all missions is for ascent, acceleration, deceleration and descent. This is illustrated in a characteristic MDC given in Table 5.3.

TABLE 5.3

CHARACTERISTIC OMLFV MISSION DUTY CYCLE

Maneuver	Burn Time (Seconds)	Throttle Setting (% Thrust)
Ascent and Acceleration	60-80	77-100
Accelerate to Cruise Velocity	20-40	43-62
Cruise	0-40	20-30
Deceleration and Descent	70-80	57-90
Hover and Touchdown	10-40	39-57

Although the MDC of Table 5-3 cannot be employed as a design point for propulsion system optimization purposes, the characteristic thrust level sequencing makes it a useful guideline for the propulsion system studies.

5.2 PARAMETRIC OPTIMIZATION STUDIES

Several operational considerations served to characterize the propulsion elements for a propulsion system minimum weight optimization study. The most directly affected elements were the propellant tanks and engines.

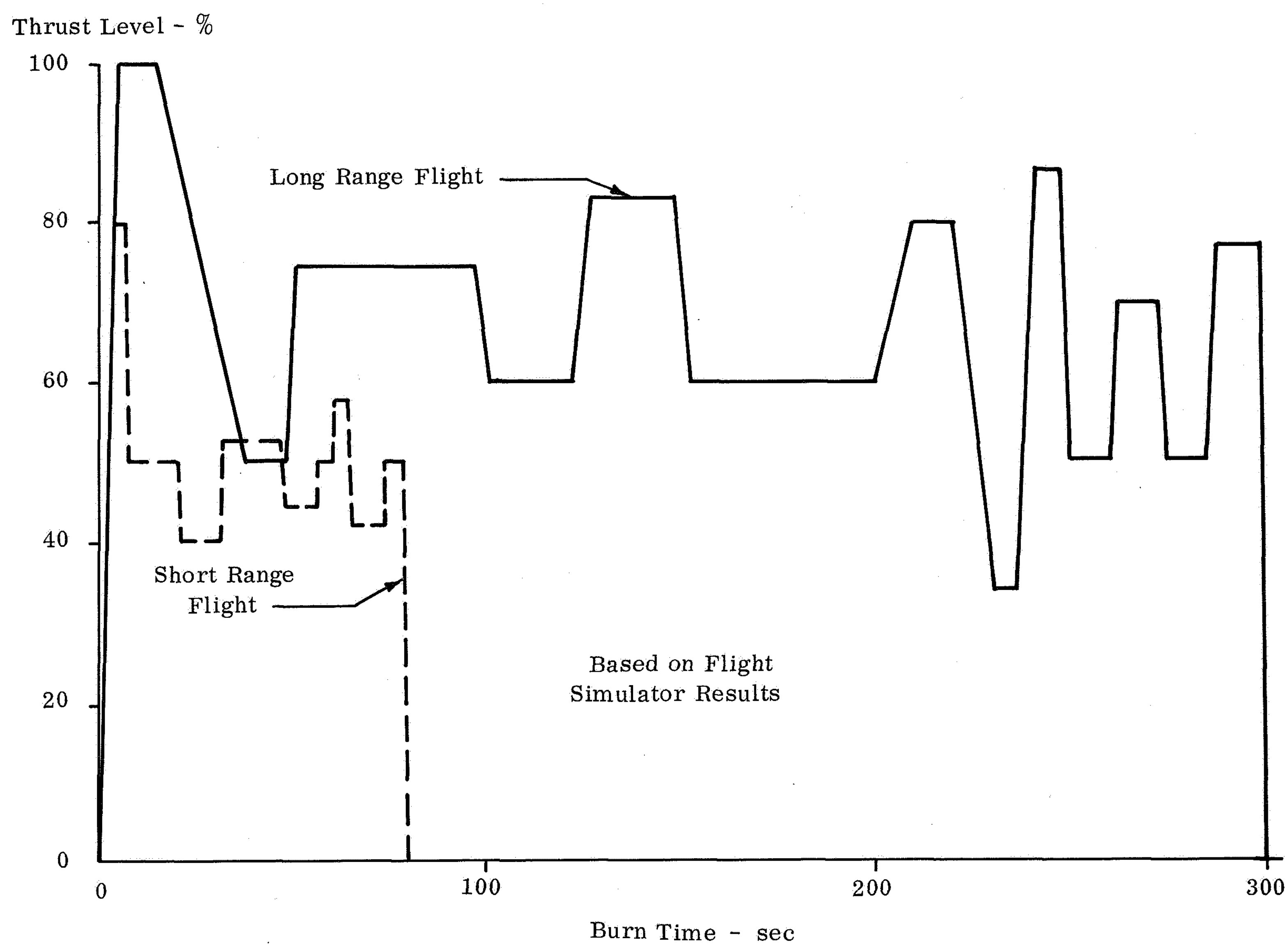


Figure 5.2. Mission Duty Cycles

5.2.1 Optimum Engine Type

The integration of propellant consumption, at all thrust levels, for the characteristic MDC of Table 5-3, with typical throttling engine performance indicates that less than 10% of the propellant is consumed at throttle settings below 40% of rated thrust. This can also be seen in the limit duty cycles of Figure 5.2. It is therefore concluded that high engine performance (I_{sp}) at thrust levels below 40% thrust is not significant and has no merit if achieved by engine reliability compromise. For this reason the throttling design features of the OMLFV engine should be optimized for maximum reliability at the expense (if necessary) of performance at low thrust settings.

The requirement for throttling also affects the engine cooling design to the extent that it severely compromizes regeneratively cooled engines for this application. Regenerative cooling is feasible at the OMLFV thrust levels and throttle ratios (see Table 5.1), but only at mixture ratios that would severely degrade engine performance throughout the throttle range. This deficiency is compounded by cooling jacket pressure drop requirements which would increase the propulsion system weight. The use of regeneratively cooled engines would therefore result in a low performance, heavy weight propulsion system.

Ablative engine cooling cannot simultaneously satisfy the requirements of 3.5 hours life in 60 to 150 firings with current technology (see Table 5.1). Furthermore, since ablative engine weight is proportional to life, relaxation of the life requirement (or periodic engine replacement) would not be competitive with refractory metal radiation cooling (e.g. Cb or Mo thrust chambers).

Only radiation cooling fully satisfies all the requirements of Table 5.1 without compromizing the propulsion system. It is therefore concluded that the optimum engine cooling approach for the OMLFV is the refractory metal radiation cooled engine.

5.2.2 Optimum Tank Type

The utilization of a continuous thrusting flight plan for the OMLFV (with no ballistic trajectory) is a major consideration in the selection of propellant tank type. Continuous thrusting during the flight produces an equilibrium propellant level normal to the thrust vector. The resulting fixed equilibrium orientation of the free propellant surface with respect to the vehicle provides continuous propellant aquisition by the simple device of locating the outlet at the "bottom" of the tank. Equilibrium propellant acquisition can therefore be provided by thrust forces to permit use of simple shell tanks that are optimum for minimum weight and maximum reliability.

Positive expulsion devices such as bladders, bellows or diaphragms increase weight and reduce reliability. However these devices can prevent momentary loss of propellant acquisition during sloshing at low propellant levels (due to maneuvers) but offer no suppression to slosh inertial effects on the OMLFV.

At low tank levels in a free surface propellant tank, momentary loss of propellant acquisition (unporting) can be induced by sloshing resulting from maneuvers. In addition, vortex formation can compromise acquisition of the last available propellant. For these reasons, a slosh baffle and a vortex baffle will be required to obtain high operational expulsion efficiency in a simple shell tank. The slosh baffles are also desireable in the OMLFV tanks in order to reduce the adverse inertial effects of the liquid, in partly filled tanks, on vehicle attitude control during flight and on vehicle stability during landing.

It was therefore concluded that the optimum propellant tank type for the OMLFV is a simple shell tank with slosh and vortex baffles. The minimum weight construction material that satisfies the requirement of current technology and compatibility with the propellants is a Titanium alloy (6Al -4V).

The optimum gas tank configuration for the OMLFV is a titanium alloy sphere for reasons of minimum weight and current technology.

5.2.3 System Optimization Procedure

These optimum engine and tank types were combined with propulsion system control components based on current technology to formulate nine candidate LFV pressure fed propulsion systems for the purpose of optimization tradeoff study.

Since vehicle configuration materially affects propulsion system design, two basic configurations were considered:

I. A twin engined system shown in Figure 5-3 with single axis gimbaling to give yaw and pitch control and differential throttling for roll control.

II. A single engine system shown in Figure 5-4 having two axis gimbaling for pitch and roll control with auxiliary thrusters for yaw control.

The effects of the following propulsion system design variations were evaluated:

1. Propellant Tank Configuration - Two or Four, Spherical or Cylindrical
2. Thrust Chamber Injector Design - Fixed or variable geometry injector (the type of throttleable injector employed materially affects the system pressure levels).
3. Manual or electrical propellant and throttle valve actuation.
4. Engine out capability with redundant thrust chambers and valves.

From these two configurations and four design variations, the matrix of nine propulsion system designs shown in Table 5-4 was selected for optimization study.

For the purposes of optimization study a number of design assumptions were made as follows:

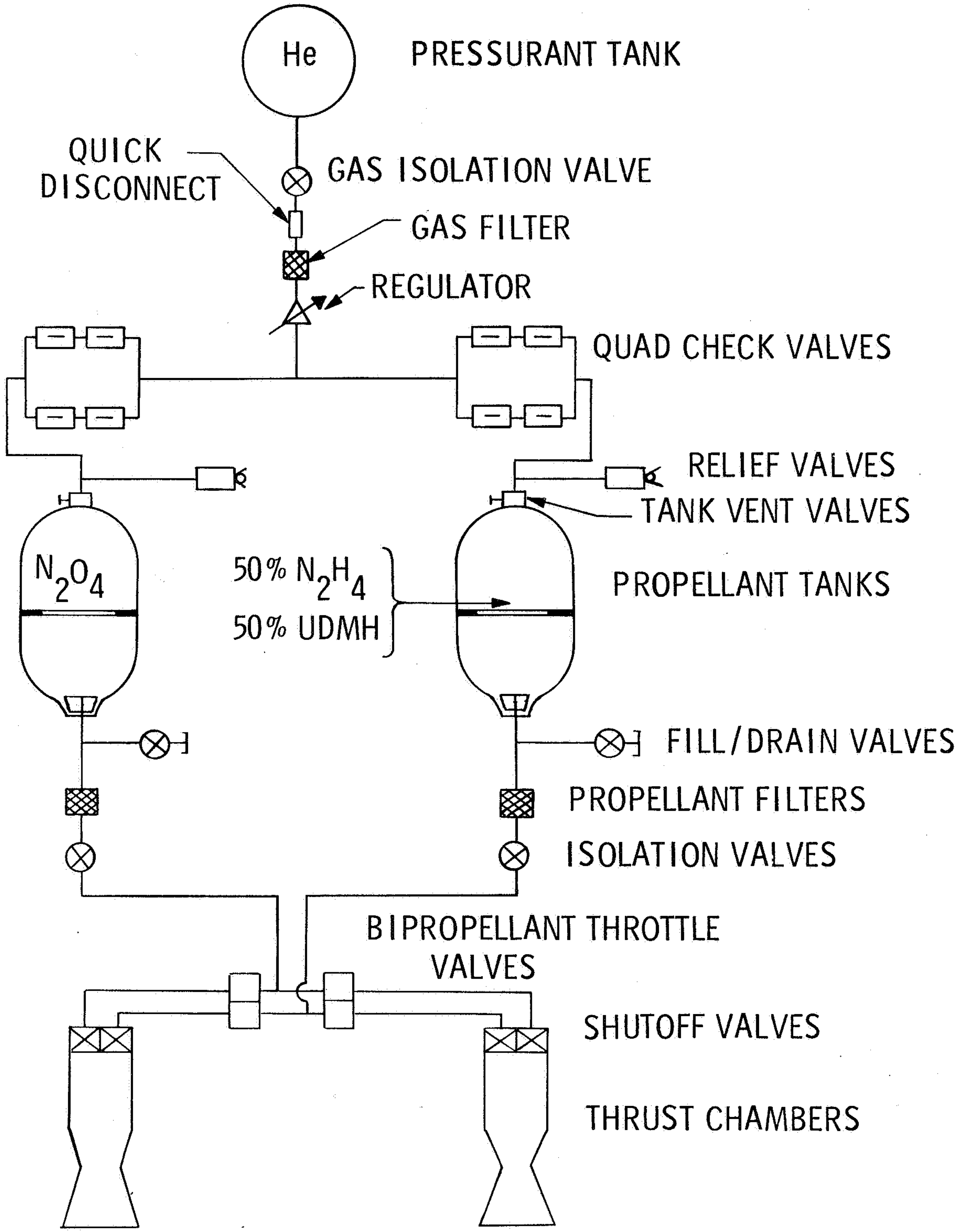


Figure 5.3. Configuration I Propulsion System Schematic - Two Outboard Engines

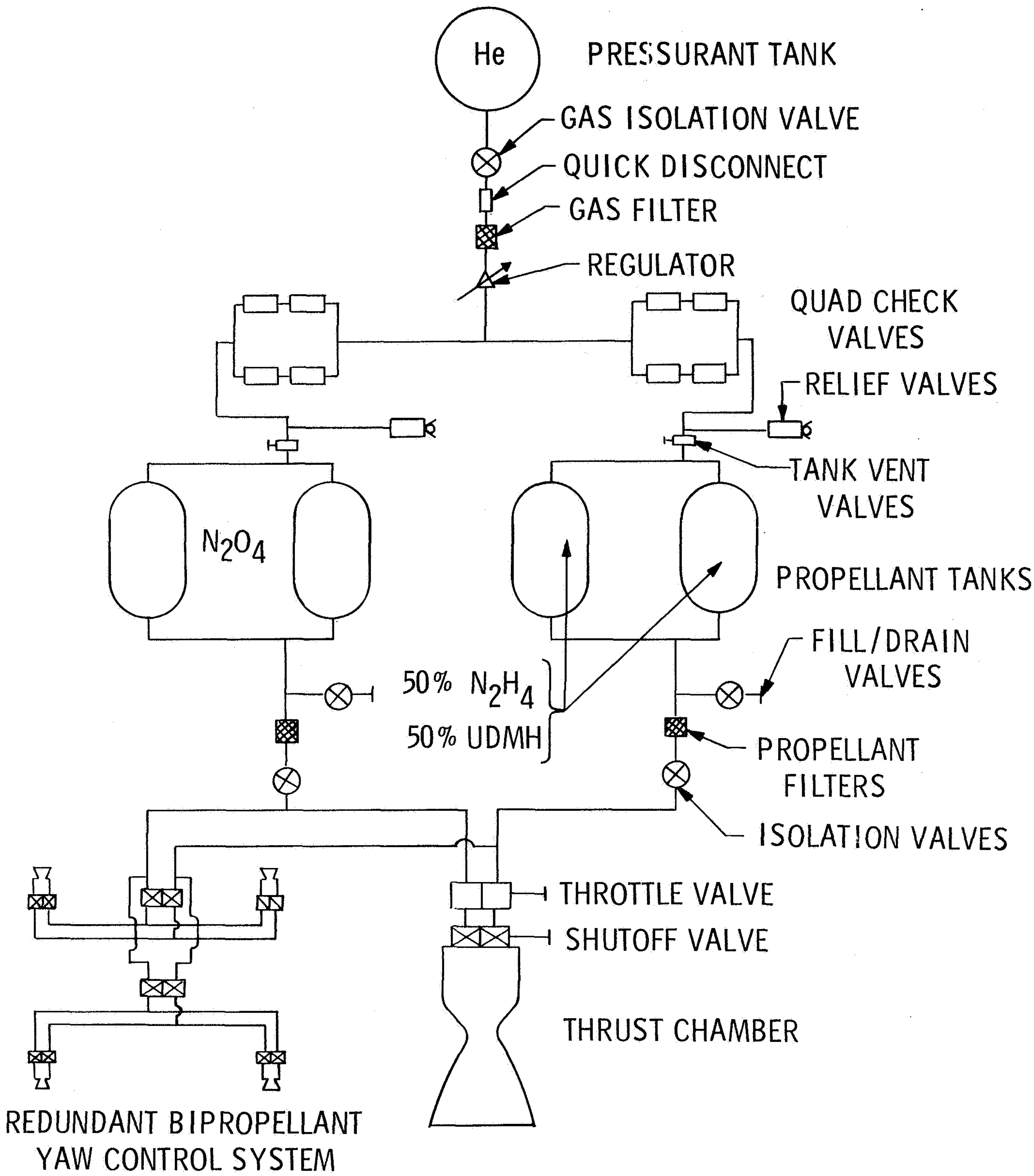


Figure 5.4. Configuration II Propulsion System Schematic - Single Central Engine

TABLE 5.4

OPTIMIZATION STUDY PROPULSION SYSTEMS

Config. No.	Design No.	Engines		Throttle and Prop. Valve Actuation	Throttling Injector Geometry	Engine Gimbal Axes	No. of Prop. Tanks	Propellant Tank Shapes	Weight** lbs
		No.	Rated Thrust-lb						
I	1	2	150	Mechanical	Fixed	Single	2	2.8 L/D Cylindrical	51.6
	2	2	150	Mechanical	Variable	Single	2	2.8 L/D Cylindrical	49.6
II	3	1	300	Electrical	Variable	Dual	4	Spherical	68.1
	4	1	300	Electrical	Fixed	Dual	4	Spherical	70.7
	5	1	300	Electrical	Variable	Dual	4	1.6 L/D Cylindrical	69.3
	6	1	300	Mechanical	Fixed	Dual	4	1.6 L/D Cylindrical	68.7
I*	7	4	112.5	Mechanical	Fixed	Single	2	2.8 L/D Cylindrical	79.8
II*	8	2	225	Electrical	Fixed	Dual	4	Spherical	87.7
	9	2	225	Mechanical	Fixed	Dual	4	1.6 L/D Cylindrical	83.7

*Engine Out Capability

**Comparative Dry Weight for 3450 fpsΔV Capability

1. Pressurant Gas

This was assumed to be helium with maximum pressure of 4,500 psia and a minimum pressure of 1.7 times engine feed pressure. In determining quantity required, compressibility was considered and a polytropic expansion coefficient of 1.2 was assumed.

2. Gas Storage Bottles

6Al-4V titanium alloy was assumed with an overall safety factor of 2.8 giving a bottle weight of 0.0182 lb/cu. in. volume for the required working pressure of 4,500 psia.

3. Propellant Tanks and Mixture Ratio

Material 6Al-4V titanium of 130,000 lb/sq. in. yield strength using a safety factor of 1.33 on yield strength, plus a scratch allowance or 0.005 inch on skin thickness. A minimum fabrication thickness of 0.017 inch was assumed.

Tank volumes were based on a total propellant load of 306 pounds at a mixture ratio of 1.6 and for densities at 120°F (to allow for propellant expansion on temperature excursions to this value during fully loaded storage of the OMLFV in the ready condition). The optimization study mixture ratio of 1.6 was selected for maximum engine performance to identify the maximum OMLFV range or ΔV. However, dry weight optimization is not sensitive to mixture ratio in the range of interest (1.3 to 2.0) and the selected 1.6 is simply a mid-range value. The optimum system design points and comparison between systems are essentially the same throughout this mixture ratio range. For stressing purposes the tank working pressure was taken to be 50 psi above nominal engine feed pressure (the engine feed pressure is dependent on injector design and rated chamber pressure).

4. Engines

The weights of the engine components were curved through several point designs in the design range of interest. Throttle and propellant valve weights are dependent upon thrust level and mode of operation (electrical or mechanical), thrust chamber weight is a function of thrust, chamber pressure and nozzle area ratio and gimbal mount weight is a function of thrust, chamber pressure and number of gimbal axes.

5. Other Propulsion System Components

It was assumed that the weight of the remaining system components, gas valves, lines, propellant lines, fill valves, etc., would be constant and 16.6 pounds was allowed for these items for the twin engine system and 27.1 pounds for the single engine system. In the case of the single engine system, 10 pounds is included for the yaw control system comprising four 5 pound thrustors, six bipropellant valves and associated lines.

An additional 5 pounds is also added on to the weight of the twin engine system with redundant engines for the failure detection/correction system.

6. Vehicle Weight

To establish the vehicle velocity increment the following vehicle weight data was assumed.

Structure Weight	100 pounds
Payload	150 pounds
Astronaut	370 pounds
Residual Propellant	6 pounds
Usable Propellant	300 pounds

7. Engine Performance

For this analysis the variation of specific impulse with chamber pressure and nozzle area ratio was based on extrapolated theoretical Bray data factored by an overall efficiency of 90%. Subsequent one dimensional kinetic performance calculations factored by 92.2% for combustion and nozzle losses give reasonable agreement with the data used. A performance variation with thrust level caused by injector scale effects was also taken into account.

Engine performance variation with throttle setting is given in Figure 5.5 for one variable injection geometry engine and two fixed geometry engines operating at the maximum performance mixture ratio of 1.6. This data was reported from fire tests by three different engine vendors.

This engine throttling data shows little difference in injector type performance (e.g. 1% average) at thrust levels above 50% where MDC evaluation indicates that most of the propellant will be consumed. An average nominal curve for throttling performance was therefore assumed.

5.2.4 Optimization Study Results

The measures of merit for the OMLFV propulsion system are minimum dry weight (to be delivered to the moon) and maximum range (with the assumed payload). Since range increases with inertial ΔV capability, optimization tradeoff data was generated by calculating the propulsion system dry weight and vehicle ΔV capability for simultaneous values of engine design chamber pressure and nozzle expansion ratio. This optimization data for propulsion system configuration I design No. 1 is given in Figure 5.6.

This figure shows that the minimum weight chamber pressure is approximately 80 psia and the OMLFV performance (ΔV) trade with dry weight that can be obtained by varying the nozzle expansion ratio. The maximum ΔV that can be obtained at favorable rates of change with weight is approximately 3,480 fps on Figure 5.6,

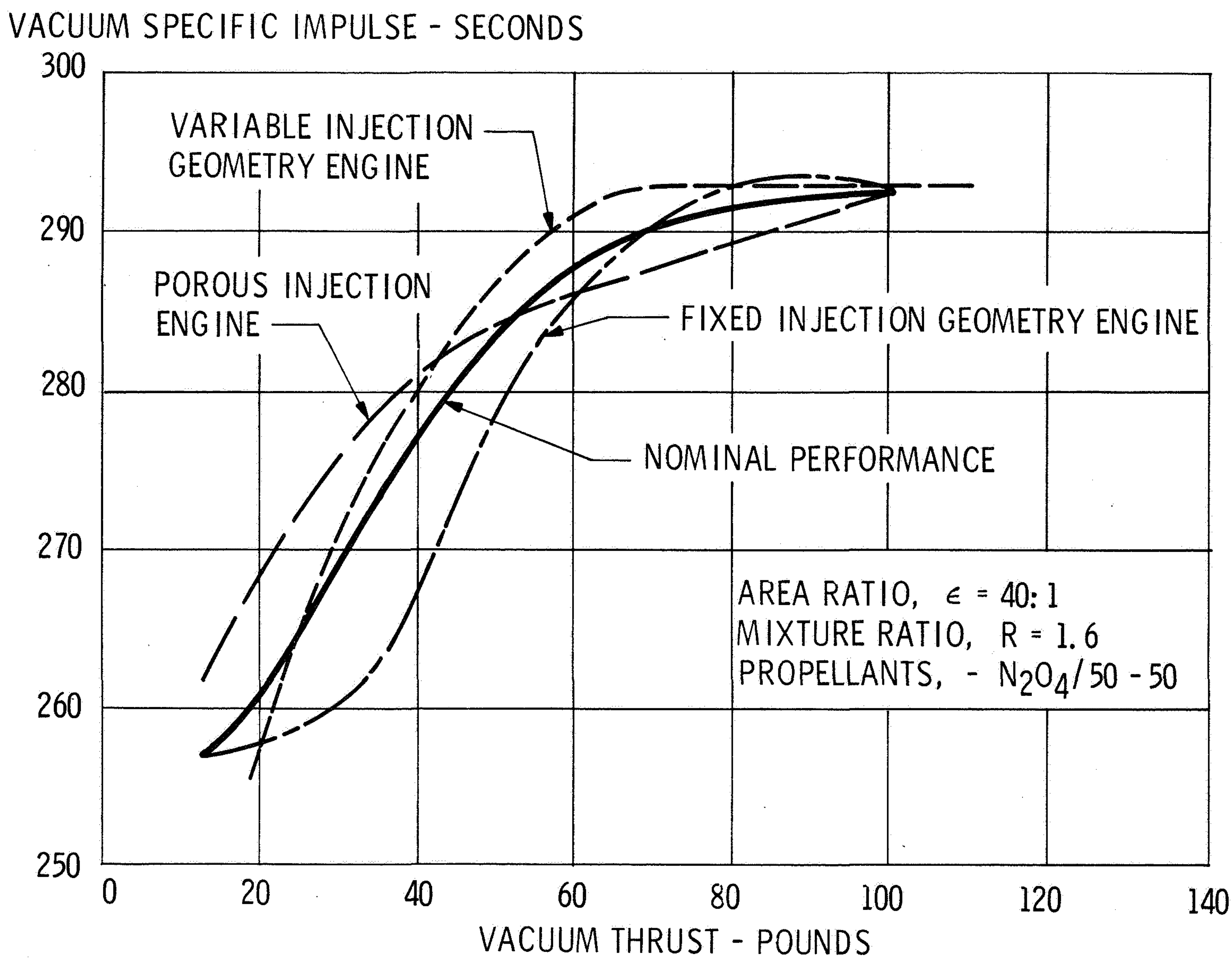


Figure 5.5. Throttleable Engine Performance Experience

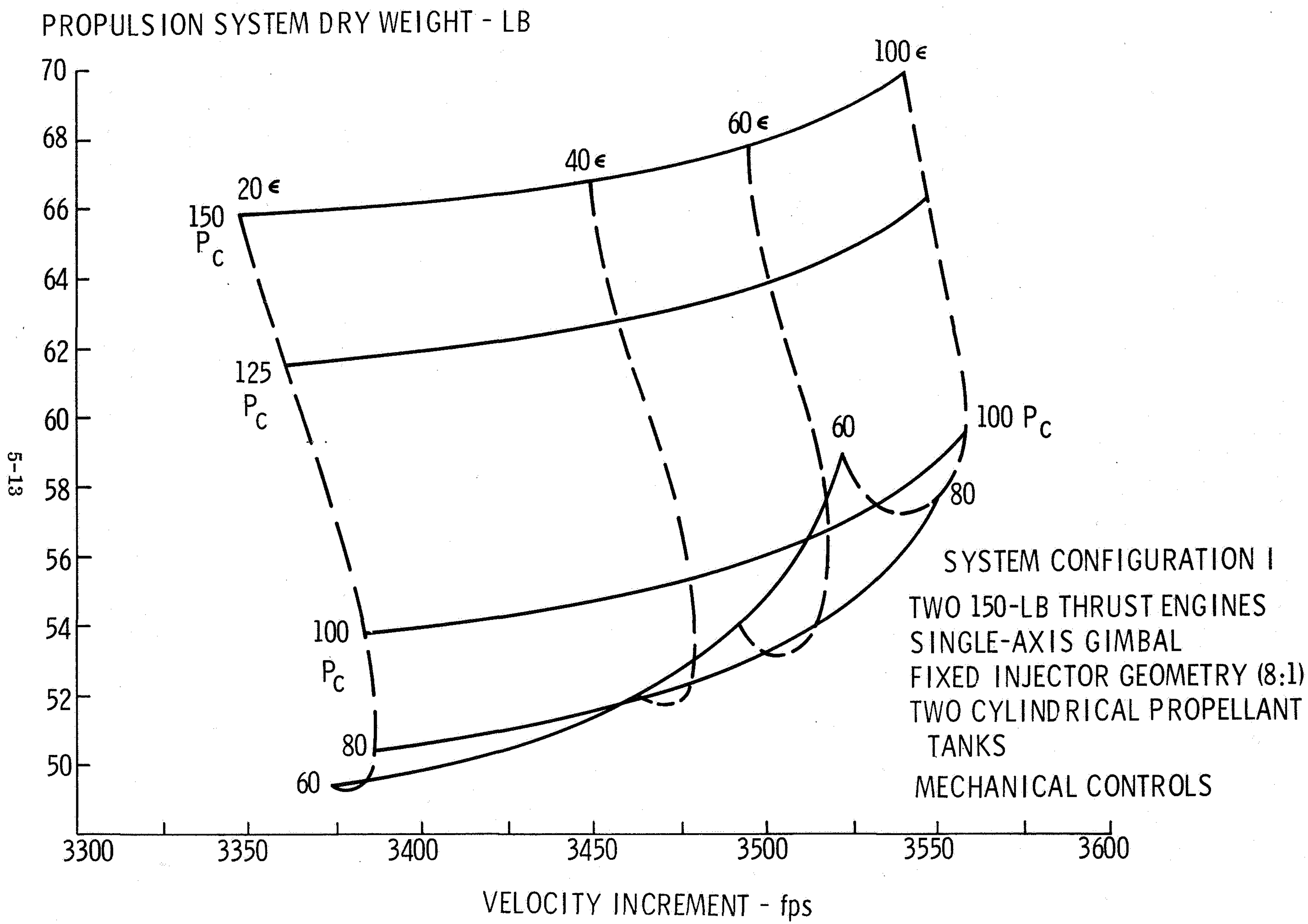


Figure 5.6. Propulsion System Optimization

indicating that the optimum nozzle area ratio is 40:1. Similar results for configuration I system design No. 6 are shown in Figure 5.7.

The inertial ΔV for this tradeoff data was calculated for engine operation at full thrust specific impulse. Since ΔV is directly proportional to I_{sp}, this data may be adjusted to the effective I_{sp} of any MDC by reference to Figure 5.5 and correcting the abscissa of Figures 5.6 and 5.7 by the ratio of the effective MDC specific impulse to the full thrust I_{sp}. However, this will not affect either the character or the magnitude of the tradeoff data for systems No. 1 and 6 or the direct comparison with simular tradeoff data for the seven other systems.

These nine propulsion system designs are compared in dry weight on the common denominator of equal ΔV or range capability in Table 5.4. Company design numbers 1 with 2 and 3 with 4 indicates a weight saving of on the order of two pounds with variable injection geometry. On this basis it was concluded that the reliability compromise (attendant to the increased complexity) of variable geometry injection was not justified by this small weight reduction and that conversely, fixed injection geometry is optimum for the OMLFV propulsion system.

Comparison of design numbers 3 and 5 indicates that cylindrical tank weight penalties are low enough to leave this selection to the option of the vehicle designer. Manually operated mechanical valves are lighter and more reliable than electrical valve actuation but this selection must also be made at the vehicle level.

The configuration I designs (1 and 2) are on the order of 20 pounds lighter than configuration II designs (3 through 6) as shown in Table 5.4. The table also shows that redundant engine versions of configurations I and II (designs No. 7, 8 and 9) involve unattractive weight penalties of 16 to 28 pounds.

Therefore, since adequate engine reliability can be provided by design simplicity and demonstrated by thorough engine development, it is concluded that the optimum propulsion system for the OMLFV is design No. 1. This is the optimum propulsion system for the two principal measures of merit; namely minimum dry weight delivered to the moon and maximum OMLFV range capability as shown in Table 5.4 and Figure 5.6.

The definative features of this optimum propulsion system are summarized as follows:

Two Engines - Providing all thrust and attitude control forces required.
Simple Basic Propellant Supply System - Per Figure 5.3
Radiation Cooled Engines - Designed for 80 P_c and 40 ϵ per Figure 5.6.
Fixed Injector Geometry - Reliability outweighs throttling efficiency
Spherical Gas Tank - Titanium Alloy (6Al-4V)
Simple Ti Shell Propellant Tanks - Slosh and vortex baffles
Options - Manual versus electrical valves and spherical versus cylindrical propellant tanks

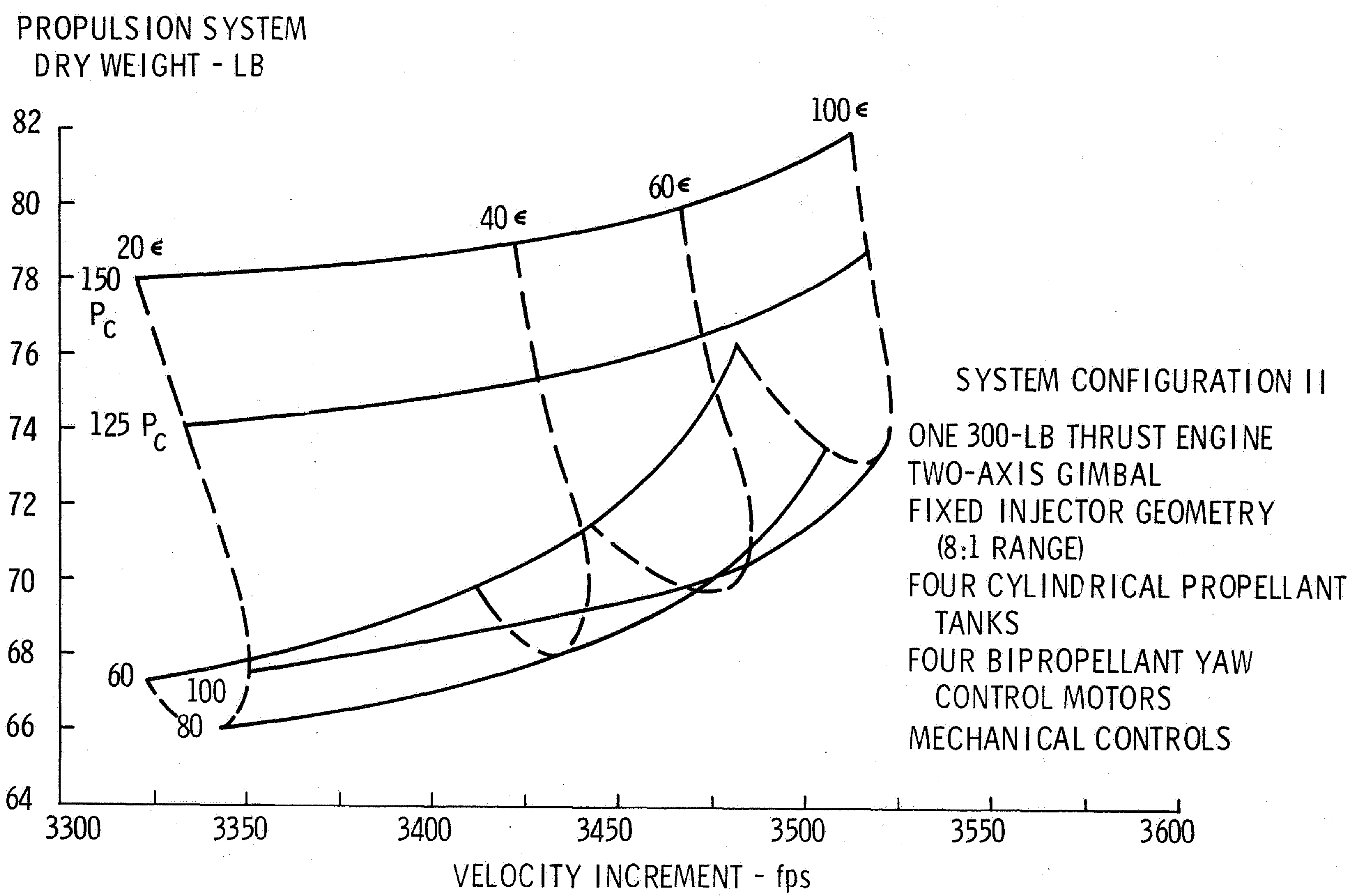


Figure 5.7. Propulsion System Optimization

5.3 CONCEPTUAL DESIGN

The results of the parametric studies serve the purpose of identifying the optimum system to establish a standard for conceptual design of the OMLFV propulsion system. However, the OMLFV configuration requirements and propulsion system operational considerations such as flight safety and reliability must first be satisfied.

5.3.1 Configuration

The non-optimum configuration II designs were therefore reviewed for any compensating advantages in the configuration and operational areas. The four tank arrangement of configuration II offers somewhat denser packaging of the propellant supply subsystem but the central location of the engine under the OMLFV is a significant disadvantage. To provide adequate ground clearance for the engine without large landing gear penalties, the size of the engine must be minimized. This can be done by reducing the nozzle expansion ratio (ε) to 20 and increasing the design chamber pressure (P_c) to 125 psia. However, Figure 5.7 shows that this non-optimum P_c and ε design point involves a weight penalty of 6 pounds and a ΔV penalty of 100 fps.

Therefore at design points selected for OMLFV installation, the most promising configuration I and II propulsion system designs compare as shown in Table 5.5.

TABLE 5.5

OMLFV PROPULSION SYSTEMS COMPARISON

Measure of Merit	Weight (lb)	ΔV (fps)	Throttle Ratio
Configuration I, Des. No. 1	52.4	3478	6/1
Red. Des. No. 7	71.6	3374	9/1
Configuration II, Des. No. 6	74.1	3334	6/1
Red. Des. No. 9	81.8	3283	9/1

This table shows that both the basic configuration I design (No. 1) and the redundant engine configuration I design (No. 7) are lighter in weight and provide higher performance than either varient of configuration II. The configuration I design was therefore selected.

Table 5.5 also shows that, in addition to weight and performance penalties, the redundant engine design requires a greater throttle ratio to provide true engine-out capability. This will impact engine development risk, time and cost. It follows that final selection between configuration I design numbers 1 and 7 must be based on comparable flight safety and reliability.

5.3.2 Flight Safety and Reliability

The nature of the Lunar Flying Vehicle mission places a premium on engine reliability. Engine performance and system weight may be traded off against other measures of merit. Reliability however, is not negotiable. The OMLFV engines must provide lift, acceleration, attitude control, safe deceleration and soft landing for a manned vehicle.

The redundancy approach to high reliability is feasible, for this mission, but this can be compromised by the resulting requirement to sense and diagnose failures and initiate corrective action before critical OMLFV attitudes or flight paths are reached.

The best approach to high reliability is intrinsic engine design reliability. High intrinsic reliability is achieved by minimizing the number of components that must operate successfully and by providing large operating margins on the design capabilities of these components. In other words design simplicity and design margin.

Design simplicity can be maximized for this application by selecting inherently simple engine design features such as manually operated propellant and throttle valves, fixed injection geometry and radiation cooling. The valves and injector can be over designed to provide large operating margins on design capability. However, the flight safety and reliability of the radiation cooled thrust chamber is dependent on thermal margin.

The radiation cooled engine maximum operating temperature experience of Bell Aerosystems is given in Figure 5.8 as a function of engine performance (c*). This family characteristic data is based on six different engine designs operating on N_2O_4 - MMH and N_2O_4 - 0.5 N_2H_4/0.5 UDMH propellants. The observed spread in maximum operating temperature at any given performance level is attributed to design and operating variations in mixture ratio, chamber pressure, engine geometry (e.g. L*) and the extent of barrier or film cooling. Published data from other engine vendors (TRW and Marquardt) also appears to fall within the envelope shown in Figure 5.8.

The limit operating temperature of refractory metal (i.e., C_b and M_o) radiation cooled thrust chambers is defined by the 3,100°F capability of the silicide coatings required for oxidation protection. A 600°F margin on this capability has proven satisfactory for man rated applications. This is substantiated by Bell experience with the fabrication and test of over 500 radiation cooled engines (C_b). No failure was ever observed during operation at or below 2,500°F.

Increasing this man-rated temperature margin by 50% to 900°F will, in the experience and judgement of Bell, provide equal safety, longer life and lower probability of engine failure, than will redundant thrust chambers. Bell engine operations at and near 2,200°F indicated unlimited life to the extent tested (perhaps due in part to the fact that this is below the 2400°F melting temperature of any Columbium

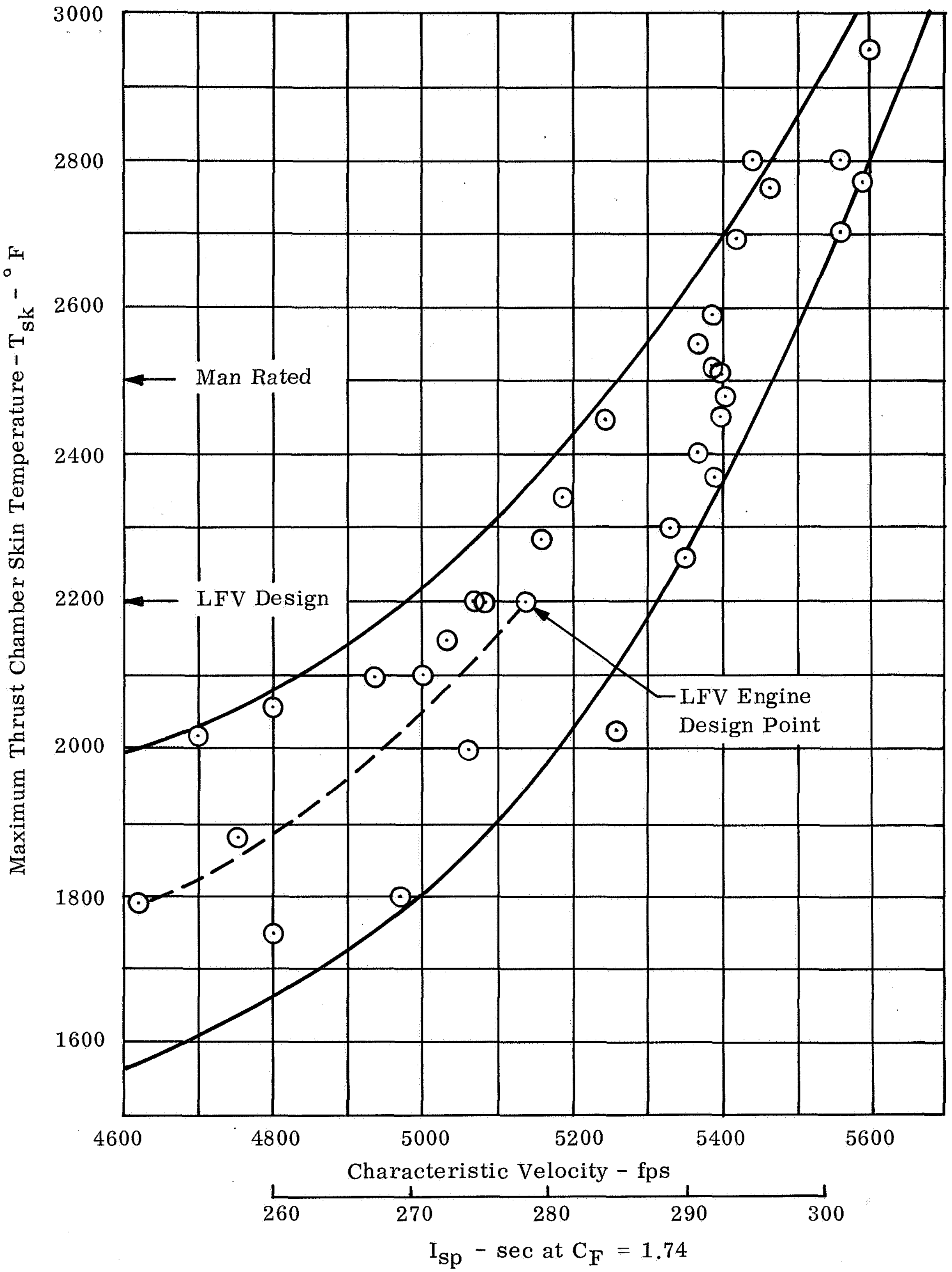


Figure 5.8. Engine Operating Temperature versus Performance

oxide that may form). Published data for TRW and Marquardt Engines appears to confirm this indication. However, as shown in Figure 5.8, this will necessarily involve reduced or derated engine performance. This figure also shows the trend (dotted line) of operating temperature as performance degrades with thrust during throttled operation of the engine.

The characteristic reflection of performance in operating temperature during throttling operation of the Bell engine is shown in Figure 5-9 for a design operating temperature of 2,200°F. This data was used to generate a typical engine time-temperature life history - for the life and mission duty cycle requirements of Table 5.1, using the short and long range mission duty cycles given in Figure 5.2. The results are plotted in Figure 5.10 for comparison with demonstrated engine life data from 3 engine vendors and silicide coating life data.

The silicide coating curve defines the potential life of these radiation cooled engines as a function of operating temperature. The line of engine data points defines the portion of this potential life that has been demonstrated by engine test without failure. The figure shows that the time-temperature life required of a OMLFV engine designed for 2,200°F maximum operating temperature lies well within currently demonstrated technology.

Figure 5.10 also indicates that design for maximum operating temperature at the currently man-rated level of 2,500°F (e.g. for maximum performance with redundant engines) would demand time-temperature life right up to the limit demonstrated. It follows that design for a maximum operating temperature of 2,200°F also provides an attractive life margin.

It is therefore concluded that the 900°F thermal margin provided by design for a maximum operating temperature of 2,200°F (throat section at maximum thrust) provides equal safety, longer life, and lower probability of engine failure than thrust chamber redundancy.

Low engine operating temperatures can be achieved by several means including reduced chamber pressure, mixture ratio or combustion efficiency and by barrier or film cooling on the combustion chamber wall. The propulsion system optimized at the low rated chamber pressure of 80 psia to provide most of the benefits of this effect without penalty. Reduced combustion efficiency is unattractive because it maximizes the performance loss attendant to the operating temperature reduction. Barrier or film cooling can be very effective without excessive engine performance derating as shown in Figure 5.11. However, the most reliable and straight forward method of operating temperature reduction is by mixture ratio reduction.

5.3.3 Mixture Ratio

The calculated change in maximum engine operating temperature with mixture ratio is shown in Figure 5.12 for the existing Bell 100 pound throttleable engine. This curve is indexed to the measured current operating temperature of this engine

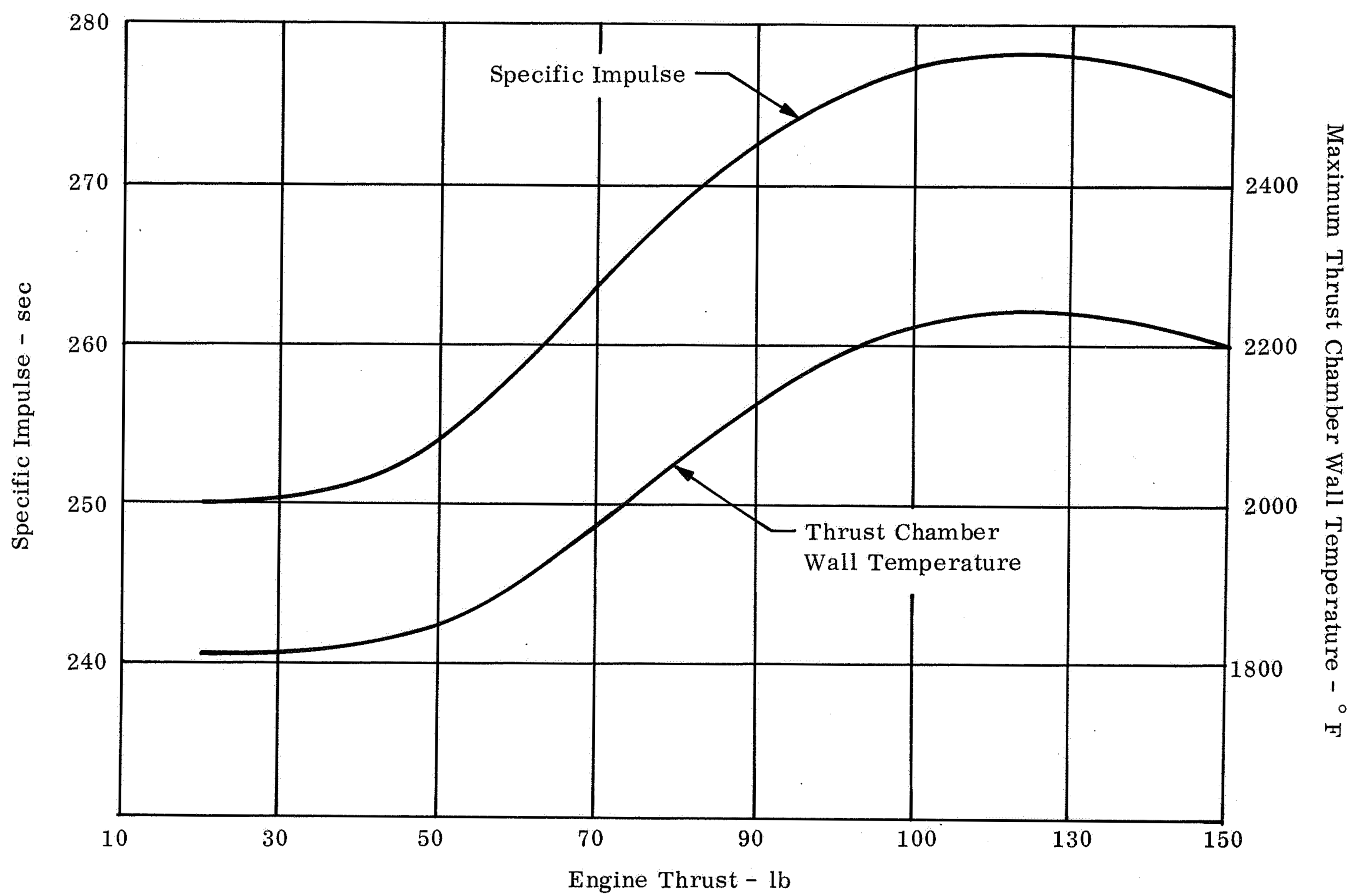


Figure 5.9. 150 lb Thrust Engine Performance and Maximum Wall Temperature

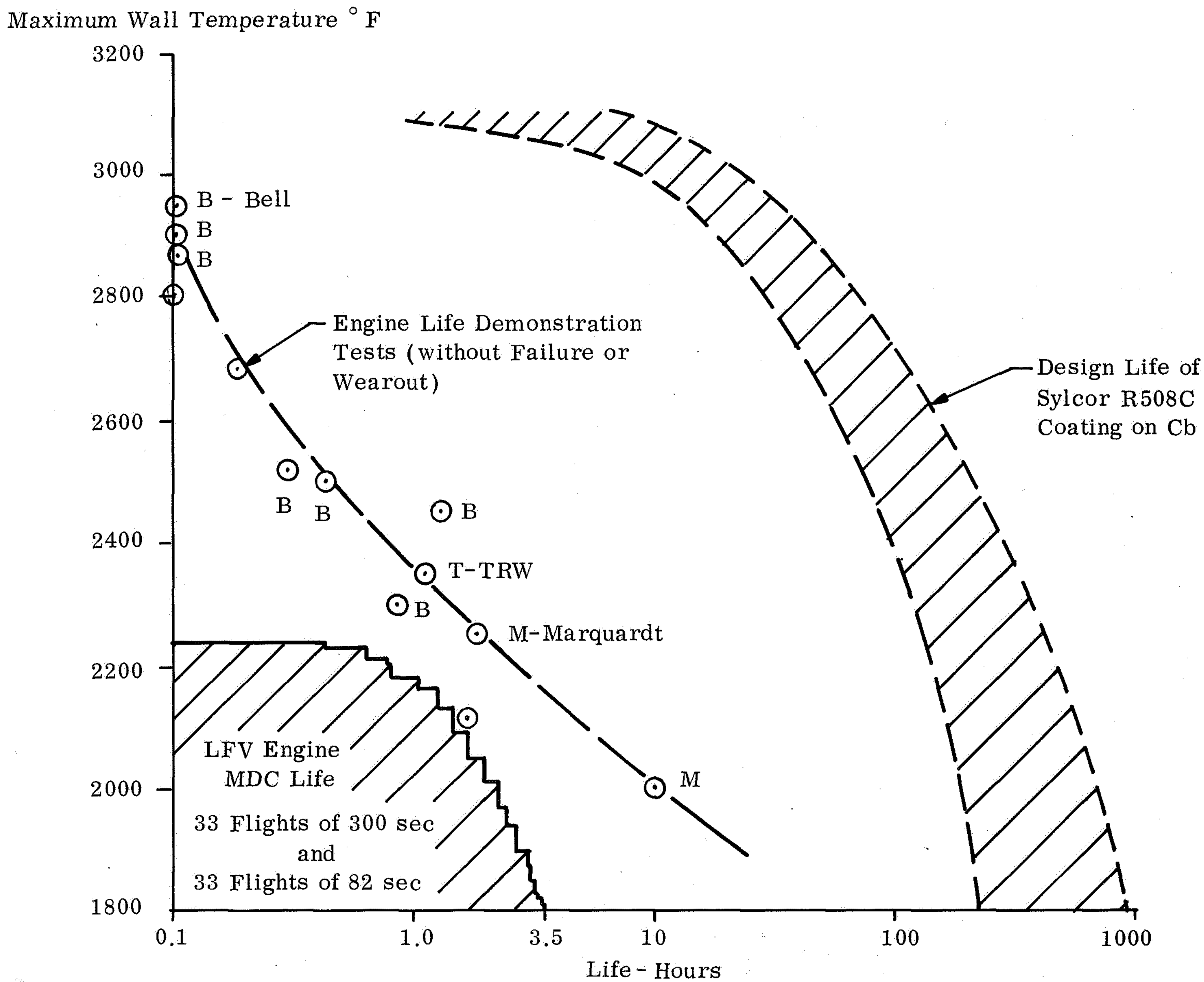


Figure 5.10. Engine Life versus Operating Temperature

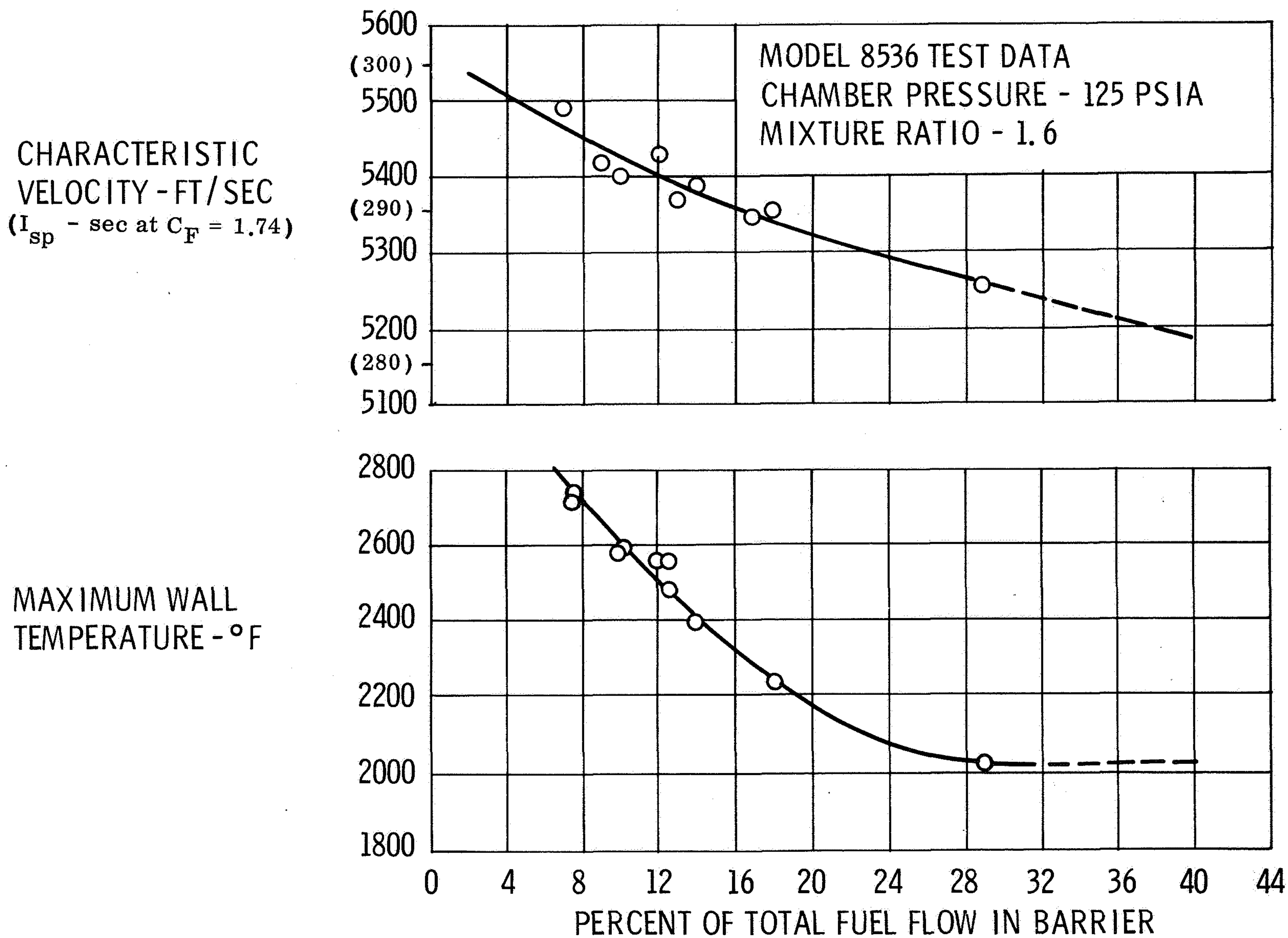


Figure 5.11. Fuel Film Cooling - 300 lb Engine

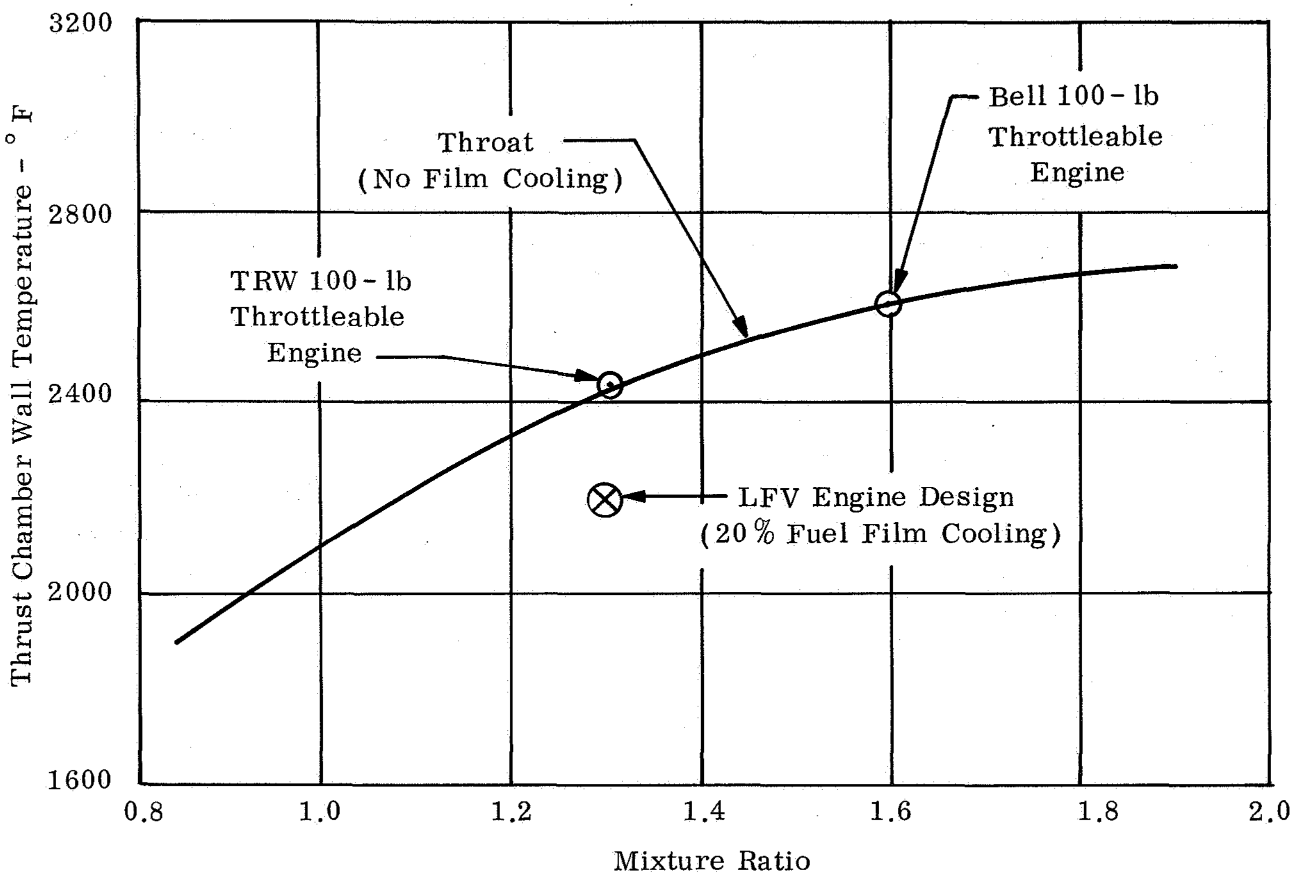


Figure 5.12. Steady-State Wall Temperature versus Mixture Ratio

at a mixture ratio of 1.6 where performance is maximum and confirmed at a mixture ratio of 1.31 by test data from the TRW 100 pound throttling engine. This figure indicates that the maximum operating temperature can be reduced to 2420°F by reducing the operating mixture ratio to 1.3. The corresponding theoretical performance loss is only four seconds of specific impulse. The figure also shows that the 2,200°F temperature objective can be obtained by operation at a mixture ratio of 1.1. However, this involves a larger performance penalty (~10 sec I_{sp}) and does not permit use of all of the availabile propellants from the LM descent stage.

The design combination shown in Figure 5.12 appears to be the most promising design approach to a maximum operating temperature of 2,200°F. In this case the engine would be designed for an overall mixture ratio of 1.3 with a film cooling circuit to assure attainment of this temperature. Existing film cooled engine test data indicates that the maximum film cooling that will be required is 20%. The predicted performance and operating temperature of this engine design is given in Figure 5.9.

The selection of an operating mixture ratio of 1.3 also provides the significant advantage of maximizing the propellants available from the residuals of the LM descent stage. Grumman (GAEC) studies of the utilization of LM for advanced applications defined the available propellant residuals for five methods of propellant transfer as shown in Table 5.6. This is average data for a 0 to 15° tilt angle after a landing that consumed the navigation and trajectory error reserves. The residual mixture ratio is 1.43 for the GAEC recommended transfer method.

TABLE 5.6

AVAILABLE PROPELLANT IN
LM DESCENT TANKS AFTER LANDING (GAEC*)

Propellant Transfer Method	W_P - lb	R
Feed Line Tap	545.3	1.44
Pressure Port Tap (GAEC Recommended)	558.7	1.43
Feed and Balance Line Tap	554.9	1.43
Feed Line and Pressure Port Tap	567.7	1.44
Feed and Balance Line and Pressure Port Tap	600.4	1.46

*Reference: "Apollo applications program study - Utilization of LM for advanced applications, final report on tasks terminated during November 1967", Grumman Aircraft Engineering Corporation, Report No. ARP325-3, 15 December 1967.

This GAEC data was generally confirmed and expanded by recent studies conducted by TRW in support of the Bell OMLFV study. The TRW analysis was based on the landing engines for LM-6 through LM-10 and considered the limit cases of 6688 fps mission ΔV for direct flight to touchdown (no lovering) and 7180 fps mission ΔV for touchdown after maximum hover. The results of this study are shown in Table 5.7.

TABLE 5.7

RESIDUAL PROPELLANTS AVAILABLE FOR USE BY THE LUNAR FLYER (TRW)

LM Descent ΔV (fps)	Propellants (lb)	Mixture Ratio
6688	1442.49	1.4841
7180	596.55	1.3278

Since the impact of LM residual mixture ratio on OMLFV operation is maximized when the LM residuals are minimum (i.e. at 7180 fps ΔV) the optimum mixture ratio for propellant utilization is 1.33 as shown in Table 5.7.

An additional advantage of a 1.3 mixture ratio lies in OMLFV tank sizing. The oxidizer and fuel tanks sized for a mixture ratio of 1.3 will provide a mixture ratio of 2.0 if the tank functions are reversed. A mixture ratio of 2.0 is recommended by Marquardt for low temperature operation of the LM RCS engines. Thus, the same tanks can be used for any one of four available engine designs.

It was therefore concluded that the optimum mixture ratio is 1.3 from the combined standpoints of propellant availability and engine life, flight safety and reliability (a conclusion shared, in both respects by TRW).

5.3.4 Final Selection

This discussion, to this point, has shown that derated engines designed for a mixture ratio of 1.3 to operate at a maximum temperature of 2200°F are comparable in flight safety and reliability to redundant engines operating at a mixture ratio and temperature of 1.6 and 2500°F respectively to develop maximum performance.

On this common denominator of equivalent safety and reliability, a tradeoff analysis was conducted to compare the redundant and derated engine designs in weight and range capability. The results are shown in Figure 5.13. The same techniques and assumptions as the optimization study were employed here, except that the work was limited to 80 psia chamber pressure because that was previously

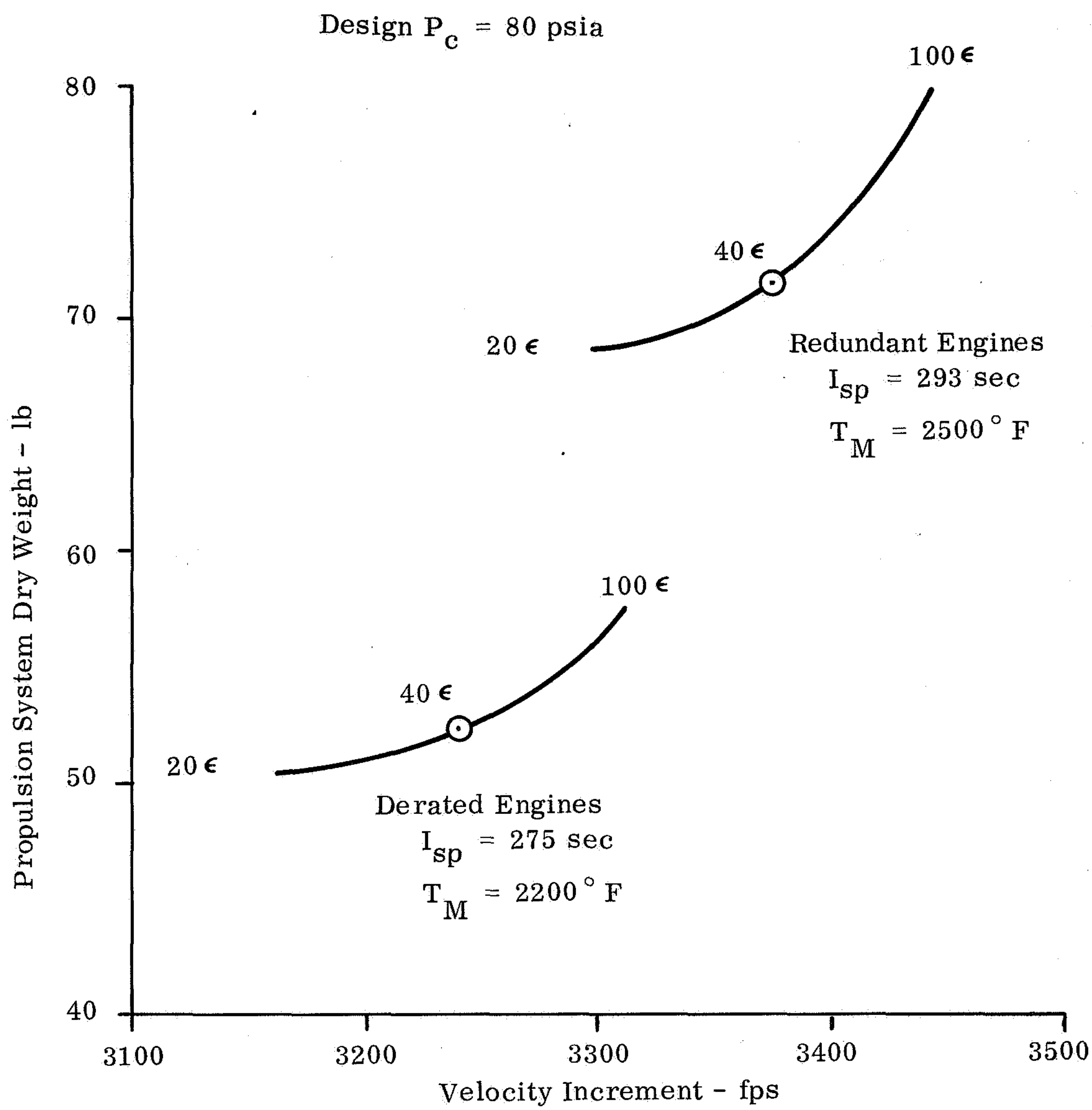


Figure 5.13. Engine Reliability Options

shown to be optimum (Ref. Figure 5.6). Figure 5.13 shows that, compared at the equal range or ΔV point, the derated engine design is 12 pounds lighter than the redundant engine design. Compared at the optimum vehicle design point (i.e., with 40/1 nozzle expansion ratios) the derated engine design is 19 pounds lighter, but 134 fps (4%) lower in ΔV capability.

The correction of these optimization study weights with detailed estimates of fixed weight elements such as tank baffles, mounts, lines and malfunction detection equipment, yields the weight comparison given in Table 5.8 for the derated and redundant engine systems.

TABLE 5.8

PROPULSION SYSTEM COMPARISON

	2 Derated Engines	4 Redundant Engines
Dry Weight - lb	62.9	81.1
Velocity Increment - fps	3240	3374
Required Throttle Ratio	6/1	9/1
Specific Impulse - sec	275 → 250	293 → 256
Approx. Cost Ratio	1	2*

*Due to: Greater Throttle Ratio Development
Malfunction Sensor/Actuator Development
Double Number of Eng. per System
Higher Engine Performance Development

This table shows that the derated engine system is 18.2 pounds or 23% lighter than the redundant engine system for a ΔV penalty of only 4%. The development risk of the derated engine is significantly reduced by the reduced performance, operating temperature and throttle ratio. The ultimate development and operational cost of the derated engine is estimated to be approximately half the cost of the redundant engine case. This is due to double the number of engines per system for the redundant case, increased development testing for higher performance and deeper throttling and the additional development of a malfunction sensor and actuator subsystem.

It was concluded that these quanitative advantages, compounded by the qualitative advantages given in Table 5.9, make the derated engine system the clear choice for the OMLFV.

The selected conceptual design is therefore the optimum propulsion system identified in Paragraph 5.2.4 with derated engines to provide flight safety and reliability equivalent to engine redundancy.

TABLE 5.9

DERATED ENGINE QUALITATIVE ADVANTAGES OVER REDUNDANT ENGINES

A. Greater Safety and Overall Reliability

1. Derated 2200°F failure rate is much less then failure rate at 2500°F operating temperature.
2. No requirement to sense failure and mechanize reaction.
3. Intrinsic reliability of design simplicity with large design margin on current proven technology.

B. Higher Confidence Level

1. Low temperature index of safety and reliability is measurable on each individual engine.
2. Economically limited life and failure rate data from development program will be adequate.

C. Operational Safety Assurance

1. Slow development of failures makes pre-flight inspection effective.
2. Initial failure cannot become critical in single flight (dev. test requirements).

5.4 SUPPORTING STUDIES

Four supporting studies were conducted to make a preliminary evaluation of design options in the major subsystems of the conceptual design.

5.4.1 Engine Control Considerations

The OMLFV selected configuration requires propellant flow modulation control located on a control quadrant that is relatively remote from the outboard engines For differential throttling for roll control, the best location of manually operated engine throttle valves is on the control quadrant. However for reliable and reproducible engine start and shutdown operations, the propellant "manifold" volume between the shutoff valve and the injection face should be held to a minimum. Therefore the best location of the shutoff valves is at the thrust chamber. It was concluded that the best control arrangement for the OMLFV outboard engines is separate throttling and shutoff valves.

The most reliable propellant flow control is a simple bipropellant variable area cavitating venturi valve. Cavitation decouples flow control from engine operating variables for simple open loop calibrated thrust control. Throttle valve design simplicity. the use of large design margins and the operational considerations shown in Table 5.10 , indicate that throttle valve redundancy will not be required for flight safety and reliability in the OMLFV application.

The propellant shutoff valve should be a normally open device to preclude thrust loss due to valve failure in flight. Since post flight shutoff redundancy is provided by the master propellant isolation valves, design for simplicity with large margins, together with the operational considerations summarized in Table 5.11, indicates that OMLFV flight safety and reliability can be provided without propellant shutoff valve redundancy.

TABLE 5.10

ENGINE THROTTLE VALVE CONSIDERATIONS

1. Throttle valve and actuation system exercised under operating pressures in preflight checks - abort ignition if binding is detected.
2. Any contamination cleared by wide open throttle operation during ascent.
3. In flight binding precluded by preflight check.
4. Degraded flow modulation due to misalignment or hard contamination compensated by trim to match second throttle valve output.
5. Internal leakage - not applicable.

TABLE 5.11

ENGINE SHUTOFF VALVE CONSIDERATIONS

Failure Mode	Cause	Effect
Fails to Open for Start-up	Stuck Closed, Actuator Jammed or Broken	Safe - Abort Flight
Actuation Failure In Flight	Broken Actuator	Safe - Valve Normally Open. Continue Flight
Fails to Close on Shutdown	Stuck Open, Actuator Jammed or Broken	Safe - Close Isolation Valve or Run to Prop. Depletion
Internal Leakage	Seat Scratch or Contamination	Safe - Close Isolation Valves

5.4.2 Pressurization Options

The number of possible pressurization options to maximize flight safety and reliability is quite large - including complete pressurization subsystem redundancy, quad-redundant regulator, and large propellant tank design ullage for unregulated blowdown operation capability. Some of the more attractive options are summarized in Table 5.12. However all of these options penalize the system on a major measure of merit-propulsion system dry weight.

The pressurization system, shown in Figure 5.3, provides safety in the OMLFV application. Each flight will be preceded by the pressurization system checkout inherent in system activation (by opening gas isolation hand valve) and readout of resulting gas and propellant tank pressures. All takeoffs will therefore be with normal pressurization system operation.

The only moving parts during flight are the check valves and the regulator. The check valves are quad-redundant and the regulator is backed up by the relief valves and the tank ullage volume. The immediate effect of regulator failure-open is overpressurization of the propellant tanks to the relief valve setting and venting of the gas tank down to propellant tank pressure (through relief valves). The propellant tank pressure will then begin to decay by blowdown of both the gas tank and propellant tank gas as propellant is consumed. The resulting excursion in propellant tank pressures can be compensated by throttle settings for a safe landing. In the case of regulator failure-closed, propellant tank pressure blows down from the regulator setting as propellant continues to be consumed. The resulting tank pressure

TABLE 5.12

PRESSURIZATION OPTIONS

Method	Design Provisions
Backup Blowdown Operation with ullage gas in prop. tanks.	Provide ullage volume in propellant tanks sufficient to allow a safe landing at any point in a flight.
Quad-redundant (series parallel) Regulators.	Provide four regulators in series parallel arrangement.
Regulator bypass with orifice to restrict flow to the rate necessary to provide adequate thrust.	Bypass and flow restricting orifice. Shutoff valve on line to regulator. Valve on bypass line.
Regulator bypass providing manual pressure control.	Bypass line with manual gas control valve allowing operator to control pressure. Shutoff valve on regulator line.
Two parallel regulators with manual control valve to protect against regulator failed open.	Add second regulator and manual control.

excursion can be compensated by throttle setting changes to produce the thrust sequence required for a safe emergency landing.

There is intrinsic compensation in this scheme of backing up the regulator with blowdown operation. As the burn time required for a safe emergency descent and landing increases during the mission (with altitude and velocity) the propellant consumed to get to increasing altitudes and velocities has left increasing tank ullage volumes of pressurized gas for increased blowdown operation capability. However, the exact value of the design ullage volume required to provide safe emergency landing in all portions of all possible flights must be determined by operations analysis.

5.4.3 Yaw Control Options

Several yaw control concepts have been studied which are applicable to OMLFV configurations with both single and multiple engine systems. These studies defined the basic merits and problem areas of each concept. Yaw control subsystems have been sized for the range of vehicle weights and yaw moment requirements generated in configuration studies. The yaw subsystem weight and delivered yaw moment for each system considered is listed in Table 5-13.

TABLE 5.13

SUMMARY OF YAW CONTROL DEVICES

Yaw Control Device	Subsystem Weight Estimate (lb)		Yaw Moment (in-lb)		Remarks
			Main Engine Thrust		
	One Sortie	Five Sortie	Max.	Min.	
Four Freon Gas Five-lb Motors	25.0	101*	150	150	Applies to all vehicles Requires resupply tanks
Four N_2 Cold Gas Five-lb Motors	35.0	167*	150	150	Integrated with main pressurization tanks. Applies to all vehicles.
Four Bipropellant Five-lb Motors	13.0	25	150	150	Applies to all vehicles. Main tanks supply propellant.
Jet Vanes on Single Engine	13.0	42	>110	~10	Yaw force decreases with thrust. Limited vane life in engine exhaust.
Jetavators on Two Engines	13.0	33	-	-	Yaw force decreases with thrust. High temperature hinge point.
Secondary Injection on Two Engines	15.0	31.0	160	136	Major design and development effort. Secondary valves required.
Two Differentially Pivoted Engines	~6	~8	400	140	Straight forward development. Large yaw moments at all thrust levels.

*Gas bottles replaced prior to each sortie.

Of the yaw control methods investigated, cold gas, liquid bipropellant and pivoted thruster systems present no development problems. Jet vanes and jetavators effect engine performance and have inherent thermal problems in the areas of control surface durability and hot hinge design. Analyses indicate that jet vane and jetavator control forces are significantly reduced at reduced thrust levels. In preliminary investigations the control effectiveness loss in an engine using secondary injection for yaw control was found to be small at reduced thrust levels.

In the critical areas of weight, control effectiveness, and development risks, differentially pivoted engines are superior to the other yaw control schemes. However, this system is limited to vehicle configurations with two or more engines. A bipropellant yaw control system is the best selection for single engine vehicle designs. Differentially pivoted thrusters are used for yaw control on the selected OMLFV.

5.4.4 Industry Survey

A survey of components used on existing space systems was completed for the major propulsion system components for the OMLFV. Designs that could be used directly or easily modified for the OMLFV application were considered. This work has been reported in the "Lunar Flying Vehicle Component Study," Bell Aerosystems Company, Technical Note 980:69:0205-1: CES, February 27, 1969, classified confidential. An unclassified summary of that work and the basic finding will be presented here.

The major propulsion system components included in the survey are the rocket engine assemblies (including throttle and shutoff valves), fuel and oxidizer tanks, pressurization gas storage tanks, and pressure regulator. Attitude control engines were included in the survey, although none are required in the selected OMLFV design. Typical designs for valves (other than engine valves), connectors, etc., were chosen to facilitate the preparation of the system design and installation drawing.

The limits set in the survey on the basic design parameters of major components are listed below:

Main Engine Assembly	50-500 lb thrust, N_2O_4/50-50 or similar propellants
Propellant Tank	200-400 psia working pressure, 1000-7000 cubic inch volume
Pressurant Tank	2000-5000 psia working pressure, 200-2000 cubic inch volume
Pressure Regulator	2000-5000 psia source pressure 190-275 psia regulated pressure 3-18 scfm flowrate range

Possible sources for each of these components for the selected OMLFV configuration are summarized in Table 5.14. This table indicates the availability of components or design technology required for the OMLFV propulsion system design. The characteristics of the particular components are discussed in the following subsection.

The industry survey indicates that qualified hardware is available for the pressure regulator, gas storage tank, and for most valves and flow control components. Minor modifications are required in some cases. A new propellant tank design, optimized for the OMLFV application, is necessary but the design can be based on proven technology. An engine based on the designs listed in Table 5.14 must be qualified for the OMLFV application.

TABLE 5.14

AVAILABILITY OF MAJOR OMLFV COMPONENTS

Component	Component Identification or Space Program Source
Engine Assembly	MIRA 150R Engine Developed for Surveyor (TRW) Model 8414 Engine Demonstrated for Manned Flying System (Bell) R-4D Engine for Apollo LM, SM and for Lunar Orbiter (Marquardt) Hypermet Engine (Aerojet)
Propellant Tanks	Apollo Command Module RCS Apollo Service Module RCS Apollo Lunar Module RCS
Pressurant Tanks	Minuteman III Post Boost Propulsion System Saturn II (700 in.^{3}) Saturn IVB (900 in.^{3} and 1000 in.^{3})
Pressure Regulators	Minuteman III Post Boost Propulsion System Lunar Module RCS Gemini/Lunar Orbiter Agena Secondary Propulsion System

5.5 PRELIMINARY DESIGN

A preliminary design study of the propulsion system was conducted, based on results from the optimization and conceptual design studies (Ref Paragraphs 5.2 and 5.3), to support the preliminary design of the OMLFV. The approach was to design as close to the optimized conceptual design as use of current operational or demonstrated technology would permit. The definitive features of this optimum conceptual design are summarized in Table 5.15.

TABLE 5.15

OPTIMUM CONCEPTUAL DESIGN FEATURES

Two Engines	-	Providing all thrust and attitude control forces required.
Radiation Cooled Engines	-	Derated for low operating temperature for safety and reliability.
Fixed Injection Geometry	-	Reliability outweighs throttling efficiency.
Simple Propellant Supply System	-	Regulated pressure-fed.
Spherical Gas Tank	-	Titanium alloy (6 Al-4V).
Simple Shell Propellant Tanks	-	Cylindrical shape, with slosh and vortex baffles (Ti, 6 Al-4V).
Mechanical Valves and Controls	-	Manually operated.

5.5.1 Propulsion System Schematic

Figure 5.14 presents a pictorial schematic of the OMLFV propulsion system. The arrangement of components shown in this schematic corresponds to their location when installed in the vehicle.

A cold gas pressurization system is used to pressurize the propellant tanks and feed propellants to the thrust chambers. Two engine assemblies consisting of thrust chambers, shutoff valves, and throttle valves are used. The engines are mounted outboard with the throttle valves remotely located and the shutoff valves close coupled to the engine.

The pressurization system consists of a high pressure helium tank, isolation valve, connector, filter, regulator, quad-redundant check valves, and pressure relief valves. The helium tank is replaced as a module during refueling. The isolation valve seals the helium tank during storage and initiates gas flow in preparation for

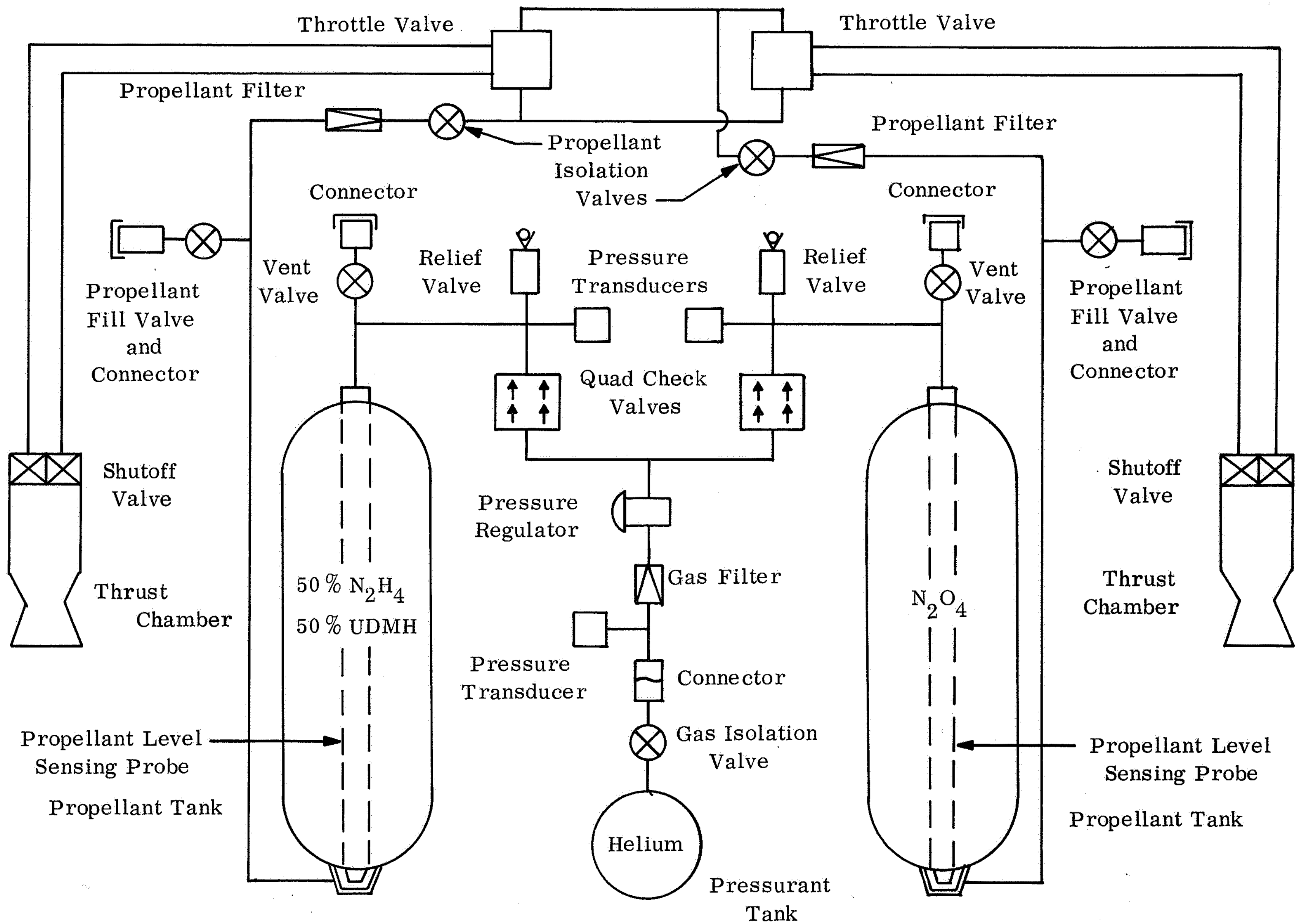


Figure 5.14. Schematic Diagram - Lunar Flying Vehicle Propulsion System

flight. The pressure transducer upstream of the regulator allows monitoring of helium tank pressure before and during flight. The quad check valves are needed to prevent propellant vapor from entering the common pressurization lines. This is a possibility as the tanks have no bladders to contain the propellants and vapors. The relief valves located in the gas lines of each propellant tank prevent over-pressurization in case of regulator failure. They also guard against over-pressurization due to temperature increases which might occur between vehicle flights.

The propellant supply system consists of the propellant tanks, fill and vent valves, filters and isolation valves. Pressure transducers and propellant quantity sensors are included to indicate the status of the propellant supply. The propellant tanks are cylindrical with slosh and anti-vortex baffles to insure propellant acquisition. No positive expulsion devices are required since thrust will be applied throughout the flight. The propellant surface will always be oriented perpendicular to the thrust vector and the propellant will be settled in the bottoms of the tanks. The isolation valves downstream of the tanks provide a backup in case of engine shutoff valve failure and prevent propellant leakage through throttle valve seals. Vent and fill valves, with quick disconnect couplings, are provided on each propellant tank to allow refilling with propellants from the LM descent stage.

The engine assemblies include the thrust chambers, bipropellant shutoff valves, and throttle valves. The throttle valves are located remote from the thrust chambers. Manually controlled valves are used on the engines and throughout the propulsion system.

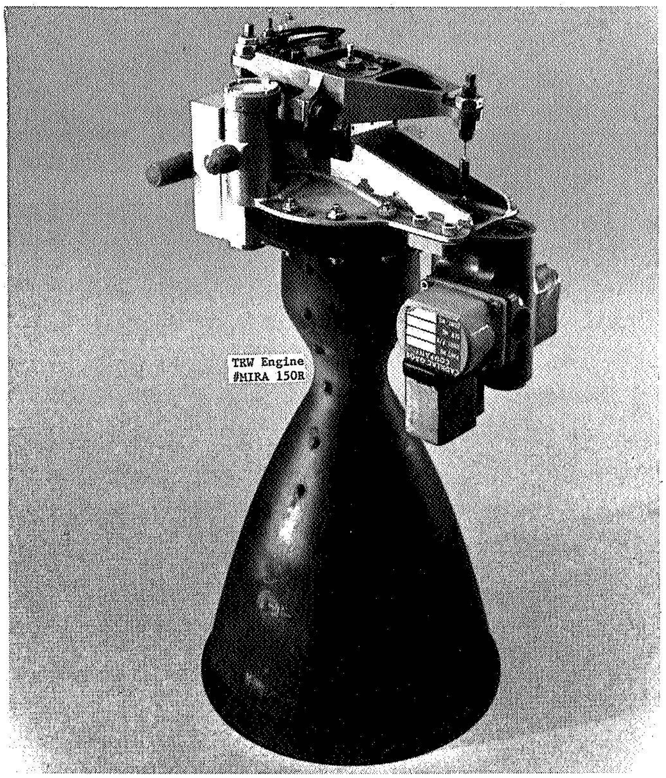
Requirements were determined for each system component shown in the schematic. Typical designs were chosen from available hardware for use in preparing an installation drawing and weight statement. An investigation was made into throttleable engine designs as this is the most critical propulsion system component.

5.5.2 Engines

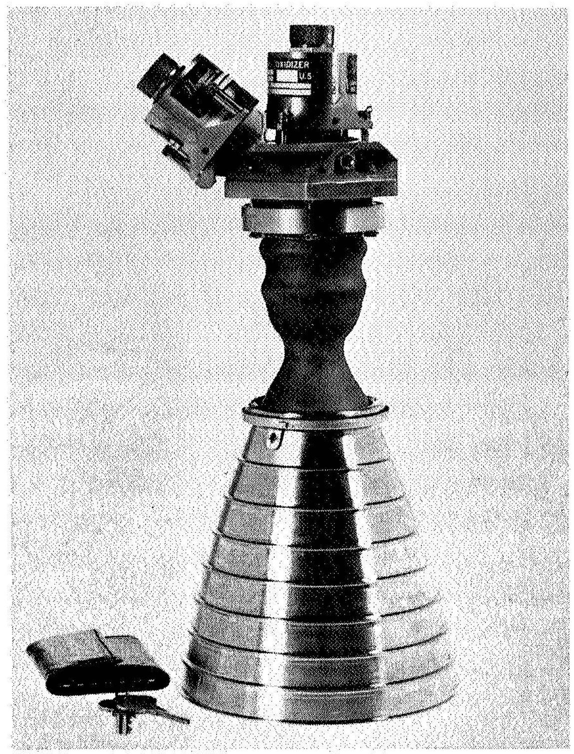
An industry survey of rocket engines, and contacts with engine suppliers, has produced four known candidates for the OMLFV application. The four engine candidates and manufacturing companies are:

a. 150 lb MIRA 150 R - TRW Inc.
b. 100 lb Model 8414 - Bell Aerosystems Company
c. 100 lb Model R-4D - Marquardt Corporation
d. 100 lb Hypermet - Aerojet-General Corporation

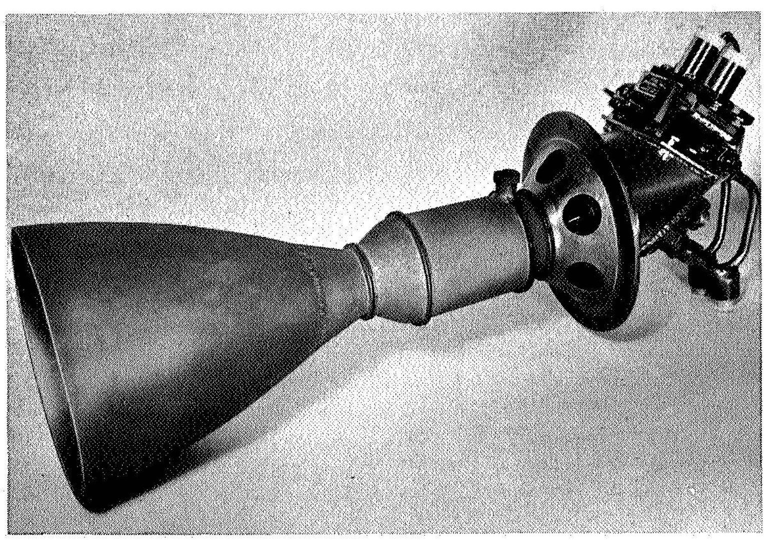
Each of these engine designs require modification and additional development. Photographs of the engines are shown in Figure 5.15. The physical and operating characteristics, required modifications for OMLFV, and development status will be discussed for each of these engines. Data on the candidate engines has been taken from reports and correspondence provided by the respective engine manufacturer.



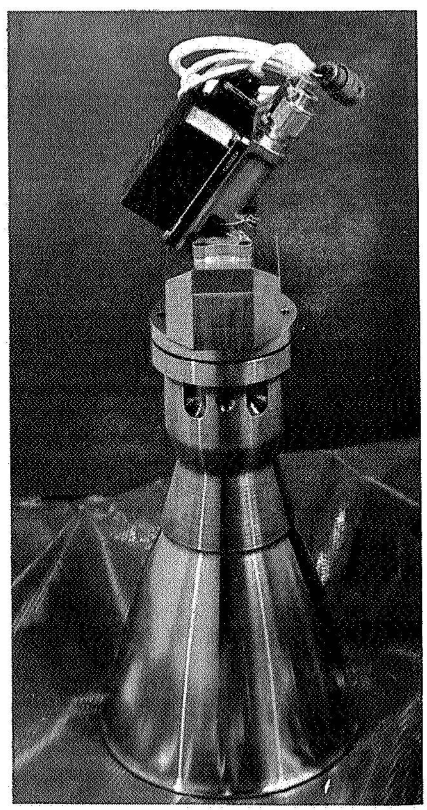
(a) TRW Mira 150R



(c) Marquardt R-4D



(b) Bell Aerosystems Model 8414



(d) Aerojet Hypermet

Figure 5.15. Candidate Engine Photographs

a. MIRA 150 R

The TRW MIRA 150 R is a throttling engine using propellant flow modulation and variable injector geometry to accomplish the desired thrust reduction. For the OMLFV application a coated columbium thrust chamber would be used. (See References 44 and 45.)

The injector uses circular slots to produce intersecting cones of fuel and oxidizer. The flow areas of both slots are varied by a short stroke sliding sleeve arrangement. Propellant flow rate modulation is by variable area cavitating venturi throttle valve and shutoff is by a separate pilot operated poppet valve.

The throttle valve and injector are currently driven by a servoactuator through a common actuator arm. The injector, throttle valve, shutoff valve and actuator are integrated into the "head-end" assembly which can be seen mounted on the engine in Figure 5.15a.

The electrically controlled version of the MIRA engine has been developed under JPL Surveyor Contract 950596 and NASA Manned Flying System Contract NAS 8-20248. The throttling range required for the OMLFV duty cycles has been demonstrated. Operating life in excess of one hour has been demonstrated at a mixture ratio of 1.31 and 100 pounds maximum thrust with a radiation cooled Cb chamber. Engine development required for the OMLFV application includes uprating to 150 pounds thrust, conversion to manual control, demonstration of 3.5 hour life, and man rating.

b. Model 8414

The Bell Aerosystems Model 8414 is a fixed injection geometry, throttling engine with a coated columbium radiation cooled thrust chamber. For the long life required for OMLFV, a fuel film would be incorporated to insure a large thermal margin. The main injector elements are balanced triplets. Flow control is by a variable area cavitating venturi throttle valve and separate shutoff valve. This engine design is shown in Figure 5.15b. (See References 46 and 47.)

The Model 8414 design was developed for the Manned Flying System under NASA Contract NAS 8-20086. The required OMLFV throttling range and typical duty cycles were demonstrated under this contract. The demonstration engines had electrically operated valves. Engine development required on the Bell Model 8414 for the OMLFV application includes scaling to 150 pound thrust, conversion to manual valve control, demonstration of 3.5 hours life, and man rating.

c. Model R-4D

The present design of the Marquardt Model R-4D is a fixed thrust, radiation cooled engine. The thrust chamber is of coated molybdenum construction with an L605 nozzle extension. The injector uses eight doublets as the main elements plus a single doublet in a preignition chamber. Fuel film cooling is provided to insure low

wall temperatures. The design presently incorporates separate fuel and oxidizer solenoid shutoff valves. (See References 48 and 49.)

The fixed thrust R-4D engine has been used operationally on the Apollo Service Module and Lunar Module reaction control systems and on the Lunar Orbiter. The R-4D is shown in Figure 5.15.c. Pressure throttling has been demonstrated in test by varying propellant supply tank pressure. The OMLFV engine life requirement has been demonstrated at a mixture ratio of 2.0.

For the OMLFV application the Model R-4D must be adapted to throttling operation and a throttle valve developed. To meet the 150 pound thrust and feed pressure requirements, Marquardt has suggested injector modifications (removal of preigniter and scaling up injection orifices) to minimize presure drop and use of low pressure drop manually operated throttle and shutoff valves. The engine, already qualified as a 100 pound fixed thrust man rated engine, must be requalified to the OMLFV requirements.

d. Hypermet Engine

The Aerojet Hypermet engine has an Inconel chamber liner and nozzle extension with a copper outer shell on the chamber. The injector face is a porous surface containing small slots several thousands of an inch wide. Each slot is alternately connected to the fuel and oxidizer manifolds. Chamber cooling is achieved by a fuel film and the high conductivity copper shell. A fixed thrust version of the Hypermet engine is shown in Figure 5.15.d.

A throttling version of this engine incorporates a variable area cavitating venturi throttle valve for flow modulation. A combined shutoff valve and throttle valve has been suggested by Aerojet. The injector flow passages are designed for laminar flow for throttling applications. As a result, the injector pressure drop is proportional to the flow rate.

The slotted injector has demonstrated throttling over a 10:1 thrust ratio. The Hypermet bimetallic thrust chamber has been demonstrated with N_2O_4/MMH propellants. Engine development required on the Hypermet engine for the OMLFV application includes scaling to 150 pounds thrust, demonstration of 3.5 hour operating life, man rating, and development of a manually operated throttle valve.

All four existing engine designs discussed require modification and additional development for the OMLFV application. However, the existence of four demonstrated candidate engine designs for the OMLFV indicates that the requirements can be met with existing design technology.

The characteristics and performance of all four candidates, as modified for the OMLFV application, are presented in Table 5.16. Due to the long operating life requirement, engine designs with low wall temperatures (high thermal margin) were requested from the engine suppliers. The design and performance numbers in

TABLE 5.16

CHARACTERISTICS OF THROTTLING ENGINES

Engine Name or Model Number	MIRA 150 R Engine	Model 8414 Engine	R-4D Apollo RCS Engine	Hypermet Engine
Engine Manufacturer	TRW, Inc.	Bell Aerosystems	Marquardt Corp.	Aerojet-General
Engine Description - Injector	Variable orifice area, impinging cones.	Balanced triplets with fuel film.	Doublets with fuel film.	Narrow slots with fuel film.
- Throttling	Variable area cavitating venturi, variable geometry injector.	Variable area cavitating venturi. Fixed injector.	Throttle valve design not selected. Fixed injector.	Variable area cavitating venturi, laminar flow injector.
- Chamber	Coated columbium.	Coated columbium.	Coated molybdenum.	Inconel liner with copper shell. Inconel extension.
- Cooling Concept	Radiation	Radiation and fuel film.	Radiation and fuel film.	Conduction and fuel film in the chamber. Radiation cooled extension.
Propellants	N_2O_4/50-50	N_2O_4/50-50	N_2O_4/50-50	N_2O_4/50-50
Vacuum Thrust, Rated, F - lb	150	150	150	150
Minimum, F - lb	25	25	25	25
Chamber Pressure at Rated F, P_c - psia	100	80	144	125
Area Ratio, ϵ	40:1	40:1	40:1	40:1
Mixture Ratio, R	1.3	1.3	2.0	1.6*
Vacuum Specific Impulse, I_{sp} at Rated F, sec	279	276	272	292*
I_{sp} at Minimum F, sec	256	250	244	266*
Characteristic Velocity, c* - ft/sec	5160	5100	5000	-
Vacuum Thrust Coefficient, C_f	1.74	1.742	1.75	-
Minimum Engine Feed Pressure, psia	235	200	235	275
Barrier Fuel Flow, Percentage of Total Fuel Flow	None	20%	-	24%
Maximum Thrust Chamber Wall Temperature - °F	2350	2250	2000	~ 1600
Engine Assembly Weight - lb	6.43	7.3	5.0	~ 8.2
Engine Envelope, Excluding Throttle Valve:				
Overall Length - in.	13.45	20.56	13.42	~13.0
Exit Diameter - in.	6.00	7.62	5.74	~ 6.0

*Data for a long life version of this engine is not available. Data presented here is for an engine designed to deliver maximum performance.

the table reflect this conservative design approach, except in the case of the Aerojet Hypermet engine. Design and performance numbers on the Hypermet engine were not available for a long life, low temperature version. Therefore, performance is quoted for an engine optimized for maximum specific impulse. This is noted on the table where appropriate.

5.5.3 Propulsion System Status Sensors

Propulsion system instrumentation is limited to that necessary for safe operation. The parameters to be measured are: helium gas supply pressure, fuel and oxidizer tank pressures, and fuel and oxidizer quantity. The helium supply pressure must be measured so that it may be checked before the start of a flight. During a flight it can serve as a backup check on the propellant quantity sensing system. The propellant tank pressures should be measured so that the astronaut can abort the flight if falling tank pressures indicate a propellant or gas leak. The tank pressure reading is also useful during refueling as it is desirable to fill the tanks against controlled back pressure to minimize propellant evaporation and loss to space. Propellant quantity sensing is necessary to refueling and safe flight of the vehicle. A continuous level sensing system is preferable to a low level warning. With continuous sensing the pilot always knows the amount of propellant remaining and can make decisions accordingly.

For pressure measurement, bourdon tube and strain gage pressure sensors were considered. The bourdon tube requires a flexible line from tank to gage and complicates the display panel design. A preliminary selection of strain gage pressure transducers was made based on these considerations. This transducer produces an electrical output signal and only small wire leads are required to the display panel. Space qualified pressure transducers, calibrated to the appropriate pressure levels, are available from the Minuteman III PBPS program for use on both the helium supply tank and propellant tanks. The overall accuracy of the pressure transducer measurement is better than 2% of full scale.

A study was made of propellant quantity sensing systems which might be applicable to the OMLFV. Continuous sensing is required and a space qualified design is desirable. Two space qualified systems are available which provide continuous quantity sensing. These designs are a nucleonic fuel gaging system similar to that developed by General Nucleonics Corporation and a capacitance measuring system developed for Apollo by Trans-Sonics Inc. The latter system is preferable for the OMLFV application because it is already qualified and in use on LM.

The selected quantity sensing system is presently operational in the LM descent stage of Apollo. The system measures the liquid remaining in the descent stage tanks as a percentage of full tank capacity. A sensing probe is mounted in each tank and one control unit transforms the output signal into the desired form. Information furnished by the supplier indicates an accuracy better than one percent of full scale throughout the measurement range. This design could be modified to the OMLFV application.

5.5.4 Valve and Components

A preliminary selection of typical components are made. Space qualified hard ware was used with minimum modification wherever possible. Table 5.17 presents a list of the basic requirements for each component and systems using components which meet these requirements. Only the propellant tanks and engines must be designed and optimized for the OMLFV applications; all other components require only minor modifications to available qualified hardware.

For the sake of completeness and easy reference, all system components have been noted in Table 5.17. Where more than one component is available for the application, the design used is underlined in the table. The components listed in the table are shown on the system installation drawing.

5.5.5 System Installation

The propulsion system installation drawing, Figure 5.16, shows the arrangement of components within the central structure of the OMLFV. The outboard engine assemblies are shown in Figure 5.17. The component designs listed in Table 5.17, along with an optimized tank design, were used to obtain the configuration shown in the installation drawing. The Model 8414 engine was employed in this preliminary design effort (Figure 5.17) because it is currently configured for the OMLFV. However, reconfigured versions of the other engine candidates also apply, without change in system design.

Of particular importance in selecting the arrangement and mounting components are the considerations of system operation during flight and during refueling. The design shown in Figure 5.16 allows all refueling operations to be done from the front of the vehicle. The front panel may be opened to give access to the helium bottle and gas isolation valve and to the propellant fill and vent valves. As the first step in refueling, the helium isolation valve is closed and the helium tank removed by unlatching the tank straps and disconnecting the gas supply line. The fuel tank is refilled from the LM descent stage by connecting the resupply line to the OMLFV fuel fill connector. The tank is filled by opening the fill valve and venting the tank. The oxidizer tank is filled in the same manner. The last step in the refueling procedure is replacement of the helium bottle with a charged bottle stored in LM.

Immediately before a flight the propellant tanks are pressurized by opening the helium isolation valve. The propellant isolation valves, shown on the top panel of the OMLFV body, are opened in preparation for starting the engines.

The location of the engine components is dictated by ignition and thermal requirements and control linkage considerations. The throttle valve is located in front of the pilot near the vehicle controls. This allows minimum complexity and backlash in the throttle control. The shutoff valve is located on the thrust chamber so that the propellant lines are full when the system is activated and for start and shutdown re-

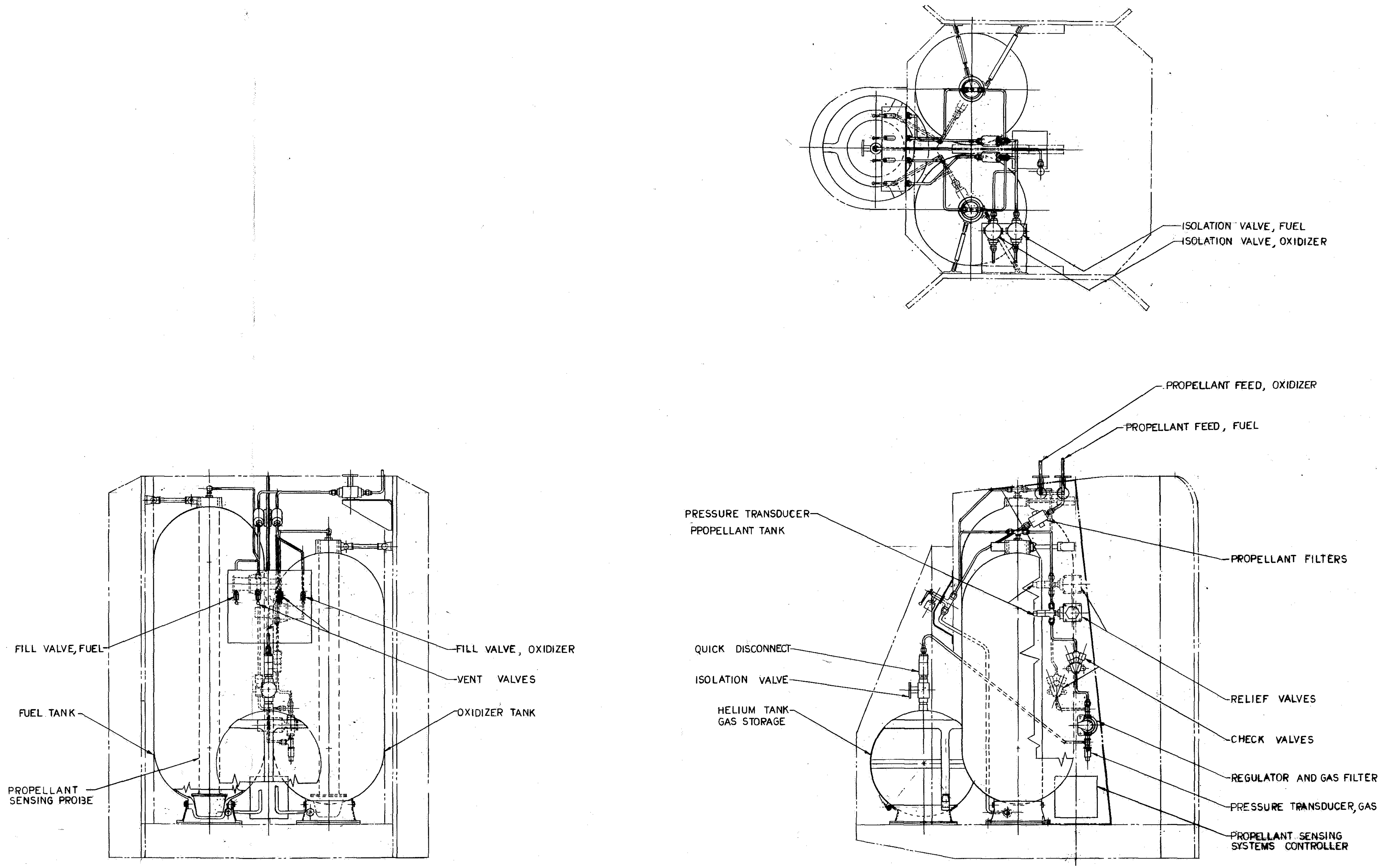
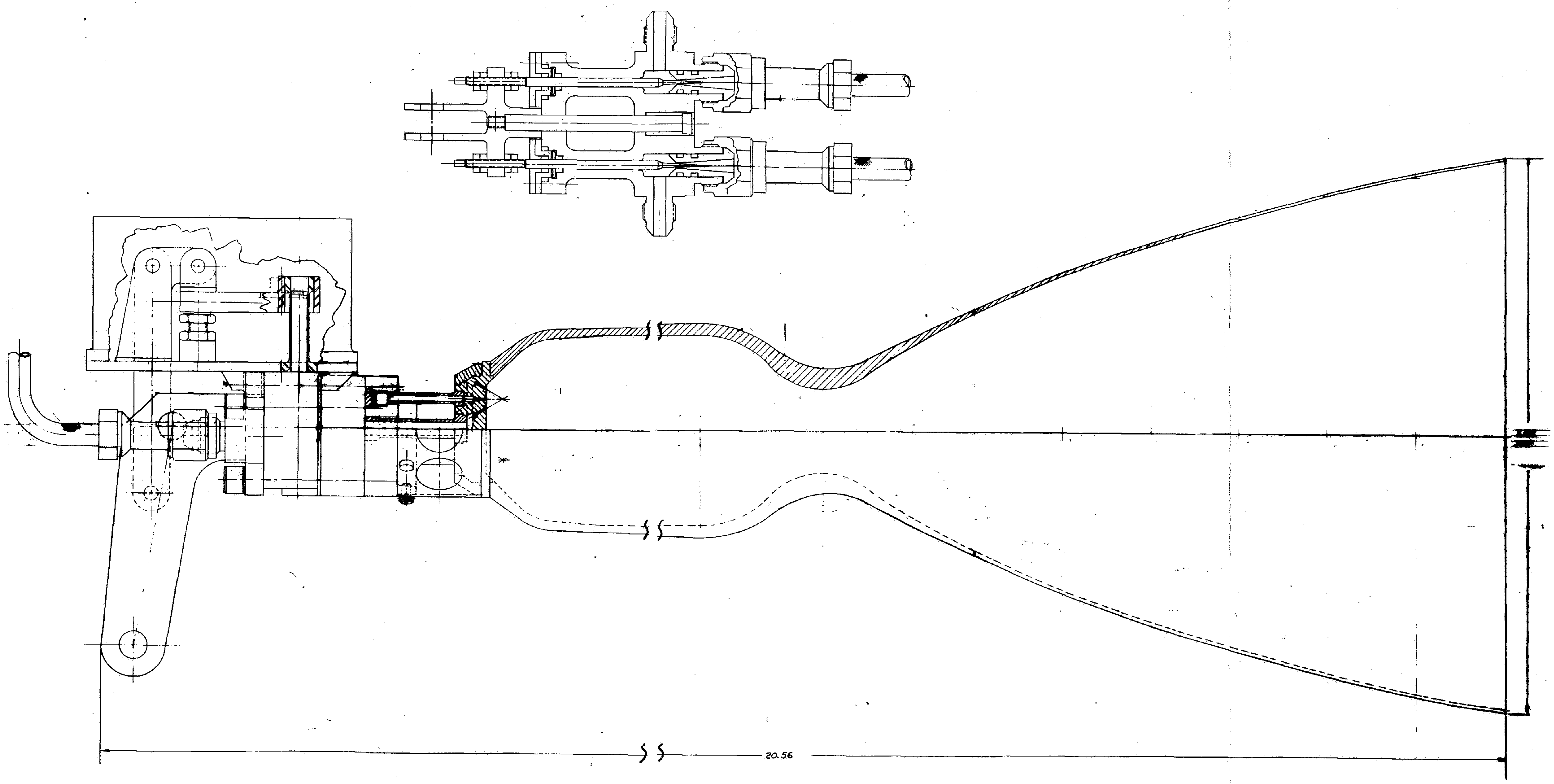


Figure 5.16. One Man Lunar Flying Vehicle - Propulsion System Installation



Remotely Located Throttle Valve
Single Axis Gimbal
Manual Valve Operation

Figure 5.17. 150 Pound Throttleable Engine

TABLE 5.17

LUNAR FLYING VEHICLE COMPONENT SUMMARY

Component	Typical OMLFV Requirement	Available Components Considered
Helium Tank	Volume - 950 cu. in Working Pressure - 3700 psia	Saturn II (2 required) Minuteman III PBPS (1 required) Saturn IVB (2 required) Saturn IVB (1 required)
Gas Isolation Valve	Pressure Rating - 3700 psia Low Leakage	Mercury (8060-472001)-Typical
Gas and Propellant Quick Disconnect	Gas Pressure - 3700 psia Propellant Pressure - 300 psia Line Size - 1/4 inch	Snap-Tite (Dwg. No. 3544) Hansen Manufacturing Co. Aeroquip Corporation
Pressure Regulator and Gas Filter	Source Pressure - 3700 psia Regulated Pressure - 190-275 psia Flow Rate Range - 3 to 18 scfm	Minuteman III PBPS Lunar Module RCS Gemini/Lunar Orbiter Agena Secondary Propulsion System
Gas Pressure Transducer Propellant Tank Pressure Transducer	Pressure Range - 400 to 3700 psia and - 0 to 300 psia	Minuteman III PBPS (Statham) Typical
Quad Redundant Check Valve	Line Size - 1/4 inch Low Pressure Drop	Lunar Module (Accessory Products)
Relief Valve	Cracking Pressure - 275 psia Maximum Flow Rate - 18 scfm	Minuteman III PBPS
Fill and Vent Valves	Line Size - 1/4 inch Flow Rate - 20 lb/min	Agena Target Vehicle - Typical
Propellant Filters	Line Size - 3/8 inch. Total flow without clogging - 5000 lb	Minuteman III PBPS - Typical
Propellant Isolation Valves	Pressure Rating - 300 psia. Low Leakage	Mercury (8060-472001)-Typical
Propellant Quantity Sensing	Sensing Range - 0 to 100% Accuracy - 1.0% maximum error. Continuous sensing required.	LM Descent Stage (Trans-Sonics) Nuclear (General Nucleonics Corp.)
Propellant Tanks	Design Pressure - 315 psia Volume - 2.0 ft^3 oxid. 2.4 ft^3 fuel	New, optimized design required
Thrust Chamber Assembly Including Shutoff Valve Throttle Valve	150 lb Thrust 6/1 Throttle Ratio 3.5 Hr Life	See Table 5-16

produceability. The lines are insulated and the heat sink of the trapped propellants is used to limit the temperature excursion to the 70 ±30°F range specified for the propulsion system. It is necessary to maintain the engine feed lines in this temperature range to preclude admitting propellants into lines which might be below the propellant freezing point. This condition might allow the propellants to freeze when flow is initiated and clog the lines.

With the lines insulated and filled with propellants, the worst thermal situation would be one in which the lines and trapped propellants would be at 40°F or 100°F and the propellant in the tanks being near the 70°F nominal. (Possible variations in propellant tank temperature are fully discussed in Section 7.0, Thermal Analysis.) With this condition, it is possible to start the engine with 40°F or 100°F propellants and then rapidly go to 70°F propellants as the propellants trapped in the lines are used. It was found that the maximum thrust variation due to this effect, with cavitating venturi flow control, is less than 3.4% and the mixture ratio variation is 3.9%. In addition, the off-nominal operation encountered is limited to a two second period after engine start at minimum thrust. This period would probably be used for system checkout before flight. Thus, these variations would not be encountered during flight.

The weight and performance characteristics were generated for the selected propulsion system design. A pressure schedule, specific impulse performance summary, and weight statement are presented in the following paragraphs.

5.5.6 Pressure Schedule

The system pressure schedule is determined by the feed pressure required by candidate engines. The survey of engine candidates indicates that a feed pressure of 235 psia could be used as typical for the system preliminary design. Therefore, the OMLFV propulsion system pressure schedule is based on this feed pressure. Table 5.18 presents the complete pressure schedule with a chamber pressure of 80 psia. The schedule is essentially the same for different engines with different chamber pressures.

The propellant tank design pressure is set equal to the full open pressure of the relief valve. This insures that the tanks will not over-pressurize when the system is activated with full propellant tanks.

TABLE 5.18

ONE MAN LUNAR FLYING VEHICLE
TYPICAL PRESSURE SCHEDULE

Chamber Pressure, P_c, psia	80
Injector Pressure Drop, ΔP_I, psi	78
Shutoff Valve Pressure Drop, ΔP_{SV}, psi	35
Throttle Valve Pressure Drop, ΔP_{TV}, psi	35
Engine Feed Pressure, P_F, psia	228
Nominal Propellant Tank Pressure, P_T, psia	235
Regulated Pressure, psia	240 ±7
Regulator Lockup Pressure, psia	247
Relief Valve Reseat Pressure, psia	255
Relief Valve Cracking Pressure, psia	275 ±7
Max. Relief Valve Pressure, Full Open, psia	315
Propellant Tank Design Pressure, psia	315
Proof Pressure, psia (Apollo F.S.)	420
Burst Pressure, psia (Apollo F.S.)	472
Minimum Gas Bottle Pressure, psia	400
Gas Tank Loading Pressure, at 60°F, psia	3200
Max. Gas Tank Operating Pressure, at 125°F, psia	3640
Gas Tank Proof Pressure, psia	5460
Gas Tank Burst Pressure, psia	7280

5.5.7 Performance and Weight

The delivered performance of the propulsion system is the engine thrust and specific impulse. Data was requested from engine suppliers on high reliability, low wall temperature versions of candidate engines. The throttling specific impulse performance of the engine candidates is summarized in Figure 5.18. The engines are all 150 lb thrust rated with 6:1 throttling. Engine wall temperatures are limited at the expense of performance to insure a 3.5 hour operating life with high reliability. The Aerojet Hypermet engine is not included in this summary since the data for a long life version is not presently available.

A detailed propulsion system weight statement is presented in Table 5.19. The weights include only propulsion system components. Brackets, tank mounts and batteries are excluded from this weight statement to be covered in the overall vehicle weight. Component weights are based on the designs listed in Table 5.17 with suitable modifications. The engine and mount weights used are representative of the candidate engines considered.

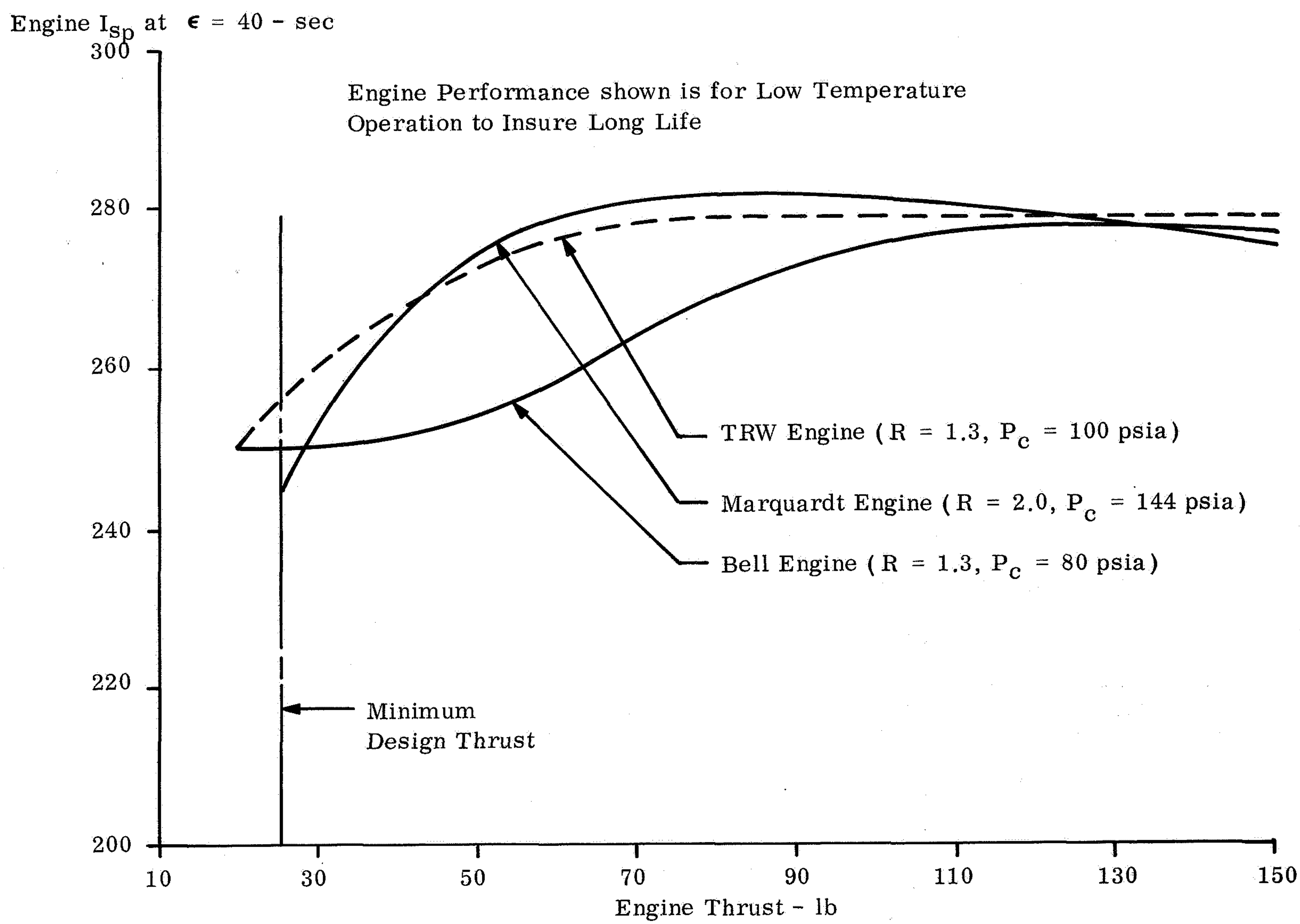


Figure 5.18. Candidate Engines Throttling Performance Summary

TABLE 5.19

PROPULSION SYSTEM WEIGHT

Item	Pounds
Pressurant Tank	12.9
Gas Isolation Valve	0.6
Gas Connector	0.5
Gas Pressure Transducer	0.2
Regulator and Filter	1.2
Quad Check Valves (2)	0.8
Prop. Tank Press. Transducers (2)	0.4
Relief Valves (2)	2.0
Vent Valves (2)	0.4
Vent Connectors, Flight Half (2)	0.4
Oxidizer Tank	9.4
Fuel Tank	10.3
Quantity Sensing Probes (2)	6.0
Quantity Sensing System Controller	3.0
Propellant Fill Valves (2)	0.4
Propellant Resupply Connectors (Flight/Half) (2)	0.4
Propellant Filters (2)	0.9
Propellant Isolation Valves (2)	1.2
Throttle Valves (2)	2.4
Bipropellant Shutoff Valves (2)	2.0
Thrust Chambers (2)	9.6
Engine Mount (2)	0.6
Rigid Lines	2.0
Flex Lines and Fittings	3.2
Miscellaneous Fittings, Etc.	2.0
Total Propulsion System Dry Weight	72.8

Excluded: Tank mounts, brackets, structural web, gauges, battery. These items are included in the vehicle structure weight.

6.0 CONTROL SYSTEM ANALYSIS AND SIMULATION

6.1 GENERAL

An extensive analysis and simulation program was performed to evaluate alternative control system configurations and characteristics, and to select, optimize and reevaluate a final recommended control system.

6.1.1 Program Phases

The control system analysis and simulation program was conducted in four phases. Each phase is briefly summarized below and is the subject of a subsequent major subparagraph, noted in parenthesis.

The first phase (see 6.3) was a comparison and selection between kinesthetically and thrust vector controlled types of vehicles. The comparison was based on a comprehensive qualitative analysis. As a result of this study, thrust vector control was selected for exclusive consideration in the remainder of the program.

The second phase (6.4) consisted of parametric studies of various handling qualities aspects to develop basic trends and comparisons over a broad range of control system characteristics. A six-degree-of-freedom motion/visual simulation, described in Appendix C, was used extensively during this phase of the program. Best values and ranges of control sensitivity, maximum control deflection, and throttle gradient were determined for a typical control system configuration. Then alternative control configurations (pivots above the cg, pivots below, and differential throttling in various axes) were evaluated and compared. The results of these parametric studies were used in the vehicle configuration design and selection study described in Sections 2.0 and 3.0. The ability of a space-suited operator to control the simulated vehicle was also verified and compared with shirt sleeve operation during this phase of the program.

The third phase (6.5) of the analysis and simulation program was concerned with the analysis, optimization, and reevaluation of a control system for the selected vehicle configuration. During this phase the weight, inertia, cg variations, control linkage ratios and the resulting control sensitivity and trim characteristics of the selected vehicle configuration were calculated and simulated in more detail. The refined vehicle model and equations of motion used for the expanded simulation are described in Appendix C. Simulation studies were conducted to refine the control system design characteristics and evaluate them under realistic conditions of weights, sensitivities, and trim variations, control and dynamic couplings, and propellant limitations.

The fourth phase (6.6) was an evaluation and comparison of alternative augmentation systems, used in conjunction with the selected vehicle and control system. A rate gyro stability augmentation system and an all mechanical control augmentation

system were simulated and compared with each other and with an unaugmented control system. A three-axis Apollo hand controller and a handle bar controller were each evaluated with and without augmentation during this phase.

6.1.2 Evaluation Techniques

Alternative control system characteristics were evaluated under four different conditions of simulated flight, using various combinations of four different quantitative measures. The flight conditions consist of five-mile flights, with profiles ranging from high altitude arcing to low altitude nap-of-the-moon; 1600-foot nap-of-the-moon flights; hover at a fixed point near the surface; and 3-axis attitude hold, with the translational degrees of freedom disabled. The quantitative measures, described in detail in Appendix C, include both subjective pilot (Cooper) ratings and objective workload, accuracy, and ΔV measures. Other quantitative measures, such as average controller displacement, were tried and abandoned. Table 6-1 is a summary of the flight conditions and quantitative measures used in connection with the various study tasks, for which data is presented in this section.

6.2 SUMMARY OF RESULTS AND CONCLUSIONS

The primary results and conclusions of the control system analysis and simulation program are presented below. References are made (in parenthesis) to figures, tables, or paragraphs which elaborate on each item.

Kinesthetic versus Thrust Vector Control

Marginally low control power results in poor handling qualities (Para. 6.3.2).

Vehicle simplification results in a weight reduction of 7.8 pounds (Table 6.3).

Because of inferior handling qualities and difficulties in earth simulation and training for kinesthetic control, it is recommended that kinesthetic control be dropped from consideration for the OMLFV.

Parametric Studies

Parametric shirtsleeve and spacesuit multi-axis control sensitivity studies indicated the following preferred control sensitivities (Cooper rating ≤ 3.5) in deg/sec^2/deg (Figure 6.17).

	Shirtsleeves	Spacesuit
Pitch	3	1.5
Roll	2	1.5
Yaw	0.5	0.5

TABLE 6-1

EVALUATION TECHNIQUES USED FOR VARIOUS STUDY ASPECTS

	Flight Condition and Quantitative Measure									
	5-Mile		1600-Foot			Hover			Attitude Hold	
Study Aspect	P.R.	Δ V	P.R.	W.L.	Δ V	P.R.	W.L.	Accy.	P.R.	Accy.
Parametric Studies										
Control Sensitivity			x	x		x	x	x	x	x
Max. Deflection			x	x						
Throttle Sensitivity			x	x						
Control Configuration						x	x	x		
Space Suit			x							
Final Evaluation	x	x	x	x	x	x	x	x	x	x
Augmentation Systems			x	x	x					

Notes

P.R. - Pilot (Cooper) Rating

W.L. - Work Load

Three axis attitude hold studies indicated that the simultaneous attitude hold capability for a proportional acceleration system is (para 6.4.1.5):

Pitch	±2 degrees
Roll	±2 degrees
Yaw	±2 to ±4 degrees

Control configuration studies indicated that pivot below the cg in pitch resulted in high Cooper rating (6.0) and noticeable deterioration in hover performance below a sensitivity of 1.5 deg/sec^2/deg. For all sensitivities, low pivots generally result in a slight increase in workload and slightly poorer horizontal hover performance ($\bar{r}$, $\bar{\dot{r}}$) than the other configurations. Pivots high, translating thruster and the all gimbaled configuration all showed some deterioration in hover performance below 1.0 deg/sec^2/deg but no significant preference between the three was evident, (Para. 6.4.4).

Final Evaluation

The preferred nominal control sensitivities (at thrust equal weight with 100-pound payload and tanks half full) was found to be:

Pitch	2.2 deg/sec^2 per deg	Figure 6.31 and Table 6.8
Roll	2.0 deg/sec^2 per deg	
Yaw	1.3 deg/sec^2 per deg	

These sensitivities are convenient to implement in a vehicle design.

The handling qualities for the simulated manually controlled vehicle were found to be completely acceptable and generally satisfactory. Cooper ratings by four operators ranged from 2.5 to 4.5 for all payload, propellant load and flight conditions tested (Table 6.10). Workload measurements indicated that from 20 to 50 percent of the pilot's full attention is required to fly the various conditions (Table 6.11).

The ΔV required for simulated 1600 foot flights ranged from 360 to 900 fps, with the bulk of the flights requiring 400 to 600 fps. The ΔV for 5 mile flights ranged from 1350 to 2300 fps with the bulk of the flights requiring from 1700 to 2000 fps. Considerable pilot-to-pilot variation was evident at both ranges (Table 6.12).

Maximum controller deflections of 9, 13, and 17 degrees in pitch, roll, and yaw, respectively, were used during simulated flights for combined control and trim inputs (Table 6.13). These deflections can conveniently be provided in the vehicle design.

Augmentation Studies

For both the stability augmentation and control augmentation systems, a steady-state rate command sensitivity of approximately 1.0 deg/sec per degree of controller input and a lag time constant (to build up to the steady-state rate) between 0.1 and 0.5 seconds were preferred. (Figures 6.35 through 6.38 and Paragraph 6.6.1.2.2.)

The preferred control sensitivities for use with the Apollo Block I hand controller was about two-thirds of the values preferred for the handle bar controllers and summarized above (Figure 6-41).

Direct comparison tests demonstrated the following benefits of augmentation, using the handle bar controller:

System	Cooper Rating
Manual Control	3.0 - 3.5
Control Augmentation	1.5 - 2.0
Stability Augmentation	1.0 - 1.5

Workload measurements corroborated this trend. The ΔV usage proved insensitive to the control system and controller alternatives (Figure 6.42).

The Apollo hand controller was found to be considerably less desirable than handle bars when used with manual control, but more desirable with stability augmentation (Figure 6.42). Because manual control is a backup mode to SAS, however, the handle bar controller was selected for use in either case.

A preliminary design study indicated that incorporation of a three-axis SAS would impose a weight penalty of about 30 pounds (Table 6.16). Reliability, operational, cost, and training penalties are also involved. The increased complexity of SAS was felt to outweigh the nonessential improvement in handling qualities and SAS is not recommended (Paragraph 6.6.4).

An all-mechanical control augmentation system is attractive as a potential means of obtaining most of the benefits of SAS with but a small fraction of its penalties. However, more development is required before this approach can be confidently recommended. Therefore, the unaugmented control system is recommended at the present time.

6.3 KINESTHETIC CONTROL INVESTIGATION

6.3.1 Introduction

In 1951, personnel at NACA Langley Research Center demonstrated the feasibility of controlling a small, compressed air jet supported platform, in free flight. (Reference 6) Balance is achieved by the same human reflexes used in standing, and control is achieved by leaning in the intended direction of flight. Since 1951, this feat has been repeated with varying degrees of success by other groups at Langley, (Ref. 7 and (Reference 8), Hiller, (Reference 9), Grumman, (Reference 10), North American Rockwell, (Reference 11), Bell Aerosystems, (References 12 and 13), and NASA-MSC (Reference 14).

This method is called "Kinesthetic", "balance reflex", or "body motion" control by the various researchers. Although this method of control has not been applied to any operational earth vehicle, it was considered during this study because of its potential for simplifying the lunar vehicle by elimination of engine pivots and pitch and roll control hardware.

The definition of "kinesthetic control", as compared with "thrust vector control", for purposes of this study, is shown schematically in Figure 6.1. Kinesthetic control of vehicle pitch and roll is achieved by motion of the pilot's center of gravity relative to a fixed thrust vector. The fixed thrust vector is produced by a "bolted on" engine.

Thrust vector control of vehicle pitch and roll is achieved by moving the thrust vector relative to a fixed vehicle/pilot center of gravity. The thrust vector is moved by rotation or sliding of the lift engine, or by differential throttling or firing of multiple engines.

A hybrid system which combines the best features of kinesthetic and thrust vector control was also considered. In this concept, the pilot stands on a small, low inertia platform which is pivot-mounted to the vehicle. This platform is mechanically linked to a pivoted engine. The pilot thus exercises kinesthetic control of a mass of low inertia, but effectively controls the rotation of a mass of high inertia.

Because the use of kinesthetic control would have a major influence on the vehicle configuration, it was necessary to make a decision early in the program, in order to eliminate the need for carrying two sets of configurations throughout the study. Fortunately, a considerable amount of analytical and simulation test data was available from the references cited. These data although highly subjective and qualitative, permitted an early decision to eliminate kinesthetic control from consideration for the lunar flyer, for reasons to be discussed.

Kinesthetic control handling qualities were evaluated by analyzing the individual factors which contribute to good handling qualities, and verifying by examining man-in-the-loop simulation and flight test data. The effects of kinesthetic control on

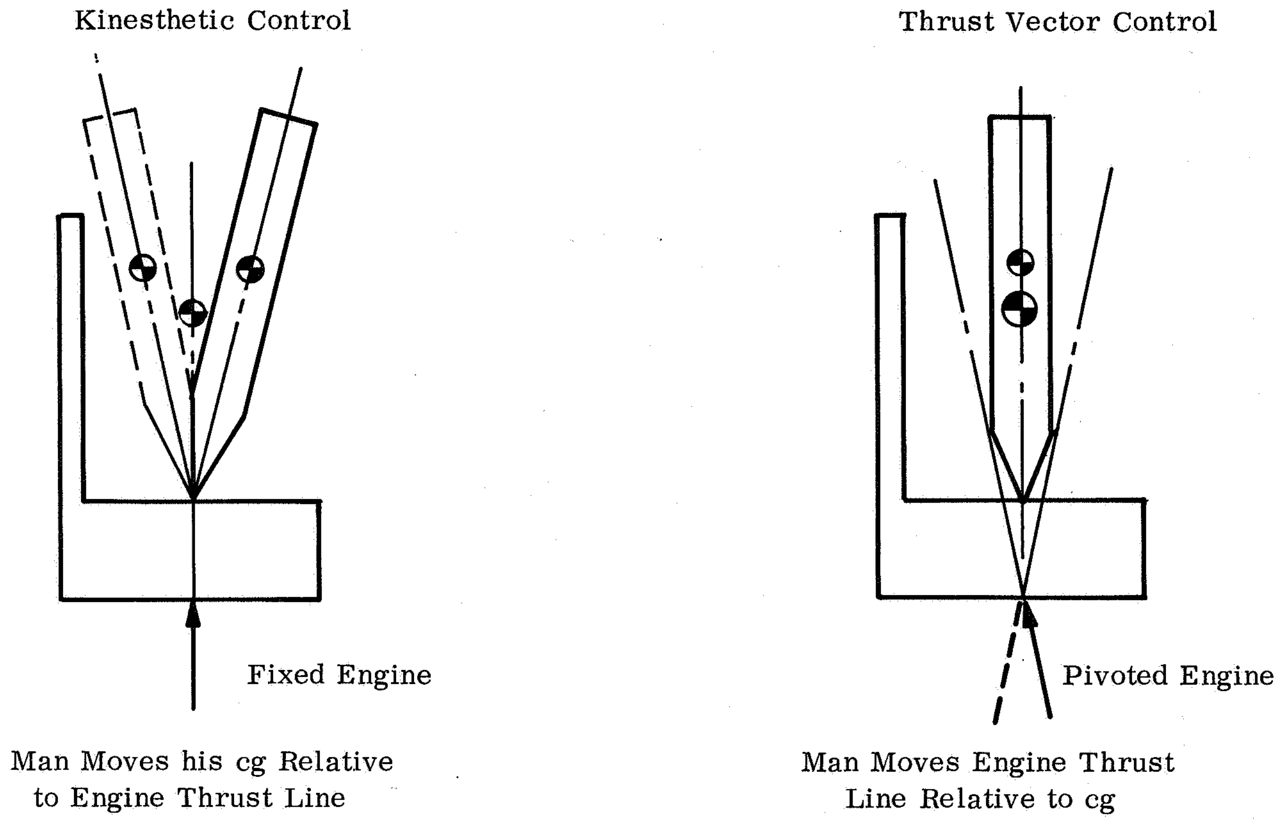


Figure 6.1. Kinesthetic and Thrust Vector Control

vehicle operation and vehicle design were also evaluated, and are discussed in the following paragraphs.

6.3.2 Handling Qualities Factors

Vehicle angular acceleration command by kinesthetic control was compared with manual thrust vector control (TVC) on the basis of maximum control power available and pilot workload, control sensitivity, vehicle response lag, and capability for incorporating control augmentation. Parameters used in the study were:

pilot, suit, and PLSS mass:	11.5 slugs (370 lb)
pilot, suit, and PLSS pitch inertia:	20 slug ft^2
pilot, suit, and PLSS cg:	3.5 ft above platform
vehicle mass range:	10-30 slugs (320-970 lb)
vehicle pitch inertia range:	30-130 slug ft^2
vehicle and pilot inertia range:	80-200 slug ft^2
vehicle cg:	variable

Maximum Control Power and Pilot Workload

It was found by test that a pressure suited pilot can move his cg a maximum of 7 inches in any direction and still keep his hand on a throttle control. At maximum vehicle plus pilot inertia of 200 slug ft^2, a vehicle acceleration of $10°/sec^2$ results. This is marginally low for precise control, as in landing, and requires gross body motion. TVC, on the other hand, can easily provide up to $50°/sec^2$ with only hand or arm motion.

Control Sensitivity

It was found by flight test that the pilot could effect small and precise inputs by small hip motion or large inputs by inputs by gross torso motion. Thus kinesthetic control provides a variable control sensitivity, at the choice of pilot. This is a desirable factor, not easily provided with TVC.

Time Lag

In practical lunar vehicle designs, the vehicle cg is below the cg of the standing pilot. With this cg relationship, the initial vehicle motion will be in reverse of the commanded direction. As the pilot leans forward, to command nose down or forward flight, the vehicle will initially pitch nose up and accelerate backward, prior to coming to the commanded nose down, forward acceleration. This will appear to the pilot as a time lag in vehicle response, of sufficient magnitude to be objectionable. This lag can be eliminated by locating the vehicle cg above the pilot cg; however it is not feasible to design a vehicle with its cg this high.

Two body equations of motions were solved by computer, considering the pilot as a rigid body, free to rotate and translate relative to the vehicle. This program

provided the vehicle time response including the initial transient reaction of the vehicle to pilot motion, and the final steady state acceleration due to the thrust vector acting on the pilot cg.

The effect of cg location is shown in Figure 6.2, for the vehicle cg 2.5 feet below and also for cg 2.5 feet above the pilot cg, for three sizes of vehicles. It will be noted that the vehicle does not move forward of the initial starting position until 2 to 3 seconds after the command, for the low cg case.

A comparable time lag comparison can be made for the TVC vehicle, where time lag is a function of the vehicle cg height in relation to the engine pivot height. **Time lag is eliminated by locating the vehicle cg below the engine pivot which is easily accomplished.**

Control Augmentation

Both the kinesthetic and TVC methods previously described are unaugmented acceleration command systems. From man-in-the-loop simulation tests, it is known that the addition of augmentation to provide rate command or rate limiting results in more pleasant handling qualities. However, the already marginal control power in the kinesthetic system limits the amount of augmentation that can be added. TVC provides sufficient control power to realize the full benefit of augmentation.

6.3.3 Simulation Results

Table 6.2 summarizes the results of manned tests of kinesthetic control conducted by various organizations, using various fixed base and free flight vehicles.

The various simulators have varying characteristics and results are subjective. which makes a direct comparison of results difficult. However, a consistent trend is noted. As vehicle inertia increases (control power decreases) handling qualities deteriorate. It has also been found that some simulators are easy to balance or maintain level, but difficult to maneuver to a precise spot, as in landing. It is often difficult to determine whether the subject was rating his ability to hover or his ability to fly a controlled flight path. It appears that the ability to hover does not deteriorate as rapidly at high vehicle inertia as does the ability to control flight path.

The subjective results of the tests summarized in Table 6.2 are plotted in Figure 6.3 in terms of "acceptable" or "unacceptable" for lunar flight, as a function of the inertia of a lunar vehicle which would give the same control power as was available in the simulation. For comparison, the expected inertia range of the one man lunar vehicles is shown. It can be seen that the average lunar vehicle inertia is far above the range judged to be acceptable based on manned simulation tests. This tends to verify the analysis which showed a marginally low control power and long time lag.

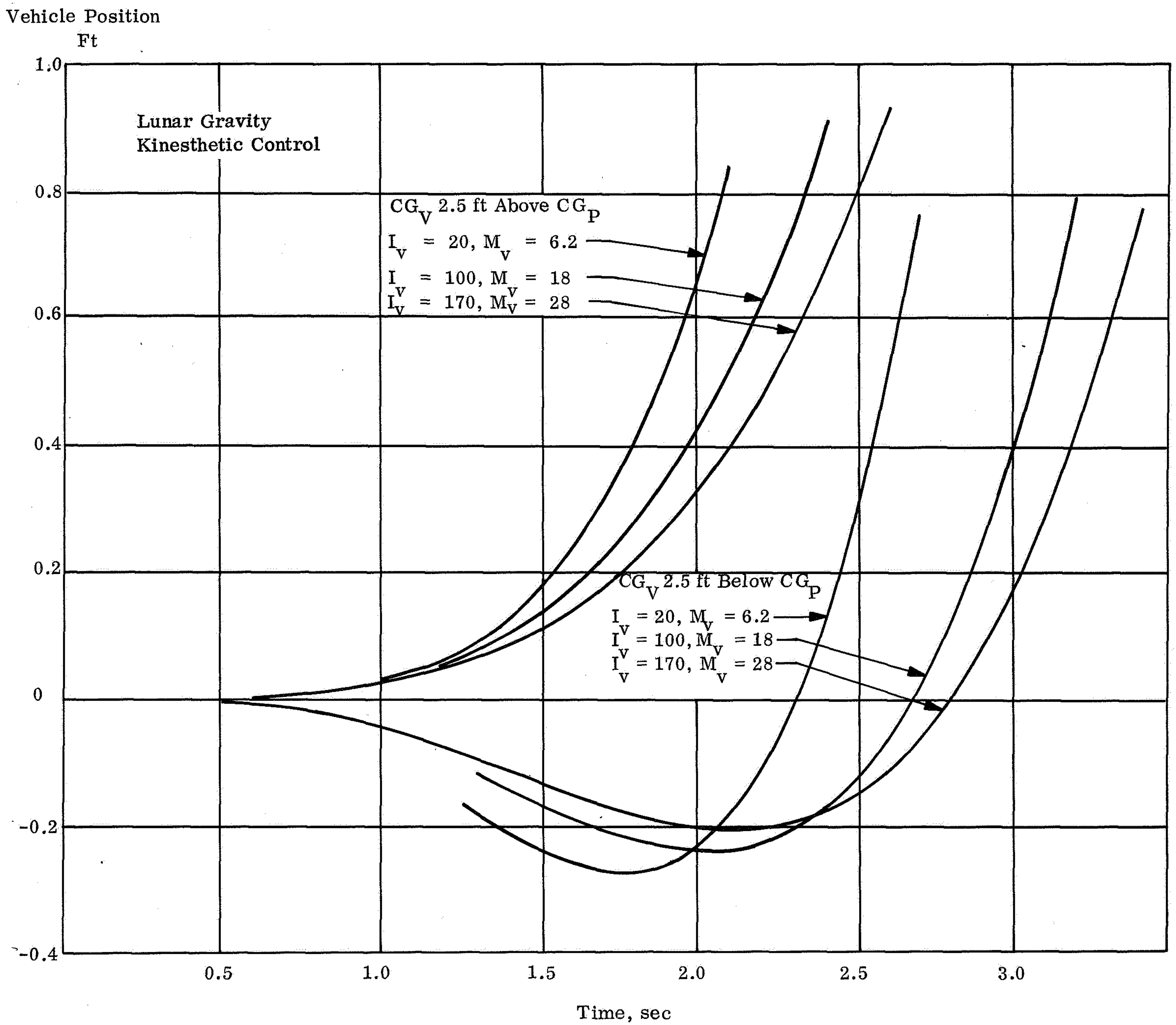


Figure 6.2. Vehicle Motion Following a Step Input Pitch Forward Command

TABLE 6.2

SIMULATION RESULTS AS OF FEBRUARY 1969

Organization	Simulator	Results: Kinesthetic	Results: TVC
1. NASA-LRC (Ref. 6)	1. Air jet supported platform in free flight, no yaw, slack safety tether, air hoses, I = 1-3 slug ft^2	Good control within limits of hoses and tether	No Test
(Ref. 7)	2. Ducted propeller supported platform in free flight, slack safety tether, I = 8 slug ft^2 (estimated)	1 subject preferred	5 subjects preferred, more precise control, faster learning
(Ref. 8)	3. "Roly-poly" platform on air bearing floor, 1 g pilot input, 1/6 g vehicle translation response, I = 300-640 slug ft^2	Easy to balance, difficult to maneuver with precision	No Test
2. Bell (Ref. 13)	1. Free flight, no tethers, H_2O_2 rocket, I = 6 slug ft^2	Good control	Good Control
(Ref. 12)	2. Free flight, slack safety tether, H_2O_2 rocket, I = 20 and 45 slug ft^2	Difficult to control	Good Control
	3. H_2O_2 rocket vehicle on 1/6 g tether, no lateral freedom, 1 g pilot input, 1/6 g vehicle response, I = 30 and 185 slug ft^2	Sluggish, difficult to maneuver with precision	Good Control
(Ref. 12)	4. H_2O_2 rocket vehicle on 1/6 g tether, no lateral freedom, pilot on 1/6 g tether, I = 30, 117, and 162 slug ft^2	Unacceptably slow response, impossible to maneuver	Good Control
	5. Platform on pitch/roll bearing, TV projected visual lunar scene computer driven, I = 145-378 slug ft^2	Hover position error increased with I, from 0.5 to 5.0 feet, difficult to maneuver	No Test
3. North American (Ref. 11)	1. Air jet supported platform in free flight, slack safety tether, air hoses, I = 40-140 slug ft^2	Unacceptable for primary mode	No Test
(Ref. 11)	2. Fixed base, TV projected visual scene, 1 g and 1/6 g	Unacceptable for primary mode	No Test
4. Grumman (Ref. 6)	1. Platform on rail, computer driven, 2 DOF, I = 1-25 slug ft^2	$<$1 slug ft^2 "good", 5-25 "barely maneuverable", $>$ 25 unmaneuverable	No Test
5. Hiller (Ref. 9)	1. Ducted propeller supported platform in free flight, slack safety tether, I = 123 slug ft^2	Could hover, but insufficient control to overcome aerodynamic moments	No Test
6. NASA-MSC (Ref. 14)	1. "Roly-poly" platform, oscilloscope display, pitch plane only, 1 g and 1/6 g, I = 20-350 slug ft^2	Data not complete	Data not complete
(Ref. 14)	2. Air bearing vehicle on sloping 1/6 g floor, pitch plane only, I = 80-360 slug ft^2	Data not complete	No Test

Note: All inertia values are stated for vehicle, not including pilot.

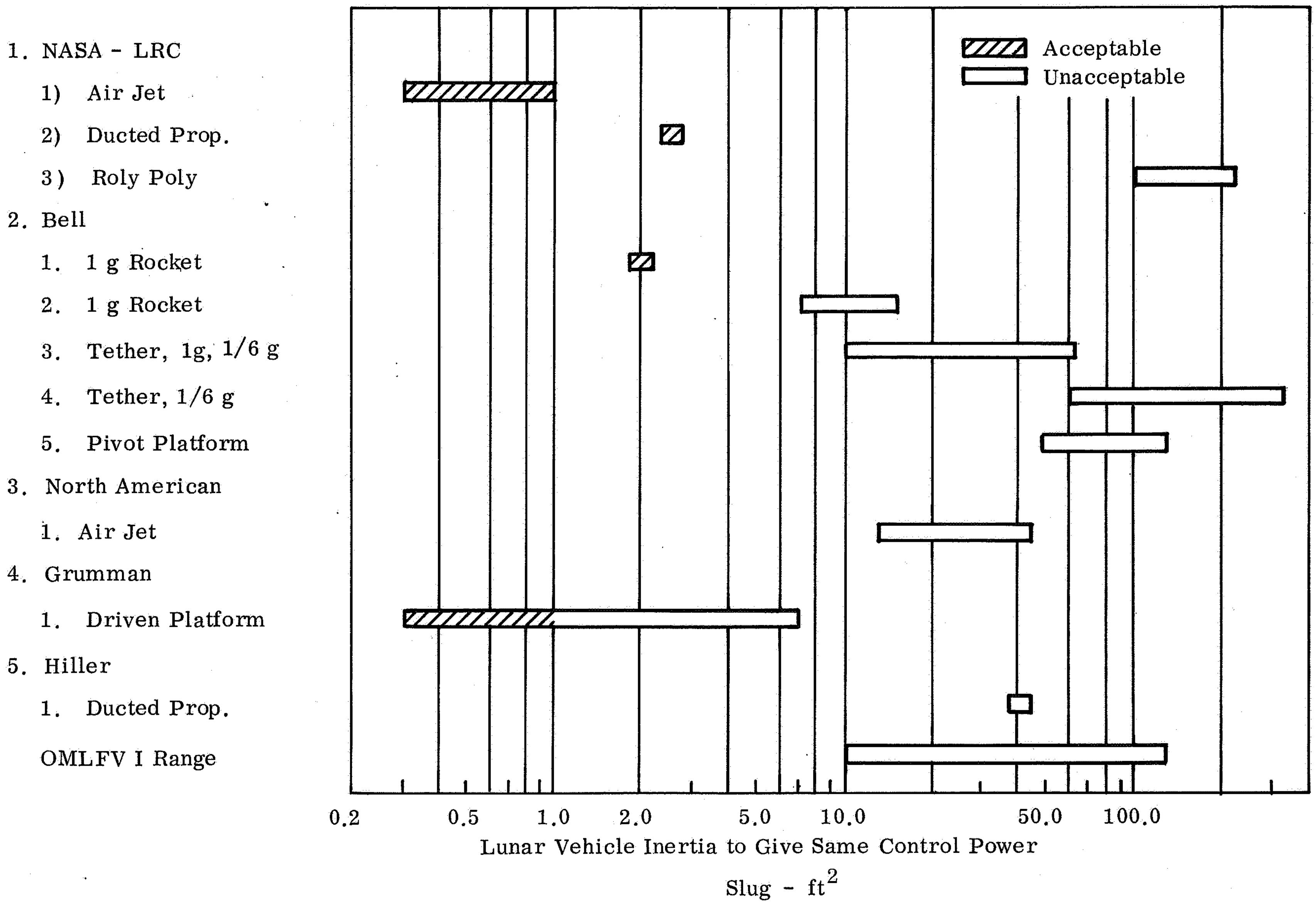


Figure 6.3. Simulation Result Summary

6.3.4 Vehicle Operation and Design Factors

Take-off and Landing on Sloping Surface

Since the kinesthetic vehicle thrust vector is in a fixed position aligned with the vehicle vertical axis, the thrust vector will not be vertical when the vehicle is standing on a sloping surface. Thus, the pilot must "blast off" at high thrust during takeoff, and must drop in at zero thrust during landing, to prevent sliding down hill with possible tip-over. In a gimballed engine vehicle, the thrust vector can be aligned approximately vertical, even though the vehicle is on a slope. The pilot can check engine operation at thrust less than weight prior to takeoff, and can "ease down" gently on landing, which is a safer mode of operation.

Takeoff and Landing on Level Surface

With kinesthetic control, the pilot will automatically trim the vehicle attitude, by shifting his body position, to compensate for vehicle cg shift due to propellant consumption. Thus, the vehicle will always be in a vertical position during hover, as prior to landing touchdown. This enhances landing visibility and capability for a gentle four leg touchdown. In a gimballed engine vehicle, the thrust vector will be vertical during hover. However, the vehicle will adjust its attitude about the engine gimbal, so that the vehicle/pilot combined cg falls on the thrust vector. Thus, although a pre-take-off trim alignment is made, the cg shift in flight due to propellant consumption will result in a small change in vehicle attitude. A TVC vehicle design consideration is to keep this angle within acceptable limits.

Downward Visibility

With kinesthetic control, the pilot cannot lean forward or sideward to get a better downward view past his feet, without putting a large control input on the vehicle. With TVC, he can lean to get a better downward view, and compensate his body cg shift by an instinctive correcting input to the TVC control system.

Ease of Simulation

In order to simulate kinesthetically controlled lunar flight, the pilot cannot be constrained relative to the vehicle. He must be free to shift the relative position between the vehicle and his body. However, a means must be provided to reduce the six times too high control input which he produces in earth gravity, if lunar gravity flight is to be simulated. Two basic methods have been developed. In one, the pilot is supported by a gimballed corset, suspended on a tether which supports 5/6 of this weight. The other 1/6 of his weight is supported by the vehicle on which he is standing. However, the complexity of the tether system, which must travel with the vehicle, but not restrain motion relative to the vehicle, introduces extraneous unwanted cues and motion limitations which impair the fidelity of simulation. In addition, the extent to which man's kinesthetic and proprioceptive senses are required, to fly by kinesthetic control, and the difficulty of transference from earth gravity cues to lunar gravity cues, are unknown.

In the second method, the man and vehicle are supported at an angle of 9° to the horizontal by a cable or low friction sloping floor system. This produces a component of earth gravity force along the vehicle/man vertical axis equal to lunar gravity force. However, this restricts motion to flight in a single plane, and places the operator at a 9° from horizontal position where he receives false visual and kinesthetic cues.

In a TVC simulator, the pilot can be strapped to the vehicle, eliminating the need for an independent pilot suspension. This simplifies the simulator and reduces the magnitude of false cues and motion limits. Also kinesthetic and proprioceptive senses are less important in a hand controlled system, so that transference from earth to lunar gravity cues should be easier.

The lower confidence in kinesthetic simulation data and reduced ability to train the astronaut on earth for lunar kinesthetic control are major obstacles to the use of kinesthetic control for the lunar flyer.

Vehicle Design Factors

In order to determine the effect of kinesthetic control on vehicle weight and complexity, a design study was made comparing kinesthetic and TVC vehicles. It was assumed that the kinesthetic vehicle must meet the requirements for payloads from 0 to 370 pounds and a propellant load of 300 pounds. It was established that the basic layout of the TVC vehicle selected in this study would also be a favorable arrangement for a kinesthetically controlled vehicle. The kinesthetic vehicle would employ two side mounted engines for reasons discussed elsewhere in this report. Yaw control is provided by a ±2° differential pivoting of these engines. This could be accomplished at lower weight and higher reliability than adding two extra yaw rockets and on-off valves. Pitch trim, to compensate for cg shifts for payload weights from 0 to 370 pounds, would have required a pilot movement of 17 inches. This is beyond his reach capability to operate the throttle and yaw controllers. The engine mount position is therefore adjustable fore and aft to permit a pre-takeoff trim adjustment dependent on payload to be carried. The pilot can compensate kinesthetically for the small in-flight cg shift due to propellant consumption. Yaw and throttle controllers are the same as used on the TVC configuration.

With kinesthetic control, the thrust vector will not move relative to the propellants in the tanks, so less anti-slosh baffling may be possible.

Table 6.3 presents the weight difference in employing kinesthetic control, based on the TVC vehicle weight presented in Section 3.0.

TABLE 6.3

KINESTHETIC VEHICLE WEIGHT DIFFERENCE

	Change for Kinesthetic Control	Weight Difference (lb)
1.	delete engine pitch pivots, mounting structure, and pitch/roll control linkage	-10.6
2.	shorten flexible lines to engines	-0.6
3.	reduce slash baffles 50%	-1.2
4.	add two rigid engine mounts	+4.0
5.	add rigid mounts for throttle and yaw controllers	+0.6
	Net Change	-7.8 pounds

In the weight analysis, no allowance was made for increase in the pilot platform area to permit pilot motion of ±7 inches in pitch and roll. Thus the actual saving could be less than the 7.8 pounds shown.

6.3.5 Hybrid Kinesthetic/TVC Vehicle

Design consideration was given to a vehicle in which the pilot stands on a small platform, pivot mounted to the main vehicle structure. This platform also carries the throttle and yaw hand controllers.

The pilot can thus kinesthetically balance this small, low inertia platform, as has been demonstrated by successful earth flights of low inertia vehicles. The vehicle lift rocket can be rigidly attached to the platform, in which case the vehicle must be loosely coupled to the platform, as by a spring/damper system to provide proper average long period vehicle attitude. The pilot can command small rapid inputs working against the spring force rather than against the higher vehicle inertia force. However, this system exhibits the same undesirable time lag as the low cg kinesthetic and low pivot TVC vehicles discussed previously. An alternate solution is to mount two pivoted engines to the main vehicle structure, mechanically linked to the pivoted pilot platform. The vehicle can be considered as a foot controlled TVC vehicle.

In either concept, no design simplification over the hand controller TVC vehicle is possible, and there is no evidence that the vehicle will be easier to fly. Therefore no dynamic analysis or simulation was conducted, and this configuration was eliminated from further consideration.

Table 6.4 is a summary of the factors which led to the choice of thrust vector control.

TABLE 6.4

COMPARISON SUMMARY

Factor	Favors KIN	Favors TVC
Maximum control power		X
Control sensitivity	X	
Time Lag		X
Control Augmentation		X
Simulation Results		X
Take-off/Landing on slope		X
Take-off/Landing on level	X	
Downward visibility		X
Ease of simulation		X
Weight reduction	X	

6.4 PARAMETRIC SIMULATOR STUDIES OF MANUAL TVC STUDIES

The parameter studies involved investigations in three categories; control parameters, vehicle control configurations, and spacesuit studies. The control parameters studied were the multi-axes control sensitivities, throttle gradient and control power (or control deflection requirements). The four vehicle configurations evaluated are summarized in Table 6.5. Configuration I was used for the control parameter and spacesuit studies.

Three basic piloted simulator tasks (short flight, hovering, and attitude hold), using a six degree of freedom motion/visual simulator with fixed vehicle mass properties (no propellant burnoff effects) and no product of inertia terms or cg trim requirements, were used for these parametric studies (Reference Appendix C). The representative vehicle properties simulated are summarized in Table 6.6. Paragraph 4 (Appendix C) summarizes the background of the simulator pilots used in the study.

TABLE 6.5

VEHICLE CONTROL CONFIGURATION STUDIED

	Pitch	Roll	Yaw
Configuration I	Pivots (above cg)	Differential Throttling	Differential Gimbaling
Configuration II	Pivots (below cg)	Differential Throttling	Differential Gimbaling
Configuration III	Translating Thrusters	Differential Throttling	Differential Gimbaling
Configuration IV	Pivots (above cg)	Pivoted above cg	Differential Gimbaling

TABLE 6.6

REPRESENTATIVE VEHICLE PARAMETERS SIMULATED FOR PARAMETRIC TVC STUDIES

Mass (M) - 28 slugs
Roll Inertia (I_{xx}) - 100 slug ft^2
Pitch Inertia (I_{yy}) - 130 slug ft^2
Yaw Inertia (I_{zz}) - 55 slug ft^2
Variable Total Thrust (T) - 50 to 300 lb
Pitch Moment Arm (ℓ_z) - 2 ft
Yaw and roll moment arm (ℓ_y-vehicle ℄ to thrust vector) 1.875 ft

6.4.1 Control Sensitivity Studies

The general method by which the pilot (Cooper) rating and workload measures are presented for the multiaxis (pitch, roll, yaw) control sensitivity studies is shown in Figure 6.4. Data plots of pitch sensitivity versus roll sensitivity for a given yaw sensitivity were constructed and compared to determine the areas of best control sensitivity. The vehicle control sensitivity variations were obtained by varying the control to thruster linkage gains (K_θ, K_ψ, and K_ϕ). The selection of the sensitivities were random and the pilot was not informed what sensitivities were being used. The pilot would perform the flight task (possibly several times) for pilot rating and performance data (in the case of the hovering task). Then the same flight task was repeated using the secondary task (Paragraph 3, Appendix C). Thus, secondary task results were obtained independently of Cooper rating and performance data eliminating the possibility of small degradations in the performance data which the pilot may inadvertently allow in order to operate the secondary task.

A portion of the Cooper rating definitions descriptive of handling qualities is presented below. (A complete Cooper rating description is presented in Appendix C):

<table>
<tr><th colspan="3">Description</th><th>Rating</th></tr>
<tr><td rowspan="5">Acceptable to Pilot
*Pilot compensation, if required to achieve acceptable performance in task is feasible.
*May have deficiencies for which pilot desires improvement, but adequate for task or flight phase.</td><td rowspan="3">Satisfactory to Pilot
*Meets all demands and expectations.
*Clearly adequate for the task or flight phase.
*Good enough without improvement.</td><td>*Excellent, highly desirable</td><td>1</td></tr>
<tr><td>*Good, pleasant, well behaved</td><td>2</td></tr>
<tr><td>*Fair. Some mildly unpleasant characteristics.
*Good enough for task or flight phase without improvement.</td><td>3</td></tr>
<tr><td rowspan="2">Unsatisfactory to Pilot
*Reluctantly acceptable.
*Deficiencies which warrant improvement.
*Performance adequate for task or flight phase with feasible pilot compensation.</td><td>*Some minor but annoying deficiencies.
*Effect on performance is easily compensated for by pilot.
*Improvement requested.</td><td>4</td></tr>
<tr><td>*Moderately objectionable deficiencies.
*Reasonable performance requires considerable pilot compensation.
*Improvement is needed.</td><td>5</td></tr>
</table>

Description			Rating
		*Very objectionable deficiencies *Requires best available pilot compensation to achieve acceptable performance. *Major improvements are needed.	6
Unacceptable to Pilot			7-9

These definitions were used in the evaluation of the configurations throughout the study. It should be noted that Cooper ratings are most useful as a tool for comparisons between configurations where relative ratings are important and absolute magnitude less important. Further, in many instances, "real world" flying has been found to be easier than simulator flying because of the absence of certain cues in simulated flight.

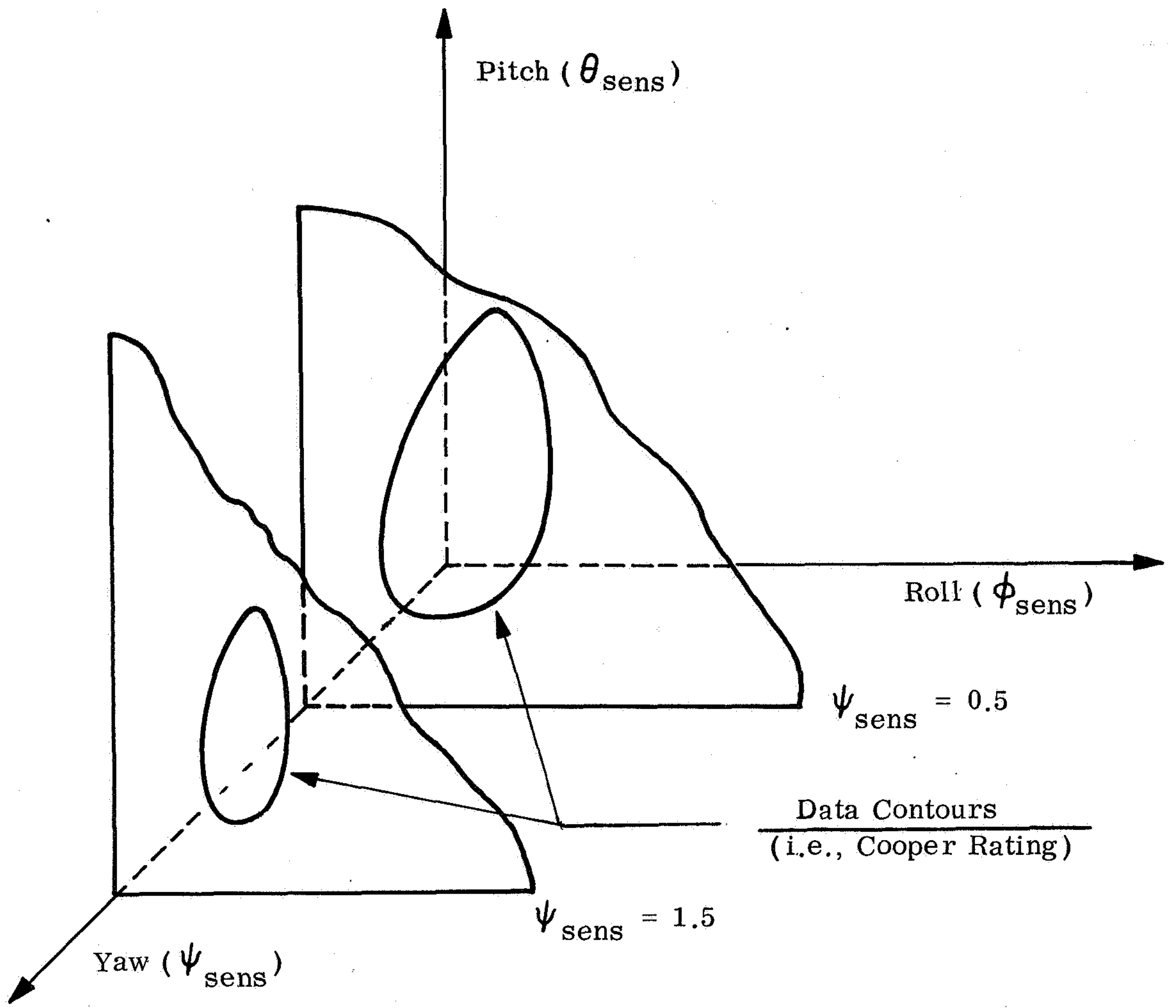
The shirt sleeve and spacesuit studies using the 1600 ft flight task were conducted to establish the most acceptable range of control sensitivies. This range of sensitivities was then evaluated for hover and attitude hold capability.

6.4.1.1 1600 ft Flight - Shirt Sleeves

Pilot (Cooper) rating data for four yaw sensitivity planes (ψ_{sens}= 0.5, 1.5, 3.0, and 5.0 deg/sec^2/deg) are shown in Figures 6.5 and 6.6 for pilot A. The boundaries of 3.5 and 6.5 shown designate the significant boundaries indicated above.

Data for two types of controllers are presented: (1) with underarm roll bars and (2) without underarm roll bars. An apparent increase in the area of the $\leq$ 3.5 cooper rating boundary was noted when the underarm bars were removed (Figure 6.5). Some of this increase was probably due to learning (Note: the initial flights were done with underarm bars). However, all pilots used finer control (smaller inputs) without the underarm roll bars and in general expressed a preference for this configuration so long as they were tightly strapped to the vehicle. The data for the yaw sensitivity planes of 3.0 and 5.0 (Figure 6.6) show no definite area for which the pilot consistently rated $\leq$ 3.5, but at the same time no unacceptable ratings $\geq$ 6.5 for controllers without underarm roll bars was obtained (generally range between 3.0 to 4.5). Note again the significant improvement over the earlier pilot ratings which were obtained with the underarm roll bars. Figure 6.7 presents pilot rating data obtained from pilot B for the yaw sensitivity plane of 0.5 deg/sec^2/deg.

Figures 6.8 through 6.10 present the corresponding secondary task data for the same operating points discussed above. The area enclosed by the 50 percent boundary



Notes: 1. All control sensitivities (θ_{sens}, ϕ_{sens}, and ψ_{sens}) are given in in deg/sec^2/deg of controller deflection and based on T = W

2. Config I in Table 6.5 used for multiaxis control sensitivity studies:

Pitch: $\theta_{sens} = \dfrac{T \ell_z K_\theta}{I_{yy}}$ Yaw: $\psi_{sens} = \dfrac{T \ell_y K_\psi}{I_{zz}}$ Roll: $\phi_{sens} = \dfrac{K_\phi \ell_y (57.3)}{I_{xx}}$

K_θ, K_ψ - Controller to thruster linkage gain (deg/deg)

K_ϕ - Total differential thrust per degree of controller (lb/deg)

Figure 6.4. General Presentation of Evaluation Data for Multi-Axis Control Sensitivity Studies

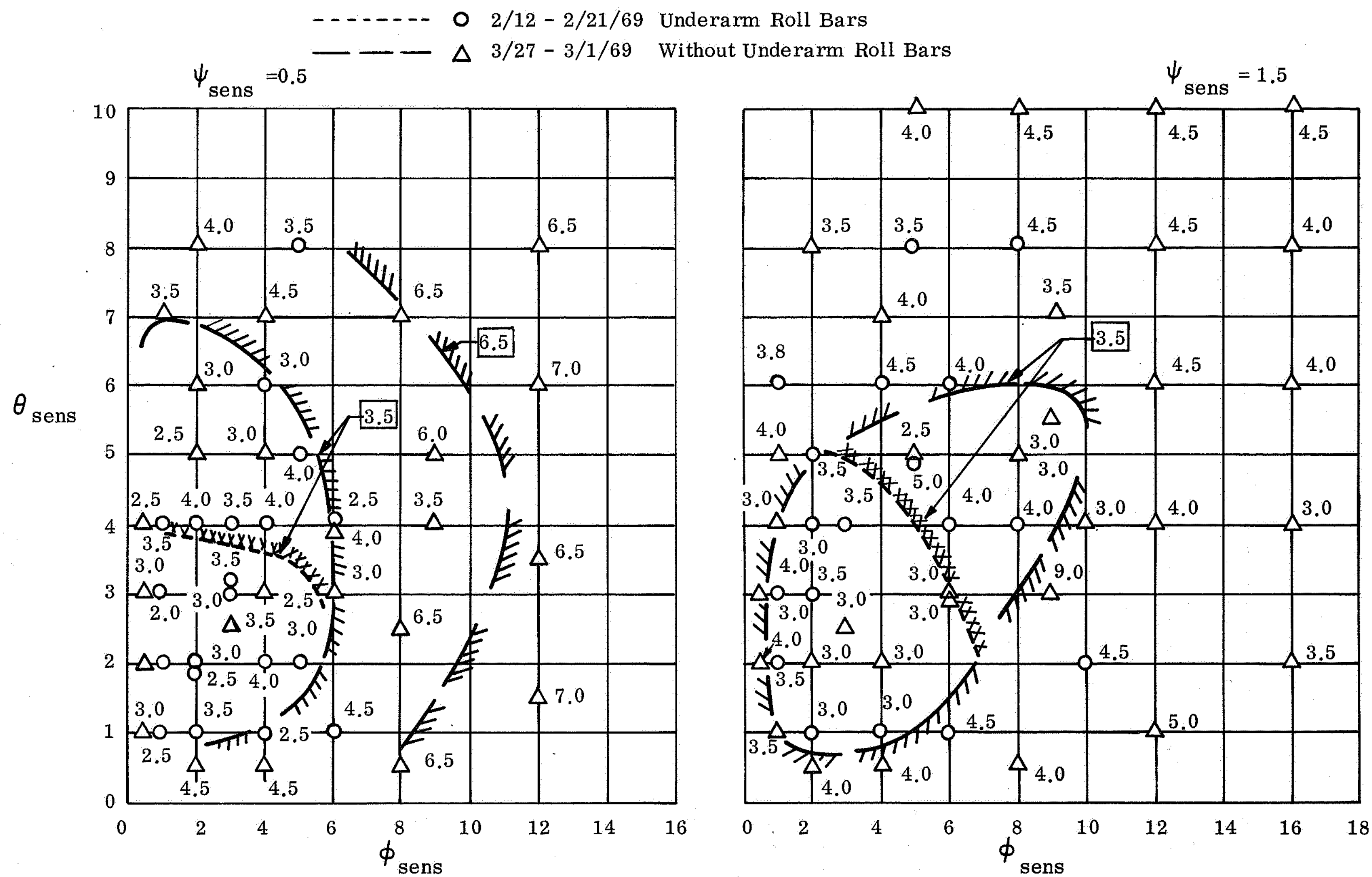


Figure 6.5. Pilot A — Task: Short Flight — Configuration No. 1 — Cooper Rating

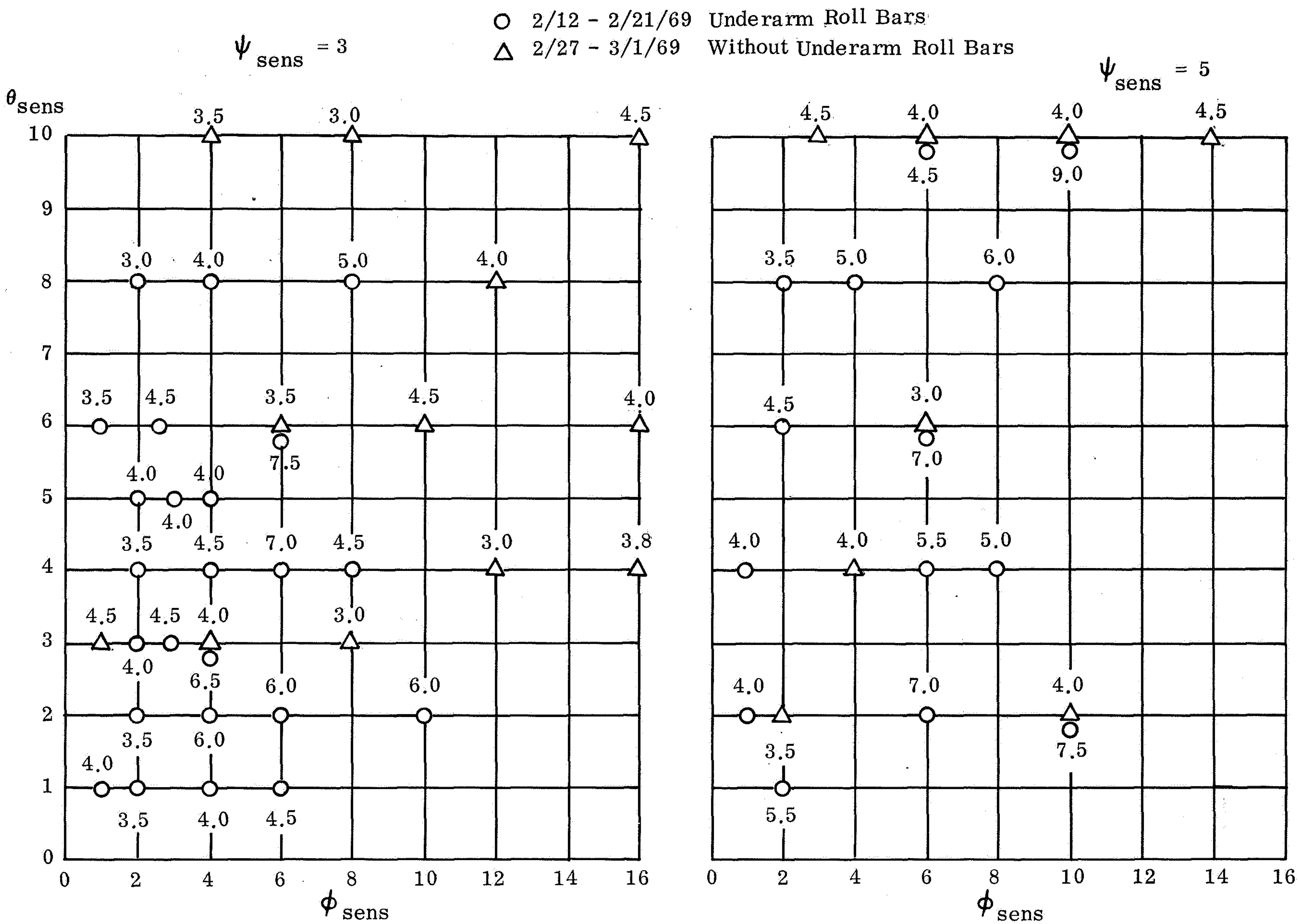


Figure 6.6. Pilot A — Task: Short Flight — Configuration No. 1 — Cooper Rating

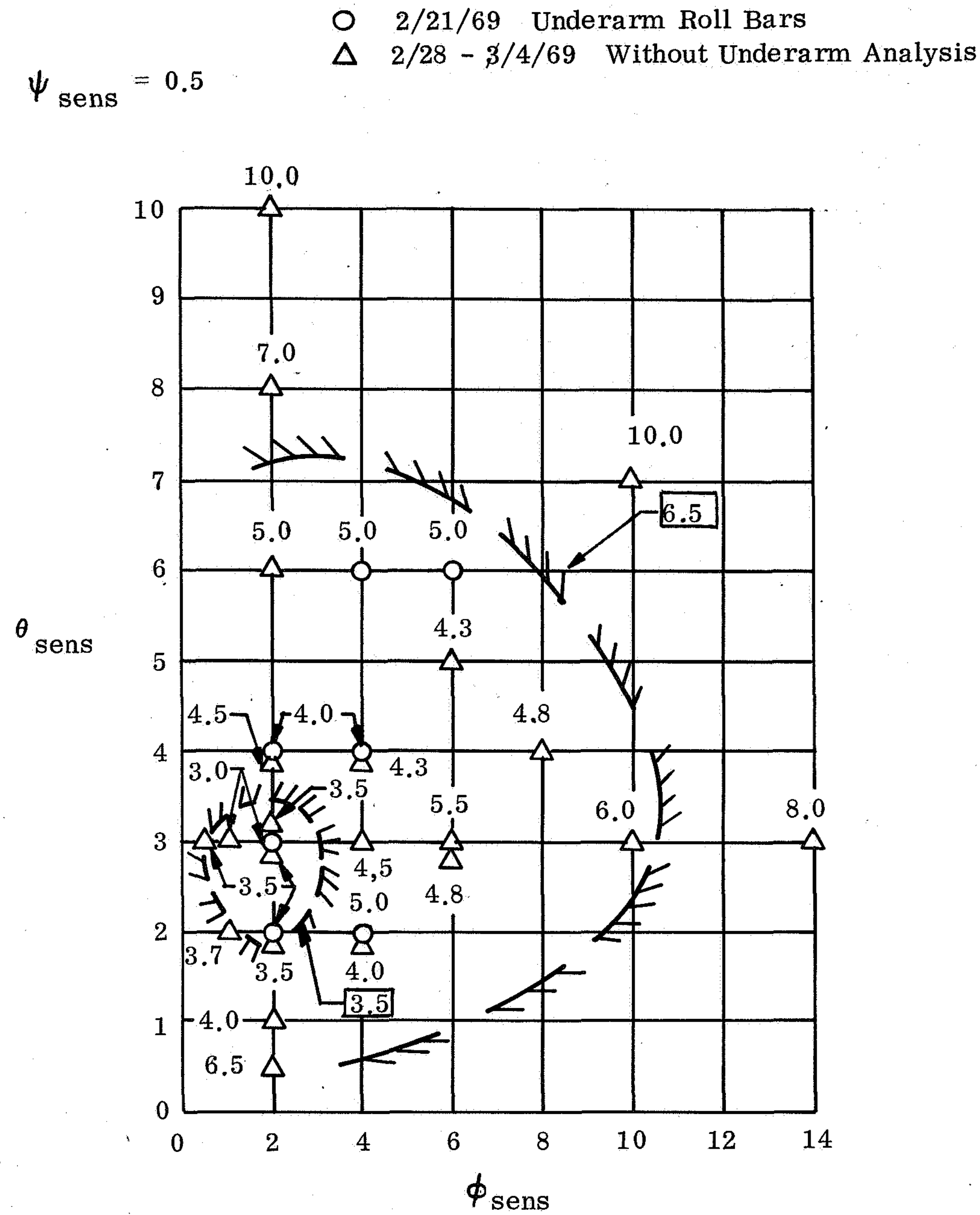


Figure 6.7. Pilot B — Task: Short Flight — Configuration No. 1 — Cooper Rating

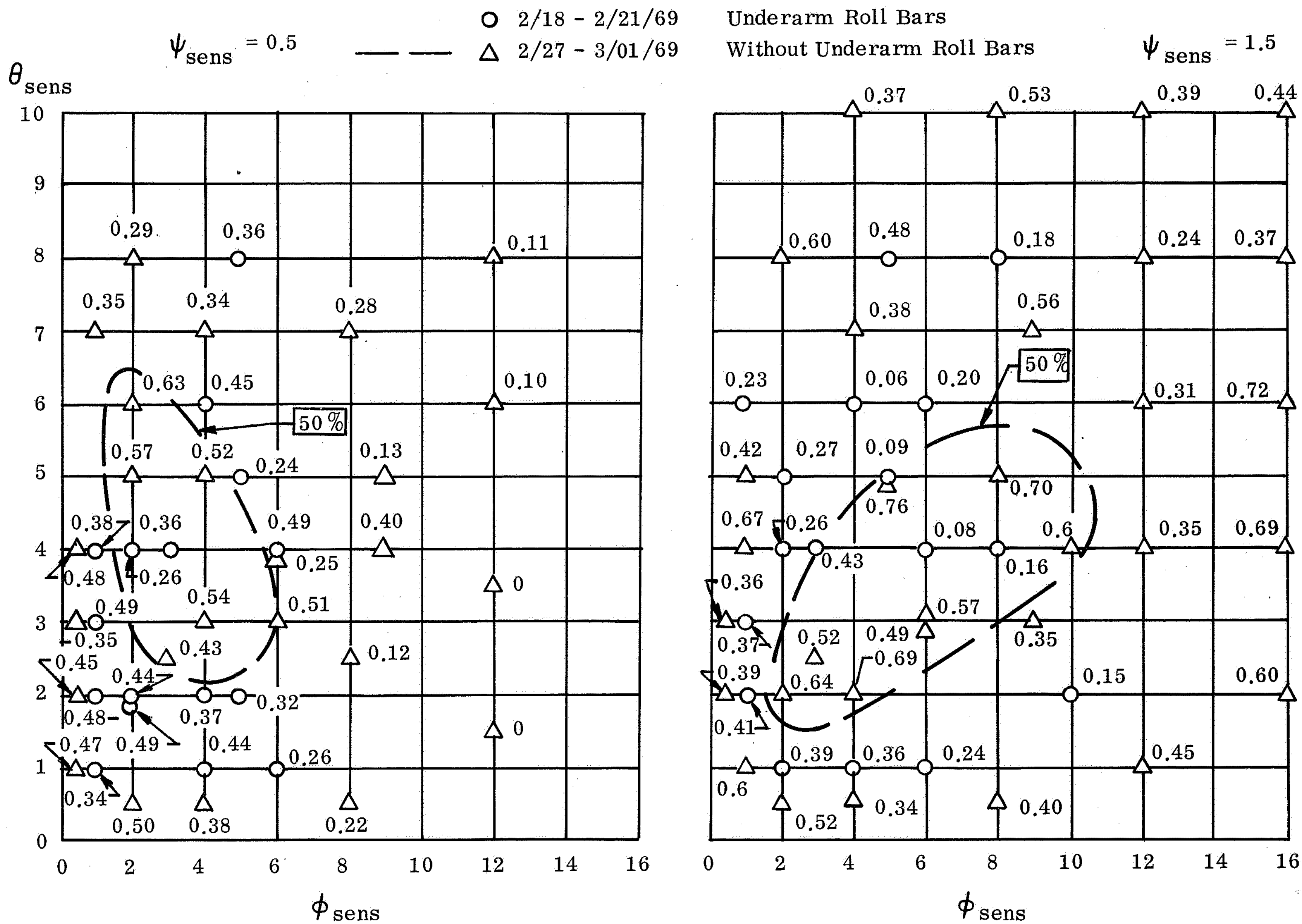


Figure 6.8. Pilot A - Task: Short Flight - Configuration No. 1 - sec. Task $= \frac{\text{resp/sec - flt}}{\text{resp/sec - calib}}$

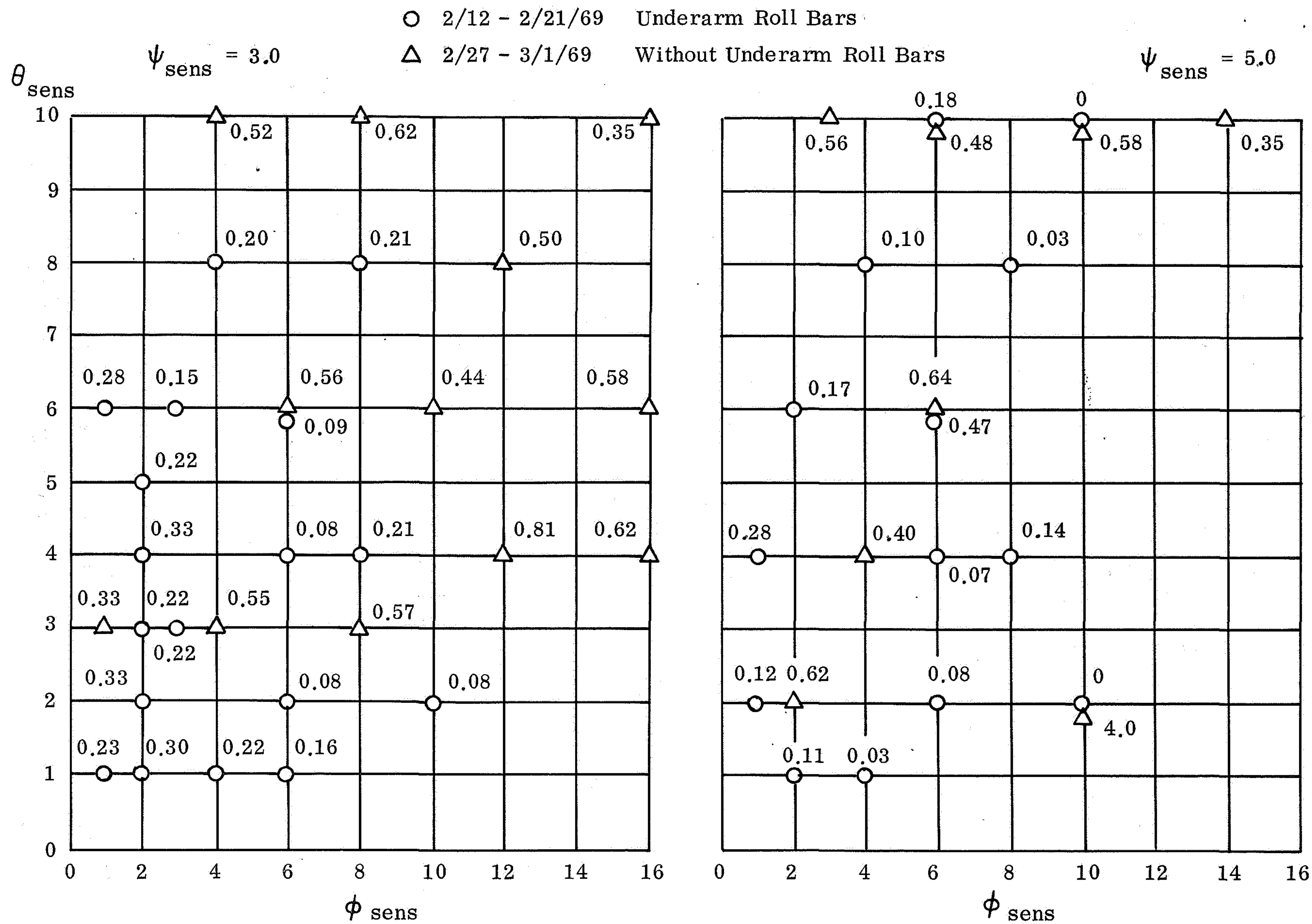


Figure 6.9. Pilot A - Task: Short Flight - Configuration No. 1 -

$$\text{Sec. Task} = \frac{\text{Resp/sec/Flt}}{\text{Resp/sec/Calib.}}$$

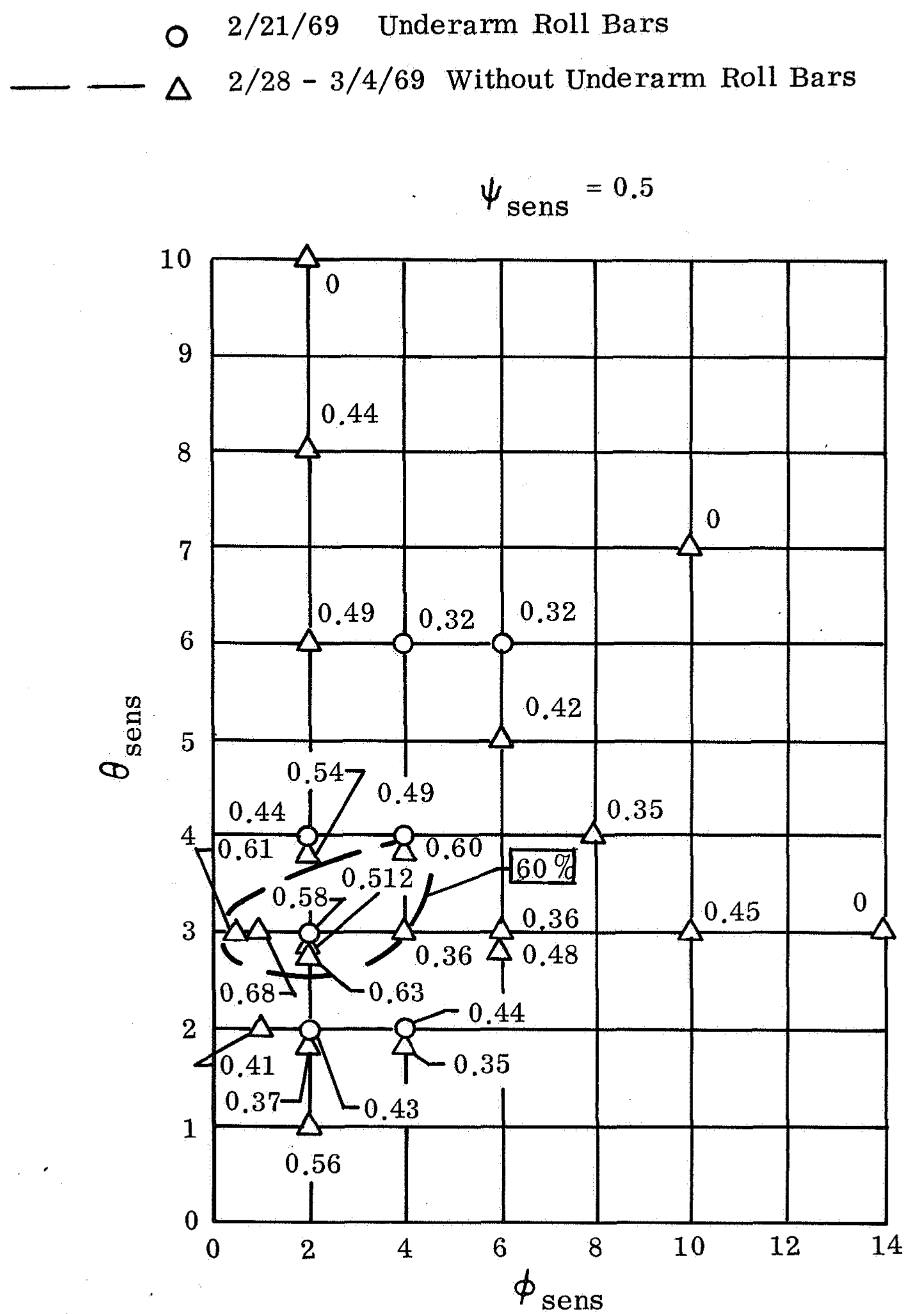


Figure 6.10. Pilot B - Task: Short Flight Configuration No. 1

$$\text{Sec Task } \frac{\text{resp/sec - flt}}{\text{resp/sec - calib}}$$

shown in Figure 6.8 indicates the general range of control sensitivities which permitted pilot A to perform the secondary task (while flying) at a response rate greater than 50 percent of the calibrated response rate (operation of secondary task only). Figure 6.11 directly compares the pilot rating with the secondary task results and indicates that in general when the response rate was 50 percent (or more) of the calibrated response rate the Cooper rating was 3.5 or less. A similar boundary is shown for pilot B (Figure 6.10), but for a 60% performance level as indicated by the comparison of pilot rating and secondary task results of Figure 6.12. Statistical analysis of the data presented in Figure 6.11 indicated a significant relationship between the secondary task performance and Cooper rating. This correlation was observed throughout the study.

6.4.1.2 1600 ft Flight-Spacesuited Subjects

The pilot rating results for spacesuited flights conducted on the simulator with pilots B and C are shown in Figure 6.13. The handle bar controller used was originally designed for shirt sleeves. The size and shape of handle grips (yaw and throttle), and the general dimensions were not optimum for comfortable reach and spread of arms; all causing interface problems with the spacesuit. However, despite the non-optimum controller design both pilots were able to fly the simulator and both pilots found the handling qualities satsifactory ($\leq$3.5 boundary) for the sensitivities shown.

Further, the amount of spacesuit simulation time for familiarization learning was short (approximately 2 hrs/day for one week). It is expected that as more experience with spacesuit was obtained, the satisfactory control sensitivity would increase and include higher control sensitivities.

6.4.1.3 Hover Task-Shirt Sleeves

A hovering task which started with a 100 ft approach was performed with pilot B. The Cooper rating results and secondary task results are summarized in Figures 6.14 and 6.15.

For additional hover performance refer to Paragraph 6.4.4 which evaluates all four control configurations (Table 6.5) in hover and presents Cooper rating, workload, average position errors, and average velocities observed.

6.4.1.4 Summary of Performance Boundaries

Figure 6.16 summarizes the Cooper rating and the secondary task boundaries of pilot A for the yaw sensitivity planes of 0.5 and 1.5 deg/sec^2/deg (Figures 6.5 and 6.8). The comparison of the 50 percent secondary task boundary established from Figure 6.11 shows good correlation with the 3.5 boundary.

Figure 6.17 summarizes the significant boundaries for pilots A,B and C. Good correlation exists between pilots A and B for the $\leq$ 6.5 boundary. The 3.5 boundary for pilot A is larger than that for pilot B; however, two factors are worth noting: (1) learning curve - Pilot A had far more experience on the simulator than pilot B. Past experience has shown that as experience is gained the area of acceptability generally increases to include a wider range of sensitivities, (2) pilot A consistently

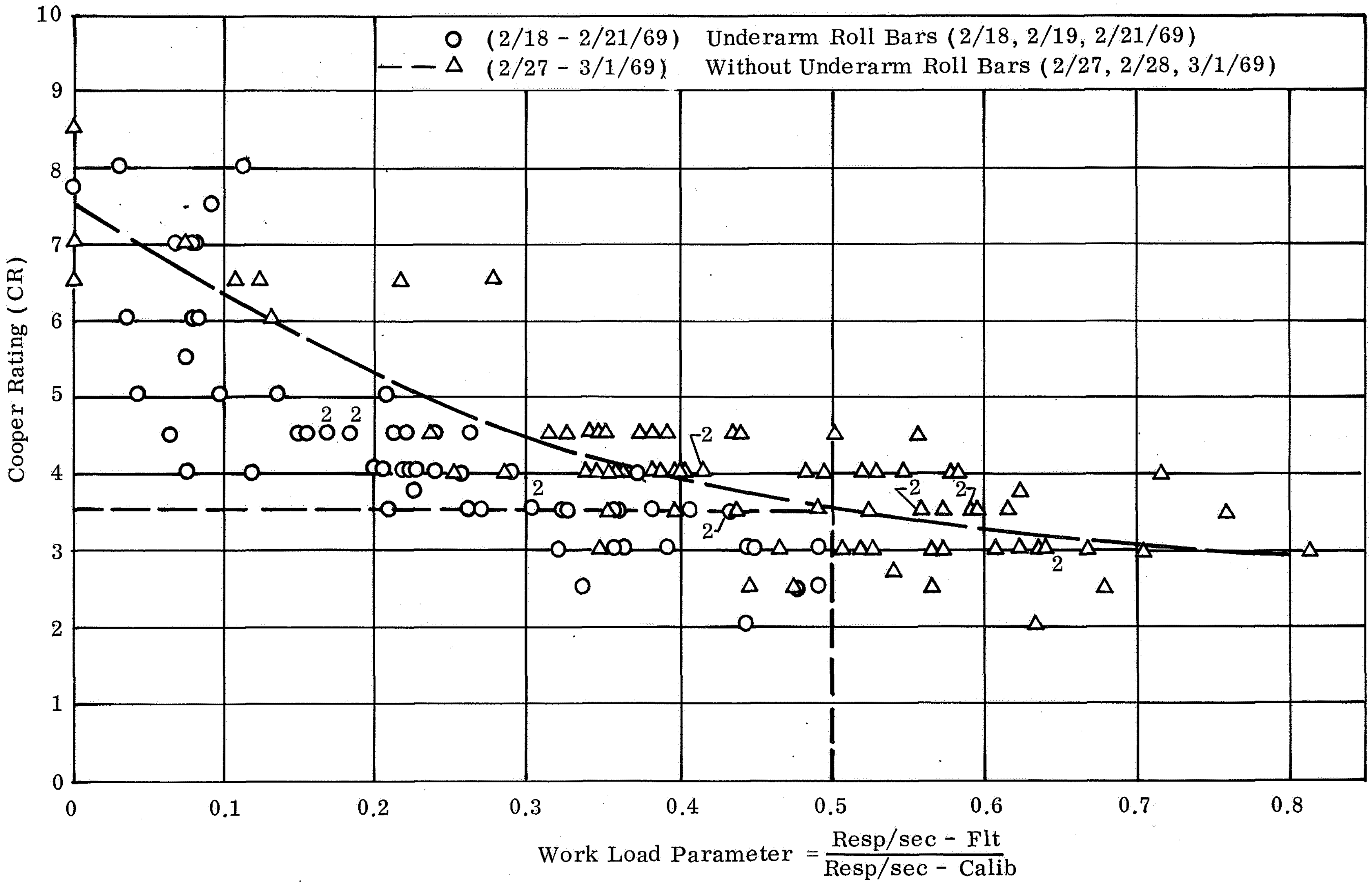


Figure 6.11. Engineer: Pilot A - Task: Short Flight - Control Mode Configuration No. 1

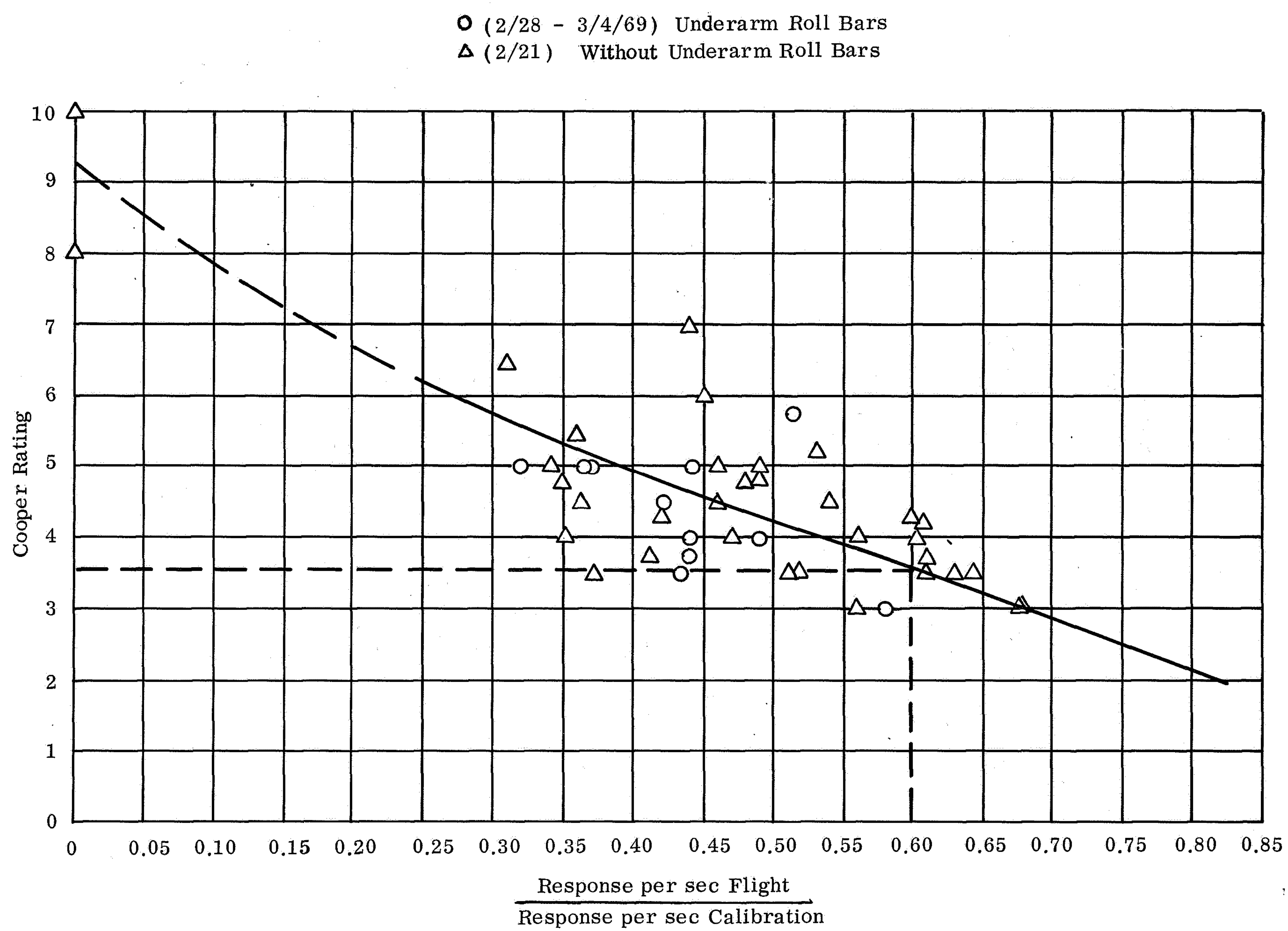


Figure 6.12. Pilot A - Task: Short Flight Configuration No. 1 Cooper Rating versus Secondary Task

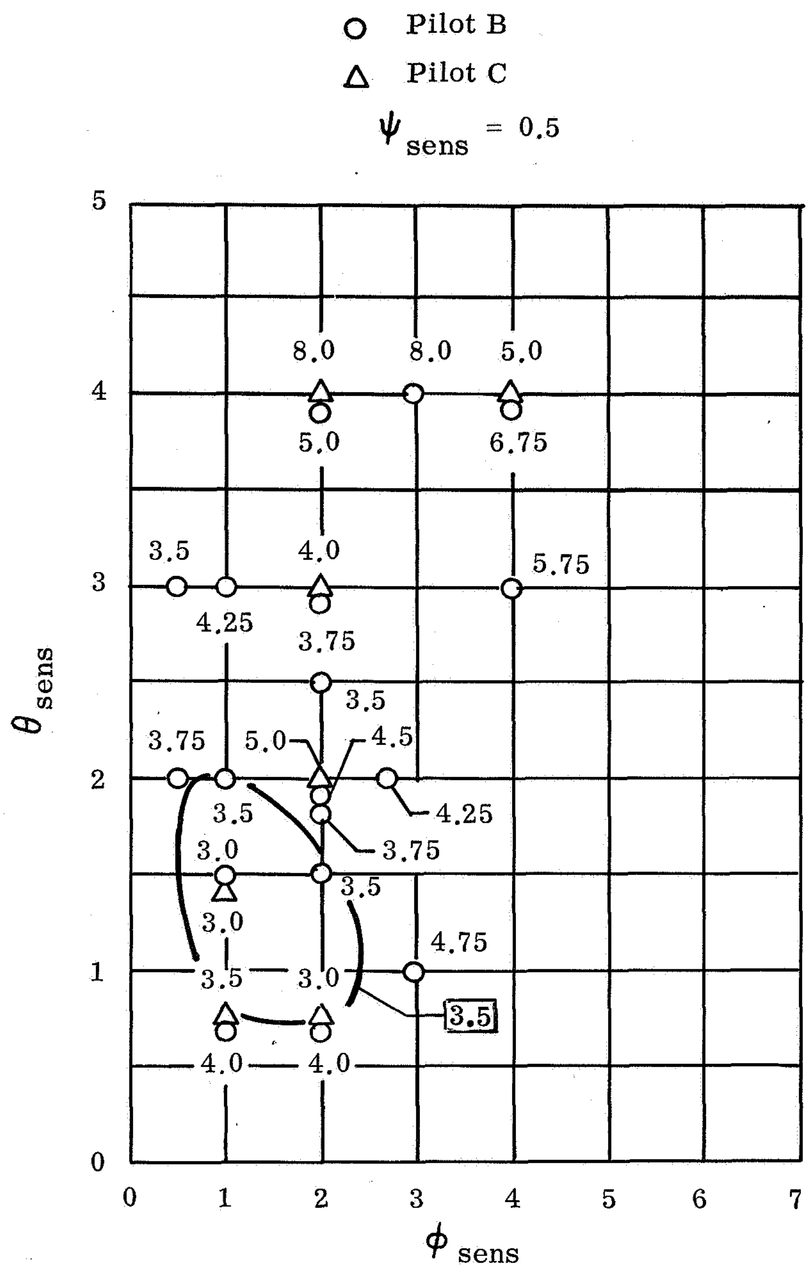


Figure 6.13. Spacesuit Studies-Task: Short Flight-Configuration No. 1-Cooper Rating

O Without Underarm Roll Bars

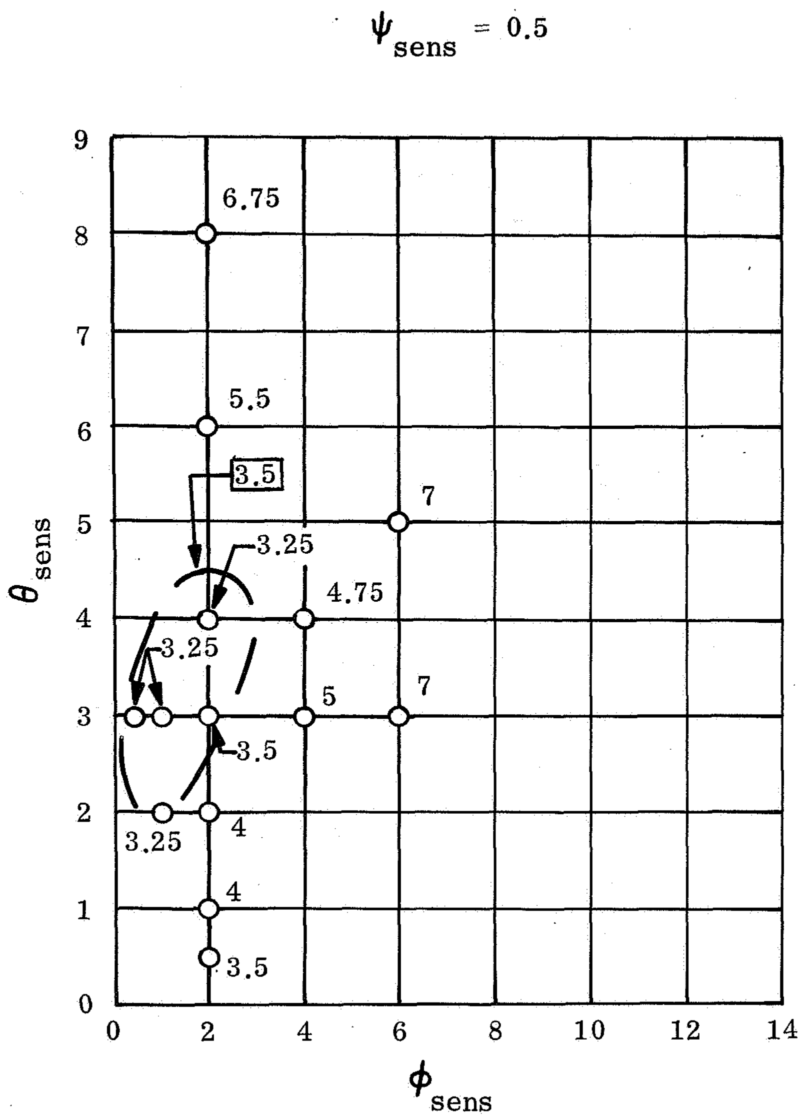


Figure 6.14 Pilot B-Task:Hover-Configuration No. 1- Cooper Rating

O Without Underarm Roll Bars

$\psi_{sens} = 0.5$

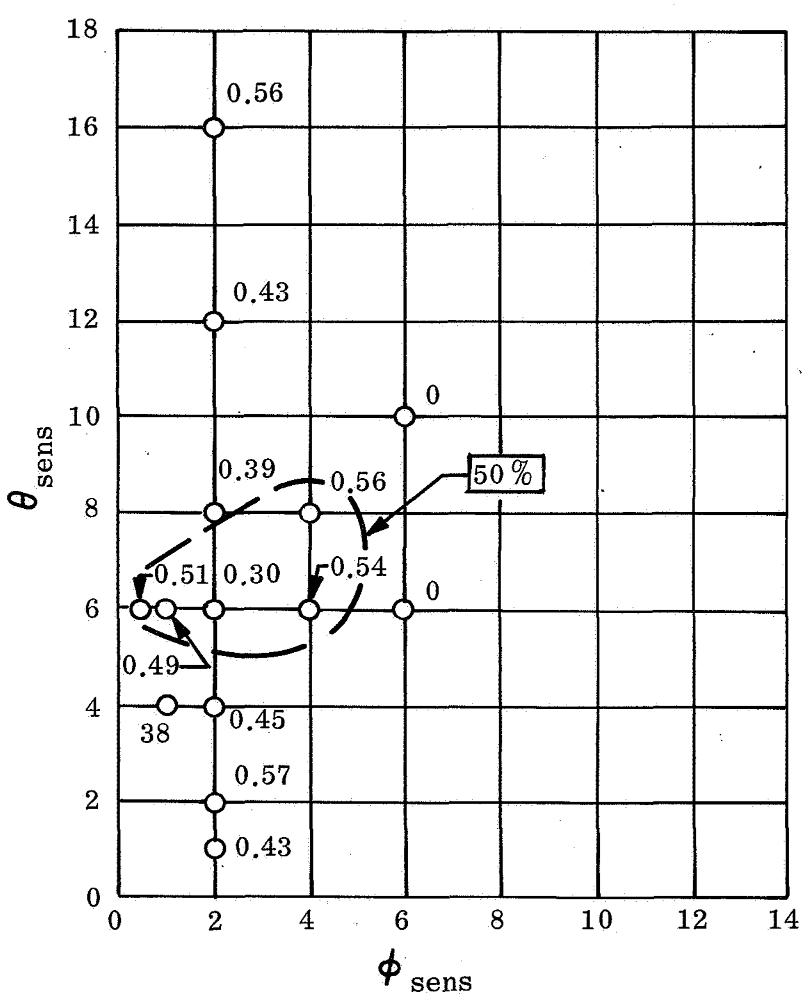


Figure 6.15. Pilot B - Task: Hover - Configuration No. 1—

$$\text{sec task} = \frac{\text{resp/sec - ft}}{\text{resp/sec - calib}}$$

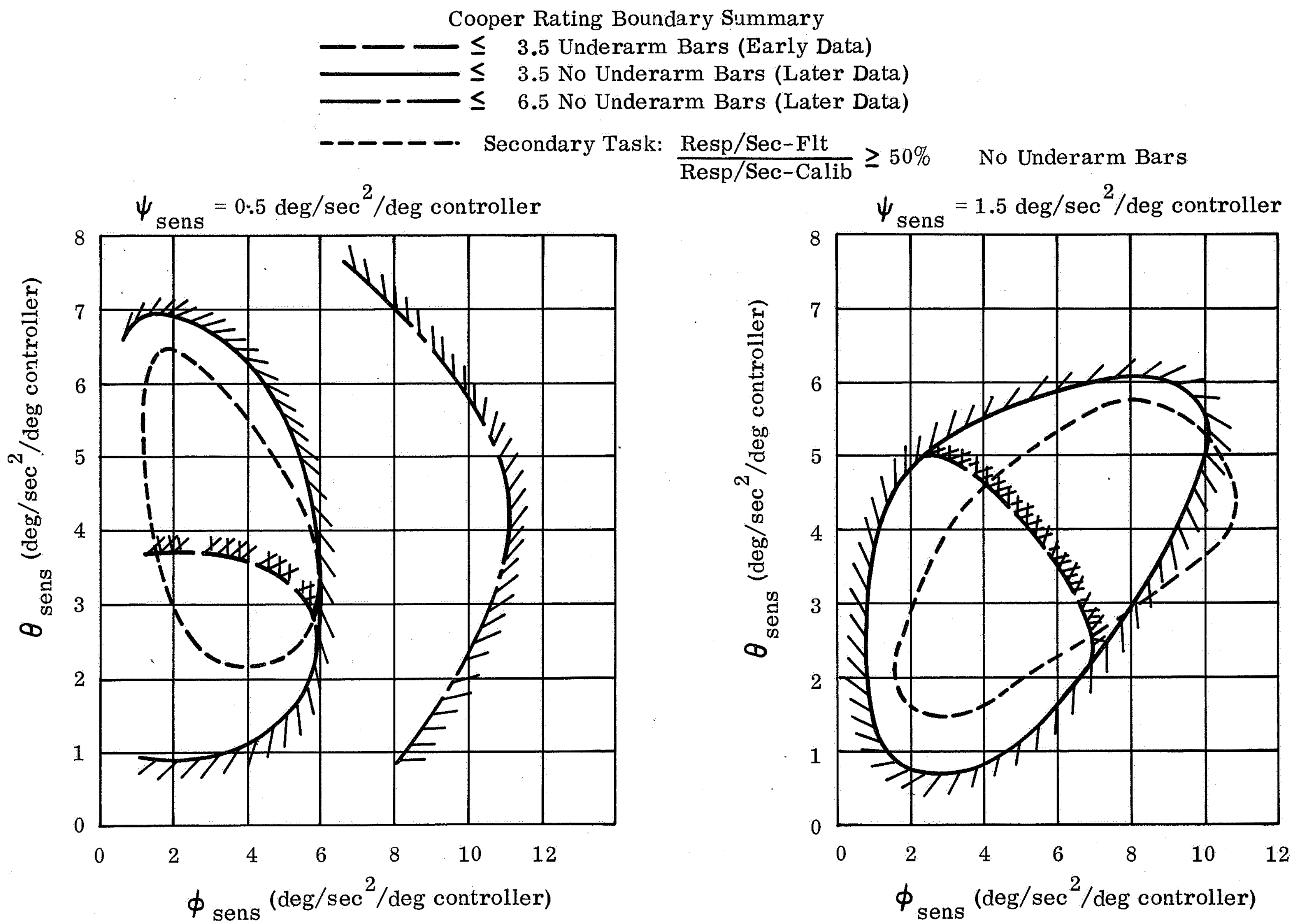


Figure 6.16. Pilot A — Task: Short Flight — Control Mode Configuration No. 1

Pitch = Pivot High
Yaw = Differential Pivoting
Roll = Differential Throttling

Yaw Sensitivity = 0.5
(degs/sec^2/deg Controller)

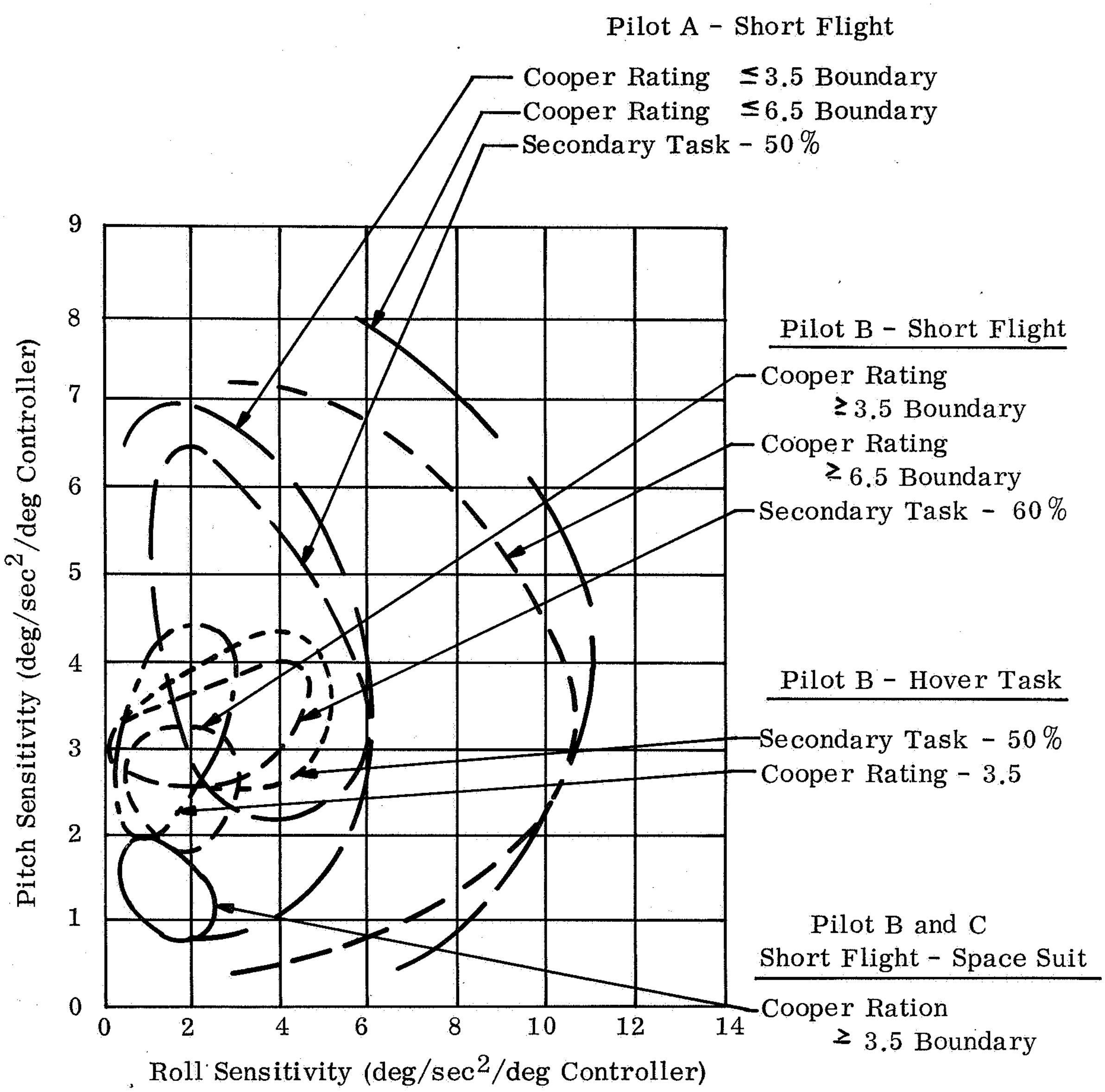


Figure 6.17. Summary of Pilot Rating Boundaries

rated the sensitivities within the indicated 3.5 boundaries of pilot B as 3.5 or better and indicated a preference along with pilot B for sensitivities as follows:

Pitch (T = W) = 2.0 to 3.0 deg/sec^2/deg
Yaw (T = W) = 0.5 deg/sec^2/deg
Roll (independent of T) = 2 deg/sec^2/deg of thrust)

Note the $\leq$ 3.5 Cooper rating boundary obtained from the preliminary spacesuit studies with pilot B and C indicated a lower pitch sensitivity was preferred. Again it is expected that with the combination of additional experience and an optimized controller (present controller designed for shirt sleeve operation caused interface problems with spacesuit) this area of acceptable sensitivities would increase.

6.4.1.5 Attitude Hold Task

An attitude hold task was performed to determine the effect of control sensitivities on the ability of the pilot to hold attitude. Since the pitch and yaw control sensitivities of Configuration I are thrust sensitive while the roll sensitivity remains relatively constant, (only effected by roll inertia changes), the attitude hold task was performed for simultaneous variations in pitch and yaw control sensitivities with a constant roll sensitivity of 2.0 deg/sec^2/deg (based on best results of Paragraph 6.4.1.4).

Using meter type attitude indicators in pitch, roll and yaw, the pilots task was to pitch forward in the simulator to +40 degrees and hold attitude for 30 seconds, return to zero pitch attitude and hold for 30 seconds, then pitch back 40 degrees and hold for 30 seconds. The roll and yaw attitudes were held to zero throughout the pitch maneuver. The results are shown in Figures 6.18 through 6.21. The shaded areas indicates the spread in the maximum (half amplitude) attitude errors observed for two different runs. The spread associated with the + and - 40 degrees pitch attitude were found to be essentially the same. A significant reduction in the attitude hold errors occurred for the attitude hold task at zero degrees pitch. The difference is attributed to the presence of gravity, which, when at a vehicle attitude of ±40 degrees attitude, resulted in a tendency for the pilot to support himself in part with the hand controllers; (even though he was strapped in), thus resulting in undesirable control inputs. Since such an effect does not exist in actual flight, the zero attitude results obtained are believed to be the most realistic evaluation of the pilot attitude hold capability which is on the order of ±1 to 2 degrees for pitch and roll and ±2 to 4 degrees in yaw.

6.4.2 Throttle Gradient

For the best pitch, roll, and yaw sensitivities (3.0, 2.0, and 0.5 deg/sec^2/deg respectively) indicated in Paragraph 6.4.14 linear throttle gradients of 4 to 17 lb/deg for a thrust range of 40 lb (T_{min}) to 300 lb (T_{max}) were evaluated using the 1600 ft flight. Pilot evaluation and secondary tasks results indicated that all gradients studied were acceptable (Figure 6.22).

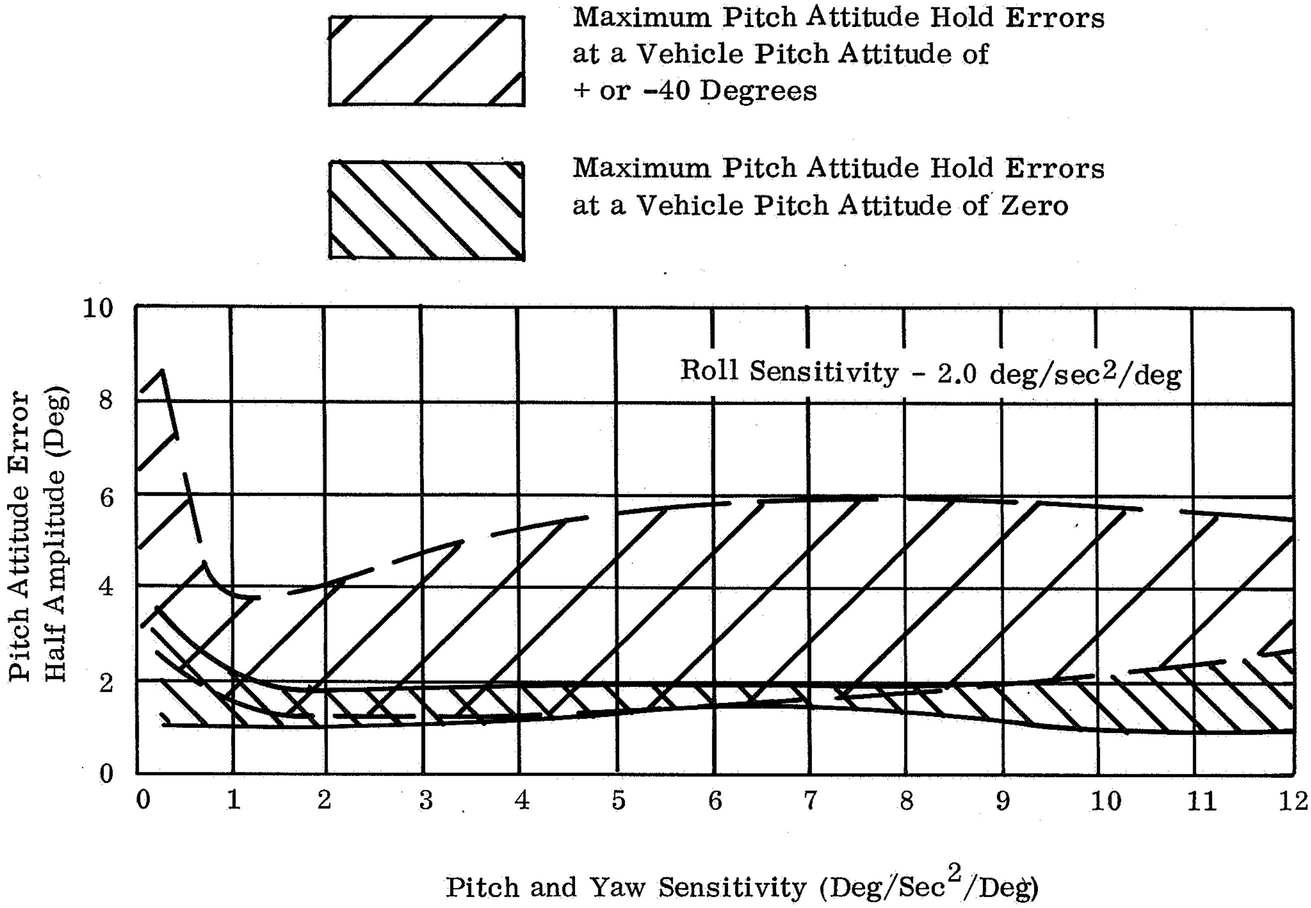


Figure 6.18. Pitch Attitude Hold Error - Pilot A

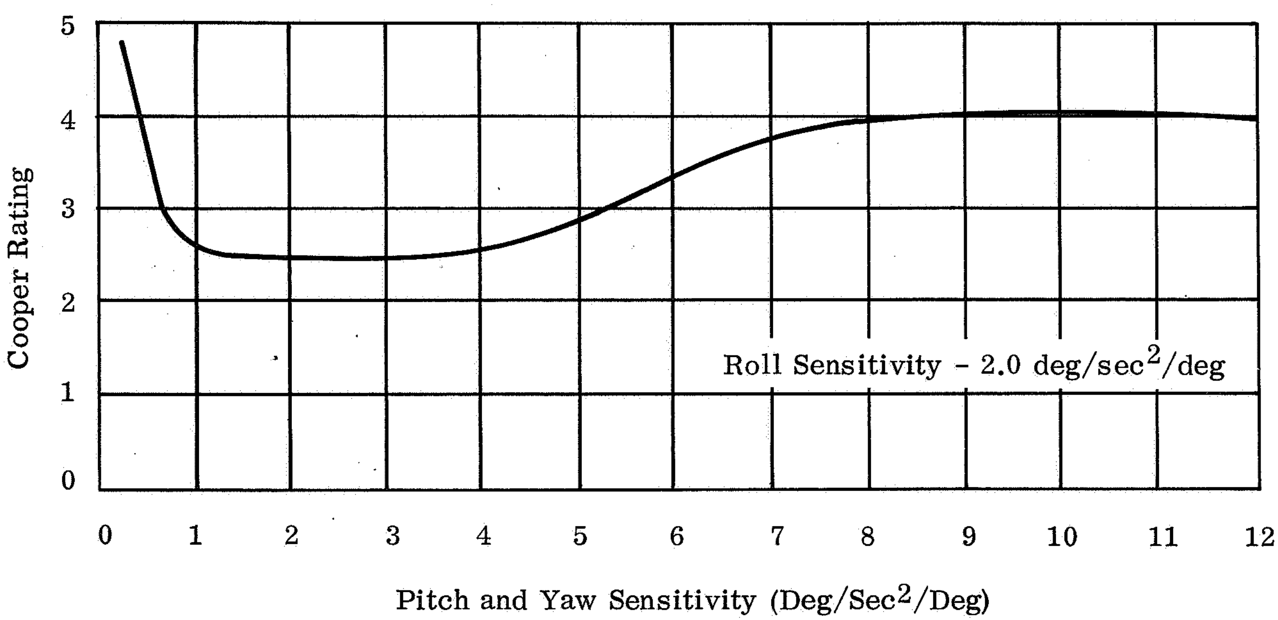


Figure 6.19. Pilot Rating for Attitude Hold Task

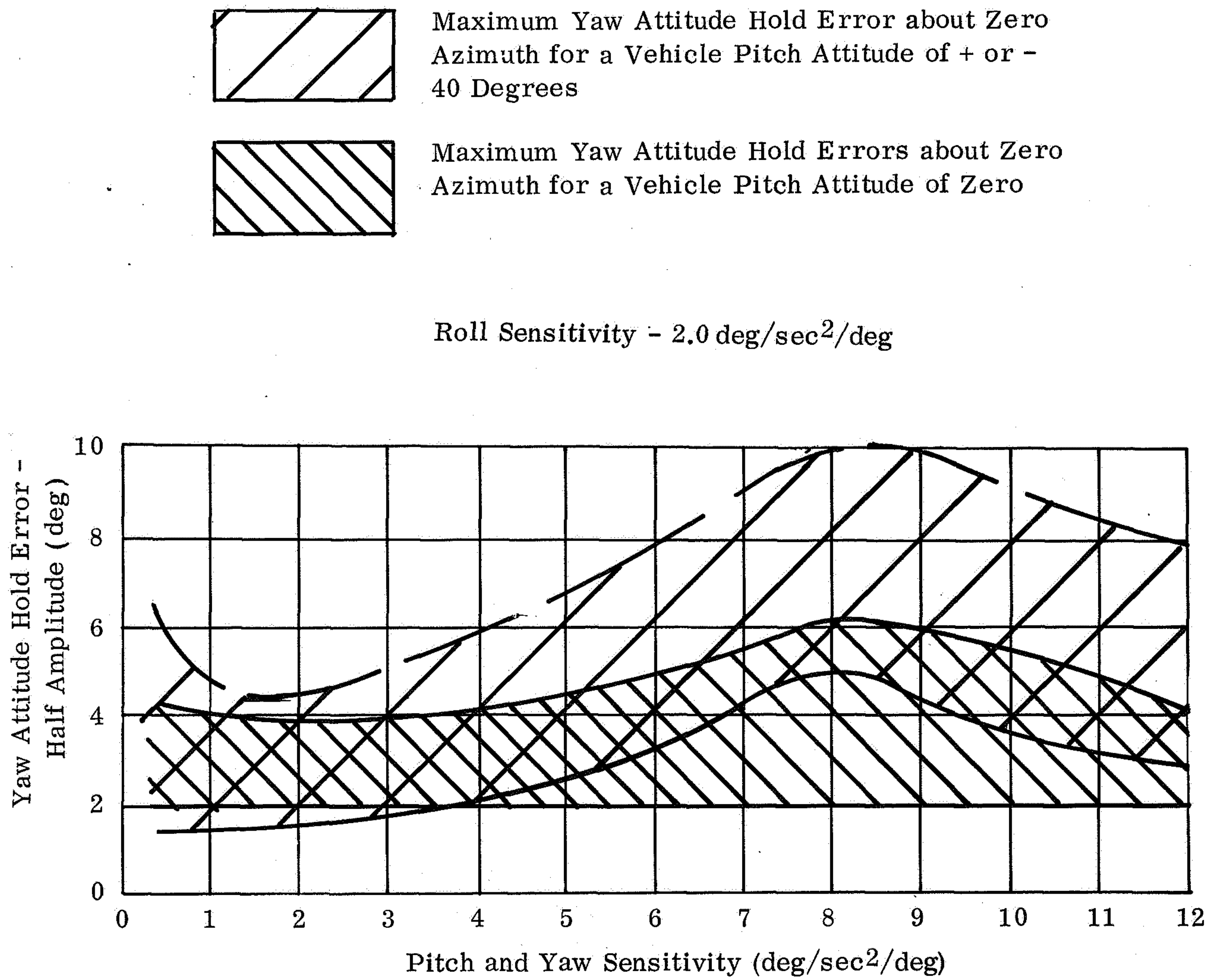


Figure 6.20. Yaw Attitude Hold Error - Pilot A

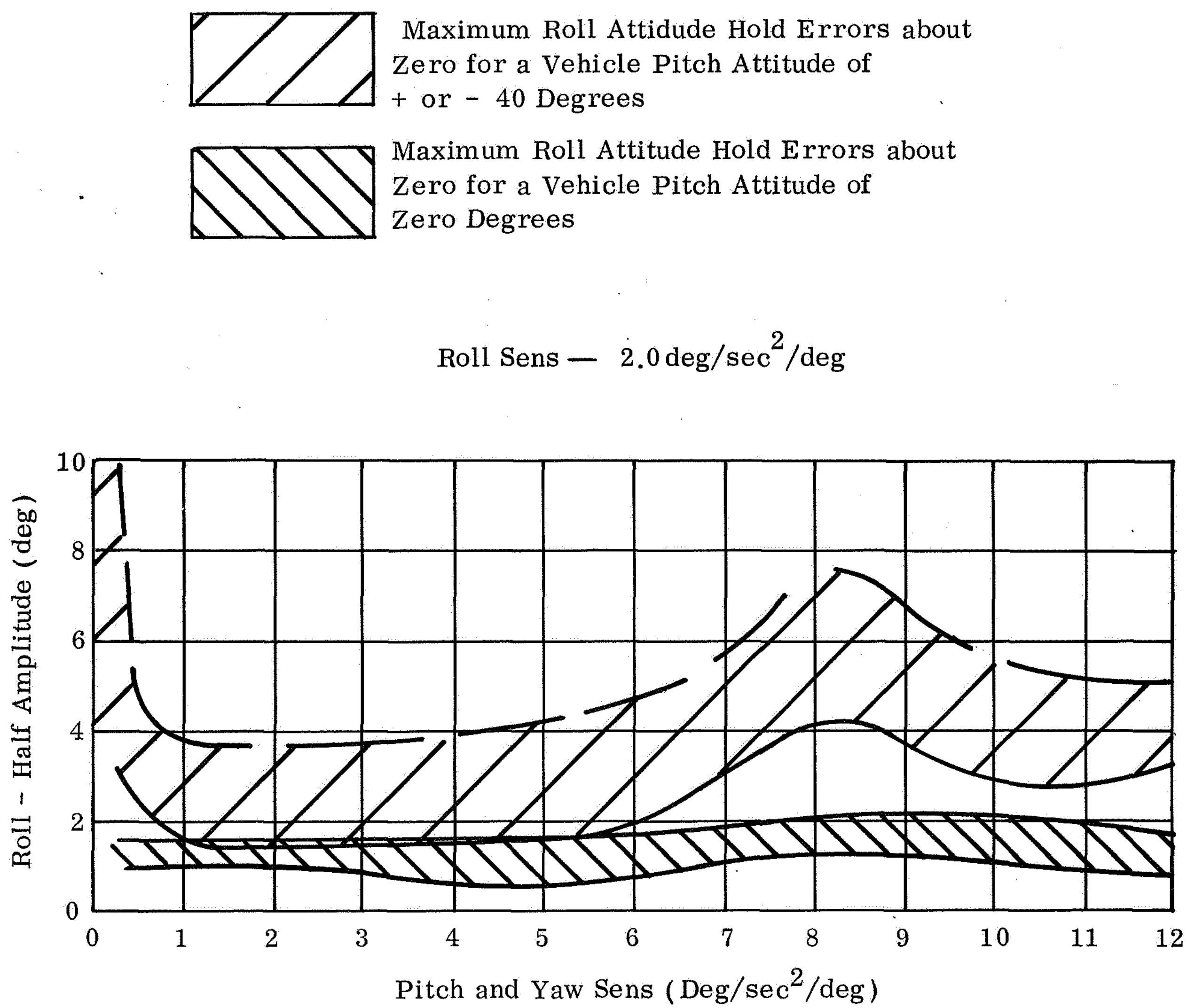


Figure 6.21. Roll Attitude Hold Error - Pilot A

Notes: $\theta_{sens} = 3.0$ deg/sec^2/deg
$\phi_{sens} = 2.0$ deg/sec^2/deg
$\psi_{sens} = 0.5$ deg/sec^2/deg
Thrust Range: 40 to 300 lb

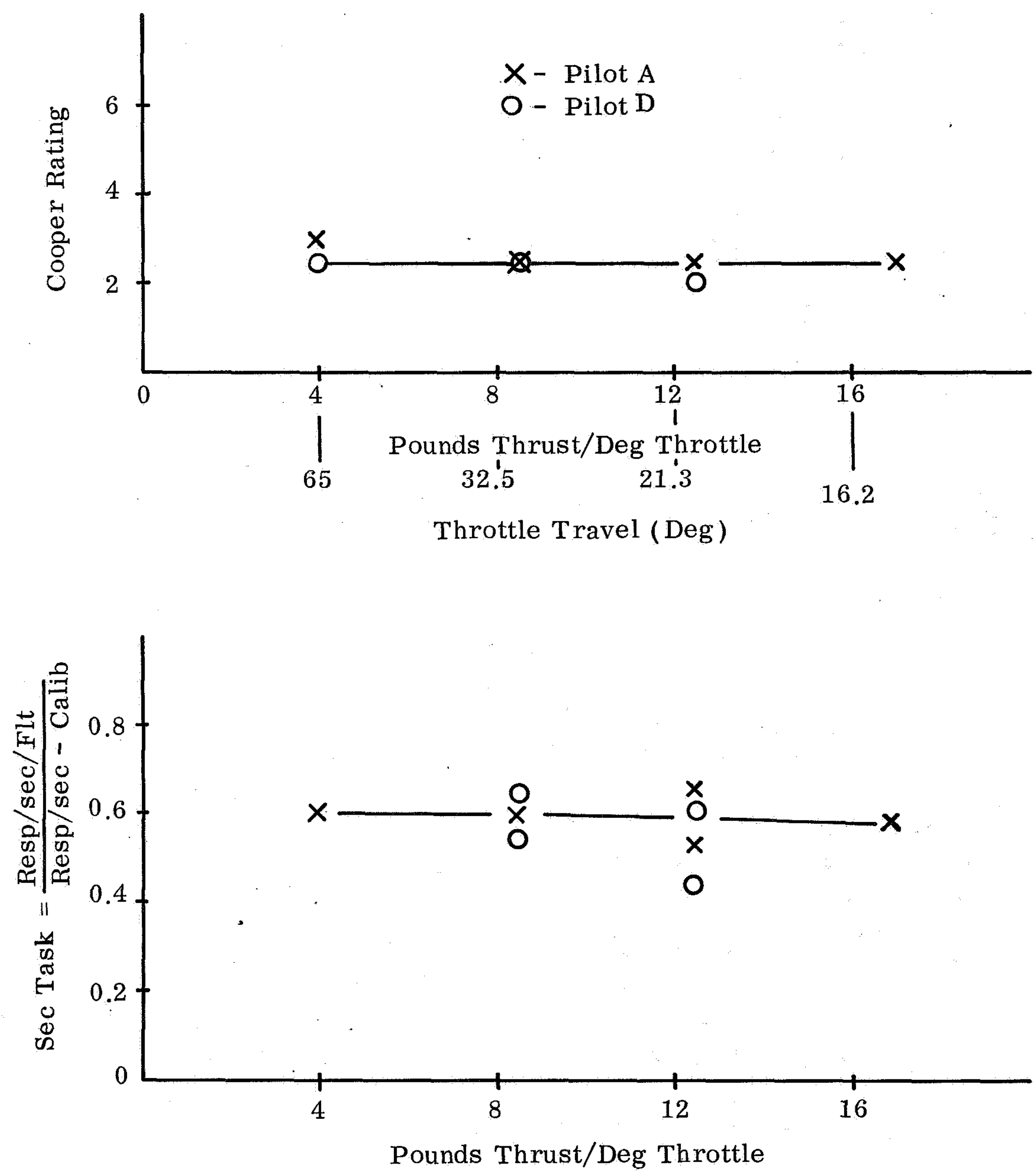


Figure 6.22. Linear Throttle Gradient Studies

6.4.3 Maximum Controller Deflection

Figure 6.23 summarizes, as a function of sensitivity, the maximum controller deflections observed during the control sensitivity studies covered in Paragraphs 6.4.1.1 and 6.4.1.2.

6.4.4 Control Configuration Studies

The handling qualities of four control configurations (Table 6.5) were evaluated on the simulator. The basic difference in these configurations from a handling qualities standpoint is their relative translational/rotational response characteristics (Figure 6.24). The equations used to generate these time histories are presented in Paragraph 4, Appendix C. The difference in the forward translational response results from the method by which thrust vector control is accomplished (translation or rotation of the thrust vector) and in the case of thrust vector rotation whether the pivot is below or above the cg.

A hover task was used to evaluate the above vehicle response characteristics since this type of simulation should be the most sensitive to the variations in translational/rotational characteristics associated with these various control systems.

Figures 6.25 through 6.30 present the cooper rating, secondary task and average horizontal position error ($\bar{r}$) and average vertical position error ($\bar{h}$) from desired hover point, and average horizontal velocity ($\bar{\dot{x}}$) and average vertical velocities $\bar{\dot{h}}$ as a function of control sensitivity. The sensitivity ranges used were based on the best indicated area of sensitivities summarized in Paragraph 6.4.1.4 ($\theta_{sens} \approx 3.0, \phi_{sens} \approx 2.0$, and $\psi_{sens} \approx 0.5$ deg/sec^2/deg). For yaw and roll sensitivity of 0.5 and 2.0 deg/sec^2/deg, respectively, the pitch sensitivity was varied from 0.5 to 10 degrees sec^2/deg and for pitch and yaw sensitivities of 3.0 and 0.5 deg/sec^2/deg, respectively, the roll sensitivity was varied from 0.5 to 12 deg/sec^2/deg. The data shown was obtained by randomly switching from one configuration to the other without informing the pilot which configuration he was flying. In general, the pilot could distinguish the configuration at the low sensitivity range of 1.0 deg/sec^2/deg or less, but for sensitivities of 2 deg/sec^2/deg and greater he was unable to distinguish any difference.

Low pivots in pitch resulted in high cooper ratings (6.0) and noticeable deterioration in hover performance at pitch sensitivities of 1.5 deg/sec^2/deg or less. For sensitivities above 1.5 deg/sec^2/deg low pivots generally resulted in a slightly higher work load and slightly poorer horizontal performance parameters ($\bar{r}$, $\bar{\dot{r}}$) then the other configurations. Pivots high, translating thrusters and the all gimbaled configurations all showed some deterioration in performance below 1.0 deg/sec^2/deg but because of data scatter no real preference could be established.

6.5 FINAL EVALUATION OF SELECTED CONFIGURATIONS

For the final evaluation of the selected configuration the vehicle control sensitivity envelopes were determined and the controller linkage gain adjusted to provide the

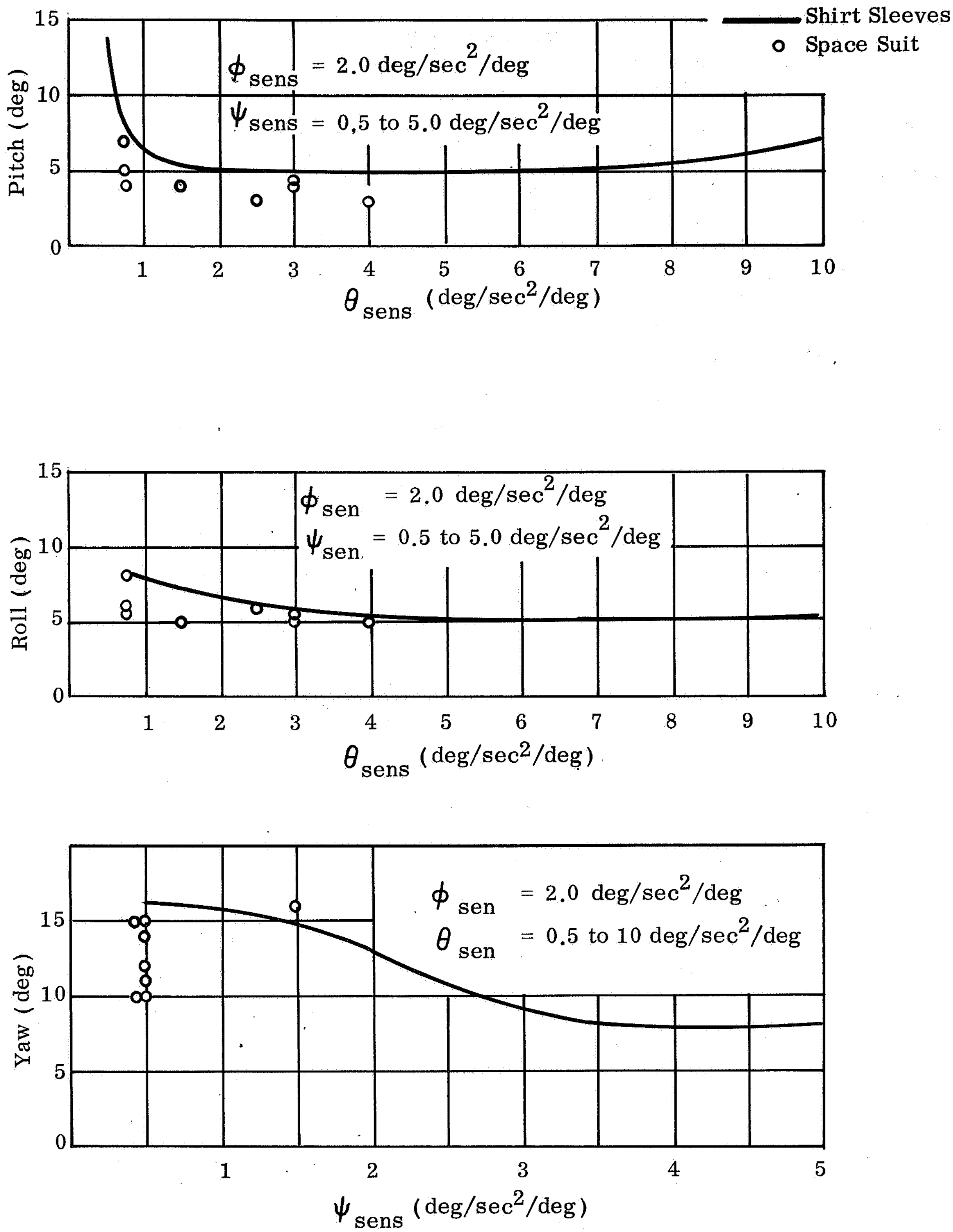


Figure 6.23. Maximum Controller Deflection

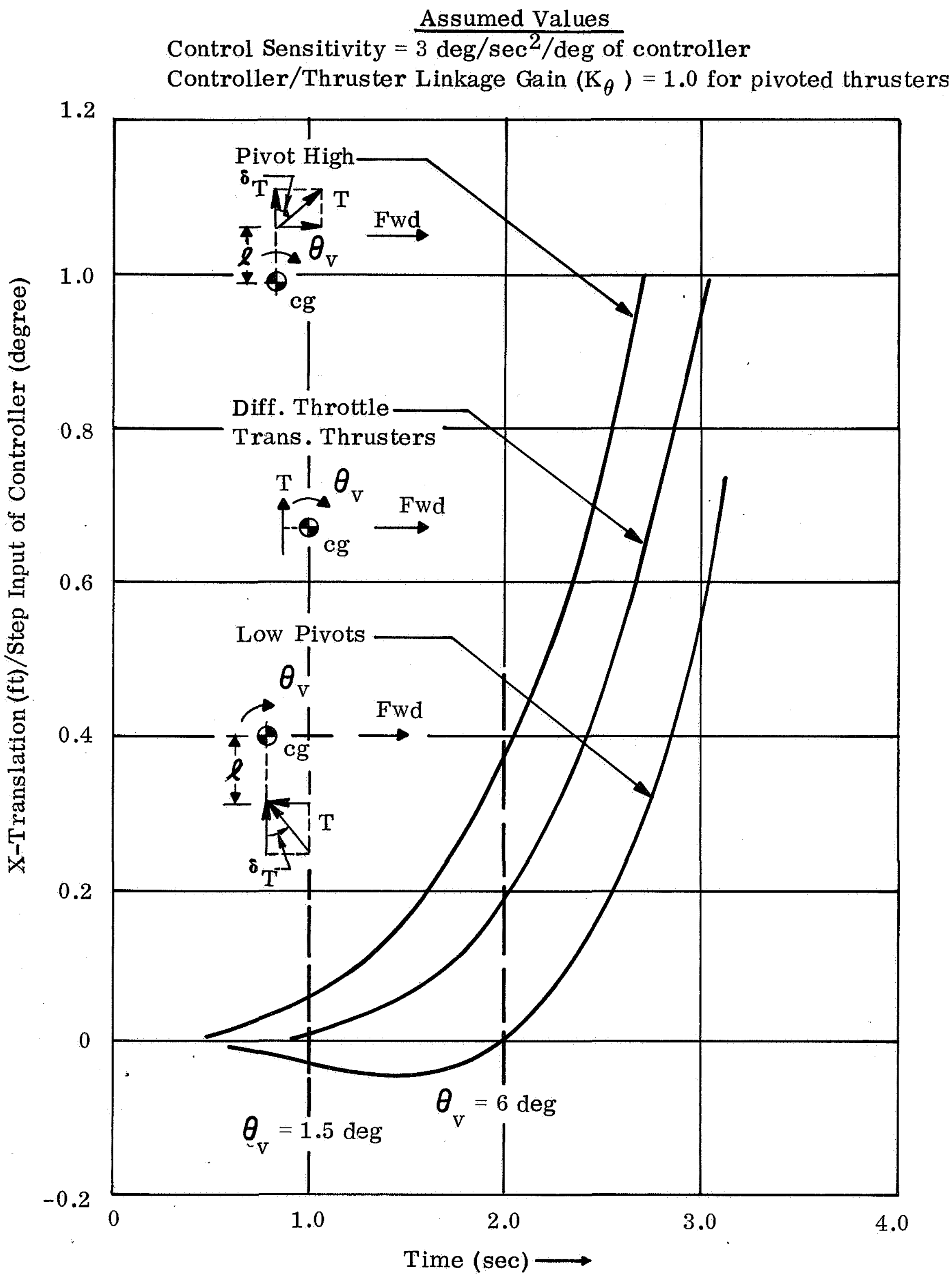


Figure 6.24. Comparison of Translation Characteristics for a Step Input of Controller

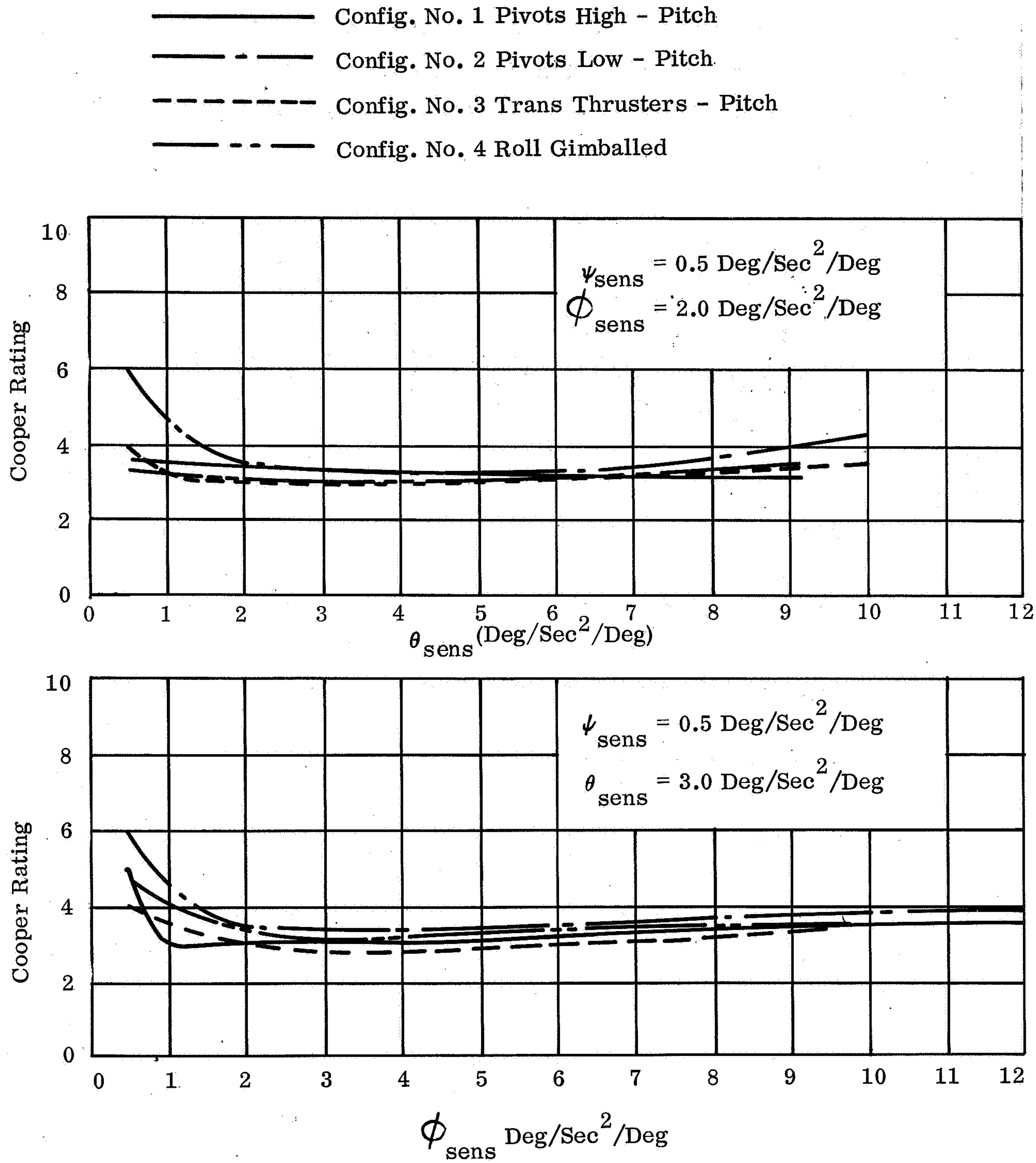


Figure 6.25. Cooper Rating Comparison For Various Control Mode Configurations

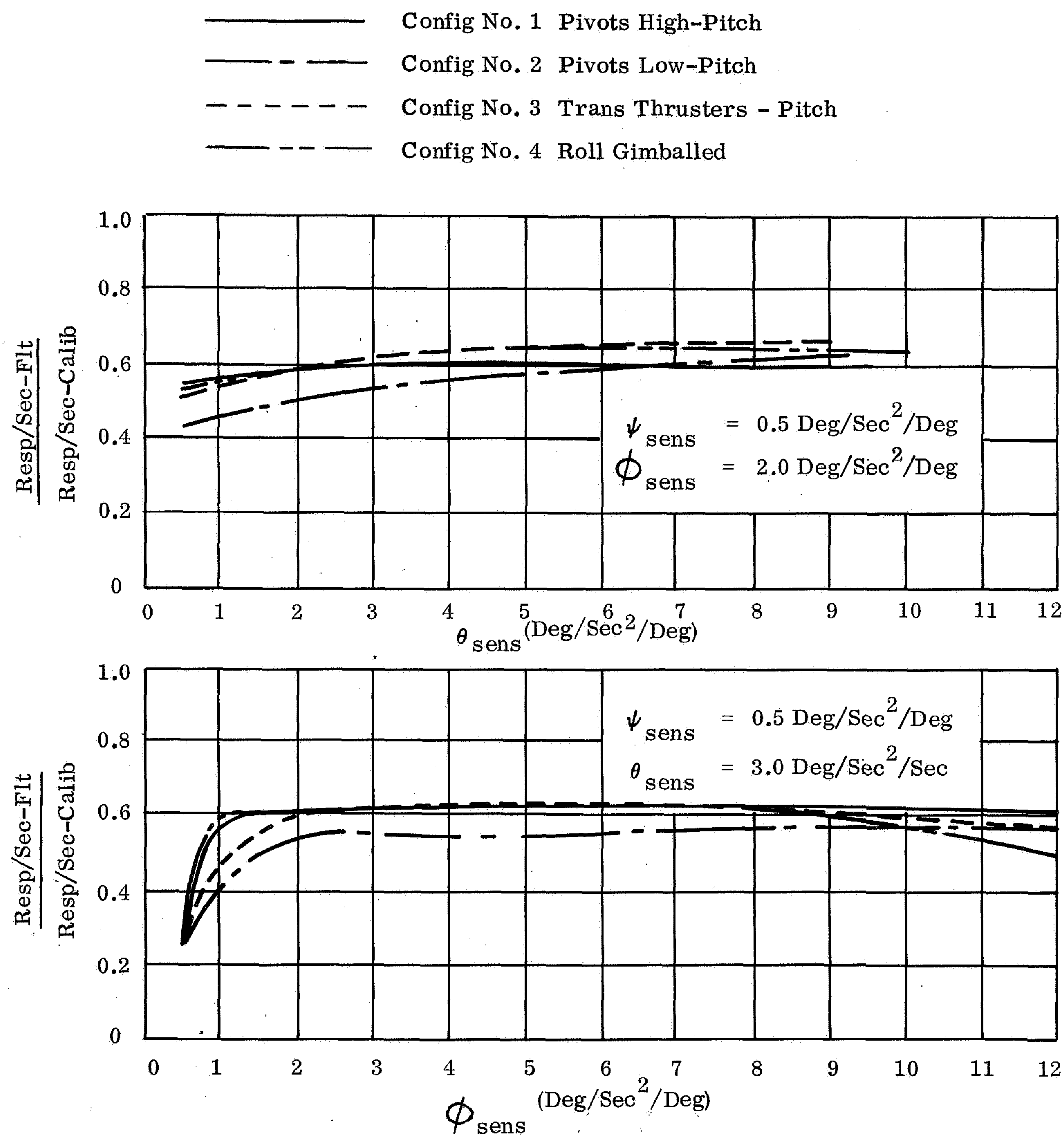


Figure 6.26. Secondary Task Comparison
For Various Control Modes

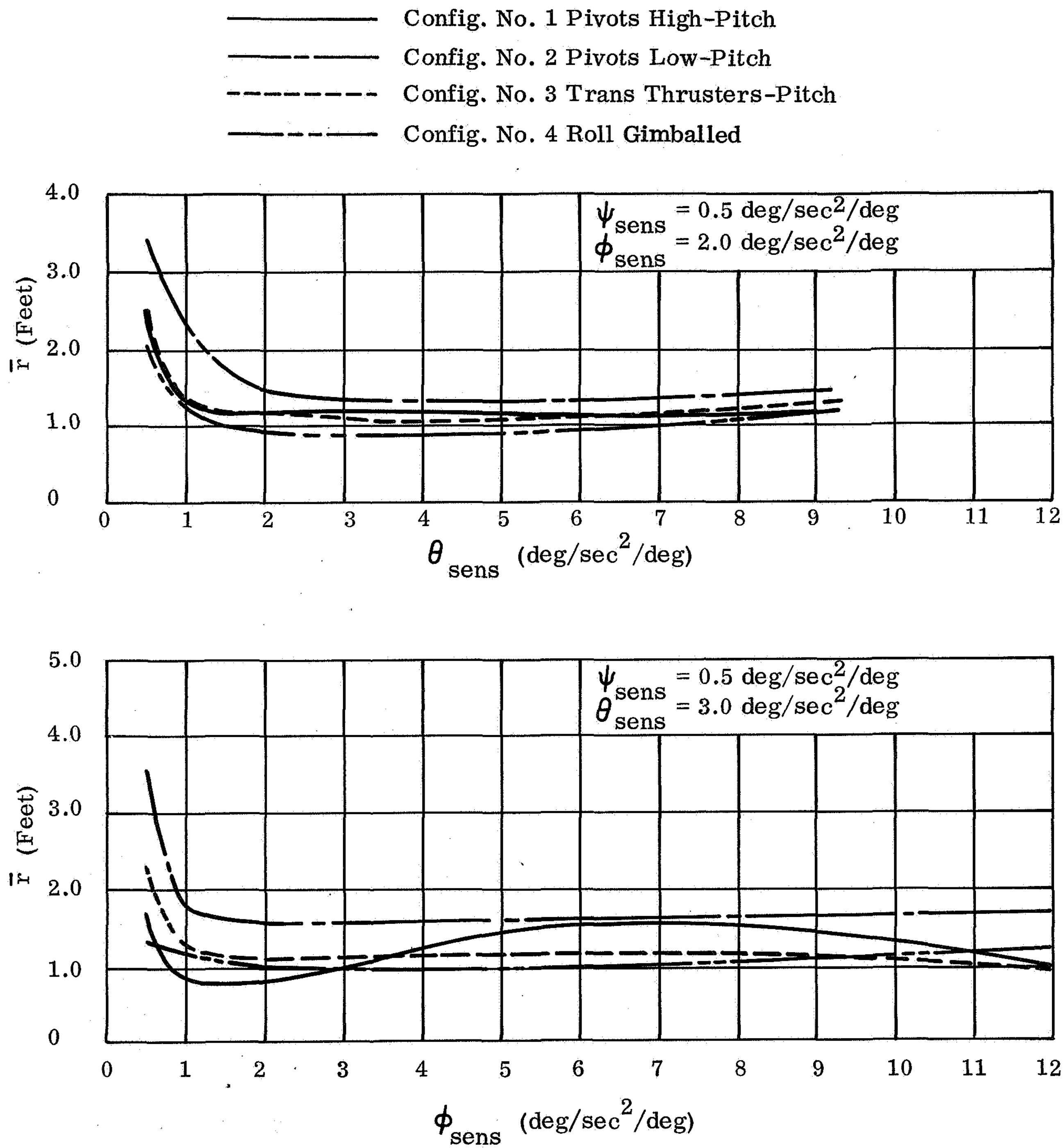


Figure 6.27. Average Position Error ($\bar{r}$) for Various Control Mode Configurations

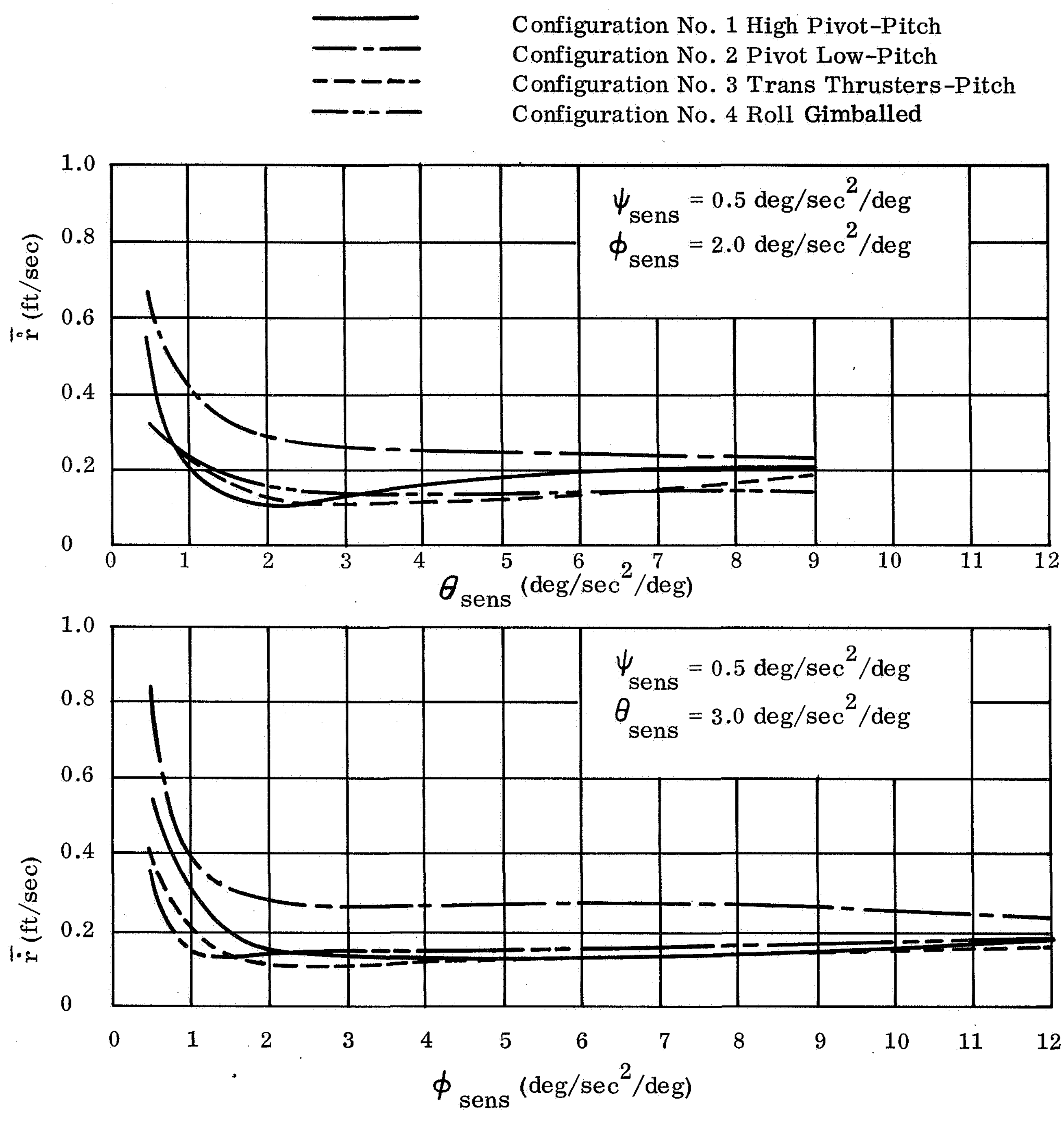


Figure 6.28. Average Velocity ($\bar{\dot{r}}$) for Various Control Mode Configurations

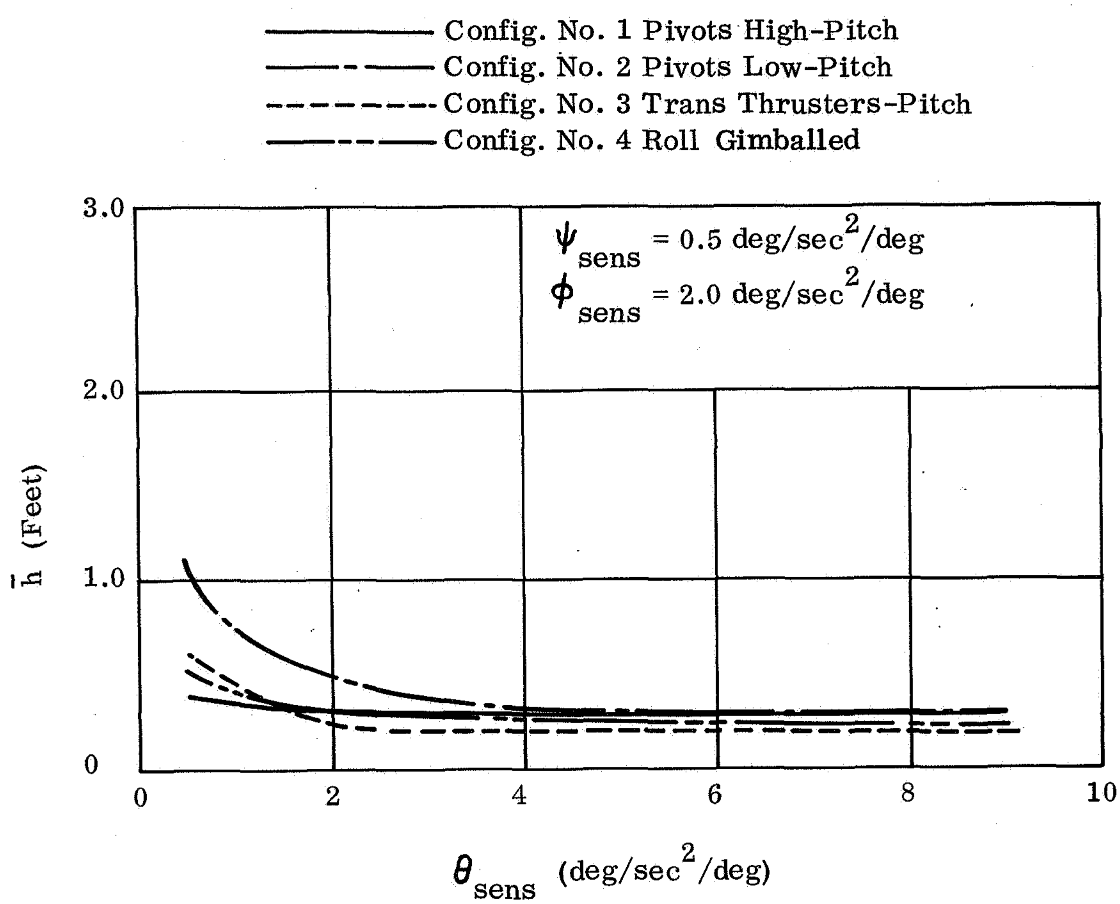


Figure 6.29. Average Vertical Position Error Z for Various Control Modes

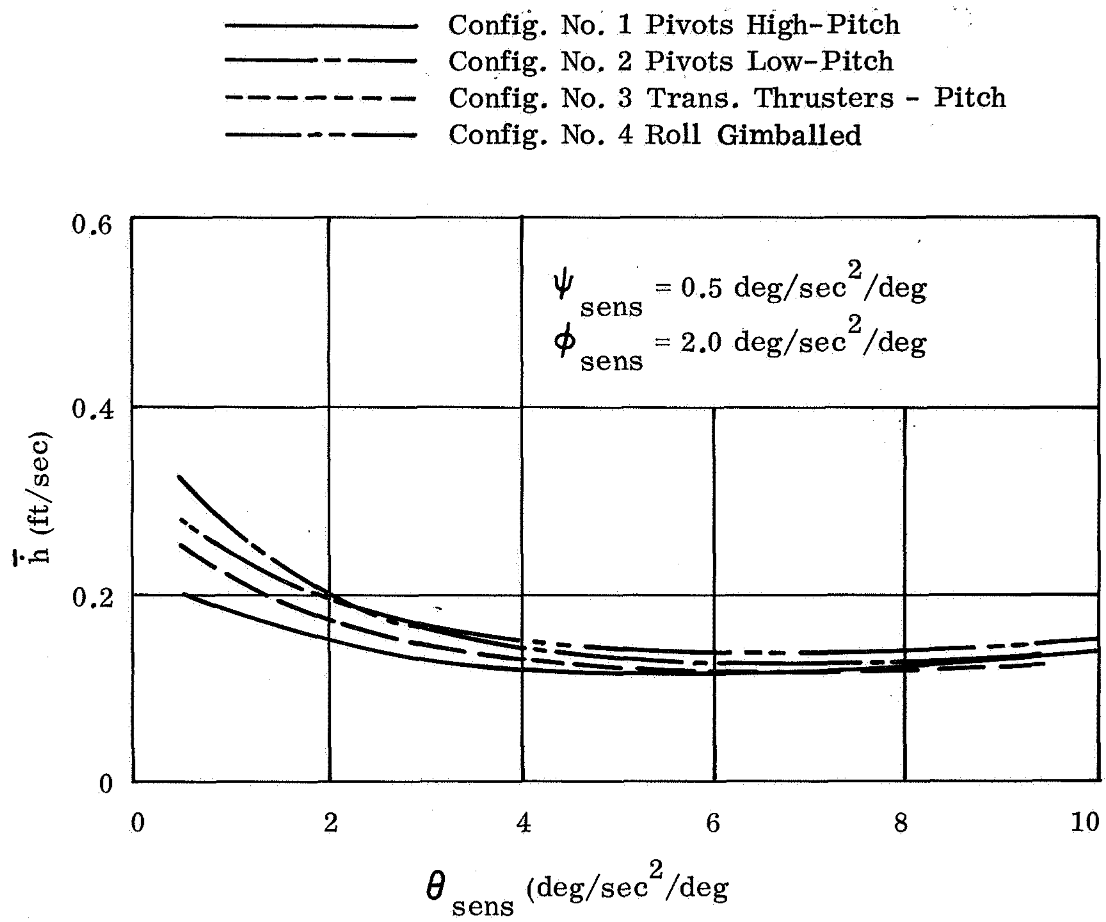


Figure 6.30. Comparison of Average Vertical Velocity for Various Control Mode Configurations

most acceptable range of control sensitivities indicated by the parameter studies of Paragraph 6.4. The control system was then evaluated using an expanded simulation (Reference Appendix C) which included all significant vehicle characteristics: propellant burnoff effects on mass properties, significant product of inertia terms, and the trim characteristics determined in Paragraph 6.5.1.2. Five mile flight profiles ranging from high altitude arcing to low altitude nap-of-the-moon, and 1600 foot nap-of-the-moon flights were made for zero, 100, and 370 pound payloads. Pilot (Cooper) ratings, work load, and ΔV were measured. In addition, the maximum controller deflections for combined trim and control were determined.

6.5.1 Vehicle Characteristics

The phyiscal vehicle properties for the selected configuration are shown in Table 6.7. The values shown without parenthesis are the original data used in this final evaluation (and subsequent stability augmentation studies) for computing control sensitivity envelopes and trim requirements. The latest estimate of vehicle properties are shown in parenthesis. These changes have been reviewed for possible impact on the study results and found to be small. With the exception of an increase in maximum roll trim requirement to 4.7 degrees from 3.7, all other changes are either neglible, self-compensating (e.g., both pitch inertia and pitch control moment arm increase in proportion), or easily compensated by minor design changes.

6.5.1.1 Control Sensitivity Envelopes

Two control sensitivity envelopes considered in the final manual control evaluation studies are shown in Figure 6.31 and Table 6.8. The equations used for computing the control sensitivities are summarized in Table 6.9. The pitch and yaw control sensitivities vary with both mass and thrust changes while the roll sensitivity varies only with vehicle roll inertia changes. The pitch and yaw sensitivity envelopes for the total thrust range (T_{min} to T_{max}) and propellant loading (full to empty) are shown for three payloads (zero, 100 and 370 lb). The T = W lines indicates the control sensitivities in cruise and hover, when thrust equals weight. Note that the T = W pitch sensitivity range, which is between 1.5 and 2.5 deg/sec^2/deg is essentially midway between the most acceptable pitch sensitivity ranges obtained from shirt sleeve and preliminary spacesuit studies (Reference Figure 6.31)

An alternate control sensitivity envelope shown in Figure 6.32 incorporated a pitch controller linkage gain of 1.4 which raised the T = W pitch sensitivities to a range of 2.5 to 3.5 deg/sec^2/deg controller (represents the best pitch sensitivities based on previous shirt sleeve simulation studies (refer to Figure 6.32). A third linkage gain of 0.75 was also evaluated.

6.5.1.2 Trim Requirements

The pitch and roll trim requirements are shown in Figure 6.33.

TABLE 6.7

LUNAR FLYER MOMENTS OF INERTIA

	0 lb Payload			100 lb Payload		370 lb Payload	
	Weight Empty	Burnout Weight	Gross Weight	Burnout Weight	Gross Weight	Burnout Weight	Gross Weight
Weight	210.5*	580.5	880.5	680.5	980.5	950.5	1250.5
	(241.7)*	(611.7)	(911.7)	(711.7)	(1011.7)	(981.7)	(1281.7)
$\overline{X}$	95.01	104.56	100.79	100.00	98.01	91.89	92.27
	(95.62)	(105.41)	(101.36)	(100.50)	(98.31)	(91.68)	(92.02)
$\overline{Y}$	100.00	100.00	99.49	100.00	99.54	100.00	99.64
	(99.64)	(99.86)	(99.37)	(99.88)	(99.44)	(99.91)	(99.55)
$\overline{Z}$	65.13	82.89	77.14	78.72	74.83	75.93	73.55
	(61.21)	(78.62)	(74.68)	(75.24)	(72.68)	(73.49)	(71.88)
I_{xx}	22.04	70.84	91.18	88.02	103.45	101.58	114.59
	(28.01)	(80.47)	(98.12)	(93.60)	(108.38)	(104.83)	(118.58)
I_{yy}	20.91	76.99	99.87	111.82	126.45	158.55	168.99
	(27.01)	(88.49)	(109.63)	(124.12)	(138.28)	(177.92)	(188.65)
I_{zz}	13.04	24.12	33.30	44.36	50.22	86.48	90.57
	(21.22)	(34.06)	(45.00)	(59.14)	(66.01)	(110.66)	(115.15)
I_{xy}	0.14	0.14	0.85	0.14	0.58	0.14	0.02
	(-0.06)	(0.24)	(1.03)	(0.15)	(0.66)	(-0.02)	(-0.12)
I_{xz}	0.58	12.69	20.67	28.93	32.65	41.11	40.33
	(-0.04)	(14.67)	(21.08)	(30.29)	(33.19)	(39.36)	(38.88)
I_{yz}	0.04	0.04	1.12	0.04	0.90	0.04	0.78
	(-0.18)	(0.15)	(3.83)	(0.08)	(3.59)	(0.05)	3.49
ℓ_z - Thruster Pitch Moment Arm (Feet)		1.33 (2.18)	1.82 (2.02)	1.71 (2.00)	2.04 (2.22)	1.58 (1.79)	1.875 (2.05)
ℓ_y - Yaw and Roll Moment Arm - Vehicle ℄ to Thruster (Feet)		1.875 (1.875)	1.875 (1.875)	1.875 (1.875)	1.875 (1.875)	1.875 (1.875)	1.875 (1.875)

NOTES: * - Includes residuals

No Parenthesis - Vehicle properties used for simulation studies. Vehicle sensitivity envelopes and trim requirements.

Parenthesis - Updated properties as of 7/3/69

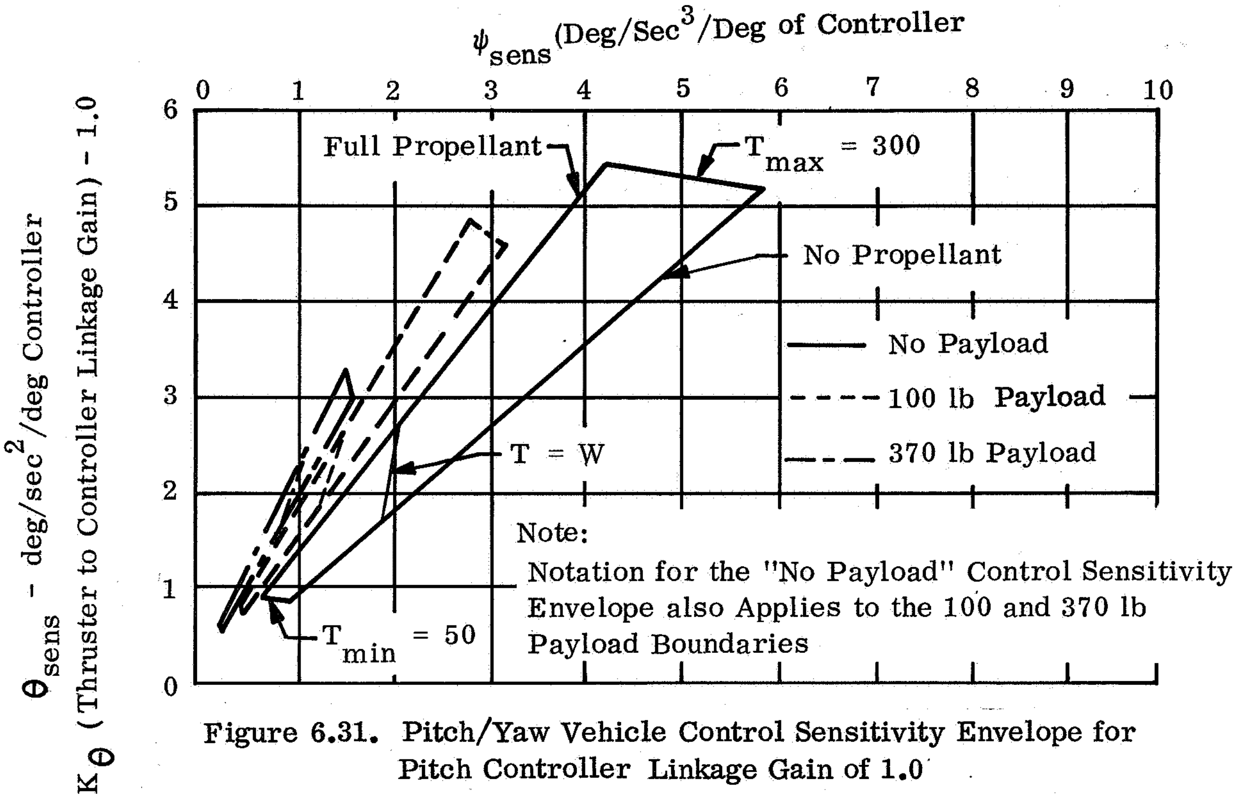


Figure 6.31. Pitch/Yaw Vehicle Control Sensitivity Envelope for Pitch Controller Linkage Gain of 1.0

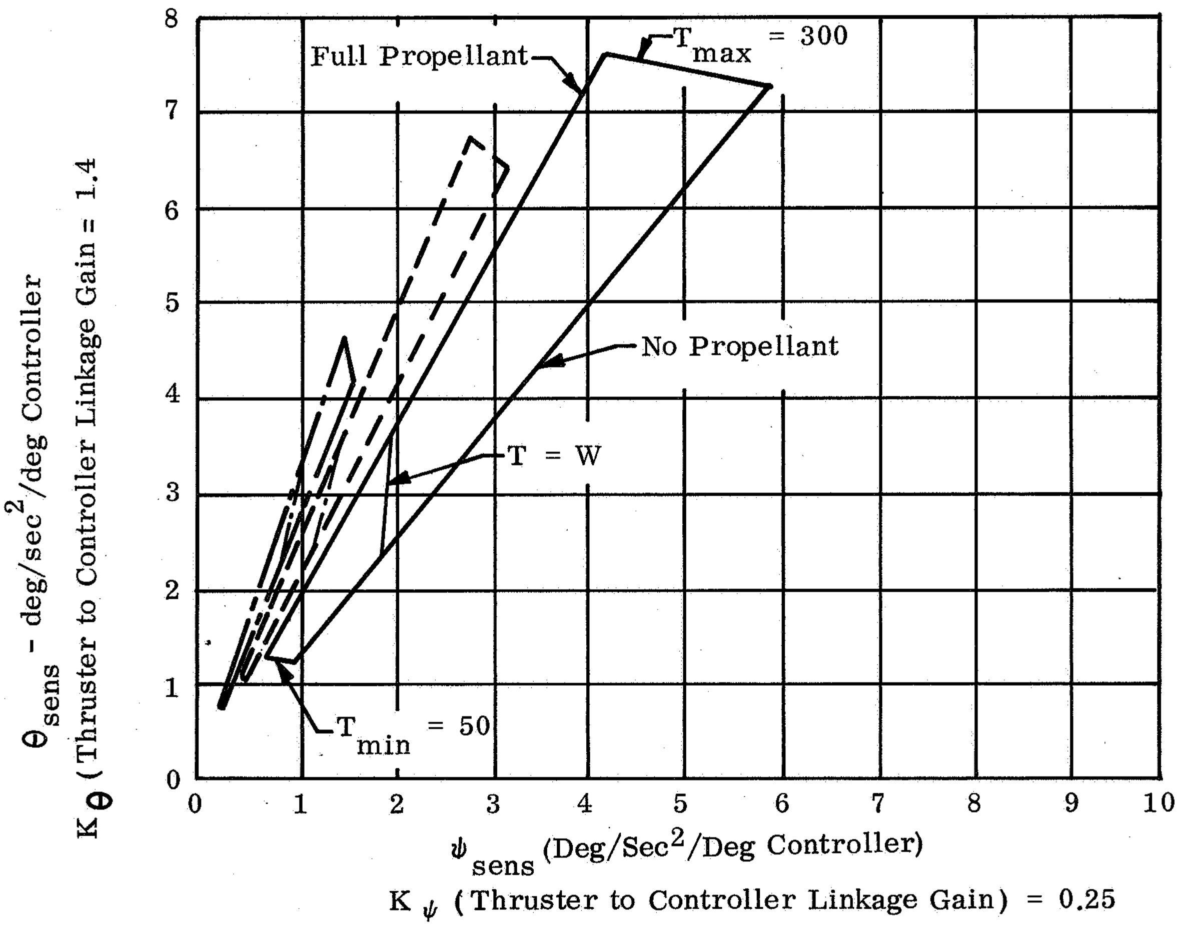


Figure 6.32. Pitch/Yaw Vehicle Control Sensitivity Envelope for Pitch Controller Linkage Gain of 1.4.

TABLE 6.8

ROLL SENSITIVITIES - DEG/SEC^2/DEG CONTROLLER

(1.75 lb of Total Differential Thrust per Deg of Controller)

Payload Condition	Propellant Condition	
	Empty	Full
No Payload	2.64 $\frac{\text{Deg/sec}^2}{\text{Deg of Controller}}$	2.06
100 lb Payload	2.14	1.82
370 lb Payload	1.85	1.65

TABLE 6.9

CONTROL SENSITIVITY DEFINITIONS (DEG/SEC^2/DEG CONTROLLER)

Pitch (Gimbaled)	Yaw (Differential Gimbaling)	Roll (Differential Throttling)
$\dot{\theta}_{sens} = \frac{\ddot{\theta}_v}{\delta\theta_c} = \frac{T\ell_z K_\theta}{I_{yy}}$	$\psi_{sens} = \frac{\ddot{\psi}_v}{\delta\psi_c} = \frac{T\ell_y K_\psi}{I_{zz}}$	$\phi_{sens} = \frac{\ddot{\phi}_v}{\delta\phi_c} = \frac{K_\phi \ell_y (57.3)}{I_{xx}}$

T - Total thrust for both engines (lb)

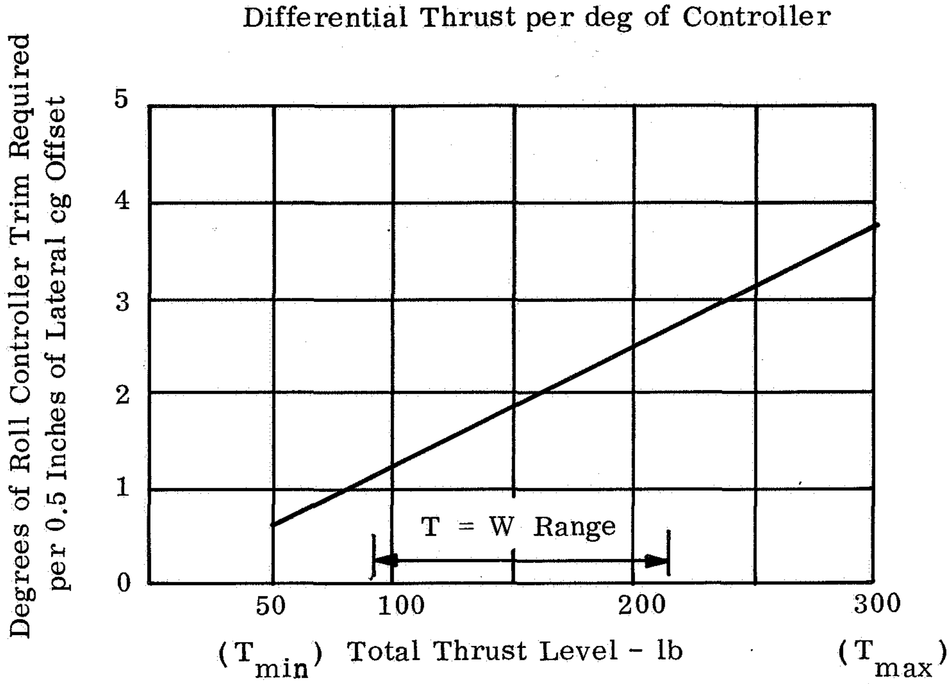
ℓ_z - Thruster moment arm in pitch (ft)

ℓ_y - Yaw and roll moment arm - vehicle ℄ to thruster (ft)

K_θ, K_ψ - Thruster to controller linkage gain in pitch and yaw, respectively (deg/deg)

K_ϕ - Total diff thrust for both engines per deg of controller (lb/deg)

Note: Trim Req based on 1.75 lb of Total Differential Thrust per deg of Controller



Note: Assumed a Ground Trim Adjustment to Midpoint of Trim Variation due to Propellant Loading

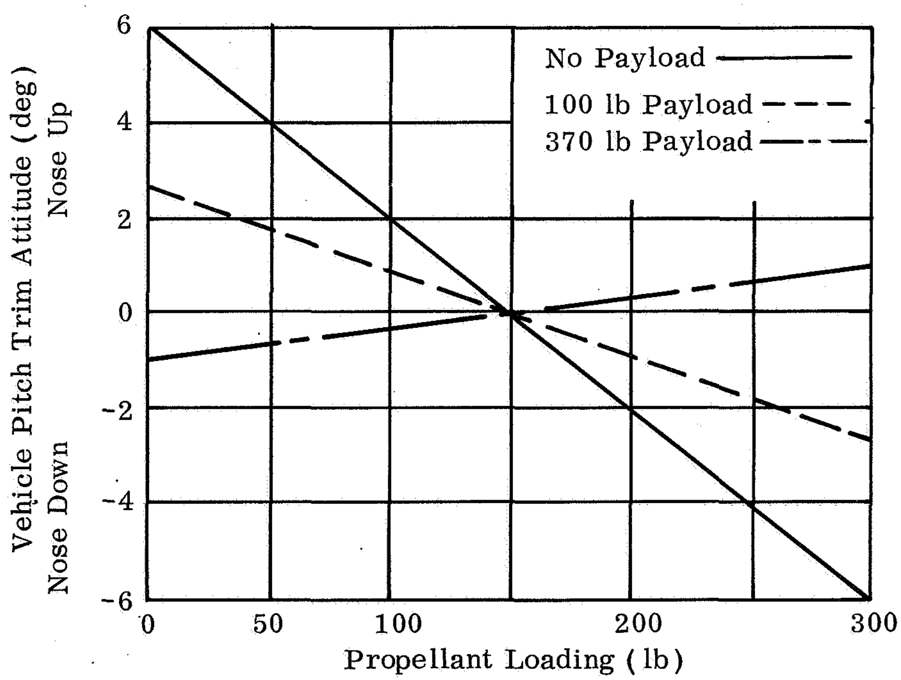


Figure 6.33. Pitch and Roll Trim Requirements

The roll controller trim requirements are a function of the operating thrust level. The maximum roll controller trim requirement occurs at T_{max} and is 3.73 deg for the maximum lateral cg offset of 0.5 inches. (For the updated vehicle characteristics, 4.7 degrees are required for the maximum 0.65 inch offset).

The vehicle pitch trim attitude is a function of payload and propellant on board. The corresponding pitch controller trim position requirements are a function of the thruster to controller linkage gain (K_θ). For a $K_\theta = 1.0$ (Figure 6.31) the required controller trim position is the same as the vehicle trim attitude, but in the opposite sense. For example, a maximum pitch-up vehicle trim attitude of 6 degrees will require a maximum downward controller trim position requirement of 6 degrees. For a K_θ other than 1.0 the controller trim position is (vehicle trim attitude)/K_θ.

6.5.2 Simulation Results

Tables 6.10, 6.11, 6.12, and 6.13 summarize the results obtained from the final evaluation study of the selected configuration.

The pilot (Cooper) ratings and workload measures are shown in Tables 6.10 and 6.11. No workload measures were obtained for the five mile flights. The pilots indicated a slight preference for the pitch linkage gain (K_θ) of 1.0 over the other two alternative values studied ($K_\theta = 1.4$ and $K_\theta = 0.75$).

Table 6.12 presents ΔV results obtained for the 1600 ft and 5 n.mi. flights. For the five mile flights, instruments were introduced because the pilots had difficulty estimating from the visual scene how fast they were going or how high they were in order to judge their final approach. The pilots essentially flew their own desired flight profiles except for several flat top cruise type trajectory performed by Pilot A (initial boost angle (θ_B) and boost time (t_B) of 40 seconds) using: (1) a timer and pitch attitude indicator, and (2) a timer, pitch attitude and altitude indicator.

The maximum control requirements observed during these final evaluations are summarized in Table 6.13.

6.5.3 Conclusions

The selected manual TVC configuration has satisfactory handling qualities over the required payload and propellant ranges. The increase in pilot (Cooper) rating observed in Table 6.7 for the five mile flights is attributed to the guidance aspects (as simulated) of the flight and not the vehicle control handling qualities.

TABLE 6.10

COOPER RATING SUMMARY - FINAL MANUAL TVC STUDY

Flight	Pitch Linkage Gain $(\dot{K}_\theta)$	Pilot	Payload: None, 100 - lb	Payload: None, Full	Payload: 100 lb, Full	Payload: 370 lb, Full	Comments
1600 ft	1.0 (Selected)	A	2.5 - 3.0	2.5 - 3.0	2.0 - 2.5	2.0 - 2.5	
		D		3.5		3.0	
		B		3.0 - 3.5	3.5 - 4.0		
		C		3.0		2.5	
			(2.75)	(3.1)	(3.0)	(2.6)	Average Value
	1.4	A	3.0			2.0	K_θ = 1.0 preferred
		B			4.0		
5 n.mi.	0.75	B		3.5 - 4.5			T_{max} - 4.0 T_{min} - 4.5
		C				3.5	K_θ = 1.0 preferred
	1.0 (Selected)	A		3.5	3.5*	3.5	
		B		3.5 - 4.0**	3.5	3.0 - 4.0	
		C			4.5	3.5	
				(3.6)	(4.0)	(3.5)	Average Value

*At T_{max} the rating was 3.5, at T_{min} the rating was 2.5

**3.5 - 4.0 rating at all thrust levels

TABLE 6.11

PILOT WORKLOAD (W.L.)** MEASURE FOR 1600 FT FLIGHTS
FINAL MANUAL TVC STUDY

Pitch Linkage Gain	Pilot	Payload			
		Zero		100 lb	370 lb
		100 lb Prop	Full	Full	Full
1.0 (Selected)	A	0.224	0.352	0.278	0.275
	B		0.35		
	C		0.50		0.22
	D		0.30		0.31
		(0.224)*	(0.38)*	(0.278)*	(0.275)*
1.4	A	0.341			0.400

* Average Value

** Reference Appendix A6.3.1 for definition.
(Each value is the average of 5 flights)

TABLE 6.12

ΔV RESULTS - FINAL MANUAL TVC STUDY

Flight	Pilot	Flight Instruments	ΔV	Number of Flights
1600 ft	A	None	360-530	23
	D	None	440-660	7
	B	None	600-900	10
	C	None	380-590	10
5 n.mi.	A	None	1750-1912	2
		Pitch Attitude and Timer	1400-1850	5
		Pitch Attitude Timer Altitude	1350-2125	7
	B	None	1730-1900	6
		All Inst Avail*	1500-2050	5
	C	All Inst Avail*	1510-2300	5

*Includes:

Altitude and Altitude Rate
Range and Range Rate
Pitch, Roll, Yaw Attitudes
Timer

TABLE 6.13

REQUIRED MAXIMUM CONTROLLER/THRUSTER DEFLECTIONS - 1600 ft and 5 mi Flights

Pilot	Pitch	Yaw		Roll	
	Controller or Thruster (K_θ = 1)	Controller	Thruster (K_ψ =0.25)	Controller*	Total Diff. Thrust (K_ϕ =1.75 lb/deg
A	12 (deg)	14 (deg)	3.5 (deg)	8 (deg)	14 (lb)
B	10	17	4.25	8	14
C	13	14	3.5	9	15.5

*Roll trim requirements for updated vehicle characteristics (Table 6.7) indicate that these values should be increased by 1.0 degree.

6.6 STABILITY AUGMENTATION STUDIES

The handling qualities for two types of stability augmentation systems (SAS) were investigated on the simulator: (1) a simple all-mechanical control augmentation system, and (2) a proportional rate command (gyro feedback) control system. Parametric studies were performed on the two systems first to determine the basic loop parameters which provided the most desirable handling qualities. This was followed by a handling quality comparison study between the best of the two augmentation systems and the selected manual TVC vehicle (Section 6.5). Two hand controller configurations were also evaluated with and without control augmentation: the handle bar arrangement and a three-axis Apollo Block I hand controller. Table 6.14 summarizes the system comparisons made.

TABLE 6.14

SYSTEM COMPARISON STUDIES

	No SAS (Manual)	All Mechanical Control Augmentation	Rate Gyro Stability Augmentation
Handle Bars	x	x	x
Apollo Hand Controller	x		x

6.6.1 Parametric Studies

Figure 6.34 summarizes the basic parameters of the two SAS systems and compares their general response characteristics with that of the manual system (proportional acceleration).

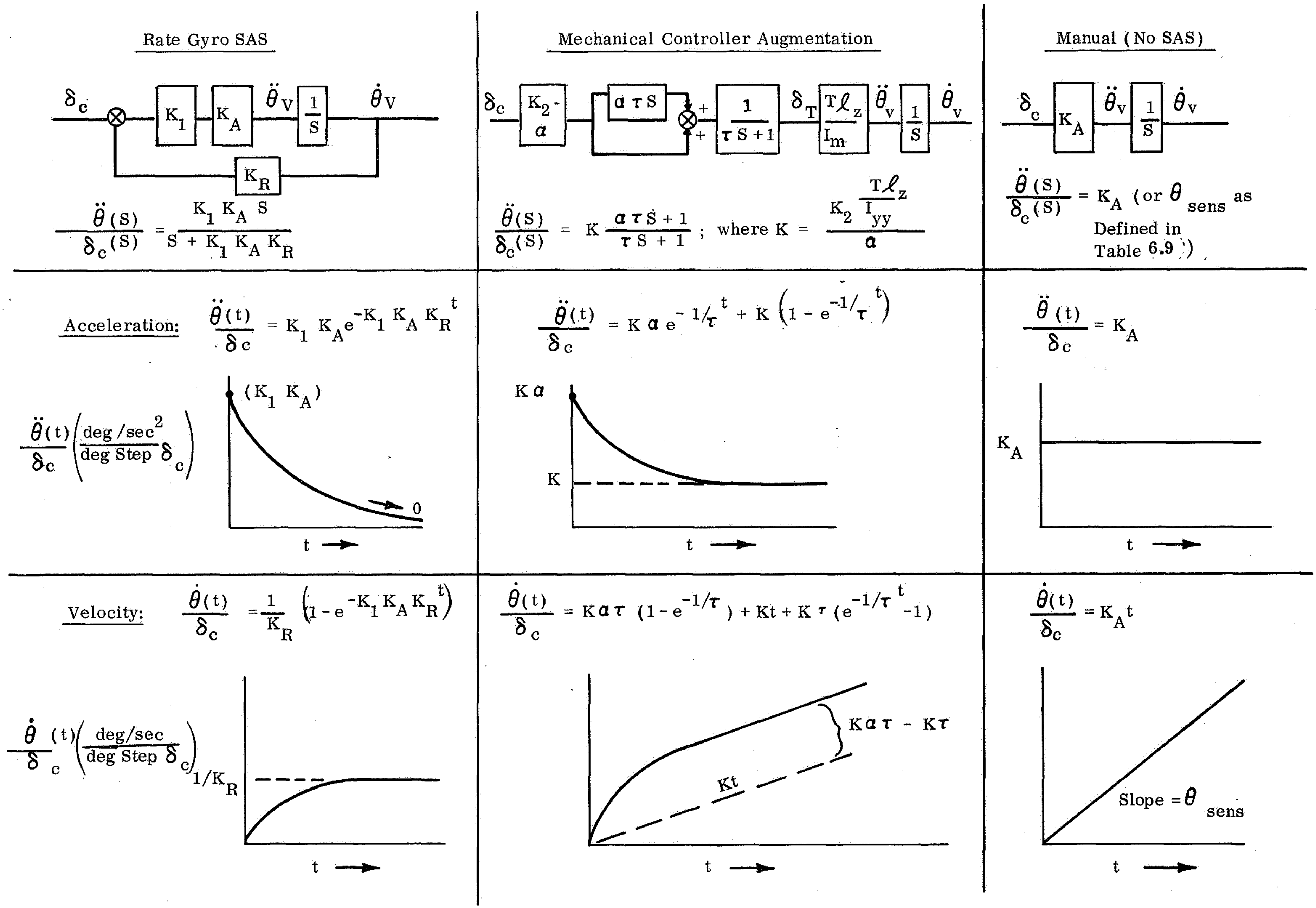


Figure 6.34. General Response Characteristics of Control Systems Studies

6.6.1.1 Rate Gyro SAS

Two parameters describe the basic response characteristics of a rate command control system (Figure 6.34):

$\frac{1}{K_R}$ - Steady state commanded vehicle rate per unit controller (δ_c) deflection. K_R is the rate feedback gain in degrees/deg/sec.

$\tau = \frac{1}{K_1 K_A K_R}$ - System time constant. The lag with which the vehicle rate builds up to the commanded rate.

K_A - Original control sensitivity associated with the selected manual configuration (Ref. Figure 6.31 and Tables 6.8 and 6.9)

K_1 - Variable gain used in parametric study to adjust forward loop SAS gain (can be thought of as a modified linkage gain).

The parametric studies were conducted by varying K_1 and K_R on the analog computer to obtain variations in the steady state commanded rate, $\frac{1}{K_R}$, and time constant τ. The same values of K_1 and K_R were used in all three axes (pitch, roll, and yaw) simultaneously. After determining the most acceptable range of values by this method, the pilots reported good harmony and indicated no desire to alter the K_1 or K_R values in any one particular axis.

The Cooper ratings obtained from two pilots using the handle bar arrangement and the three-axis hand controller are shown in Figures 6.35 through 6.38 in terms of the pitch axis parameters. Pilot C is an experienced test pilot and has flown rate command systems. This was pilot A's first experience with a rate command system.

Both pilots preferred a steady-state command rate ($1/K_R$) of 1.0 deg/sec of vehicle rate per degree of controller input for both types of controllers with time constants ranging from 0.1 to 0.4 sec. In general, pilot C preferred a shorter time constant than pilot A.

The circled points in the four figures indicate the rate gyro control loop parameters selected for the system comparison studies of Paragraph 6.6.2. It is seen in Figure 6.35 that the point used for pilot A with handle bars is actually that preferred by pilot C, and is rated 1.0 cooper level worse than his best rating. This proved somewhat unfortunate in that it slightly clouded the final comparison (see Figure 6.42).

The SAS studies with handle bars were conducted for a 1600 ft flight using the expanded vehicle simulation, (product of inertia, trim requirements, and vehicle mass property variations due to propellant burnoff). For the Apollo hand controller, studies were similar except that the trim requirements were omitted from the simulation on the assumption that another means would be used (i.e., trim button). The effects of including trim requirements when using the hand controller with the existing spring return forces was briefly investigated. In the worst case (no payload, full propellant)

Configuration	$K_A \frac{Deg/Sec^2}{Deg}$		
	Pitch	Yaw	Roll
No Payload	2.6	2.04	2.06
370 lb Payload	2.3	1.07	1.64

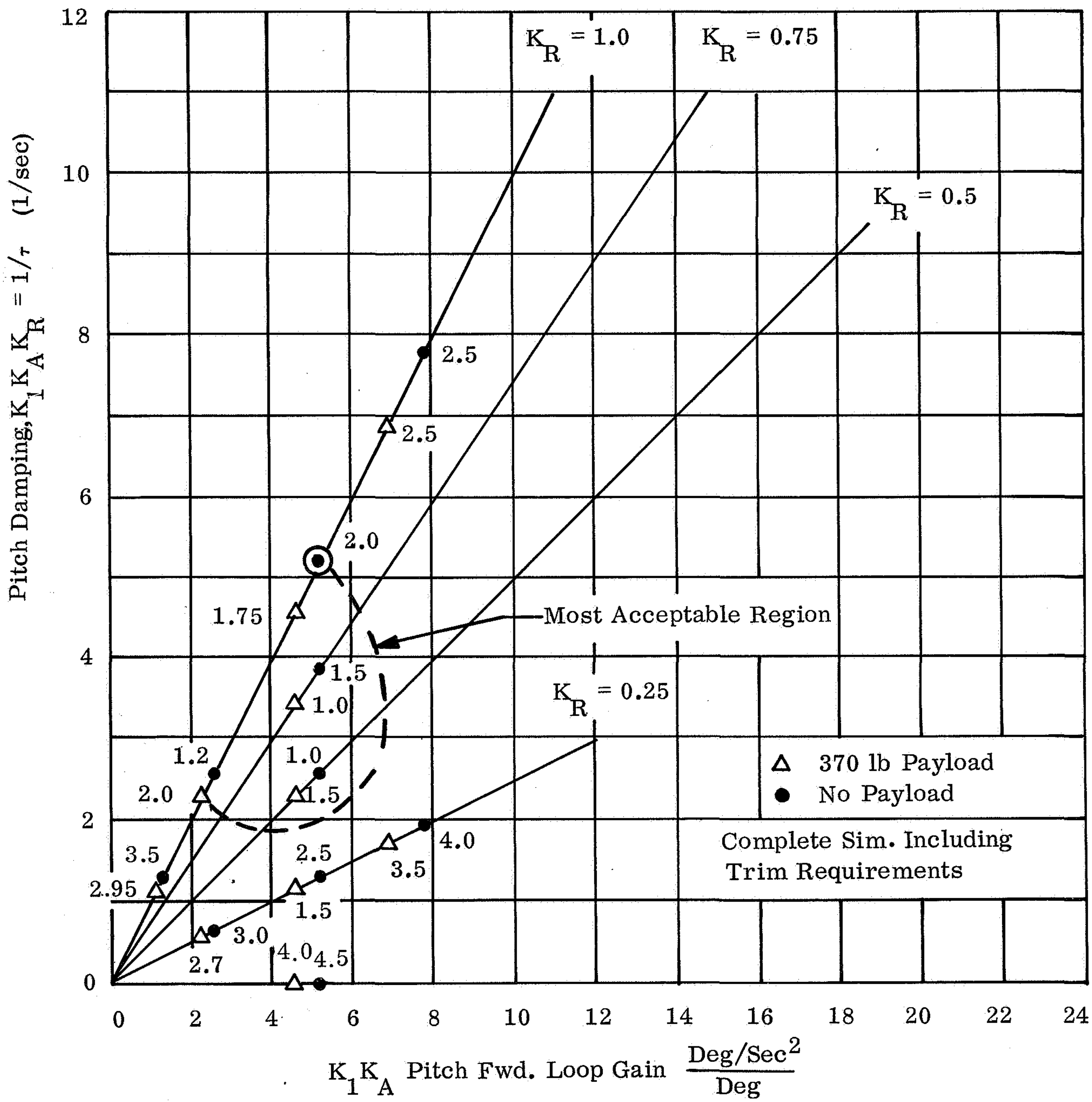


Figure 6.35. Cooper Ratings - Parametric Evaluation of Rate Gyro SAS Using Original Handle Bar (Pilot A)

Configuration	$K_A \frac{Deg/sec^2}{Deg}$		
	Pitch	Yaw	Roll
No Payload	2.6	2.04	2.06
370 lb Payload	2.3	1.07	1.64

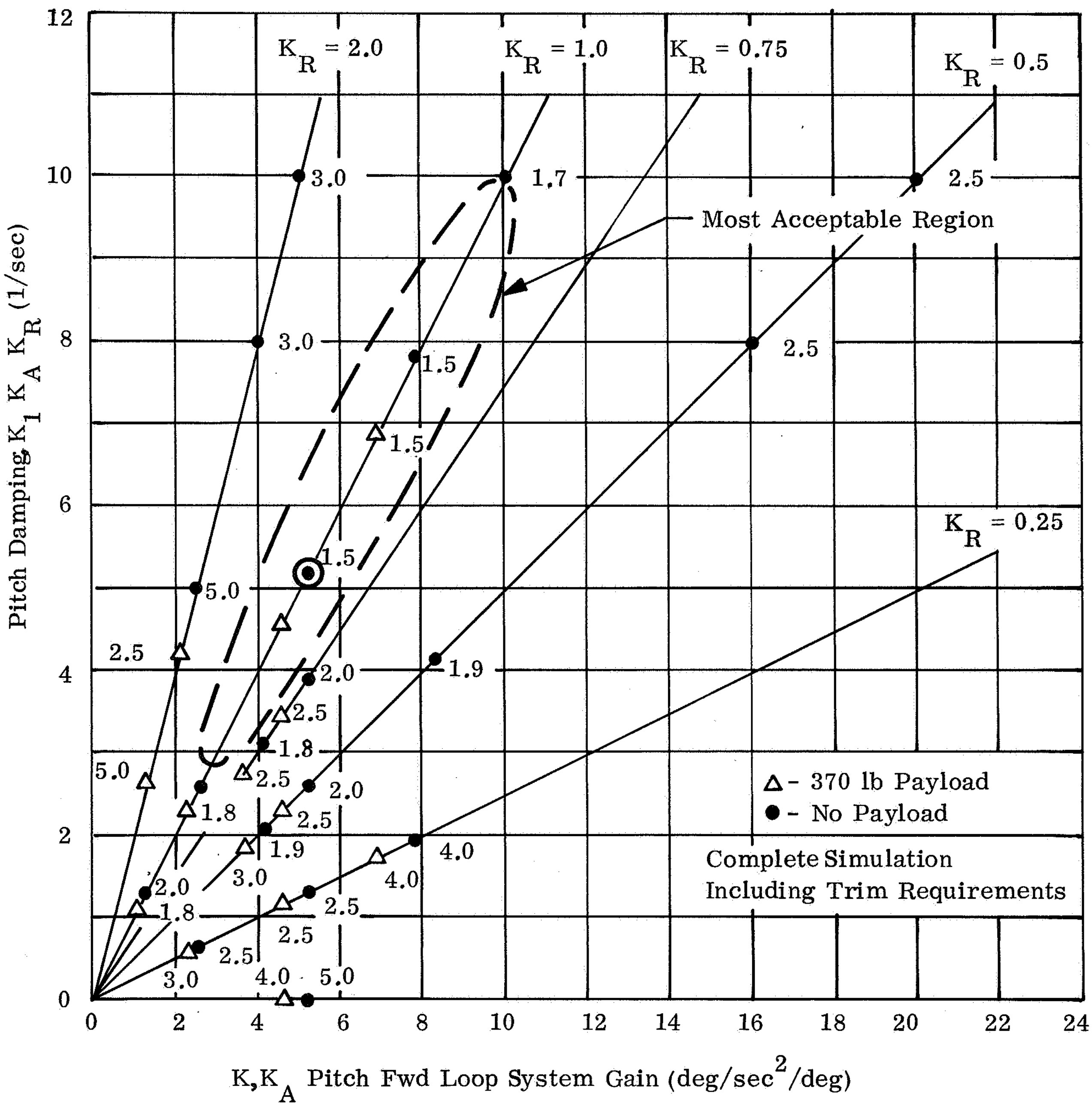


Figure 6.36. Cooper Ratings - Parametric Evaluation of Rate Gyro SAS Using Original Handle Bars (Pilot C)

Configuration	$K_A \frac{Deg/Sec^2}{Deg}$		
	Pitch	Yaw	Roll
No Payload	2.6	2.04	2.06

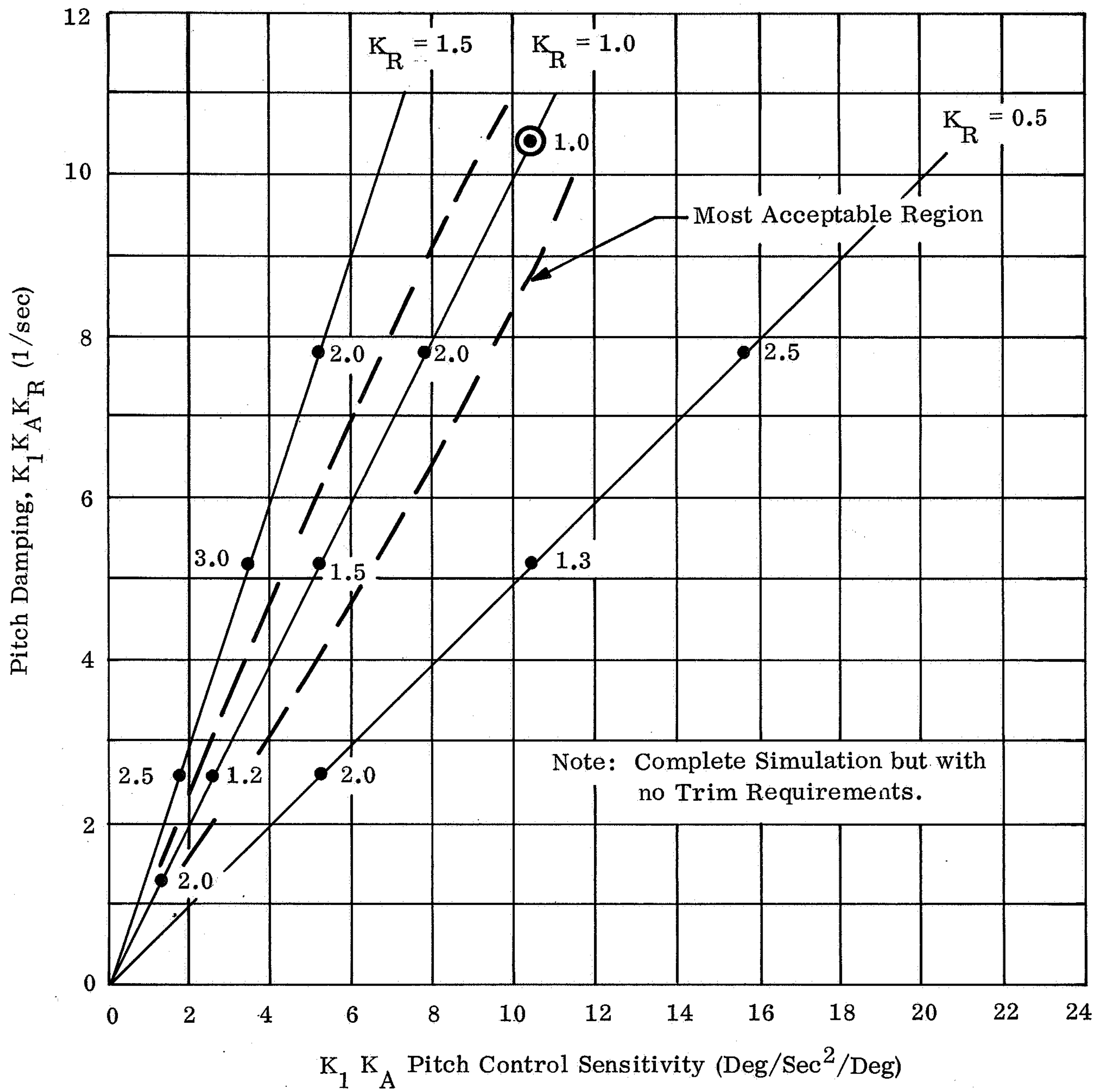


Figure 6.37. Cooper Ratings - Parametric Evaluation of Rate Gyro SAS Using Apollo Hand Controller (Pilot A)

Configuration	$K_A \frac{deg/sec^2}{deg}$		
	Pitch	Yaw	Roll
No Payload	2.6	2.04	2.06

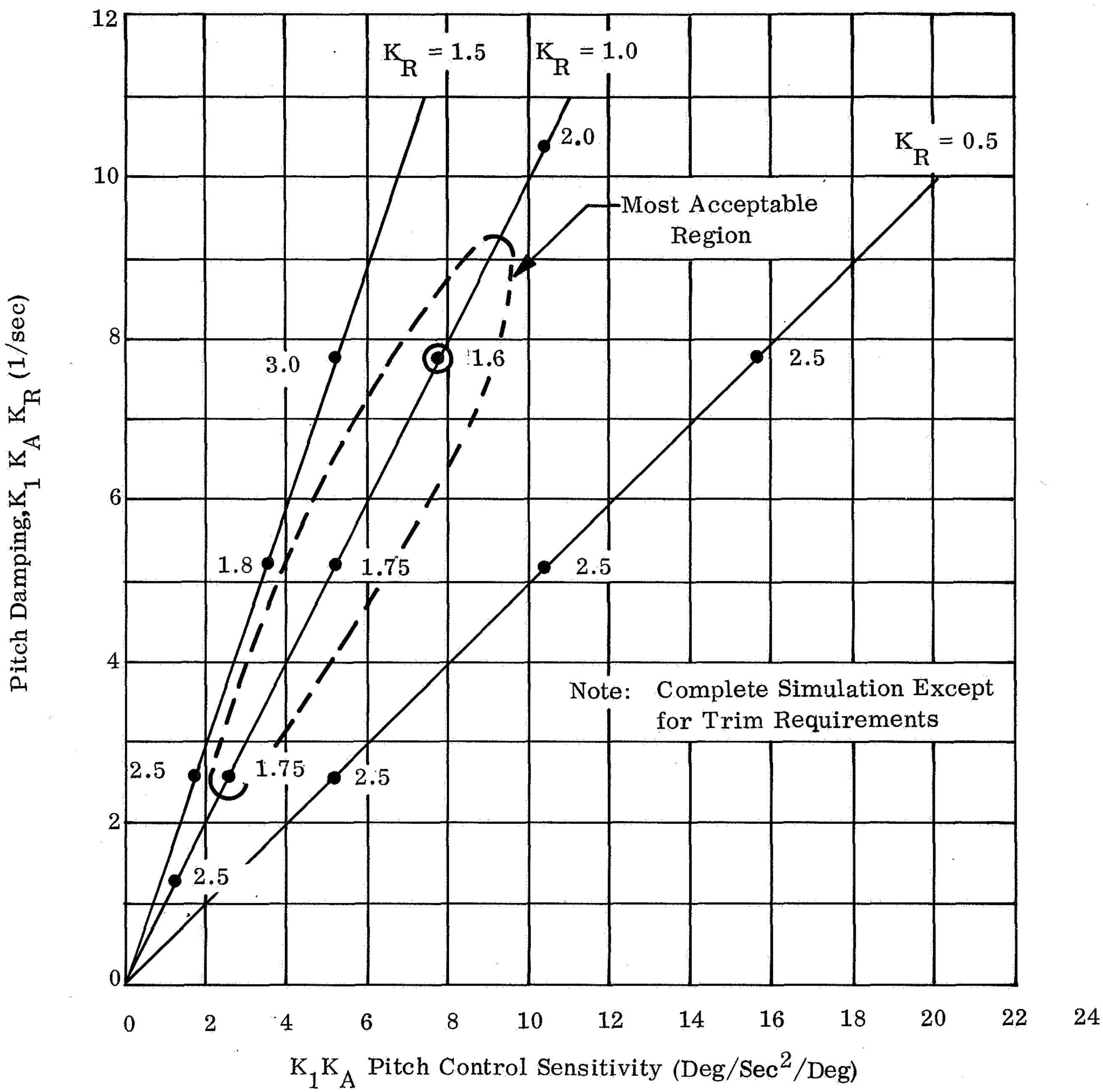


Figure 6.38. Cooper Ratings - Parametric Evaluation of Rate Gyro SAS Using Apollo Hand Controller (Pilot C)

the Cooper rating at the preference point degraded to 2.5 from 1.0 for pilot A and to 4.0 from 1.6 for pilot C.

6.6.1.2 Control Augmentation

A control augmentation system was simulated and evaluated to determine the potential of this much simpler, all-mechanical type of system to provide some or all of the benefits of an electromechanical SAS.

6.6.1.2.1 Description of Control Augmentation

A sketch of one axis of a control augmentation system and its parameters are shown in Figure 6.39. The complete equations of motion are derived in Appendix C. For this preliminary parametric study, it was assumed that the natural frequency (ω_n) of the engine spring arrangement could be made sufficiently higher than the control frequency ($\omega_c \approx 0.5$ cps) and with adequate damping such that engine inertia effects can be neglected. The transfer function $(\frac{\ddot{\theta}}{\delta_c})$ can then be approximated as follows: (Refer to Appendix C for the relation between the physical parameters and the terms of the transfer function.)

$$\frac{\ddot{\theta}}{\delta_c} = K \frac{\alpha \tau S + 1}{\left(\frac{S}{\omega_n}\right)^2 + \tau S + 1} \approx K \frac{\alpha \tau S + 1}{\tau S + 1}$$

Some general characteristics of the above transfer function are:

(a) If $\tau = 0$ (implies B = 0) or if $\alpha = 1.0$, the system becomes a proportional acceleration system (same as present manual control system) with a control sensitivity of K.

(b) If $K_T = 0$ in Figure 6.39, the vehicle response characteristics to controller inputs would be the same as that of a rate command system

$$\frac{\ddot{\theta}}{\gamma_c} = K \frac{\gamma \tau S}{\tau S + 1}$$

However, with $K_T = 0$, the control system is incapable of trimming out sustained disturbance moments.

(c) α must be greater than 1.0 in order to provide a lead-lag system for approximating the lead associated with a rate gyro feedback control system. Note if $\alpha < 1.0$, the transfer function becomes a lag-lead type network.

(d) The smaller the value of K, the smaller the proportional acceleration contribution to the rate command characteristics.

There are two incompatible requirements associated with the parameter K. To obtain controller trim position capabilities similar to that of the selected manual

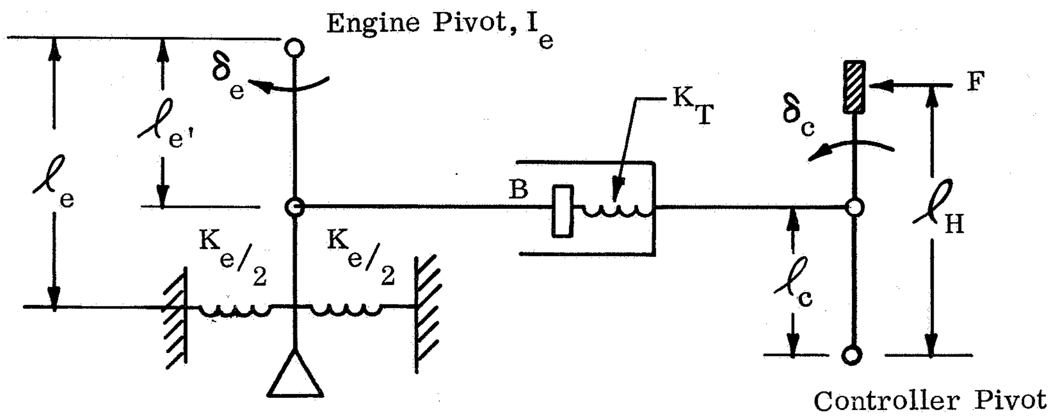


Figure 6.39. Sketch of Mechanical Control Augmentation System Concept

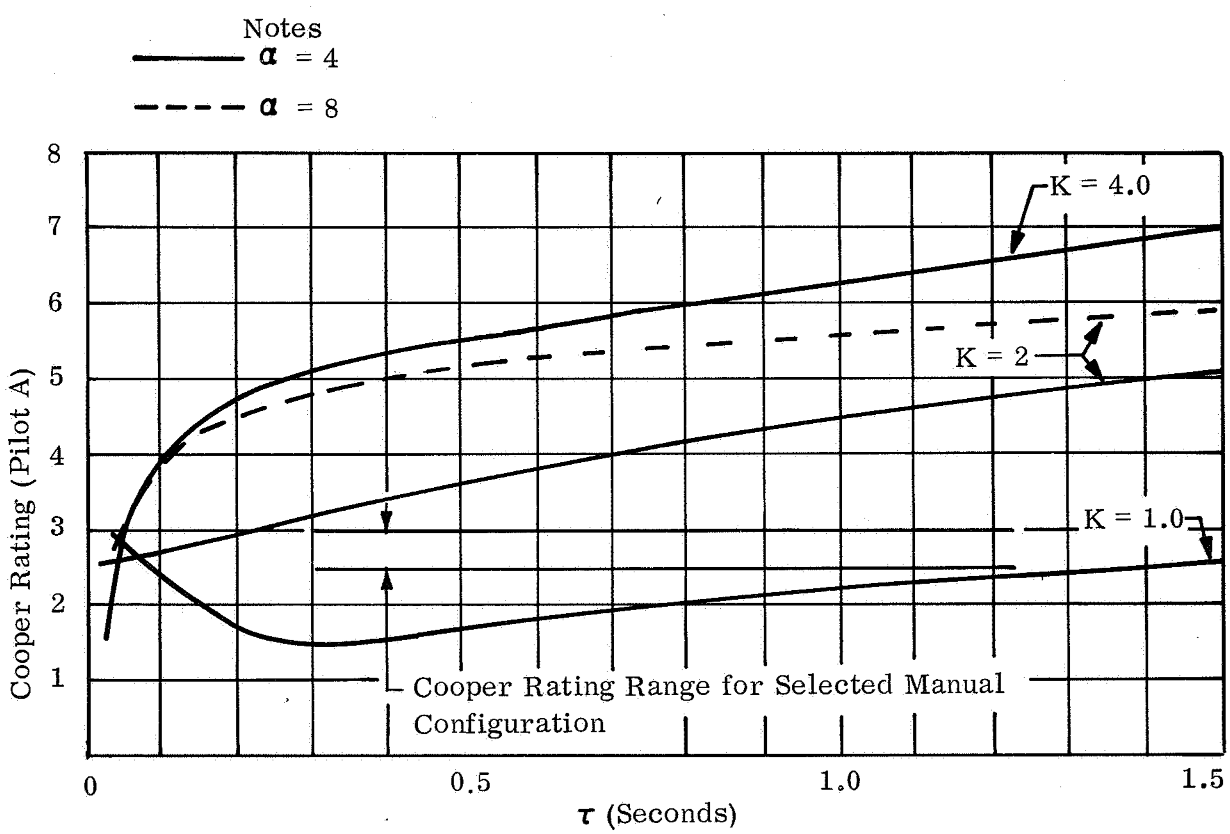


Figure 6.40. Parametric Control Augmentation Studies

vehicle implies that the value of K should be the same as the present control sensitivities. For pitch, this requires a K = 2.6 deg/sec^2/deg. However, to permit the response characteristics of this system to come closer to duplicating the desirable characteristics of a rate command system, requires K to be a minimum in order to reduce the acceleration contribution to the rate command level.(Figure 6-34).

6.6.1.2.2 Simulation Results

Piloted simulation evaluations were conducted using this type of control augmentation in the pitch axis (only) of the selected manual configuration. Figure 6.40 summarizes the cooper rating trends for several variations in α, τ and K.

As expected, a low value of K resulted in the most improvement in pilot ratings. The larger values of K resulted in such a large proportional acceleration contribution to velocity characteristics that the pilot could not accept an additional lead contribution. Thus, very small time constants were preferred at the higher values of K essentially making the system a proportional acceleration control system.

The best single axis design was used as the basis for simulating a three axis control augmentation system in pitch, yaw and roll. Two sets of system parameters were then reevaluated. One configuration used a K = 1, $\alpha = 4$, and a $\tau = 0.2$ in all three axes while a second configuration used a K = 1, $\alpha = 4$, with individual time constants (τ) of 0.2, 0.3 and 0.4 in pitch, roll and yaw, respectively. A pilot evaluation indicated he preferred the latter, which in turn was used in the system comparison study of 6.6.2.

6.6.1.3 Apollo Block I Hand Controller (with no stability augmentation)

In order to evaluate the use of an Apollo Block I hand controller, it was necessary first to determine the most suitable ranges of control sensitivity. This was done by assuming that the relative magnitudes of the pitch/roll/yaw control sensitivities would be the same as previously determined for the handle bar controller (K_A), but that a single gain factor, K_1, may be desired in all three axis.

Figure 6.41 shows pilot ratings obtained for the Apollo hand controller (with no stability augmentation). The control sensitivities investigated are shown as a fraction of the T = W control sensitivities proposed for the selected configuration using handle bars (Figure 6.31). The apparent best control sensitivities are about 2/3 of the proposed handle bar control sensitivities. These are the values which are used in subsequent comparison studies for the hand controller with no SAS.

6.6.2 Comparison of Systems

Direct comparisons were made of the selected unaugmented, stability augmented, and control augmented systems using handle bars and Apollo controllers. As mentioned previously, the use of the Apollo controller with control augmentation was not investigated. Although data on all of these system arrangements had been obtained

	Configuration Zero Payload — Full Prop. (No Trim Requirements)		
	Pitch	Yaw	Roll
K_A	2.6	2.04	2.06
K_1	Same value used in all three axes (varied from 1/8 to 1.0)		

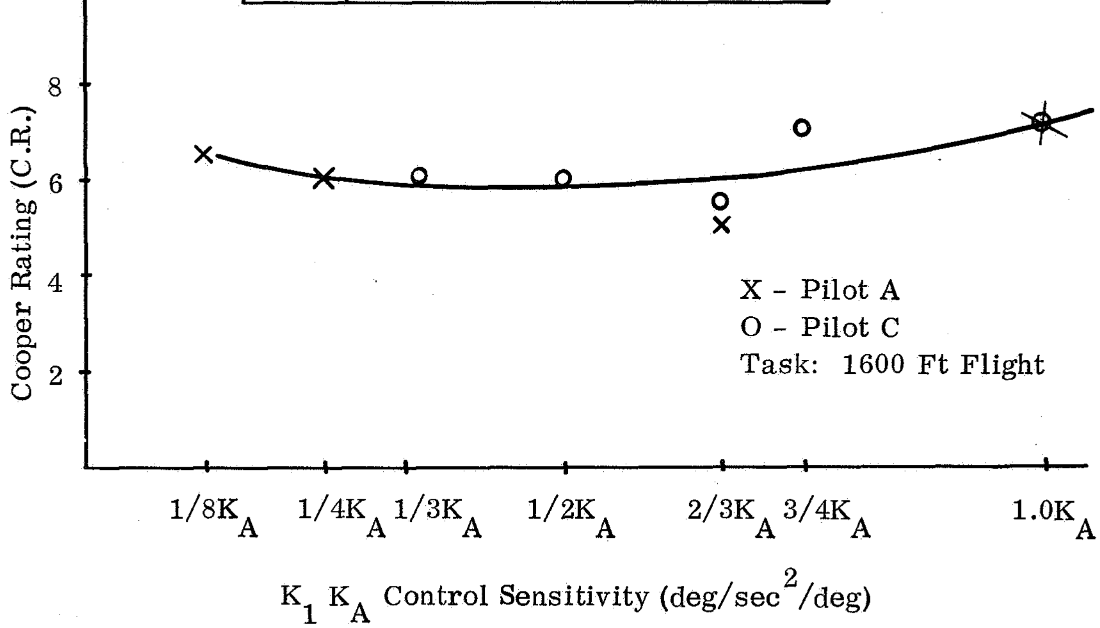


Figure 6.41. Apollo Hand Controller without Stability Augmentation (K_R = 0)

TABLE 6.15

PARAMETERS USED IN SYSTEM COMPARISON

System Configurations	K_A (deg/sec^2/deg)			K_1	K_R (deg/sec/deg)	K (deg/sec^2)/deg	a	τ (sec)		
	Pitch	Roll	Yaw					Pitch	Roll	Yaw
Handle Bars: No Augmentation	2.6	2.06	2.04	-	-	-	-	-	-	-
All Mechanical Control Augmen.	-	-	-	-	-	1	4	0.2	0.3	0.4
Rate Gyro SAS	2.6	2.04	2.06	2.0	1.0	-	-	0.217	0.243	0.245
Hand Controller: No SAS	1.73	1.36	1.37	-	-	-	-	-	-	-
Best Rated Gyro for Pilot A	2.6	2.04	2.06	4.0	1.0	-	-	0.096	0.122	0.121
Best Rated Gyro for Pilot C	2.6	2.04	2.06	3.0	1.0	-	-	0.128	0.161	0.163

during the previous parametric studies, this study permitted a direct comparison between the optimized alternatives. Comparison between systems using the same type controller were made in rapid sequence by switching on the analog computer. Comparisons between controllers involved a lapse of about a week to allow for mechanical modifications and proficiency training.

The characteristics of the systems compared are listed in Table 6.15 and the results of the comparisons are shown in Figure 6.42. Two comments should be made in explanation of Figure 6.42.

1. All comparison runs but one were made using the system parameter values preferred by the individual operator. The exception occurred in the case of pilot A evaluating handle bars with SAS. Here, he was given the somewhat longer time constant preferred by pilot C and which he had rated as much as 1.0 Cooper point worse than optimum during the previous parametric study. If an adjustment for this were in the results of Figure 6.42, both pilots would be in perfect agreement on the relative ranking of systems.

2. As described previously, the high spring centering rates of the Apollo hand controller made it necessary to eliminate trim movements from the simulation when it was being used. This complicates the comparison between the hand controller and the handle bars, since a simpler control task was involved with the former. On the other hand, both pilots felt the hand controller spring rates were too high even without trim effects. Hence, it is probable that a hand controller with much lower spring rates would still be closely comparable to the handle bars with SAS, even with trim included.

6.6.3 Hardware Implementation

A preliminary design of a typical three-axis rate gyro stability augmentation system was developed to determine representative weight, size, and electrical power requirements and verify that there are no major problems associated with its implementation and operation in the lunar environment.

6.6.3.1 System Description

The stability augmentation system is diagrammed in Figure 6.43 and 6.44 which are, respectively, a schematic diagram of the signal portions of the system and a block diagram of the associated power supply and test/monitor equipment.

The SAS attitude sensing system is based on a three axis rate gyro triad where the input voltage and output signals are 400 cycle. The gyros are supplied with a spin motor speed sensor output signal and a test torque input signal capability for assessment of their operational readiness. The output of the pitch and yaw rate gyros are fed collectively and differentially, respectively, into the two thruster deflection loops. The roll gyro output is fed directly into the roll differential throttle loop. Each of these loops contain a 400 cycle summary/power amplifier, a two-phase servo motor,

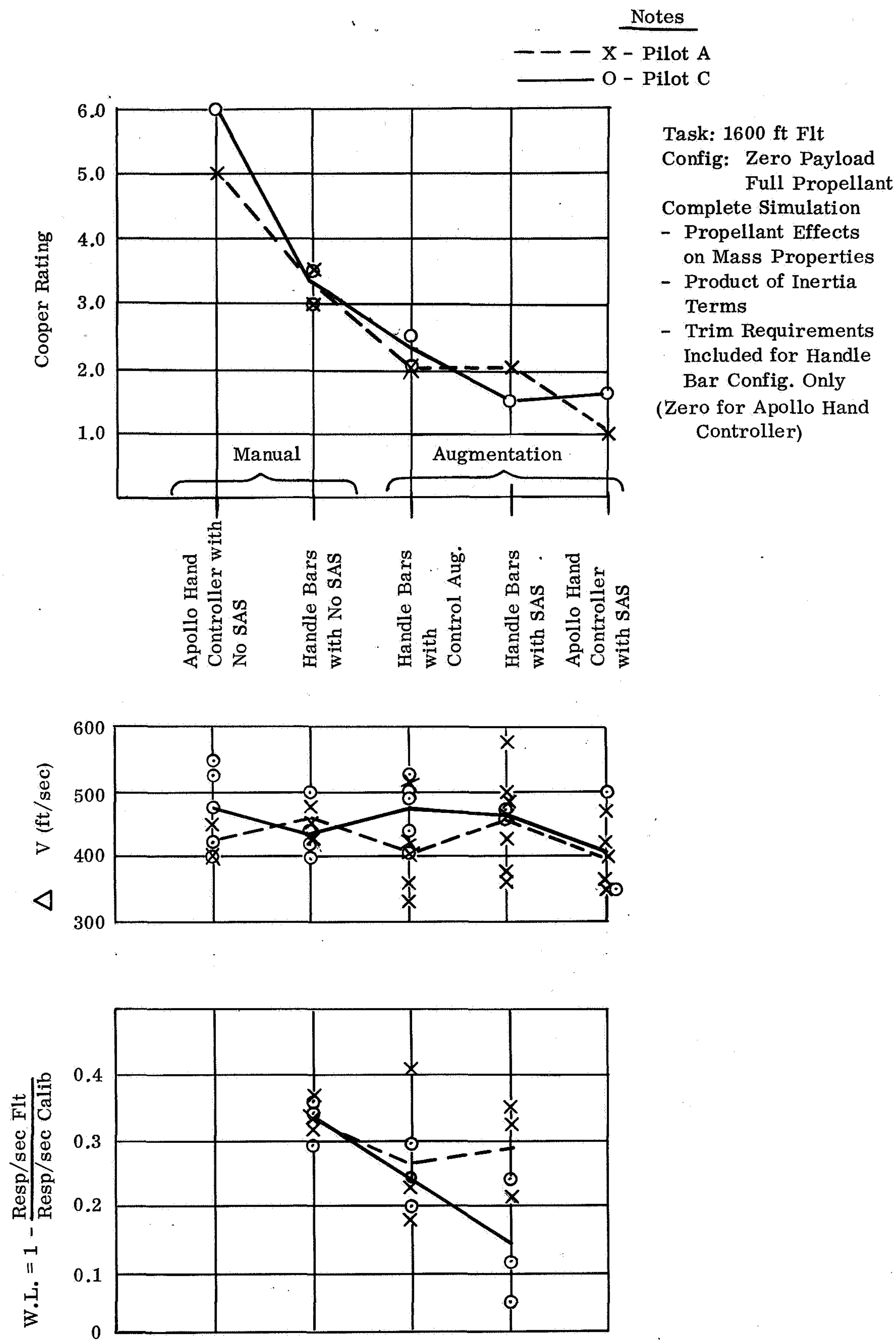


Figure 6.42. Control System Comparison Study

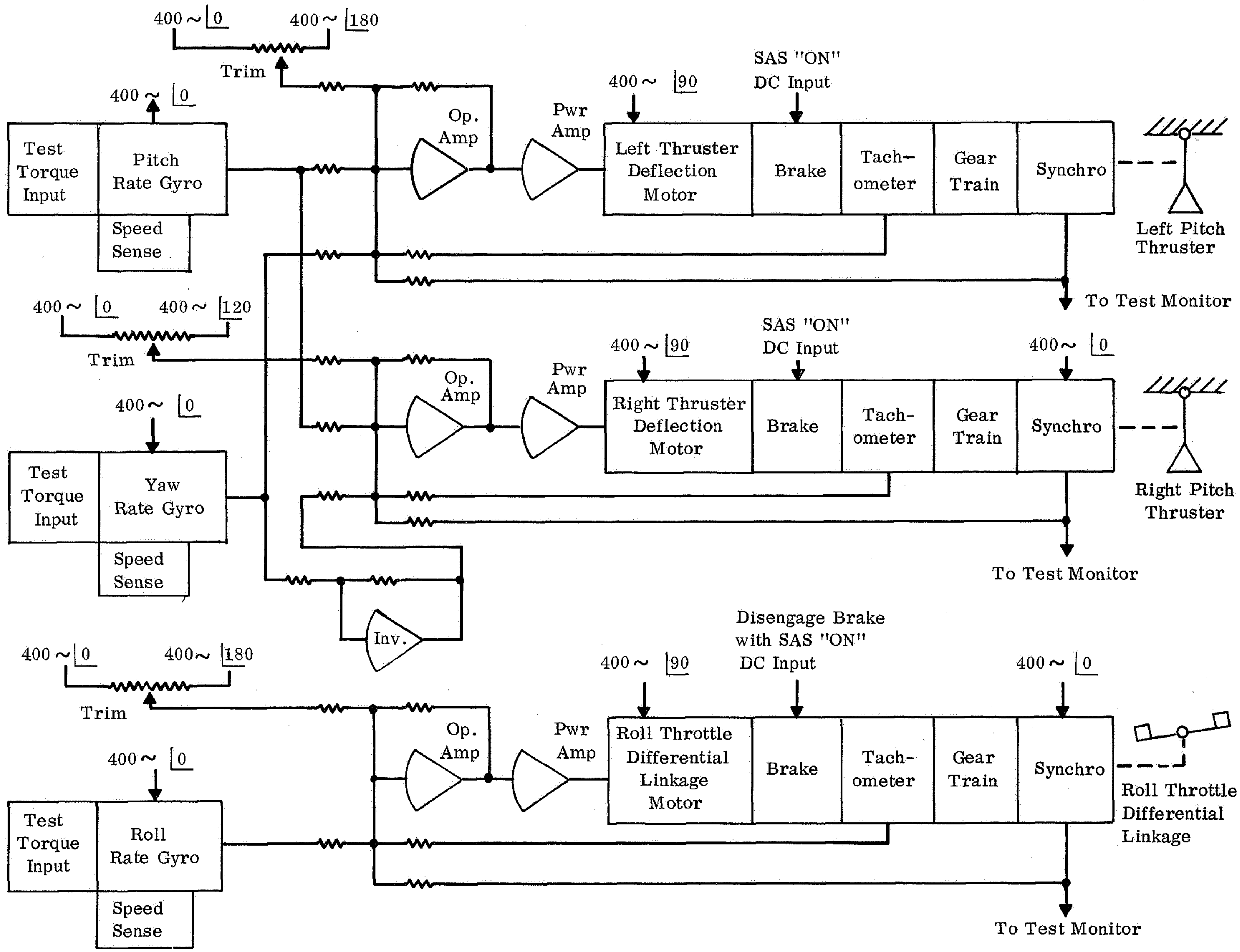


Figure 6.43. SAS Block Diagram

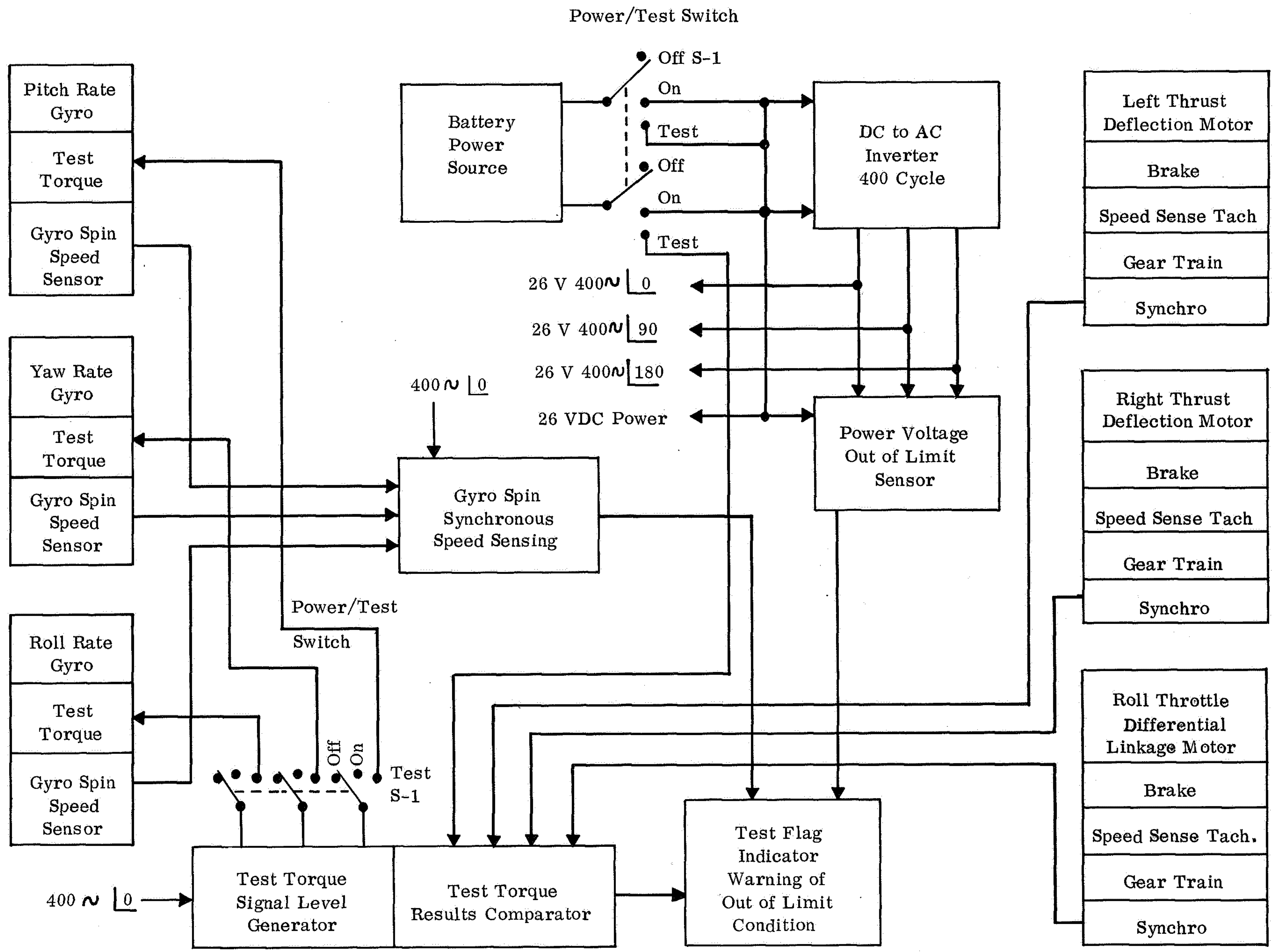


Figure 6.44. SAS Test Monitor and Power Supply Block Diagram

a tachometer to provide a damping signal, a syncro for position feedback, a normally energized brake, and a gear train.

The SAS motor/gear train outputs are mechanically summed with the manual inputs to the right and left thruster pitch deflection linkage and the roll differential throttle linkage. Anti backup devices are used as required to prevent actuator reaction forces from deflecting the manual controllers. With SAS turned off, or in the event of a power failure, the actuator brakes are engaged to prevent manual control inputs from driving into them rather than into the downstream linkages. SAS authority is limited to about one-third of the manual input authority so that adequate manual control is available with the actuators locked in any position.

The system monitor and test features include a warning flag indicator which shows the warning flag whenever it is deenergized. Thus, loss of power at any time automatically causes the warning to show. The warning also appears under any of the following conditions:

1. Gyro spin motor not up to synchronous speed (any one of the three rate gyros).
2. Any one of the supply voltages below limit (dc, 400 cycle, O phase, 90° phase and 180° phase).
3. During pre-flight switch selection to TEST, if the results of the torque level test input fail to cause any one of the SAS actuators to go to its correct test deflection value. The actual thruster deflection may also be visually monitored.

The only operating control in the SAS is a power OFF-ON-TEST switch. The switch is located on a small instrument panel adjacent to the left hand controller unit. The switch is guarded to prevent accidental activation. The TEST position is a spring loaded momentary contact position which is held manually in the TEST position for a few seconds to assure correct operation before the start of a flight.

The gyros require about 20 seconds to come up to operating speed. All other SAS components are ready for immediate operation following application of power. However, a period of one to two minutes of start up time and confidence test time is considered probable before the start of each flight.

6.6.3.2 System Components

Typical SAS equipment has been selected on the basis of state-of-the-art availability, space qualification where possible, high reliability, minimum size, weight and electrical power and compatibility with the lunar vehicle design and its mission. All electronic circuit elements are solid state and integrated circuits are used where applicable. Because of the preliminary nature of the design, conservative design practice was used. A list of components is given in Table 6.16. The total system weight increment is 30.95 pounds, including batteries to provide the required 72.8 watts of electrical power.

TABLE 6.16

SAS ELECTRO MECHANICAL COMPONENTS, TYPICAL

Part Description	No. Reqd.	Totals Power (w)	Totals Weight (lb)	Manufacturer
Gyro, Rate, Triad	1	15	3.25	U.S. Time Corp.
Trim potentiometer	6	0.5	0.4	Bourns Inc.
DC to AC Inverter 40W output	1	(10 losses)	3.0	Sanders Assoc.
Power Amplifier, AC	3	21	0.75	Weston-Transicoil
Operational Amplifier	3	0.9	0.2	Philbric Research
Servo Drive Motor, 400	3	15	2.7	Weston-Transicoil
Brake, Power on disengage	3	6	0.8	Weston-Transicoil
Tachometer, speedsensor	3	-	0.6	Weston-Transicoil
Gear train	3	-	1.2	Weston-Transicoil
Synchro sensor	3	2.7	1.2	Clifton Precision
Power/Test Switch	1	-	0.2	Honeywell
Flag Indicator	1	-	0.2	Weston Instrument
Diodes, Silicon	26	-	0.3	Texas Instrument
Battery Holder Case	1	-	1.5	Bell Aerosystems
Battery Cells, Silver Zinc	40	-	3.75	Yardney Electric Corp.
Logic microcircuit packages	41	1.7	0.5	Texas Instrument
Resistors	87	-	0.5	
Capacitors	65	-	1.4	Sprague Electric Co.
Inductors, coils, transformers	16	-	1.8	United Transformer Corp.
Circuit boards, printed	8	-	0.8	Photo Circuits Corp.
Connectors	9	-	0.9	American Phenolic Corp.
Misc. components, hardware	-	-	1.5	
Main chassis and case	1	-	2.0	Bell Aerosystems
Wire and Cable			1.5	
Totals		72.8	30.95	

6.6.3.3 Electrical Power System

The maximum mission duration for one load of propellants is approximately 10 minutes but this could be in as many as 10 separate flights of 1.0 minute each. Assuming a 2.0 minute warmup/test period with each flight, a total SAS operating time of 3.0 minutes per flight and a total time of 30 minutes is obtained. Thus, a total load capacity of 36.4 watt-hours is required.

Following standard practice, a 100 percent reserve capacity is provided using two parallel but electrically isolated banks of batteries, each capable of supplying the entire electrical load for a half hour. Each bank consists of 20 Yardney Electric Corp. Type HR-1 cells connected in series to provide a 1.4 ampere-hour rating at a terminal voltage of 28 volts, for a 39 watt-hour nominal capacity. The cells weigh approximately 1.1 ounces each including fluid electrolyte but not including interconnecting wiring assembly holder case, etc. The total cell weight for two banks is therefore 2.75 lb.

A double compartment pressurized case is used to contain the battery cells, with one battery bank in each half of the case. Each half is separately pressurized to further minimize the risk of a single failure causing loss of all power. The individual cells are semi-sealed and no leakage is evident in normal operation. They have been used and qualified on other NASA and military space programs.

The wiring diagram of the power source is presented in Figure 6.45. Isolating diodes are used to prevent the discharge of a good battery bank into a faulty bank (battery bank with a shorted cell).

6.6.3.4 System Installation

The SAS drive motor gear trains are coupled to multiturn jack screw thruster deflection linkages which have only a fraction of the total deflection capability, typically in the order of 30 percent for SAS so that in the event of a SAS malfunction, 70 percent is still available for the mechanical/manual mode of control. In the event of a loss of SAS power or when the SAS is deenergized, the SAS motors and gear trains are braked automatically. Thus, the SAS portion of the thruster deflection becomes fixed at its last set value. No sudden transient is introduced and the manual mode is immediately and continuously available.

The SAS drive motors and gear trains are located in the thruster control linkages. The remainder of the SAS equipment is located in the vehicle equipment housing adjacent to the fuel tanks for good thermal stability. The batteries are contained in a sealed pressurized container. The gyros are individually sealed units. The other SAS components do not require sealed, pressurized containers but are housed in a semi-sealed container to exclude dust and particles and afford some mechanical protection.

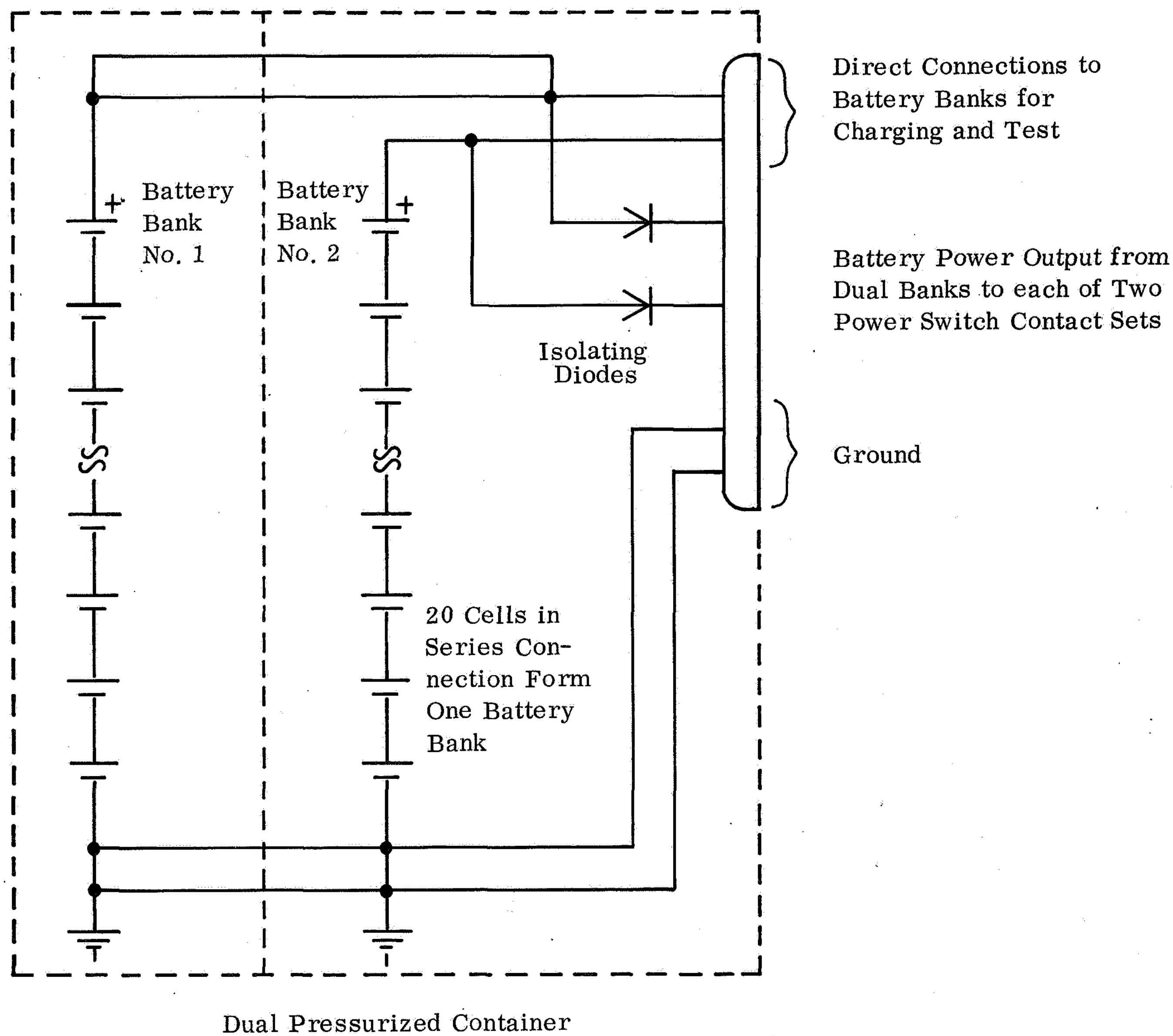


Figure 6.45. SAS Power Source Diagram

6.6.4 Conclusions

The augmentation system study has shown that while the handling characteristics of the unaugmented vehicle are satisfactory, they can be improved by the use of the relatively simple mechanical control augmentation system, and still further improved by the use of an electromechanical stability augmentation system.

However, there are also disadvantages with the use of these augmentation systems. Specifically, SAS would involve penalties of the following types:

1. Reliability - The system utilizes a number of rate gyros, servomotors, and other relatively low reliability electromechanical components, making it by far the least reliable system of the vehicle. The primary jeopardy is to mission completion and operational readiness rather than safety, because if the system fails, it reverts to a satsifactory acceleration command mode. However, flight safety is somewhat affected by the fact that the pilot (1) may not be as well trained or experienced in the acceleration mode as if SAS had not been provided, (2) may be required to transfer instantly and without warning from one set of handling characteristics (SAS) to another (no SAS), and (3) this transfer may be accompanied by a noticeable transient which must be corrected with the unaugmented system.
2. Weight - The preliminary design study has shown that the incorporation of a SAS will impose about a 30-pound weight penalty.
3. Operational Consideration - The system must be warmed up and checked out prior to each takeoff.
4. Cost - The system must be developed and space qualified.
5. Training - The operator must become proficient at flying both the augmented and the unaugmented vehicle, and at handling failure-induced transients.

These disadvantages are felt to outweigh the advantage of providing a non-essential improvement in handling characteristics. Therefore, a rate gyro type of stability augmentation is not recommended for the OMLFV.

The same disadvantages apply to the mechanical control augmentation system, but to a much smaller extent. In all areas except training, the mechanical control augmentation system will lie much closer to the unaugmented system than to SAS. Somewhat more training may be required with mechanical control augmentation than SAS, however, because two different failure modes are possible and the pilot must be proficient with each. A seized damper would leave the operator with a relatively high gain unaugmented system, while a leaky damper would leave him with a low gain system. The mechanical control augmentation system design was not carried far enough to determine an accurate weight, but a three-axis system is conservatively estimated at 5 pounds.

Although the mechanical control augmentation system has considerable promise for achieving significantly improved handling qualities with minimal disadvantages,

more development to verify that its potential can be achieved in practice is needed before it can be confidently recommended for the OMLFV. Hence, an unaugmented control system is recommended at the present time.

With regard to controller type, these studies have shown the handle bars to be preferable to the Apollo hand controller for unaugmented control and closely comparable to it when SAS is included. Because unaugmented control is used as a backup mode even in the stability augmented control system, a handle-bar type of controller was selected for the OMLFV.

7.0 THERMAL ANALYSIS

This section reports the thermal analysis of the critical areas of the OMLFV. The primary thermal design constraint requires that the propellant be maintained at 70 ±30°F at all times during the mission. Other limiting factors are controlling the helium tank temperatures below 120°F as well as protecting electrical subsystems and power supply. The present design concept is different in many thermal respects from the previous MFS design (Ref 1), i.e., the storage requirements are not the controlling factor for the selection of an outer thermal coating. (The coatings for the MFS were selected to satisfy a requirement for 180 day storage on an exposed shelf of the LM.) The OMLFV is stored within a compartment of the LM, and consequently the OMLFV follows the LM thermal cycle. Thus, the governing requirement is to select thermal coatings which will give adequate passive thermal control during lunar daytime operations.

7.1 ENGINE - SHIELD TEMPERATURE STUDY

A detailed heat transfer study has been conducted to investigate thermal coupling between the thrusters and shielding so as to select a design whereby heat radiated from the thrusters and the heat from the plume impingement will not be detrimental to the vehicle. The results of the study indicate that the maximum temperature experienced by the vehicle shield will not exceed 880°F at full engine thrust during steady state operation if the engines are canted outboard 10° at a separation distance of 13.4 inches.

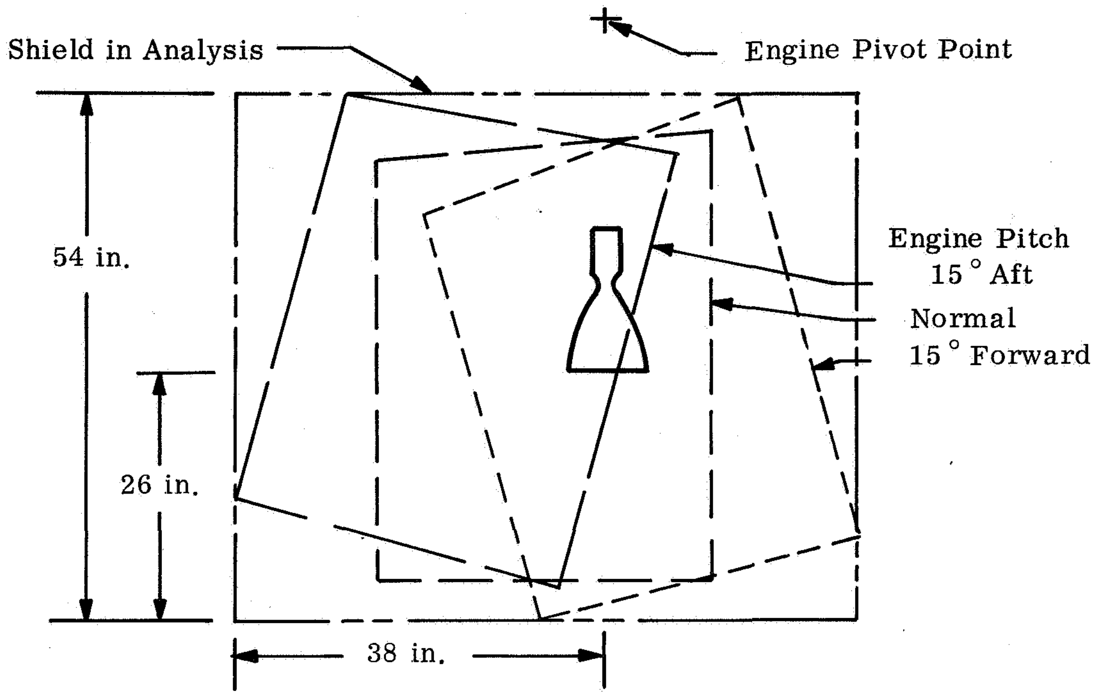
7.1.1 Heat Transfer Mathematical Model

During the course of this analytical study, various engine cant angles and various distances between the engine and the shield were considered. The positions corresponding to pivoting the thruster both forward and aft 15° were considered by analyzing one shield which encompassed all engine-shield positions. As shown in the sketch below,(a) the shield studied is somewhat larger than the actual shield, thus the temperature predicted for both the engine and shield are slightly higher than actual.

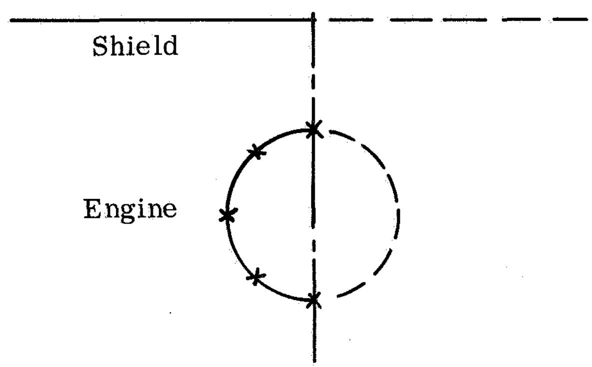
In order to compute temperatures on the shield and engine, it was necessary first to set up a mathematical model of the physical system and describe the boundary conditions influencing the temperature.

Because of similarity in geometry and the desire to minimize the total number of discrete elements or nodes, only one-half of the shield and engine was analyzed. The division was made through the engine centerline and perpendicular to the shield surface. The half shield was subdivided into 35 rectangular nodes. The engine was subdivided into nine axial stations and, with the half engine, only five equal circumferential sectors at each station were required to establish the circumferential temperature gradient as shown in the sketch below: (b)

Shield Geometry for the Analysis



(a)



(b)

The engine was further subdivided to include nodes on both the outer and inner surface to establish the thermal gradient through the wall thickness.

Conduction heat transfer was accounted for between the adjacent nodes in the engine; however, conduction was neglected in the thin Inconel shield.

Radiation interchange configuration factors were calculated relating the nodes on the shield and on the engine as well as view factors for the engines and shield nodes to space and to the lunar surface for various cant angle/separation distances. The mathematical model accounted for the radiation interchange between nodes on the engine inner surface as well as radiation out of the bell.

7.1.2 Engine and Shield Emissivities

The emissivities assumed for the inner and outer surfaces of the thruster were 0.7. This value has been found to be characteristic of the silicide coating applied to columbium thrusters to resist oxidation. An emissivity of approximately 0.8 was selected for the shield surface in order to promote heat loss by radiation as opposed to a low emissivity indicative of a highly reflective surface. Promoting radiation is particularly desirable for the shield area which is heated by the convective exhaust gas impingement. (A coating with similar properties is used on the LM spacecraft, in areas exposed to the impingement of reaction control rocket engine exhaust plumes or radiation from rocket engine components.) The emissivity assumed for the lunar surface was 0.927. An accepted value for the lunar spherical albedo is 0.073.

7.1.3 Analytical Boundary Conditions

The boundary conditions employed in the analysis accounted for thruster heating due to combustion, shield heating due to the thruster exhaust plume impingement, incident and reflected solar radiation, radiation to the lunar surface and radiation to space.

The combustion heating rates for the 150 lb thrust OMLFV engines, i.e., the local combustion gas temperatures and heat transfer film coefficients, were obtained from analytical correlations of data obtained during earlier testing of a similar 100 lb thrust engine. (Film cooling on the inner wall was not considered.) The nominal presumed for analyses were a mixture ratio of 1.3 and a chamber pressure at full thrust of 80 psia.

To obtain shield heating rates resulting from plume impingement, a flow pattern for a nozzle exhausting into a vacuum was derived from existing data reported in Reference 15. With a knowledge of the free flow Mach number before impingement on a surface, the pressure ratio, temperature ratio and Mach number change across an oblique or normal shock were predicted. This information, along with the combustion gas properties behind the shock, was employed to determine the convective heat transfer rates on the shield surface using empirical relations from Reference 15. These heat transfer rates were calculated for various thruster cant angles and distances from the shield.

The magnitude of the solar heat flux incident upon the shield was determined from a simple heat balance considering the angular direction of the solar vector. Assuming the shield to be perpendicular to the lunar surface, the maximum shield temperature will not occur with the solar vector normal to the shield surface because at the same time the shield views a relatively cool lunar surface. In the heat balance

equation which includes solar heating to the shield and to the lunar surface as a function of the solar vector angle, radiation interchange between the shield and the lunar surface, and radiation from the shield to space, it was found that the greatest shield temperature (268°F), without engine effects, occurs with the solar vector at an angle 25° above the normal to the shield, i.e., 25° above the horizon. As a result, a solar radiation heat flux rate of 320.5 BTU/ft^{2}hr was imposed upon the shield nodes. The solar absorptivity of the shield was assumed equal to the emissivity.

To account for shield and engine radiation to the lunar surface, a lunar surface temperature of 113°F was assumed, consistent with the above boundary conditions.

A value of -460°F was used for the space sink temperature.

7.1.4 Discussion of Results

The maximum temperatures on the shield occur along the line of intersection between the shield and a plane through the engine centerline perpendicular to the shield since this is where the engine is nearest the shield. Figure 7.1 shows temperature distributions along this line for various engine - shield positions. The separation distance noted on the figure is the distance from the center of the nozzle exit plane to the shield along the line perpendicular to the shield surface. In each position, two peak temperatures occur. The peak nearer the shield bottom is caused primarily by the exhaust plume impingement, whereas the peak nearer the top of the shield results primarily from the heat radiated from the thruster. Temperatures are reduced as the separation distance is increased. As the cant angle is increased, the peak temperature in the region of the plume impingement is reduced. However, at a fixed separation distance, increasing the outboard cant angle moves the engine chamber section closer to the shield and as a result, the shield temperatures near the engine increase. Since the pivot point of the thruster and arm are considered fixed, when the cant angle is increased to +10°, the separation distance changes from 9.7 to 13.4 inches. Thus a comparison of the two results for design purposes should be on the basis of 5 degree cant at 9.7 in. and 10 degree cant at 13.4 inches. For the selected configuration, the engine is canted 10 degrees outboard and the separation distance is 13.4 inches. In this position, the peak shield temperatures are 770°F in the plume region and 880°F for the thruster region.

The engine outer surface temperatures nearest the shield (computed for the selected vehicle design) are 2375, 2374, and 1302°F for the chamber section, throat station, and exit station respectively. Correspondingly, the engine temperatures on the side away from the shield are 2370, 2371 and 1280°F, giving an indication of the circumferential thermal gradient caused by the shield. If no shield was present, the same locations would be 2370, 2371, and 1277°F with no circumferential gradient. These temperatures show that the selected design shield has little effect on engine temperatures, i.e., the greatest increase is approximately 20°F at the exit station.

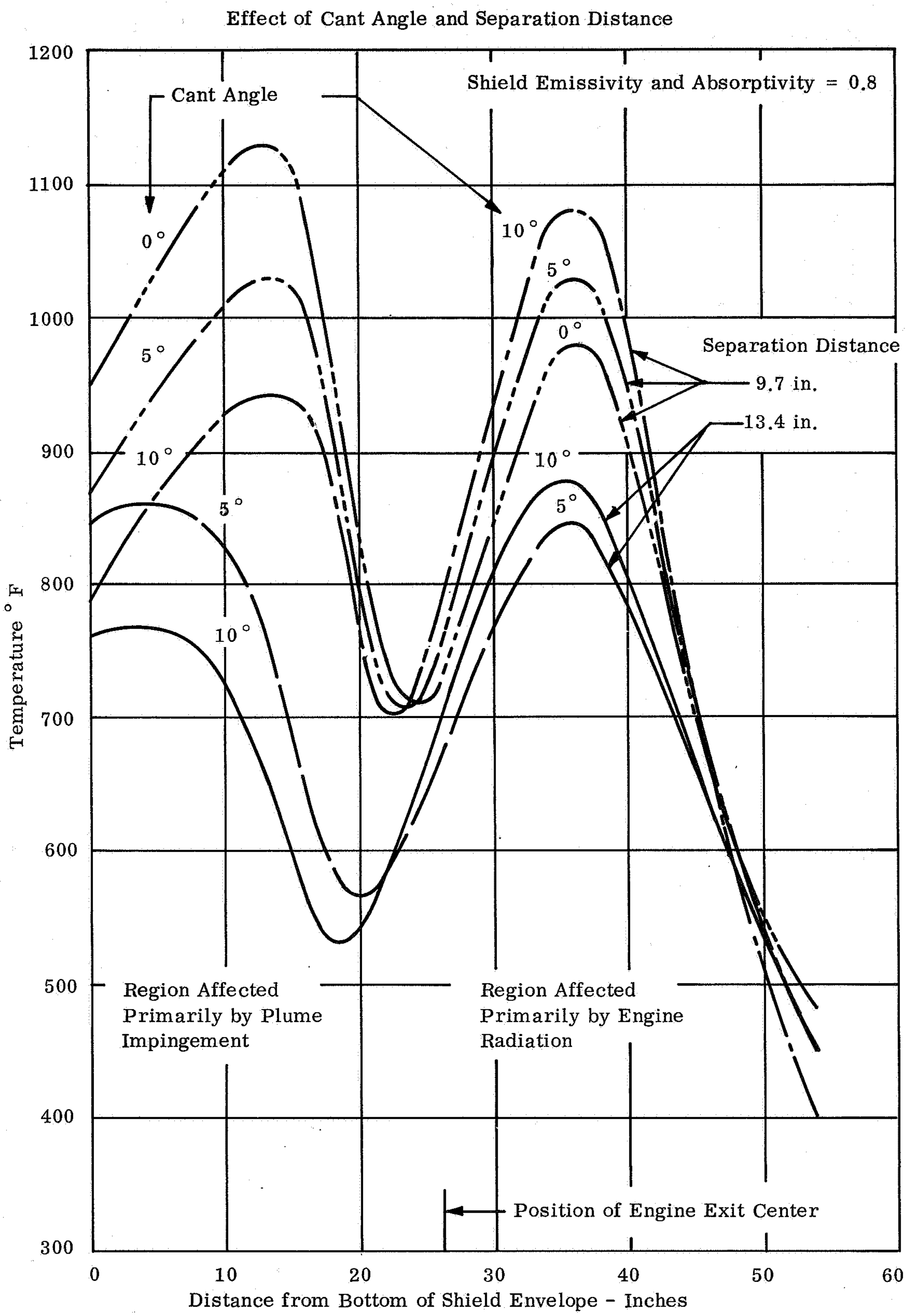


Figure 7.1. Temperature Distribution on the Shield along the Intersection of a Plane Perpendicular to the Shield through the Engine Centerline

The temperatures computed for the inner surface of this particular engine indicated temperature gradients through the wall to be 13, 80, and 1°F at the chamber section, throat station, and exit station respectively. The study indicates that these gradients are approximately constant around the engine circumference.

For the case where the thrusters are nearer the shield, i.e., a 10 degrees cant angle and a 9.7 inch distance, the engine temperatures nearest the shield increased 2.4, 3.4, and 14.7°F at the same three locations.

In the computations, a shield emissivity of 0.8 was assumed. To show the effect of reducing the shield emissivity, an emissivity of 0.25 was assumed and temperatures again computed for the selected configuration. In the plume region, the shield peak temperature increased from 770 to 1120°F. In the engine radiation region, the shield peak temperature remained about the same (880°F) but engine temperatures increased somewhat. (Heat transfer to and from this region is primarily by radiation).

Studies were conducted to determine the effect of engine mixture ratio on shield temperatures. A change in mixture ratio from 1.3 to 1.6 for the selected configuration was found to increase shield temperature in proximity to the engine approximately 20°F, whereas temperatures near the bottom of the shield were increased less than 5°F. The thruster temperatures were increased by 159, 176 and 106°F in the chamber section, throat station and exit station respectively.

Figure 7.2 presents a temperature map of the entire shield surface analyzed for the selected configuration (10 degrees cant angle, 13.4 inches separation distance). The figure illustrates that nowhere is the shield temperature less than 300°F.

7.2 DESCRIPTION OF VEHICLE THERMAL SHIELDS AND COATINGS

Thermal shields are required on each side of the vehicle as well as on the landing gear to protect critical components and primary structure from plume impingement and engine radiation heating effects.

The side shields consist of a thin Inconel 600 sheet, backed by 20 layers of high temperature multilayer insulation, e.g., Polyimide (Dupont H-Film) 0.5 mils thick and aluminized on both surfaces (emissivity = 0.05).

The outer surface of the shield is coated so as to have an emissivity of 0.8 or greater in order to keep the shield temperature at a minimum. Suitable coatings are available in either white or black paints which have been verified and tested in the space environment (Ref 16). However, these coatings would require qualification in an exhaust environment.

The recommended finish is a chromium-oxide pigmented coating which has an approximate upper temperature limit of 800°F when cured only and 2000°F when vitrified. This coating is currently used as a thermal control coating on the lunar module. The insulation blanket is attached to titanium structure which serves as a retaining wall for the vehicle propellant tank compartment.

Engine Cant Angle	10°
Separation Distance	13.4 in.
Shield Emissivity and Solar Absorptivity	0.8

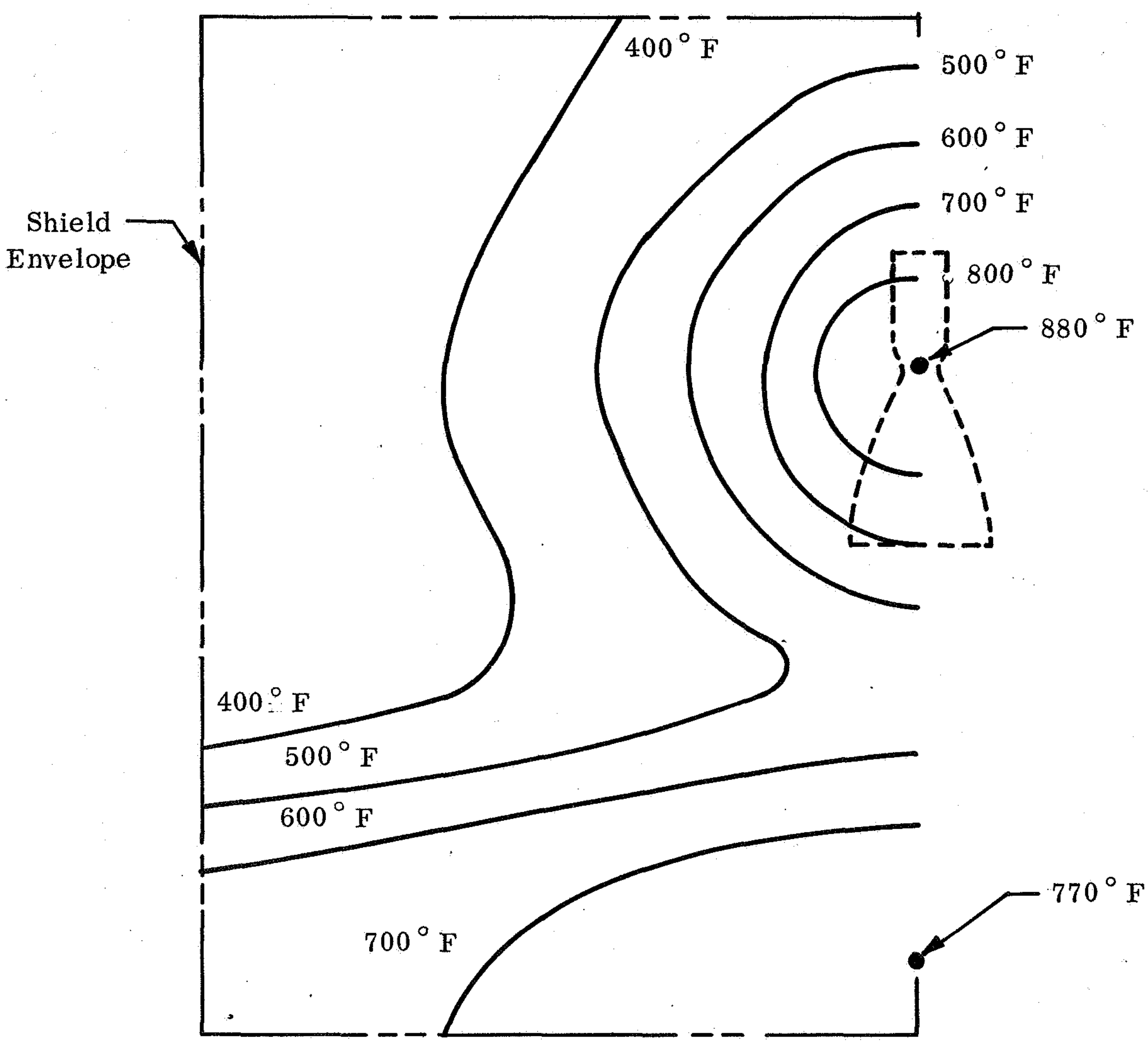


Figure 7.2. Constant Temperature Lines on the Shield Surface

The fact that the edges of the shield are so hot suggests that the thermal shield must be designed to protect the astronaut as well as the payload on the front of the vehicle against radiation and plume heating. This is accomplished by extending the edge of the shield outwards to deflect the plume.

The titanium landing gear legs are subject to exhaust plume heating and must be protected to avoid exceeding the design limit of 400°F. A titanium half shield is attached to the upper section of the landing gear legs.

When the OMLFV is carrying the maximum payload, each engine is set at its maximum forward position, i.e., the forward landing gear legs are less than 15 inches from the plume center line. Initial studies indicated that under these conditions, the peak temperature of the landing gear could be as high as 880°F when no shield was used. Further studies determined that a shield with controlled thermal coatings could reduce the temperatures significantly.

The recommended method of temperature control is to apply a coating to the outer surface of the shield (emissivity $\geqq$ 0.8). This high emissivity coating will re-radiate most of the convective heating from the plume. The inside surface of the shield has an emissivity = 0.24 ("as received" titanium). The emissivity of the upper surface of the titanium legs is required to be low to reflect the radiant heat from the shield. By applying a silicone-based adhesive backed aluminum foil, an emissivity of 0.05 can be obtained (see Ref 16). The lower surface of the landing gear legs is required to have a high emissivity coating (emissivity $\geqq$ 0.8) to radiate as much heat as possible to the lunar surface. Figure 7.3 shows that with a shield installed, the peak temperature of the landing gear legs is reduced to 420°F which is an acceptable value. This temperature reduces to 270°F at the far side around the periphery of the legs. The shield peak (near side) temperature is 880°F which is approximately the peak temperature of the unshielded legs. However, the temperatures around the periphery of the shield are reduced to values equal to the leg temperature. Accordingly, brackets should be attached from the shield extremities to the legs so as not to suffer any degradation in the thermal performance of the shield. It is also required to extend the shield to cover at least 31 inches of each leg length to maintain the maximum temperature of 400°F at all locations on the leg.

If it is necessary to provide leg temperatures less than 400°F, intermediate foil shields can be inserted between the outer shield and the titanium legs. If five intermediate shields (with emissivity = 0.05) are used, peak temperatures can be reduced to 260°F.

During a maneuver the plume axis will impinge almost directly on the landing gear legs. An estimate of the transient temperature response of the shield was made to determine the severity of heating for a short period of time. The shield is considered to be 0.01 inches thick titanium and will respond initially at a rate of 234°F/second. The temperature response of the protected landing gear will lag behind the shield response since it acts as a heat sink during transient heating.

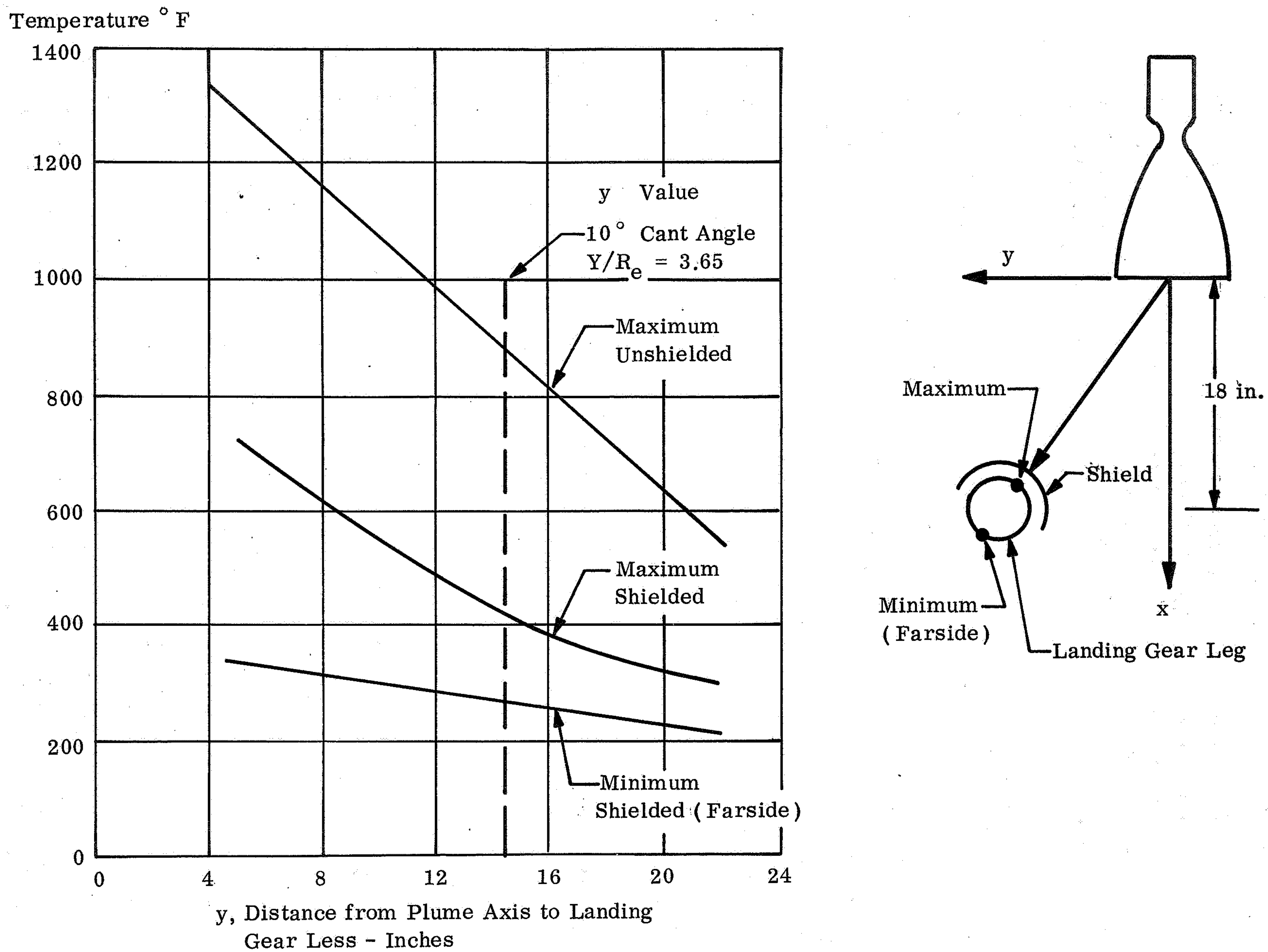


Figure 7.3. Landing Gear Leg Temperature with and without Shield

7.3 PROPELLANT TANKS AND HELIUM BOTTLE ISOLATION/THERMAL DESIGN

The problem of heat leak to the propellant tanks is less severe than in previous studies (Ref 1) wherein the lunar vehicle was presumed stored on an exposed shelf on the lunar module and had to withstand a total time of 180 days storage i.e., six complete lunar days. In the present application the vehicle is stored completely within an insulated compartment on the LM, at the same temperature as the LM (i.e., 70°F) and the associated transient temperature excursion during storage are tolerable.

The present design consists of a "bird cage" type base mount fabricated from 0.10 inch titanium. (The "bird cage" is an expression to describe the thermal standoff between the tank and the flyer deck). The titanium mount has cut-outs which serve to reduce the conduction paths and also serve to lighten the engine mounting structure. The bird cage ring is fastened to the tanks by four 1/4 inch diameter bolts with teflon washers. The bird cage flange is then bolted (six 1/4 inch diameter bolts) to the honeycomb deck structure separated by a 1/8 inch teflon pad. Analysis has shown that such a design will perform adequately during the lunar mission.

The top of each tank is supported by two hollow titanium struts to provide adequate thermal resistance. The struts are connected through a clevis joint to a collar encircling the boss on the tank top and to the titanium structure behind the vehicle heat shield. About 10% of the total heat leak to the tanks occurs through these struts.

Another source of heat exchange to the propellant tanks is by radiation to and from the vehicle inner walls. In order to minimize the heat exchange, a blanket consisting of 20 layers of aluminized mylar (approximately 0.5 inch thick) is fastened to the inner surface of the vehicle walls.

The helium tank is mounted in a similar fashion, i.e., a nonmetal strap and bird cage type saddle arrangement to provide adequate thermal resistance between the tank and the deck.

7.4 TRANSIENT ANALYSIS OF LUNAR TRAVERSE

A thermal model of the OMLFV configuration was developed to simulate the vehicle. The model employed 37 nodes which were thermally connected to account for radiation and conduction heat transfer within the vehicle. Four typical sorties were selected for study to investigate the thermal response of the vehicle during an exploration mission. Figure 7.4 shows each of the four sorties for the Hyginus Crater/Rille Exploration.

Time varying boundary conditions were accounted for in the analysis by presuming the vehicle to be oriented in the required direction and keeping track of the elevation of the sun and lunar surface temperatures. Thus, the transient thermal environment for each external surface were determined and used as input to a transient solution of the temperature response throughout the vehicle.

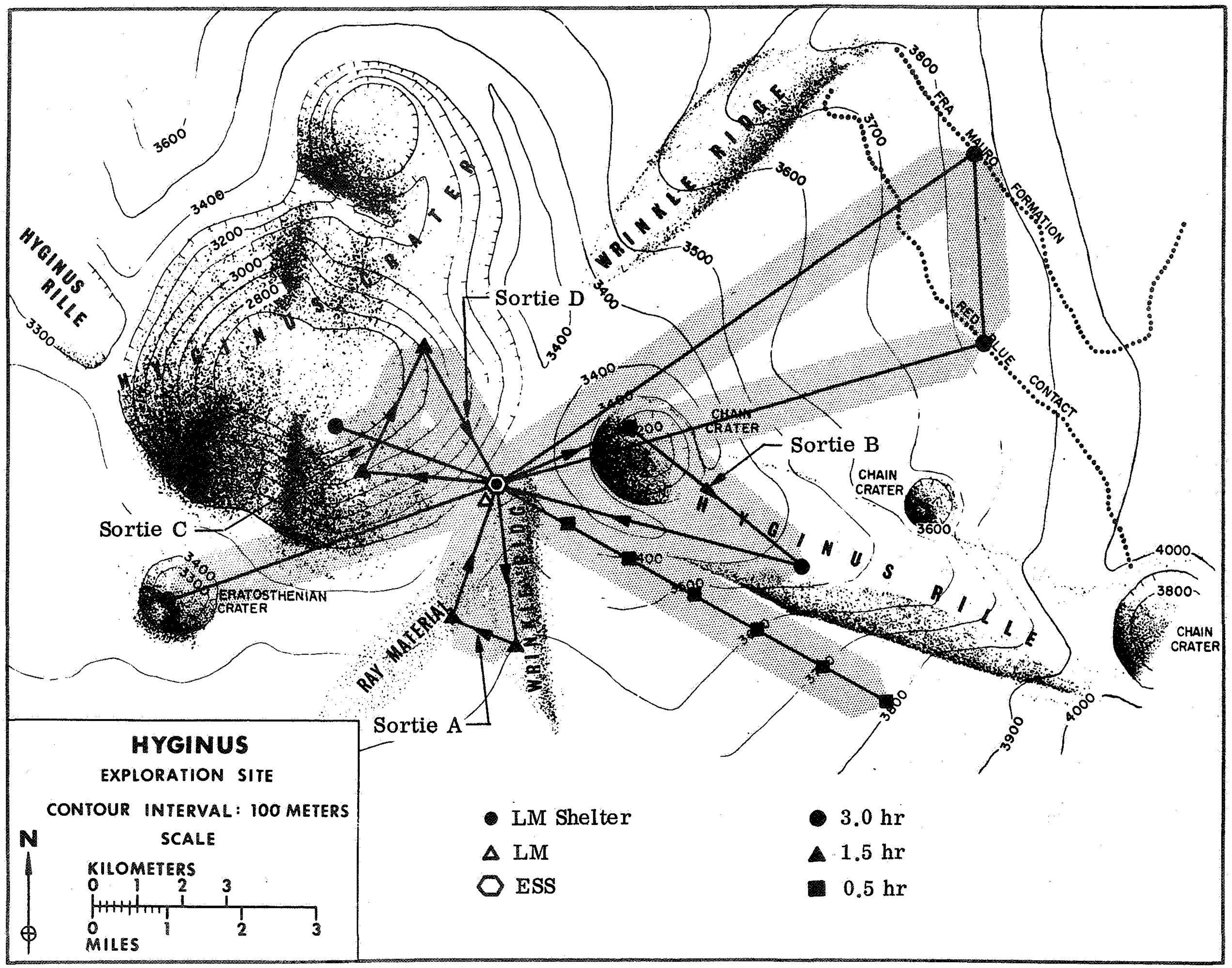


Figure 7.4. Hyginus Crater/Rille Exploration

Prior to the analysis, an outer coating must be selected such that the vehicle and the propellant tank temperatures will not exceed the specified limits, during the complete mission. Since the heating of the vehicle from solar radiation is time and direction dependent, it is necessary to make a judicious selection to cover all boundary conditions. Based on preliminary analysis, a characteristic of $\alpha/\epsilon = 1$ was selected. The coating is assumed to have an emissivity = 0.25 and a solar absorptivity equal to 0.25 (except the shield as previously discussed). Such a coating might be aluminum silicone paint (Ref. 16).

The vehicle is assumed to be stored within the LM quadrant in a 70°F environment. Thirty hours after sunrise, the vehicle is removed, prepared and fueled for flight by the astronauts. It is presumed that the astronauts will have landed in early morning. At this time the sun is approximately 15 degrees above the lunar horizon. As shown in Table 7-1, the vehicle departs at 34.0 hours on the first sortie which takes 3 hours to complete. On return, the propellant tanks are immediately refilled from the LM supply at 70°F. Previous studies indicated that it is very beneficial to keep the tanks full between sorties to take advantage of the large thermal mass.

TABLE 7.1

HYGINUS CRATER/RILLE EXPLORATION SCHEDULE

	Lunar Time Hours	Duration Time Hours
Storage in LM	30.0	30.0
Parked (No Blanket)	4.0	34.0
Sortie A	3.0	37.0
Parked (No Blanket)	17.0	54.0
Sortie B	6.0	60.0
Parked (No Blanket)	8.0	68.0
Sortie C	3.0	71.0
Parket (No Blanket)	7.0	78.0
Sortie D	2.5	80.5
Parked (With Blanket)	591.5	672.0

During flight between each stop, propellant expenditure is accounted for through a special computer subroutine to the main program which corrects the thermal mass of the propellant tanks. Sortie B is the longest sortie, and for this case it was assumed that 20% of the nominal propellant load was remaining in the tanks at the completion of the sortie.

It is assumed that the vehicle is always parked facing the direction of the next leg of the sortie or the first leg of the following sortie. No blanket is necessary between sorties.

Sorties B, C and D are flown and at 80.5 hours after sunrise, the vehicle propellant tanks are filled and a multi-layered quilted blanket is placed over the complete vehicle. (The aluminum outer surface of the blanket has an emissivity of 0.05 and a solar absorptivity of 0.12. The gold coated inner layers have an emissivity of 0.03). Figure 7.5 shows the time/temperature history of the propellants during the sorties and while stored under the blanket for the remainder of the lunar day.

During storage, the oxidizer tank temperature reaches a maximum of 87°F during the solar day and has cooled to 76°F at the end of the lunar night. The fuel tank lags somewhat due to its slightly higher thermal capacity. The helium tank temperature rises to 103°F and drops back to 70°F by the end of the lunar night. The helium tank maximum allowable temperature limit is +120°F. The results also show that the full propellant tanks increase about 17°F during the lunar day and decrease about 10°F during the night, i.e., a net gain of 7°F during one lunar cycle. This would indicate that the propellant tanks within the vehicle could stay within allowable limits for about 3 months sitting on the lunar surface under a blanket.

The selected coating (α/ϵ = 1.0) controls the outer surface temperature to a maximum of 250°F for full view to space. However, the tank mounts are connected to inner walls which are partialy isolated from the outer skin by insulation. The net effect is to considerably reduce the heat leak to and from the propellant tanks.

Figure 7.6 shows the temperature response of three typical skin temperatures during the entire lunar cycle. The top surface temperature reaches a maximum temperature of 220°F. This is indicative of temperatures that would be incurred by instruments mounted on such a surface if the coating is the same (α/ϵ = 1.0). Thus it is recommended that the instruments be recessed and/or isolated from the hot outer surface and perhaps a more beneficial coating be employed in localized areas.

The rear panel temperatures reach 355°F in direct sunlight. This high temperature can be attributed to the relatively poor view to space. It therefore may be beneficial to select a coating such that $\alpha/\epsilon < 1.0$ to reduce this peak temperature.

Damage to the astronaut's suit is not a problem since his garment can stand temperatures up to 500°F for short periods of time, and the vehicle surface temperature will drop rapidly because of its low thermal capacity in the shade of his body.

The shield average side temperatures peak at about 700°F during firing but they too drop to a relatively low equilibrium temperature upon engine shut-down.

7.5 NIGHT TIME OPERATION CONSIDERATIONS

OMLFV insulation system design as well as selection of thermal coatings were based on a daytime operation, i.e., the exposed surfaces were assumed to have an emissivity of 0.25 with the exception of the shield which is coated to obtain an emissivity of 0.80 or greater. Without solar radiation, the outer surfaces cool quickly to equilibrium temperature almost independent of the heat leakage from the propellants

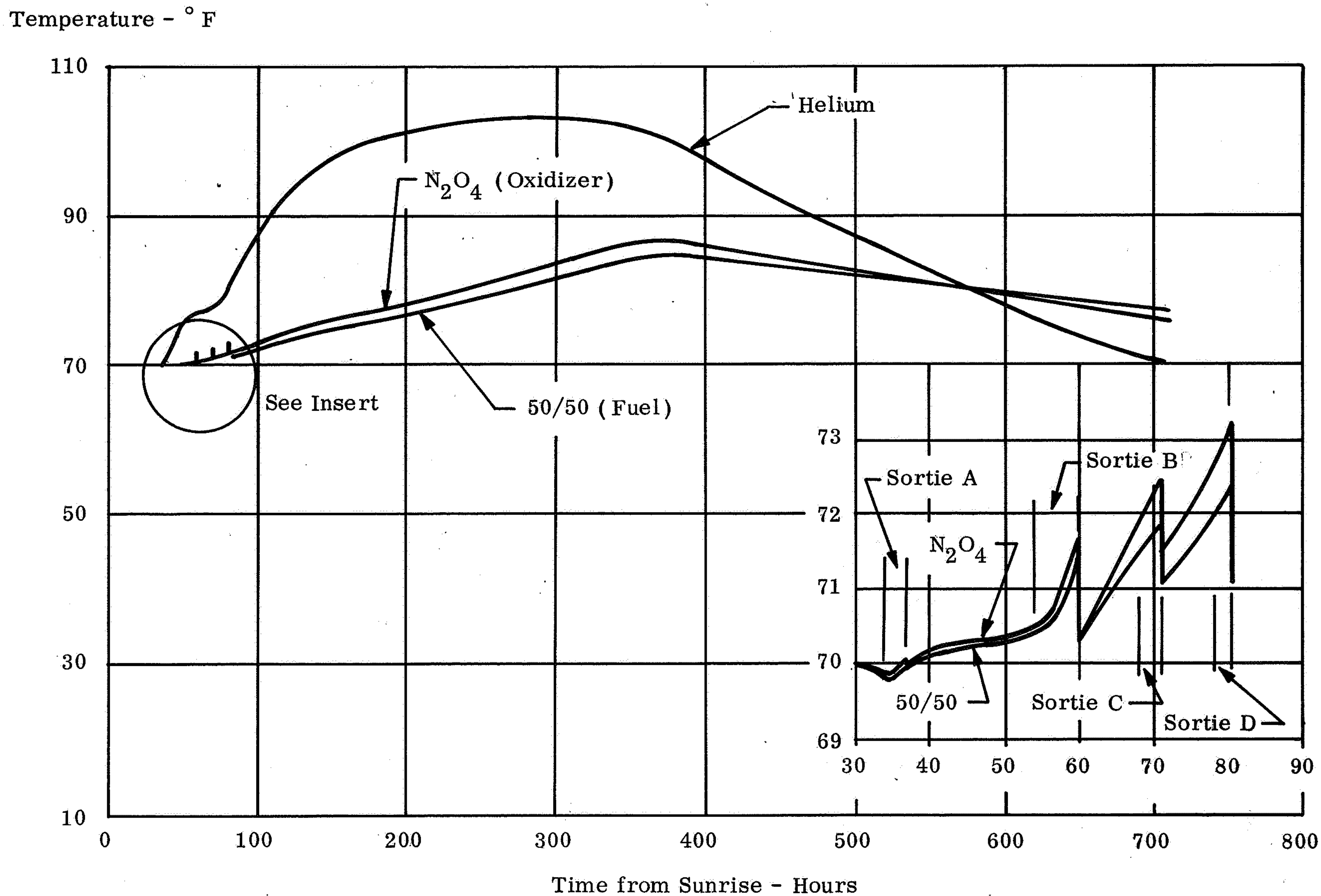


Figure 7.5. OMLFV Helium and Propellant Bulk Temperature during Hygenus Crater/Rille Exploration

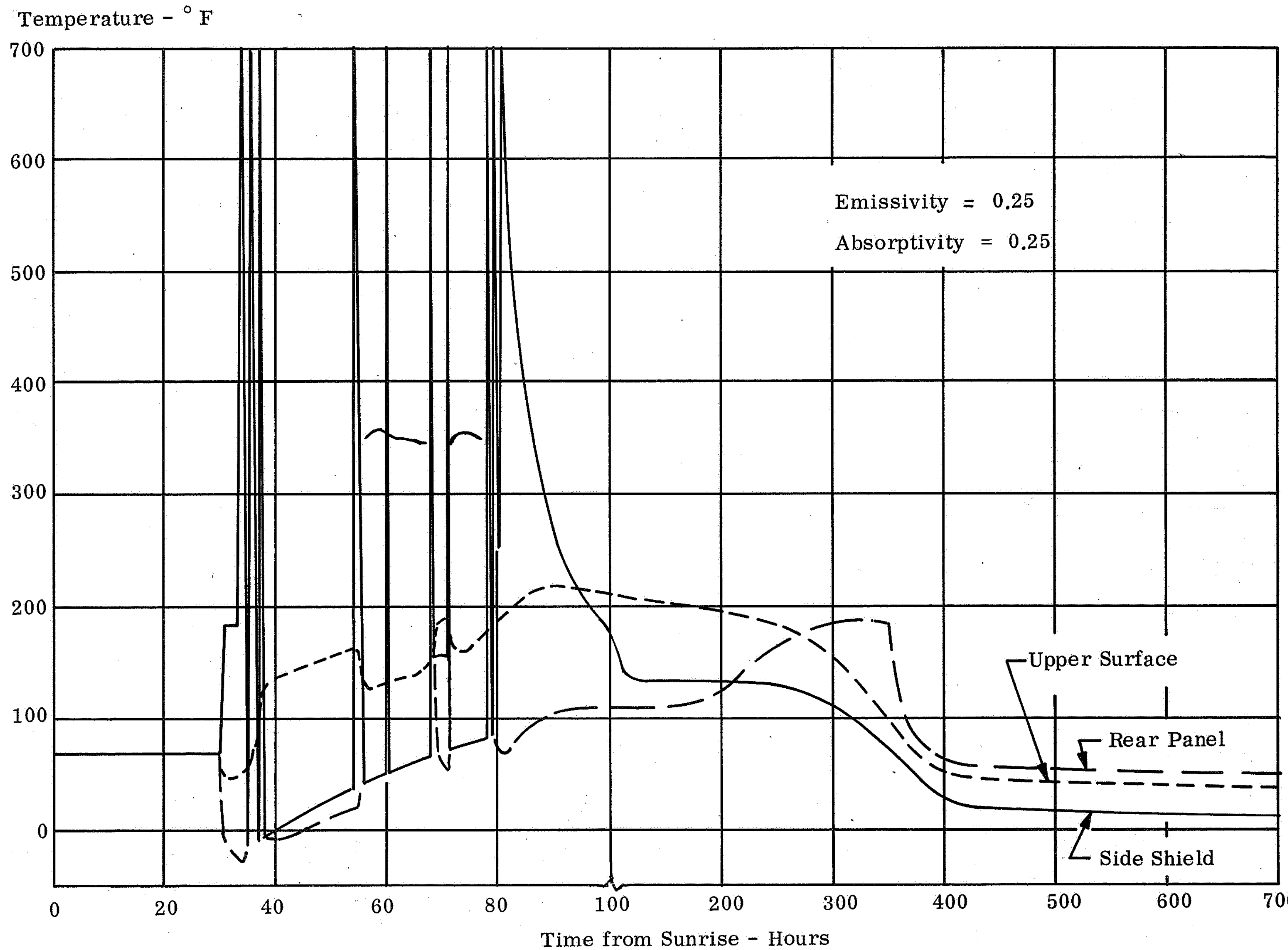


Figure 7.6. OMLFV Typical Surface Temperatures during Hyginus Crater/Rille Exploration

tanks. As a consequence of the slow heat leak, the exposed surface would reach a very low temperature during night-time operation (-300°F or less). However, if the vehicle is covered by a quilted blanket of multilayer insulation during storage or between flights, this heat leak will be considerably reduced. Since the actual total flying time may be seven minutes or less for a given sortie, the exposure without a blanket is minimal and analysis indicates that the propellants will stay within the allowable temperature limits during a 50.5 hour night operation. In the study it was presumed that the multilayer blanket covers the whole vehicle including the propellant lines and thrust-chamber valves. However, caution must be exercised or stiffening provided so that the blanket does not at any time come into contact with a hot engine, otherwise the blanket material will disintegrate.

7.6 STORAGE IN LM QUADRANT

The OMLFV is stored in a LM descent stage quadrant and is protected from the lunar environment by insulation of similar construction to that used in other areas of the LM vehicle.

If thermal coupling to the LM is insufficient, the coating on the outer surface of the insulation may have to be changed to a low emissivity coating such as gold to preclude excessive temperature excursions during the lunar day and night. The thermal coupling to the LM can also be increased by removing the multi-layer insulation on the compartment inner walls.

Since the LM-RCS engine plume will now impinge on the upper surface of the storage area, the walls in this local area will be required to withstand higher heating rates than the present LM shields were designed for. Adequate protection can be achieved by either adding wire mesh layers and nickel foils to the insulation system or by applying a thin ablative coating in those areas where the design limits of the present construction are exceeded.

Further study will be required to analyze the OMLFV storage within the LM. Such studies require consideration of the LM structural design and the thermal cycling of the adjacent LM components during a typical lunar cycle.

7.7 PROPELLANT LINE INSULATION REQUIREMENTS

It is required to insulate the propellant lines on the OMLFV so as to maintain allowable propellant temperatures during the exploration sorties. In the proposed vehicle configuration, the lines are exposed, and thermal insulation must be provided to preclude boiling and/or freezing of the propellants in the lines. Accordingly, studies were conducted to investigate the thermal effectiveness of the number of layers of multilayer insulation required to provide the necessary protection.

It is assumed that the outer surface of the insulation is alternately striped with gold (75%) and aluminum (25%), thus providing an effective solar absorptivity = 0.395 and an effective emissivity = 0.195. In direct sunlight the outer surface temperature

may cycle as high as 384°F and in the shadow as low as -260°F. However, the temperature excursion is significantly damped through the insulation. The propellant line consists of a flexible teflon hose with stainless steel braid (1/4 I.D., 0.07 inch wall thickness).

The result of the study shows that 15 layers (0.375 inch) of NRC-2 insulation are sufficient to provide enough protection for three hours of exposure (unblanketed) assuming the lines are full between each engine firing, at which time the propellants within the lines are replenished. Near the LM, judicious positioning will be employed to maintain the vehicle within allowable temperature limits.

7.8 THERMAL PROTECTION OF LUNAR GROUND SERVICE EQUIPMENT

Spare helium bottles and propellant transfer lines will be stored in the LM quadrant which will provide adequate thermal protection from the lunar environment when the equipment is not in use (as discussed in Section 5).

During servicing, the propellant transfer lines will be unreeled, dragged about 15 to 20 feet across the lunar surface, and connected to the OMLFV. Preliminary analysis indicates that if the lines are constructed like those on the vehicle, they will have sufficient thermal capacity to tolerate the short duration, i.e., 15 minutes or less anticipated exposure to the lunar environment. (After refueling is completed, the lines will be coiled back into the storage compartment.)

The helium bottles will be replaced after each sortie and it is not necessary to provide additional thermal protection during the exchange.

8.0 PAYLOAD STUDIES

Because the variation in payload weight, from 0 to 370 pounds, is a large fraction of the vehicle gross weight, the payload weight and distribution will have a significant effect on the vehicle flight trim adjustment and handling properties. The payload may consist of groupings of scientific instruments, or a second astronaut. Although the requirement to carry a disabled astronaut has been deleted, the full 370 lb payload capability is retained, and a second astronaut can, in fact, be carried on the selected vehicle.

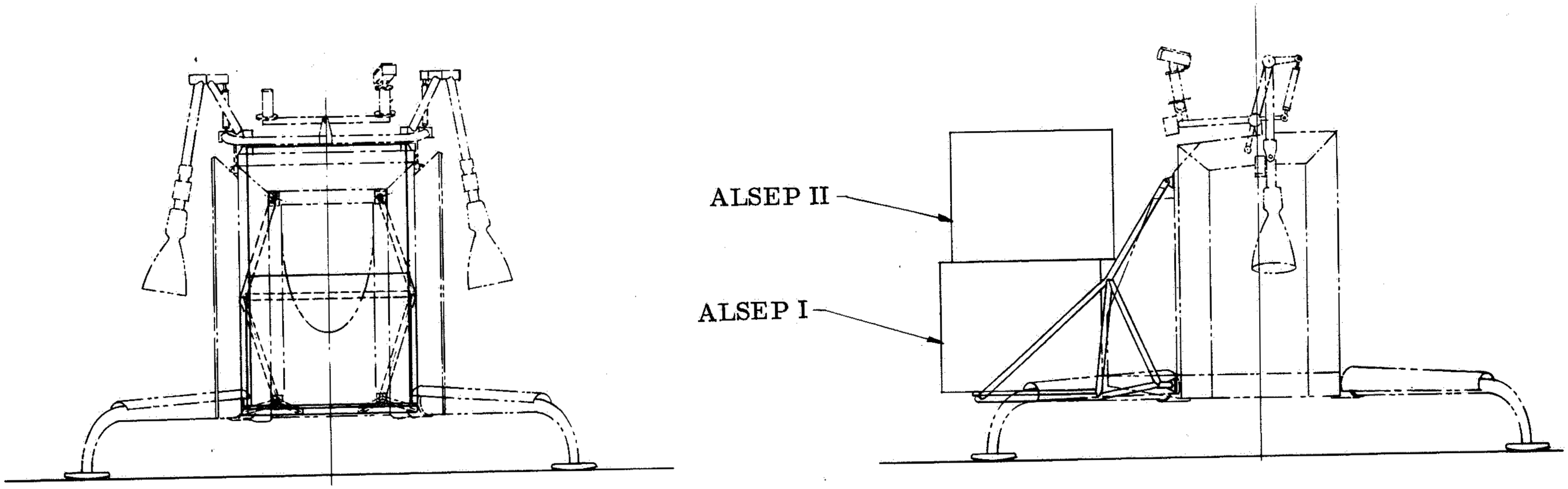
This section presents mass properties and loading techniques for typical payloads. The resulting effects on vehicle trim control is discussed in Section 3.0 and on handling qualities in Section 6.0.

As definite and specific payloads for each sortie have not been established, several typical payloads have been synthesized from References 3 and 17-20. Early in the study a 370 pound payload with a density of 25.3 lb/cu ft (average from Reference 17) was used in the configuration selection process. For study of the selected configuration two typical payloads were defined and are presented in Table 8.1. Installation of these payloads on the selected vehicle is illustrated in Figure 8.1. The package density of each group is 17 lb/cu ft for Group I and 13.7 lb/cu ft for Group II.

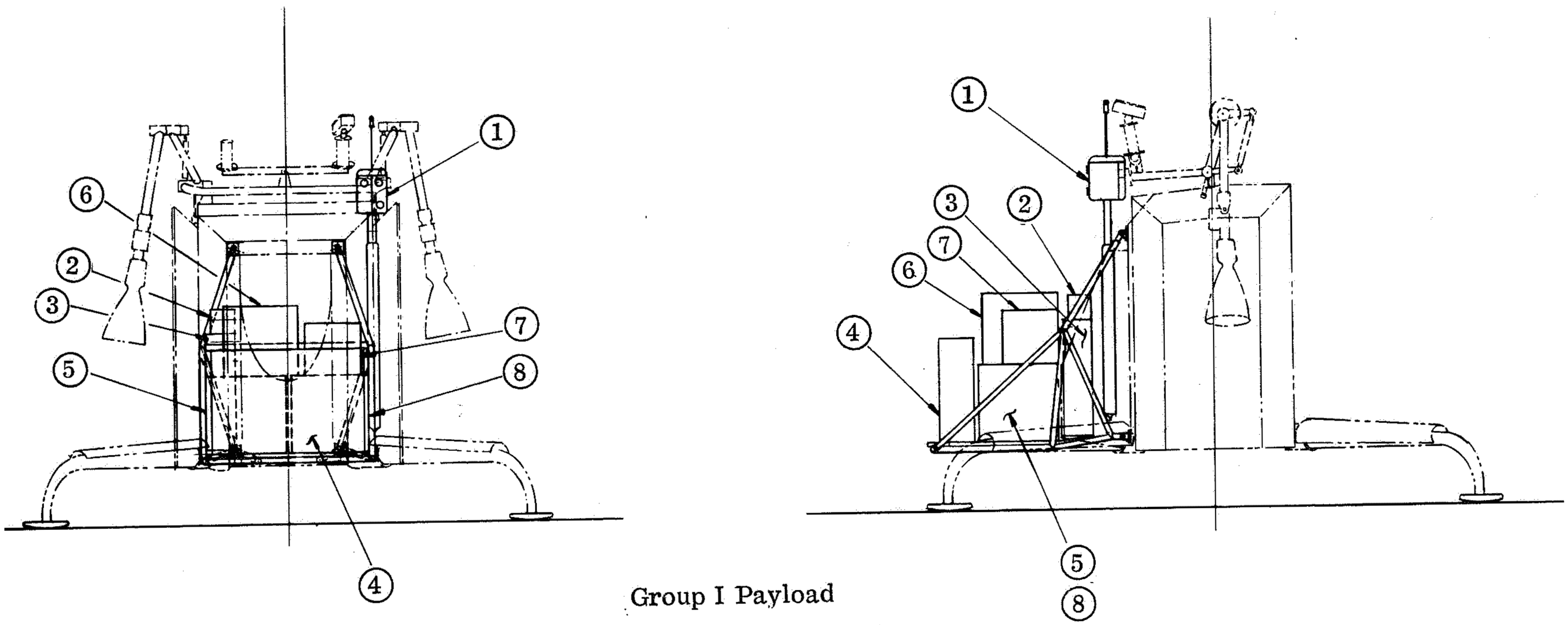
Because the selection of specific items of scientific equipment and their detail design will change as the program progresses, a flexible total systems approach to payload mounting has been adopted. Since the payload is originally stowed onboard the LM descent stage and then transferred to the OMLFV, these payload items can be combined into functional or sortie groups and then palletized. These pallets can be designed to mount on the LM descent stage and protect the payload through earth launch and lunar landing. Removed from LM and reinstalled onboard the flyer, the pallets continue their mounting and protecting function. Such an arrangement is convenient for the astronaut and saves preflight equipment loading time. Because of the dual function, this concept probably saves overall weight. A representative pallet has been illustrated in Figure 8.1.

This concept offers payload flexibility and versatility on a sortie to sortie basis. However, if the payloads are well known in advance and are packaged so that they can be distributed about the vehicle, some vehicle and operational simplification can result by eliminating the requirement for retrimming. For example, an earth pre-mission setting might be established which will accommodate the payloads carried in a particular fashion.

The mass moments of inertia of the Group I and II payloads, about their own cg's are given in Table 8.2. Table 3.4.2 presents a summary of the mass properties of the total vehicle for various payloads, including the Group I and Group II loads. The vehicle has been designed to accommodate the complete range of cg travel and inertias presented.



Group II Payload



Group I Payload

Figure 8.1. OMLFV Payload Installations

TABLE 8.1

TYPICAL PAYLOAD GROUPS

Group I

	Name	Weight (lb)	Dimensions (inches)	Ref.
1.	Staff	28.5	3.9 x 5.9 x 55	3
2.	Close Photography Camera	3.0	vol = 0.04 ft^3	17
3.	3M Drill	27.0	5 x 5 x 20	18, 20
4.	Tools and Carrier	22.0	6 x 18 x 26	19
5.	Geophysical Probe	11.2	vol = 1.2 ft^3	17
6.	Heat Flow Terminal Probe	25.6	vol = 1.4 ft^3	17
7.	Facsimile Camera	6.0	vol = 0.5 ft^3	17
8.	Mass Spectrometer	15.0	vol = 1.5 ft^3	17
		138.3	vol = 8.16 ft^3	

Group II

Name	Weight (lb)	Dimensions (inches)	Ref.
ALSEP			
Package I	127.5	22.4 x 26.7 x 27.2 vol = 18.7 ft^3	20
Package II	127.5	22.4 x 26.7 x 27.2 vol = 18.7 ft^3	20

TABLE 8.2

MASS MOMENTS OF INERTIA

	wt lb	I_{xx}	Iyy	I_{zz}
Group I	138.3	8.81	9.12	7.26
Group II				
Package I	127.5	2.89	2.95	3.54
Package II	127.5	2.89	2.95	3.54

Requirements for astronaut rescue include moving him to the vehicle, if necessary, and provision of adequate support during the flight back to LM. To meet these requirements a litter/travois concept was developed prior to the deletion of the requirement (to carry a disabled astronaut). This device, shown in Figure 8.2, is taken to the astronaut, the astronaut strapped to the litter and the litter then dragged to the flyer. The litter is attached to the vehicle by pivot fittings at the foot end, then rotated into a near vertical position and secured by struts at upper fittings. The litter can be folded for storage aboard LM and can be designed to be used as a payload rack.

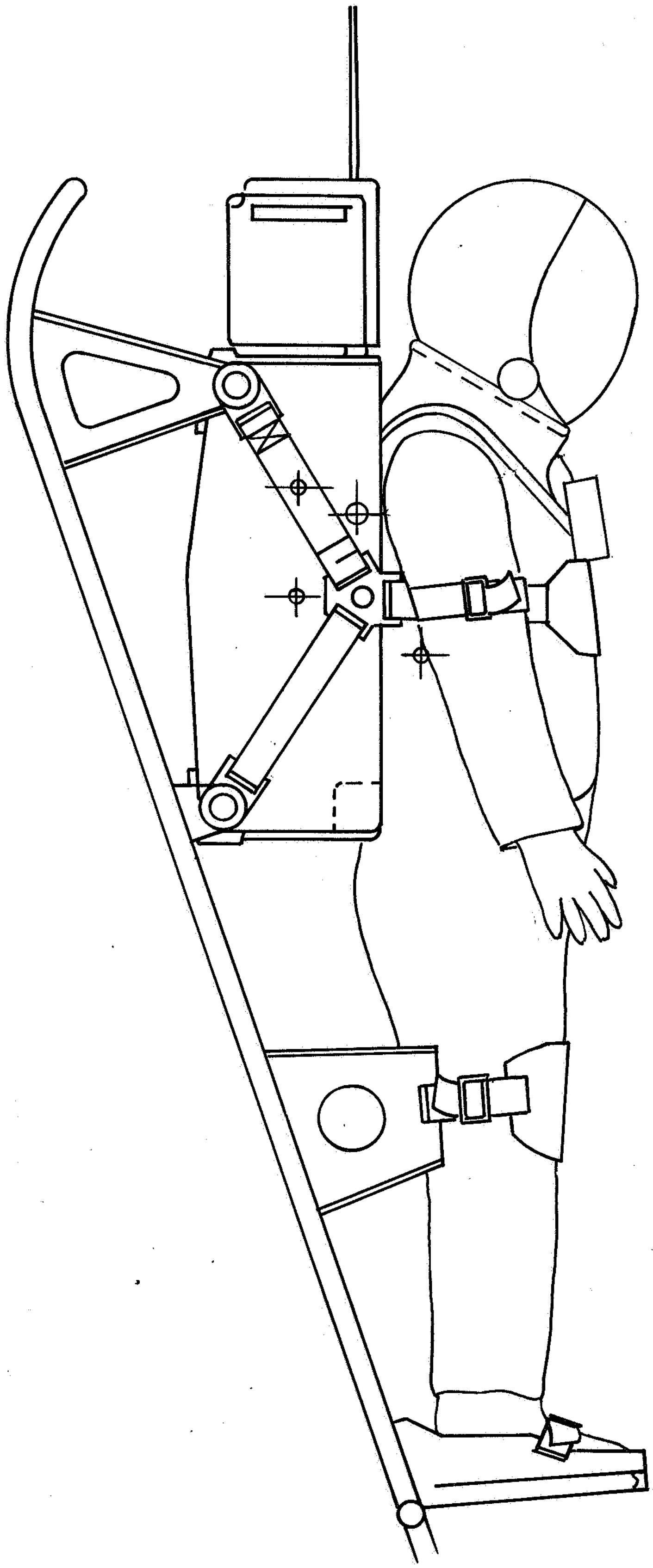


Figure 8.2. Travois Litter Arrangement

9.0 OPERATIONS STUDY

Flyer related operations have been analyzed to establish vehicle requirements compatible with other mission elements and to provide preliminary operational sequences and time lines. The results of these analyses, for both KSC and lunar surface operations, are reported in this section.

9.1 KSC OPERATIONS

The activity subsequently defined is that associated with integrating OMLFV prelaunch operations with other mission operations at KSC. The activity involved takes place at three separate physical locations; the Operations and Checkout Building, the Static Test Complex and the Weight and Balance Building.

Two flyers will be shipped to KSC in separate containers with all tanks empty. Payload pallets may be shipped separately or attached to the vehicles. The arrival of the flyers will be timed and scheduled to permit checkout, test and mating, parallel with the on-going LM checkout activity. Because of the simplicity of the flyer, its propulsion system is the only subsystem interfaced with the LM. Timely entry of the vehicles into the prelaunch cycle thus depends largely on correct scheduling of fuel system inspection and checkout.

9.1.1 Operations and Checkout Building

The vehicles will be received at the Operations and Checkout Building, where they will be unpacked and visually inspected for damage. All assemblies including deployment gear, will be examined and functionally verified. In addition, the display systems and power circuits will be checked, and low pressure leak tests of the propulsion system conducted.

Upon completion of the receiving inspection and checkout, the vehicles will be returned to their shipping containers.

9.1.2 Static Test Complex

The vehicles will be moved, in their shipping crates, from the Operations and Checkout Building to the Static Test Complex. In the Static Test Complex, the servicing interface between the LM descent stage propellant storage tanks and the OMLFV tanks will be functionally verified.

Upon completion of the LM descent stage static firing tests, and prior to draining the propellant system, the flyers will be serviced with the residual propellants, using the LM transfer lines. The pressurant bottles will be installed and the OMLFV engines static fired. Upon completion of the tests, the LM storage tanks and flyer propellant tanks, will subsequently be drained, purged and flushed. The helium bottles will be replaced and a final inspection of the flyers made.

Finally, in preparation for shipment of the complete LM assembly to the Weight and Balance Building, the flyers will be stowed and hard-mounted in Quadrants I and III of the LM descent stage. The supply of helium bottles will also be installed at this point.

9.1.3 Weight and Balance Building

The procedures conducted here do not affect the flyers. The flyers are at this point mounted in the LM to assure proper determination of the weight and cg balances for both the LM descent stage and the mated descent/ascent stages.

For the remainder of the prelaunch activities,including the final mating of the LM with the CSM and S-IVB launch adapter, the flyers will remain intact, and will not be involved in any subsystem operations. Figure 9.1 summarizes the flyer related KSC operations.

9.1.4 Ground Support Equipment

The GSE required for the KSC checkout and test of the OMLFV is:

(a) Propellant service system
(b) propellant flush and purge system
(c) Helium service system
(d) Battery service system
(e) Rocket test set (Bench type)
(f) Protective covers
(g) Test tie-down fixture
(h) Handling fixtures and slings
(i) Shipping containers
(j) Transport dollies
(k) Leak-detection units
(l) Special tool kits
(m) Maintenance kits

9.1.5 Storage Phase Checkouts

Due to the inherent simplicity of the OMLFV plus the fact that the vehicles are stowed with empty propellant tanks, no sensing equipment for monitoring vehicle subsystem status during storage or trans-lunar flight, is required.

9.2 LUNAR SURFACE OPERATIONS

The OMLFV lunar activity has been divided into two broad categories: that activity involved in initial deployment and checking of the vehicle, and that involved with its routine use. The first phase will therefore be referred to as the Activation Phase and the second as the Exploration Phase. The former phase ends, and the latter commences, at the end of the first fueling of the flying vehicles. However, preliminary

Operations and Checkout Building

1. Vehicles Received and Unpacked
2. Inspection and Functional Verification
3. Repackaging

↓

Static Test Complex

1. Propellant Interface with LM Test
2. Static Firing
3. Draining and Purging
4. Helium Replacement
5. Inspection
6. Stowage and Hard Mounting in LM
7. Helium Supply Bottles Stowed

↓

Weight and Balance Building

1. LM Ascent Stage, Mated and Unmated with Descent Stage, Checked with OMLFV Installed

Figure 9.1. OMLFV Related KSC Activity Summary

to defining each of these two phases, analysis of important operations in both, are examined.

9.2.1 Plume/Surface Effects

One of the operational factors and vehicle design considerations studied was the effect of the engine exhaust upon the lunar surface. This is important since it affects the safe minimum distance between the flyer landing site and the lunar module. The important vehicle design consideration is the minimum permissible height of the engine from the lunar surface. Earth flights of the rocket belt in sandy/dusty desert soil indicate that pilot visibility during landing is not impaired by flying and billowing dust and debris. Therefore, the area of prime interest is the possible effect of impingement of the lunar surface debris upon the lunar module. This effect can be minimized by landing some distance away from the LM and using long hoses to resupply the flyer. An alternative is to provide a means for moving the flyer to and from a refueling site closer to LM; this alternative is undesirable from a weight and operational complexity point of view. A more attractive operational alternative is to land at a refueling site close to the lunar module (within 15 to 20 feet). It then becomes a matter of determining the minimum engine to surface clearance to permit such close approaches without appreciable impingement effects. The best information available on the effect of rocket engine exhaust plume upon the lunar surface was obtained during the Surveyor VI mission when the landing engines were re-fired, moving the Surveyor craft from its initial landing position. Photographs in Reference 21 show the effect that firing of the three engines had upon the surface. There is evidence of movement of a shallow layer of soil beneath and adjacent to the spacecraft. This soil was redeposited up to several meters away from the spacecraft. Only a small amount of surface material was moved and it was judged that if such erosion were present during a lunar flyer landing within 15 to 20 feet of the LM, no problems would be encountered. Analytical and experimental studies of surface erosion effects have shown that the most significant parameter affecting surface disturbance is the stagnation pressure. Dynamic pressure, which affects viscous erosion, is directly proportional to stagnation pressure.

Table 9.1 shows the theoretical surface pressures experienced on the surface during the Surveyor VI hop.

TABLE 9.1

SURVEYOR VI SURFACE STAGNATION PRESSURE

Time from Ignition (sec)	Engine Number		
	1	2	3
0.2	0.83 psia	1.35	1.96
1.0	0.19	0.18	0.18

These were obtained using Robert's theory (References 22 and 23) and telemetered flight and thrust chamber information. Using the same theory and the characteristics of potential lunar flying vehicle engines, the stagnation pressure as a function of engine height above the surface was calculated. This data is shown in Figures 9.2 and 9.3 for one and two engine configurations. Figure 9.2 shows the stagnation pressure for a 300 lb thrust engine at its rated thrust and throttled down to 175 pounds. The band to the left indicates the range of initial (peak) pressures experienced by the Surveyor engine firings. For the criteria indicated (P_s = 1.0 and T/W = 1.5), the minimum allowable engine height above the surface is 20 inches. If the engine is at a distance of 20 inches or more above the surface, the stagnation pressure will be less than the pressure experienced during the Surveyor VI hop. These figures indicate that there will be little or no surface disturbance at engine heights greater than 4 to 5 feet. Thus, surface disturbance will occur only during the portion of flight when the engine is from 20 inches to 5 feet from the surface. Figure 9.3 shows that the engine height used on the selected vehicle (two engines) is well above the allowable minimum for a two engine configuration. Analyses conducted by Grumman, reported in Reference 24, also indicate that flyer landings within 20 feet of LM are acceptable.

9.2.2 Task Time-Line Analysis

In order to understand the details of the mission, attach times to each of the operations and define EVA and training requirements, three important areas are examined in detail: (1) Deployment of the flyers, (2) Initial fueling operations, (3) Refueling operations.

Tables 9.2, 9.3, and 9.4 present the results of this analysis. Each of the three operations analyzed is made up of simple task elements, amenable to standard training procedures, and time analysis.

9.2.3 Activation Phase

The Activation Phase consists of the deployment and initial fueling operations. Table 9.5 shows the entire Activation Phase sequence for two flyers. Of the total EVA time available, 83 minutes is required for activating two flyers. On early missions, some of the excess time could be used to conduct checkout and familiarization flights and reservicing of the vehicle used. A similar analysis for the activation of a single flyer is shown in Table 9.6.

9.2.4 Exploration Phase

Table 9.7 defines the procedure for operating the flyer routinely, once it has been activated in a previous EVA cycle. Table 9.8 shows a summary grouping of the functional times.

All the times attached to each of the operations examined are conservative. Significant reductions in times are therefore possible and likely. It may also be possible to improve the productive to non-productive time ratio, by simplifying and

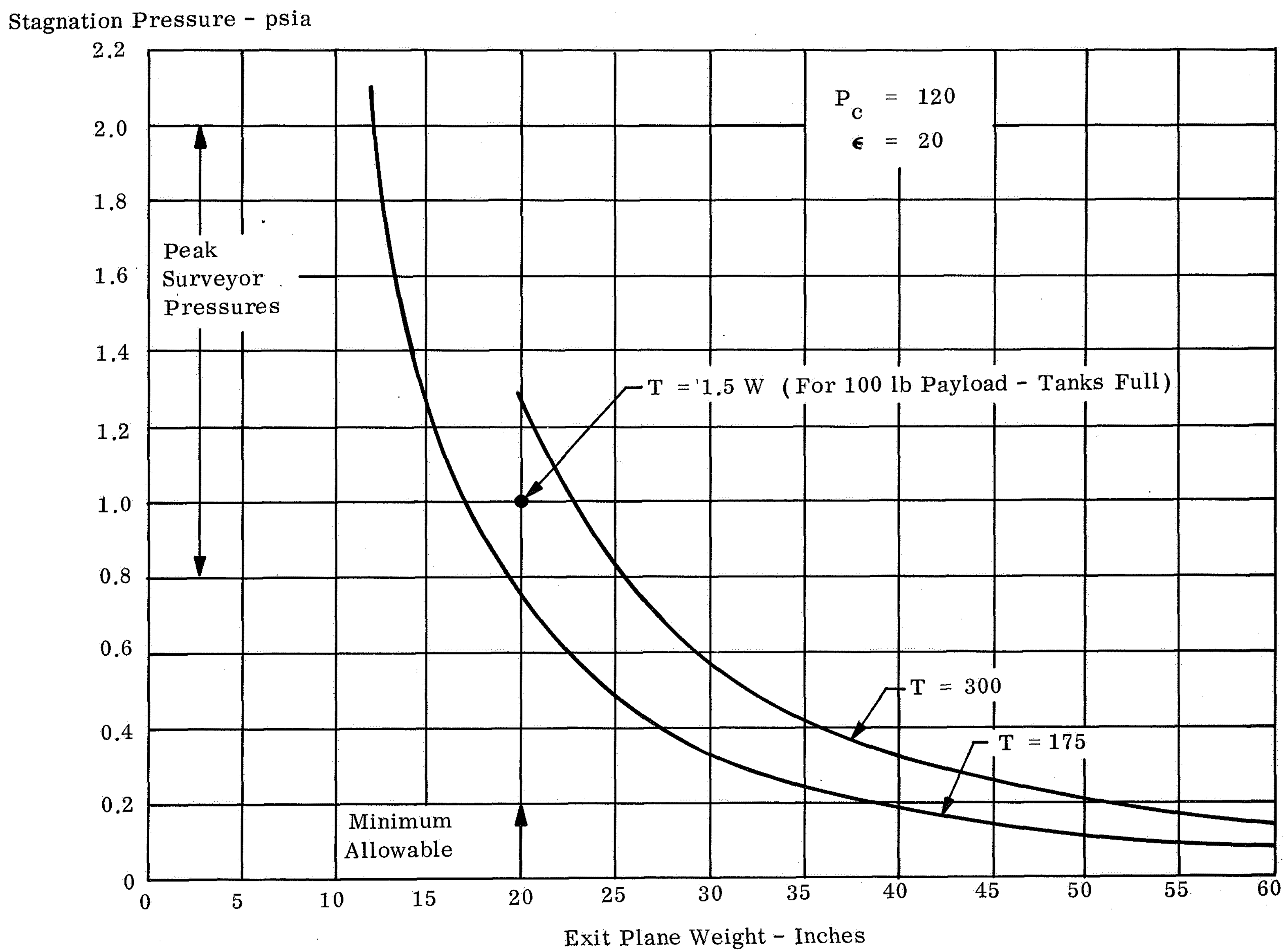


Figure 9.2. Surface Pressure - Rated Thrust 300 lb (One Engine Case)

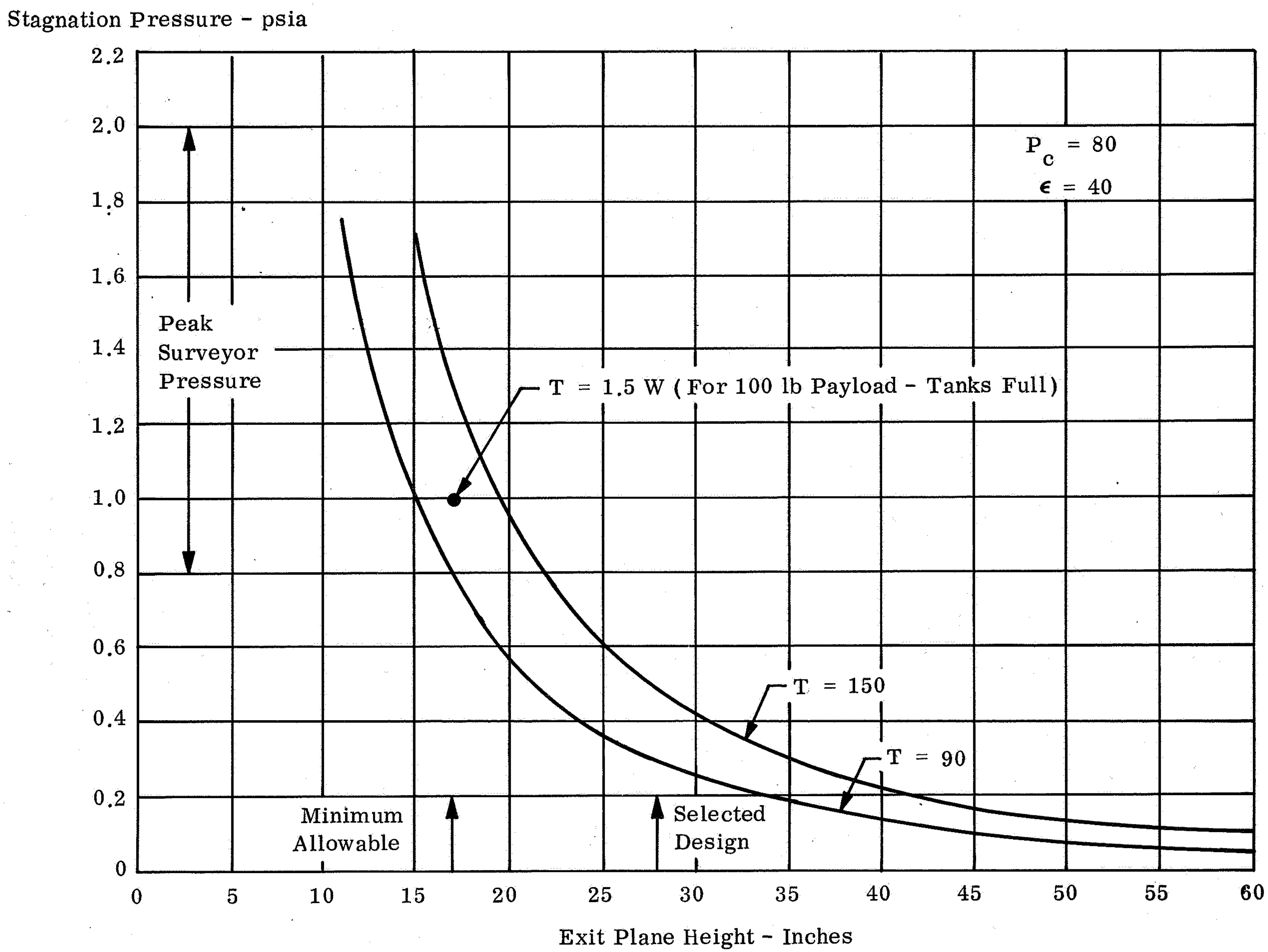


Figure 9.3. Surface Pressure - Rated Thrust 150 lb (Two Engine Case)

TABLE 9.2

FLYER DEPLOYMENT TIMELINE

	Function	Time Seconds	Cum. Time Seconds
	Start: Astronaut has left LM		
1.	Walk to R side of LM quadrant 1.	10.0	10.0
2.	Grasp door lanyard.	5.0	15.0
3.	Turn, walk back 8 feet to LM.	10.0	25.0
4.	Pull lanyard to open compartment door.	5.0	30.0
5.	Walk to LM, release lanyard.	10.0	40.0
6.	Grasp landing gear lanyard stowed in compartment.	5.0	45.0
7.	Turn, walk back 8 ft, turn to LM.	10.0	55.0
8.	Pull lanyard to unlock forward landing gear legs.	5.0	60.0
9.	Controlling rate of landing gear descent, lower legs.	10.0	70.0
10.	Walk to LM, release lanyard.	10.0	80.0
11.	Grasp one of forward legs and push down to lock.	5.0	85.0
12.	Walk to other leg.	10.0	95.0
13.	Lock second leg.	5.0	100.0
14.	Grasp boom lanyard stowed in compartment.	5.0	105.0
15.	Turn, walk back 16 feet from LM and turn back.	35.0	140.0
16.	Pull lower portion of lanyard until unlocking takes place.	5.0	145.0
17.	Continue pulling until boom and OMLFV fully extends.	10.0	155.0
18.	Pull upper portion of lanyard to lower flyer to surface.	10.0	165.0
19.	Release boom lanyard and walk to flyer.	10.0	175.0
20.	Walk to left rear leg.	10.0	185.0
21.	Unlock leg, lower and lock in position.	10.0	195.0
22.	Walk to other leg.	10.0	205.0
23.	Unlock leg, lower and lock in position.	10.0	215.0
24.	Walk to front of OMLFV.	10.0	225.0
25.	Pick up boom lanyard.	5.0	230.0
26.	Turn, walk 8 feet from flyer and turn back	15.0	245.0
27.	Complete lowering of flyer to surface with upper lanyard.	10.0	255.0
28.	Walk to flyer and release lanyard.	10.0	265.0
29.	Mount vehicle	10.0	275.0
30.	Pull pin to disconnect cable and sway brace lanyard.	5.0	280.0
31.	Dismount, holding swaybrace/boom retract lanyard.	10.0	290.0
32.	Release lanyard and walk to front of vehicle.	10.0	300.0

TABLE 9.2 (cont)

	Function	Time Seconds	Cum. Time Seconds
33.	Disconnect lower loop of boom-lanyard from vehicle front.	5.0	305.0
34.	Turn, walk back and pick up S.B./B.R. lanyard.	10.0	315.0
35.	Retract boom.	5.0	320.0
36.	Walk to flyer rear leg and release lanyard.	10.0	330.0
37.	Walk to front of vehicle.	10.0	340.0
38.	Release landing gear lanyard from flyer and hold.	5.0	345.0
39.	Clip lanyard to suit.	5.0	350.0
40.	Walk 10 feet to front of flyer and turn.	15.0	365.0
41.	Drag flyer to fuelling site and turn it 90°.	180.0	545.0
42.	Walk to flyer.	10.0	555.0
43.	Release landing gear lanyard from vehicle.	5.0	560.0
44.	Release lanyard from belt and drop.	5.0	565.0
	End of deployment operation.		
	NOTE: Total time for deployment operation is 9 minutes 25 seconds.		

TABLE 9.3

INITIAL FUELING TIMELINE

	Function	Time Seconds	Cum. Time Seconds
	Start: OMLFV Assembled		
1.	Walk to LM from OMLFV.	20.0	20.0
2.	Access LM store.	10.0	30.0
3.	Check T line valves closed.	5.0	35.0
4.	Open LM T valves.	5.0	40.0
5.	Remove T lines.	5.0	45.0
6.	Walk T lines to OMLFV.	30.0	75.0
7.	Access OMLFV valves and ports.	5.0	80.0
8.	Uncap fuel T line nozzle.	2.5	82.5
9.	Uncap OMLFV fuel fill-port.	2.5	85.0
10.	Connect fuel T line to OMLFV fuel fill port.	5.0	90.0
11.	Uncap Ox. T line nozzle.	2.5	92.5
12.	Uncap OMLFV ox. fill port.	2.5	95.0
13.	Connect ox. T line to OMLFV ox. fill port.	5.0	100.0
14.	Walk back to LM.	20.0	120.0
15.	Remove coiled vent lines.	10.0	130.0
16.	Walk to vent area.	37.5	167.5
17.	Place fuel dump nozzle in fuel dump area.	10.0	177.5
18.	Place ox. dump nozzle in ox. dump area.	15.0	192.5
19.	Walk to OMLFV while paying out vent lines.	60.0	252.5
20.	Uncap fuel and ox. vent line nozzles.	5.0	257.5
21.	Uncap OMLFV fuel vent port	2.5	260.0
22.	Connect fuel vent line to fuel vent port.	5.0	265.0
23.	Uncap OMLFV ox. vent port.	2.5	267.5
24.	Connect ox. vent line to ox. vent port.	5.0	272.5
25.	Open OMLFV fuel vent port valve.	5.0	277.5
26.	Wait for fuel tank to vent.	1.0	278.5
27.	Open fuel T line valve.	5.0	283.5
28.	Open OMLFV fuel fill port valve.	5.0	288.5
29.	Adjust fill rate with fuel vent valve.	5.0	293.5
30.	Wait for tank to fill.	450.0	743.5
31.	Cut fuel flow with fuel fill port valve.	5.0	748.5
32.	Close fuel vent port valve.	5.0	753.5

TABLE 9.3 (cont)

	Function	Time Seconds	Cum. Time Seconds
33.	Close fuel T line valve.	5.0	758.5
34.	Disconnect fuel T line and cap.	7.5	766.0
35.	Open OMLFV ox. vent port valve.	5.0	771.0
36.	Wait for ox. tank to vent.	1.0	772.0
37.	Open ox. T line valve.	5.0	777.0
38.	Open OMLFV ox. fill port valve.	5.0	782.0
39.	Adjust fill rate with ox. vent valve.	5.0	787.0
40.	Wait for tank to fill.	450.0	1237.0
41.	Cut ox. flow with ox. fill port valve.	5.0	1242.0
42.	Close ox. vent port valve.	5.0	1247.0
43.	Close ox. T line valve.	5.0	1252.0
44.	Cap OMLFV fuel fill port.	2.5	1254.5
45.	Disconnect ox. T line and cap.	7.5	1262.0
46.	Cap OMLFV ox. fill port.	2.5	1264.5
47.	Place lines on hook.	7.5	1272.0
48.	Disconnect fuel vent line and cap.	7.5	1279.5
49.	Cap OMLFV fuel vent port.	2.5	1282.0
50.	Disconnect ox. vent line and cap.	7.5	1289.5
51.	Cap OMLFV ox. vent port.	2.5	1292.0
52.	Remove vent lines from OMLFV vicinity.	15.0	1307.0
53.	Place vent lines on ground.	1.0	1308.0
54.	Return to OMLFV.	12.5	1320.5
55.	Close OMLFV valves and ports panel.	5.0	1325.5
56.	Remove T lines from hook.	5.0	1330.5
57.	Walk to LM with lines.	32.5	1363.0
58.	Stow lines.	10.0	1373.0
59.	Close LM fuel and ox. T valves.	10.0	1383.0
60.	Close storage hatch.	10.0	1393.0

TABLE 9.4

REFUELING TIMELINE

	Function	Time Seconds	Cum. Time Seconds
	Start: Pallet removed.		
1.	Access He valve.	10.0	10.0
2.	Close He valve.	5.0	15.0
3.	Close He access panel.	10.0	25.0
4.	Access on OMLFV valves and ports.	5.0	30.0
5.	Access and remove tongs.	15.0	45.0
6.	Walk to vent lines.	15.0	60.0
7.	Pick up vent lines.	5.0	65.0
8.	Return to OMLFV with lines.	17.5	82.5
9.	Place vent lines on OMLFV hook.	5.0	87.5
10.	Replace tongs on pallet.	10.0	97.5
11.	Return to OMLFV.	7.5	105.5
12.	Remove vent lines from hook.	5.0	110.0
13.	Uncap vent lines.	5.0	115.0
14.	Uncap OMLFV fuel vent port and connect lines.	7.5	122.5
15.	Uncap ox. vent port and connect line.	7.5	130.0
16.	Walk to LM	22.5	152.5
17.	Access LM store.	10.0	162.5
18.	Check T line: valves closed, no leaks.	7.5	170.0
19.	Open LM T valves.	5.0	175.0
20.	Remove T lines.	5.0	180.0
21.	Walk T lines to OMLFV.	30.0	210.0
22.	Uncap fuel T line nozzle.	2.5	212.5
23.	Uncap OMLFV fuel fill port.	2.5	215.0
24.	Connect fuel T line to OMLFV fuel fill port.	5.0	220.0
25.	Uncap ox. T line nozzle.	2.5	222.5
26.	Uncap OMLFV ox. fill port.	2.5	225.0
27.	Connect ox. T line to OMLFV ox. fill port.	5.0	230.0
28.	Open OMLFV fuel vent port valve.	5.0	235.0
29.	Wait for fuel tank to vent.	1.0	236.0
30.	Open fuel T line valve.	5.0	241.0
31.	Open OMLFV fuel fill port valve.	5.0	246.0

TABLE 9.4 (cont)

	Function	Time Seconds	Cum. Time Seconds
32.	Adjust fill rate with fuel vent valve	5.0	251.0
33.	Wait for tank to fill	450.0	701.0
34.	Cut fuel flow with fuel fill port valve	5.0	706.0
35.	Close fuel vent port valve	5.0	711.0
36.	Close fuel T line valve	5.0	716.0
37.	Disconnect fuel T line and cap	7.5	723.5
38.	Open OMLFV ox. vent port valve	5.0	728.5
39.	Wait for ox. tank to vent	1.0	729.5
40.	Open ox. T line valve	5.0	734.5
41.	Open OMLFV ox. fill port valve	5.0	739.5
42.	Adjust fill rate with ox. vent valve	5.0	744.5
43.	Wait for tank to fill	450.0	1194.5
44.	Cut ox. flow with ox. fill port valve	5.0	1199.5
45.	Close ox. vent port valve	5.0	1204.5
46.	Close ox. T line valve	5.0	1209.5
47.	Cap OMLFV fuel fill port	2.5	1212.0
48.	Disconnect ox. T line and cap	7.5	1219.5
49.	Cao OMLFV ox. fill port	2.5	1222.0
50.	Place lines on hook	7.5	1229.5
51.	Disconnect fuel vent line and cap	7.5	1237.0
52.	Cap OMLFV fuel vent port	2.5	1239.5
53.	Disconnect ox. vent line and cap	7.5	1247.0
54.	Cap OMLFV ox. vent port	2.5	1249.5
55.	Remove vent lines from OMLFV vicinity	15.0	1264.5
56.	Place vent lines on ground	1.0	1265.5
57.	Return to OMLFV	12.5	1278.0
58.	Close OMLFV valves and ports panel	5.0	1283.0
59.	Remove T lines from hook	5.0	1288.0
60.	Walk to LM with lines	32.5	1320.5
61.	Stow lines	10.0	1330.0
62.	Close LM fuel and ox. T valves	10.0	1340.0
63.	Close storage hatch	10.0	1350.0
64.	Access He bottle storage	10.0	1360.0
65.	Remove fresh He bottle	20.0	1380.0

TABLE 9.4 (cont)

	Function	Time Seconds	Cum. Time Seconds
66.	Close He bottle storage	10.0	1390.0
67.	Return to OMLFV	30.0	1420.0
68.	Extend bottle handle	5.0	1425.0
69.	Set bottle down	5.0	1430.0
70.	Access old bottle	10.0	1440.0
71.	Check He valve closed	5.0	1445.0
72.	Disconnect old bottle	5.0	1450.0
73.	Open He valve on bottle to vent	10.0	1460.0
74.	Release old He bottle and jettison	10.0	1470.0
75.	Pick up fresh bottle	5.0	1475.0
76.	Remove handle and jettison	10.0	1485.0
77.	Place bottle in position	10.0	1495.0
78.	Connect OMLFV He line .	10.0	1505.0
79.	Tie bottle down	10.0	1515.0
80.	Cover He bottle enclosure	10.0	1525.0

TABLE 9.5

ACTIVATION PHASE FUNCTIONAL FLOW FOR TWO FLYERS

	Function	Time Minutes	Cum. Time Minutes
	Start: LM landed.		
1.	Activate LSE.	5.00	5.00
2.	Exit LM.	5.00	10.00
3.	Remove and assemble OMLFV No. 1.	9.25	19.25
4.	Move OMLFV No. 1 away from LM.	2.00	21.25
5.	Fuel OMLFV No. 1.	23.25	44.50
6.	Mount pallet and payload No. 1.	5.00	49.50
7.	Remove and assemble OMLFV No. 2.	9.25	58.75
8.	Move OMLFV No. 2 away from LM.	2.00	60.75
9.	Fuel OMLFV No. 2.	23.25	84.00
10.	Mount pallet and payload No. 2.	5.00	89.00
11.	Conduct preflight checks on OMLFV No. 1.	2.00	91.00
12.	Conduct preflight checks on OMLFV No. 2.	2.00	93.00
13.	Time available for non-flyer related activities.	77.00	170.00
14.	Enter LM.	5.00	98.00
15.	Deactivate LSE.	5.00	103.00
	End activation phase.		

TABLE 9.6

ACTIVATION PHASE FUNCTIONAL FLOW FOR ONE FLYER

	Function	Time Minutes	Cum. Time Minutes
	Start: LM landed.		
1.	Activate LSE.	5.00	5.00
2.	Exit LM.	5.00	10.00
3.	Remove and Assemble OMLFV.	9.25	19.25
4.	Move OMLFV away from LM.	2.00	21.25
5.	Fuel OMLFV.	23.25	44.50
6.	Mount pallet and payload.	5.00	49.50
7.	Conduct preflight checks.	2.00	51.50
8.	Time available for non-flyer related activities.	118.50	170.00
9.	Enter LM.	5.00	175.00
10.	Deactivate LSE.	5.00	180.00
	End activation phase.		

TABLE 9.7

EXPLORATION PHASE FUNCTIONAL FLOW FOR LUNAR OPERATIONS

	Function	Time Minutes	Cum. Time Minutes
	Start: Activation phase complete.		
1.	Activate LSE.	5.0	5.0
2.	Exit LM.	5.0	10.0
3.	Conduct preflight checks.	2.0	12.0
4.	Conduct OMLFV sortie.	123.8	135.0
5.	Conduct post-flight procedures.	2.0	137.0
6.	Remove payload and pallet.	2.5	139.5
7.	Refuel OMLFV.	25.5	165.0
8.	Replace pallet.	2.5	167.5
9.	Pick up payload and return to LM.	2.5	170.0
10.	Enter LM.	5.0	175.0
11.	Deactivate LSE.	5.0	180.0
	End of sortie.		

possibly eliminating some operations. Examples would be (1) the use of prepackaged propellant tanks automatically filled from the main LM tanks after landing, and (2) initial connection of the fuel and oxidizer servicing lines on earth prior to launch. These lines would be automatically deployed as the vehicle is deployed to the lunar surface. None of these alternatives has been investigated in sufficient detail to establish the technical feasibility, and so are not incorporated in the present operational sequence.

Safety during servicing operations will be maximized by the use, where possible, of equipment which allows only one mode of operation, the correct/safe one, and by training for the correct sequencing of these operations in a simulated lunar environment. An example of the approach to equipment selections aimed at safe operation is to be found in the fuel/refuel system. Any error allowing the admixture of fuel with oxidant, other than in the engine could be disastrous. Thus, fuel and oxidant line nozzles will be so constructed as to preclude mating with other than the appropriate vehicle port. Thus, human error is eliminated by eliminating any alternative to correct procedure.

However, it is inevitable that some aspects of system operation could be dangerous in the event of operator error, e.g., the operation of a quick disconnect on the fuel or oxidant lines, prior to closing the appropriate transfer line valve. Two approaches to eliminating hazards of this sort exist: First, the design of new equipment which precludes the error (in this case, by making a disconnect physically impossible while transfer line valves are open), and secondly, by ensuring correct operational sequence by the use of itemized checklists monitored by a second person. Both approaches will be used to provide maximum safety.

TABLE 9.8

TYPICAL EVA CYCLE TIME USAGE

Function	Time (Minutes)
LM Egress/Ingress	20
Flyer Servicing and Check	30
Scientific Activity (Payload Handling, Flying and Exploration)	130
Total (3 Hr EVA Cycle)	180

10.0 VEHICLE PERFORMANCE ANALYSIS

10.1 GENERAL

The objectives of the performance analysis are to (1) show the nominal propellant requirements for the OMLFV, (2) determine the effects on propellant requirements of flying off-nominal conditions, (3) show the effects on propellant requirements of various navigation aids, (4) establish propellant requirements for a conservative low altitude, terrain following (nap-of-the moon) guidance technique and (5) identify the effects of payload and range on the amount of propellant required. The performance analysis shows a comparison of the propellant requirements obtained analytically with those obtained from a piloted simulation study.

The analytical study employed a digital computer program modeling a simple guidance law which is easy for a pilot to execute. The program provided for the introduction of errors, representing errors in pilot judgment, hardware and vehicle characteristics. Nominal trajectories were established by exposing each candidate trajectory to 100 sets of random errors (Monte Carlo analysis) and trading off propellant requirements with the ability to safely accommodate reasonable errors. This same technique was used to evaluate and compare navigation aids. Nap-of-the moon performance data represents the propellant required to fly the contour of the lurain (within approximately 100 ft) at relatively low velocities. This flight technique is expected to be representative of the type of flights to be flown on the earliest of the lunar exploration missions.

10.2 BASIC PERFORMANCE DATA

10.2.1 Analytical Procedure

10.2.1.1 General Approach

A major objective of the analytical studies was to determine nominal propellant requirements for various payloads and ranges. A means for defining a nominal flight was required in order to establish these requirements. The following approach was used.

(1) Develop the equations of motion and the guidance laws required to define the desired flight profile.

(2) Define vehicle mass and thrust characteristics together with their expected uncertainties.

(3) Based on the equipment carried on board the flyer and the capabilities of the astronaut, determine the expected value of errors and uncertainties in the guidance parameters required for flight path control.

(4) Develop a computer program of the math model, including a means for accounting for uncertainties and errors in vehicle characteristics and guidance parameters.

(5) Using the program, vary the trajectory control parameters (thrust duration, magnitude and orientation) to obtain a range of flight profiles. Accept those profiles which can accommodate the expected errors and reject all others.

(6) For each range, select one of the acceptable profiles as a nominal trajectory. This selection is based on ΔV and the sensitivity to errors. The other acceptable profiles are used as the basis for determining performance requirements for flying off-nominal conditions.

The following sections describe the guidance technique used in the study and defines the mathematical model.

10.2.1.2 Guidance Technique

To estimate the performance requirements analytically requires the definition of a guidance law and the specification of the type of flight profile to be flown. With this information, a mathematical model can be developed and a parametric study conducted.

Earlier studies have shown ballistic trajectories (a maximum thrust boost phase followed by a zero thrust coast phase and finally a maximum thrust braking phase) to be efficient but very error sensitive. A more desirable trajectory provides an optimum speed (which result in low ΔV requirements) while providing a terminal approach phase which allows for easy accommodation of errors. This modified flight profile was used in this study and is described in more detail in the next section. The reader is directed to References 2 and 26 for a more extensive discussion of the various types of flight paths.

A guidance technique suitable for flying the flight profile was defined. The selected guidance technique was divided into six distinct phases for purposes of analysis. A summary description of these phases is given in Table 10.1 and a sketch of a typical flight trajectory is presented in Figure 10.1.

Phase 1 - Lift-Off and Boost

Phase 1 consists of a vertical lift-off followed by a smooth transition to a sloping straight line flight path. Lift-off occurs at $t = 0$ when maximum thrust is applied and held constant for a preselected boost duration (t_1). At the same time the vehicle pitch attitude is changed at a constant pitch rate from the initial zero value to the preselected boost attitude (θ_B) and then held constant at this value for the duration of boost.

TABLE 10.1

SUMMARY DEFINITION OF FLIGHT PHASES

Phase	Thrust	Pitch Attitude	Termination Parameter
1	T_{max}	Constant	Preselected Time
2	T_{min}	Zero	Zero Rate of climb
3	$T = W$	Zero	Range to Destination
4	T_{min}	Zero	Closure Condition $\rightarrow$ 75 % T_{max}
5	Variable	Variable	Range of 200 ft
6	Variable	Variable	Acceptable Touchdown Conditions

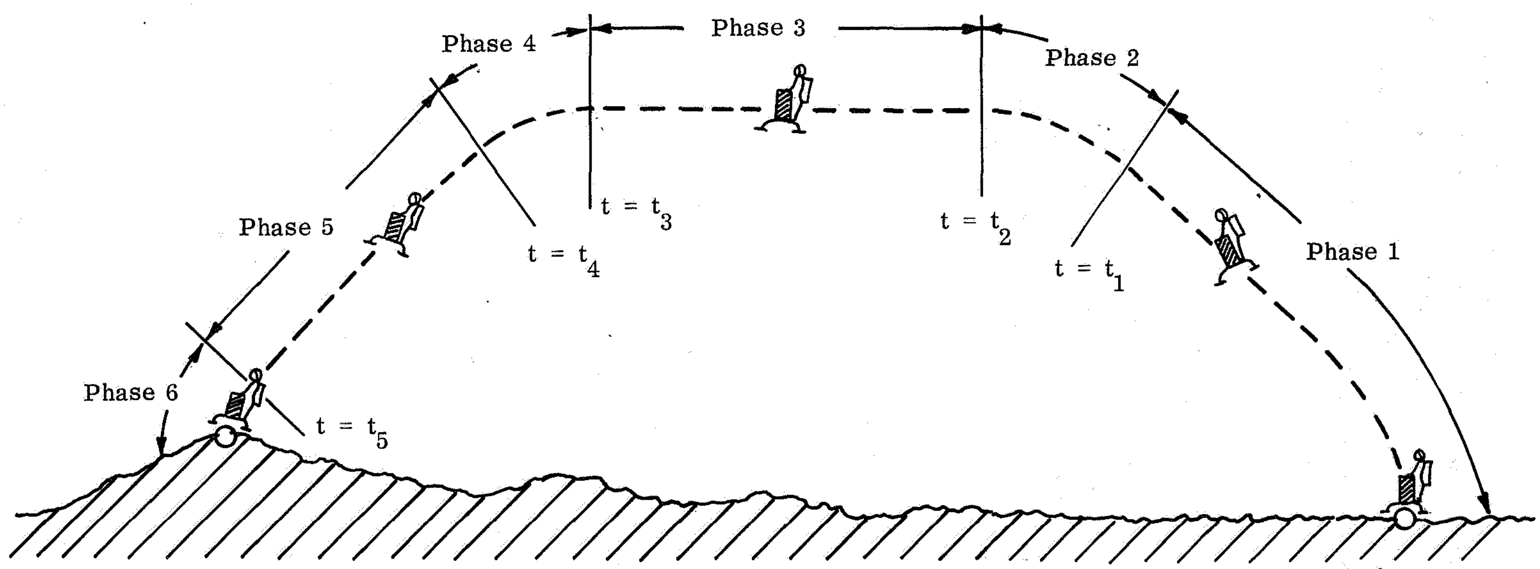


Figure 10.1. Typical Flight Profile

Phase 2 - Transition to Cruise

The flight during this phase is a near-parabolic transition from the point of boost termination to a horizontal flight path. The duration of this phase is from t_1 until the moment when the altitude rate is observed to have been brought to zero ($\dot{h} = 0$) at $t = t_2$. At time $t = t_1$ the thrust is cut to the minimum setting, and the vehicle pitch attitude is returned at a constant pitch rate to the vertical (zero) position. At minimum thrust, gravity gradually reduces the altitude rate to zero and phase 3 begins.

Phase 3 - Cruise

During this phase, the flight path is horizontal, constant velocity; the duration of this phase is from t_2 until the time when the vehicle is observed to have reached a selected distance to the target ($x = x_{des}$) at $t = t_3$. This phase is flown VFR. The vehicle pitch attitude is maintained at zero, and the thrust is controlled to maintain the vehicle at constant altitude.

Phase 4 - Transition to Terminal Approach

The flight during this phase is a near-parabolic transition from the horizontal to a downward sloping flight path. The duration of this phase is from $t = t_3$ until a constant deceleration terminal approach can be made with a 25% T_{max} thrust reserve (T_{des} = 75% T_{max}) at $t = t_4$. At $t = t_3$ the thrust is cut to the minimum level, while the vehicle pitch is maintained at zero. During this transition phase, the position and velocity, with respect to the destination, is monitored and used to estimate the point where 75% of T_{max} is required to reach the destination.

Phase 5 - Terminal Approach

The flight during this phase is approximately a straight line flight path to the destination. The duration of this phase is from $t = t_4$ until the range to target is within 200 ft ($R < 200$), at $t = t_5$. This phase is flown VFR. Based on the observed range and line-of-sight angle and their rates, appropriate commands for thrust and pitch attitude are constantly generated such that the resulting flight path will approach the destination along a constant deceleration flight path. At time $t = t_4$ the pitch attitude is changed to the commanded pitch attitude; following this the applied thrust is brought up to the commanded thrust level.

Phase 6 - Touchdown

The flight for this final phase is a smooth transition from the terminal approach to a near vertical flight path having very low vertical and horizontal velocity residues at the final touchdown at $t = t_6$. During this phase, also flown VFR, commands for thrust and pitch attitude are generated based on observed altitude and vertical and horizontal velocities such that these three variables can be brought to near-zero values simultaneously. At time $t = t_5$ the vehicle is pitched at a constant pitch rate

to the proper attitude. Following this maneuver, the applied thrust is modulated to set up acceptable closure conditions. Termination of this phase occurs when the altitude reaches one foot or when the altitude rate goes to zero ($h \leq 1$ or $\dot{h} \geq 0$).

10.2.1.3 Mathematical Model

A description of the digital computer model used in the performance study is given in Appendix B. Equations defining the variable mass in-plane equations of motion, as well as the flight procedures and constraints imposed in each flight phase are presented.

The manner in which error effects were modeled is also described in Appendix B . Vehicle/mission parameters subject to variation include initial range, initial mass, maximum and minimum thrust levels, and specific impulse. Guidance parameters subject to error in measurement or estimation include instantaneous range, range rate, altitude rate, and pitch attitude. The criteria for determining whether a given set of errors leads to a successful or unsuccessful flight is also given in the Appendix.

10.2.2 Basic Vehicle Performance

Vehicle performance is measured in terms of the characteristic velocity, ΔV, required to fly a specified range. The performance data presented in this section are based on the guidance law described above, and on the following vehicle characteristics.

T_{max} = 300 lbf

T_{min} = 50 lbf

m_o = 29.9 slugs plus payload

I_{sp} = 280 sec

In addition, maximum pitch rate limits, based on piloted simulation studies, were imposed to include the effects of finite control capabilities in the performance data. These limits were set to 5 deg/sec during boost and 10 deg/sec for the remainder of the flight.

10.2.2.1 Nominal Performance

Figure 10.2 shows the nominal vehicle performance and several significant guidance parameters as functions of range for three different payloads. In comparing the no payload data with the 370 lb (a change of about 42% in takeoff weight) payload data, it is seen that the boost duration is increased on the average by 51%. Thus, almost the full increase in boost duration is directly due to the increased mass (reduced T/W).

It is also seen from this figure that the pitch attitude is nearly constant for the longer ranges (greater than about 4 n.mi.) but takes on different values for different payloads. Cruise altitude is also a function of payload weight, decreasing with payload in an almost exponential manner.

Flight profiles for flights of 1, 2, 5 and 10 miles are presented in Figure 10.3.

10.2.2.2 Off-Nominal Performance

For the earliest exploration flights, the astronauts may elect to fly at speeds and altitudes lower than those required for good efficiency. The performance penalty resulting from flying these off-nominal conditions is shown in Figure 10.4 through 10.6. These figures present constant ΔV contours in the cruise velocity - cruise altitude plane. The contours are nearly vertical lines at the lower speeds, indicating that the propellant requirements are virtually independent of altitude. At cruise speeds approaching the optimum, the effect of cruise altitude on ΔV becomes more pronounced and the effect of cruise velocity on ΔV becomes less pronounced. A good approximation to the ΔV requirements for the low speed conditions can be made by assuming that the total range is flown at the cruise speed chosen. At the lower speeds, a large percentage of the flight is flown at cruise conditions (T = W). The terminal approach is flown at a thrust level only slightly greater than T = W. Further, the boost phase together with the two low-thrust transition phases approaches an average thrust of T = W. Thus the ΔV can be approximated by

$$\Delta V = gt = g\frac{x_o}{V_c}$$

The upper boundary lines shown on some of the figures, identify the velocity/altitude conditions which result in flights having no cruise phase. The lower boundary represents the limit of the combinations of altitude and velocities which can be attained with the guidance laws and constraints assumed in this analysis.

10.2.2.3 Effects of Elevation on Performance

Since the mission of the flyer is to aid in the exploration of interesting features on the lunar surface (crater floors and rims, rilles, etc.) it is highly likely that the takeoff site will be at a different elevation from the landing site. The effects of this elevation difference on performance was investigated. Five differences in elevation, each 500 feet higher (or lower) than the next, were considered for a flight having a range of 2.0 n.mi. There was virtually no difference (less than 5 fps) in the ΔV requirements of these flights.

10.2.3 Evaluation of Navigation Aids

The above performance data is based on a guidance system which utilizes the capabilities of the astronaut to the maximum possible extent, providing him with visual cues and a timer for in-flight guidance information. This section identifies

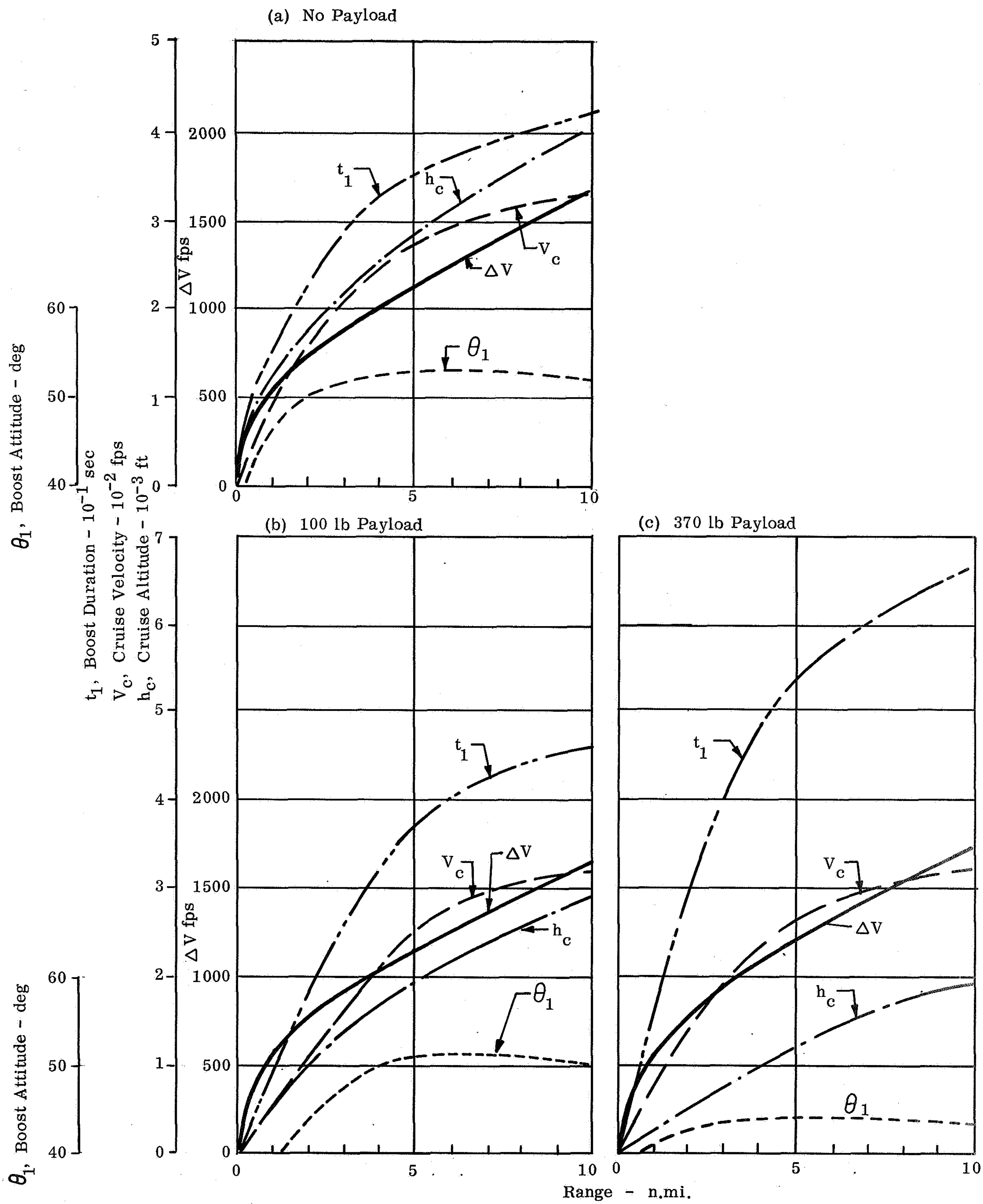


Figure 10.2 Nominal Trajectory Characteristics

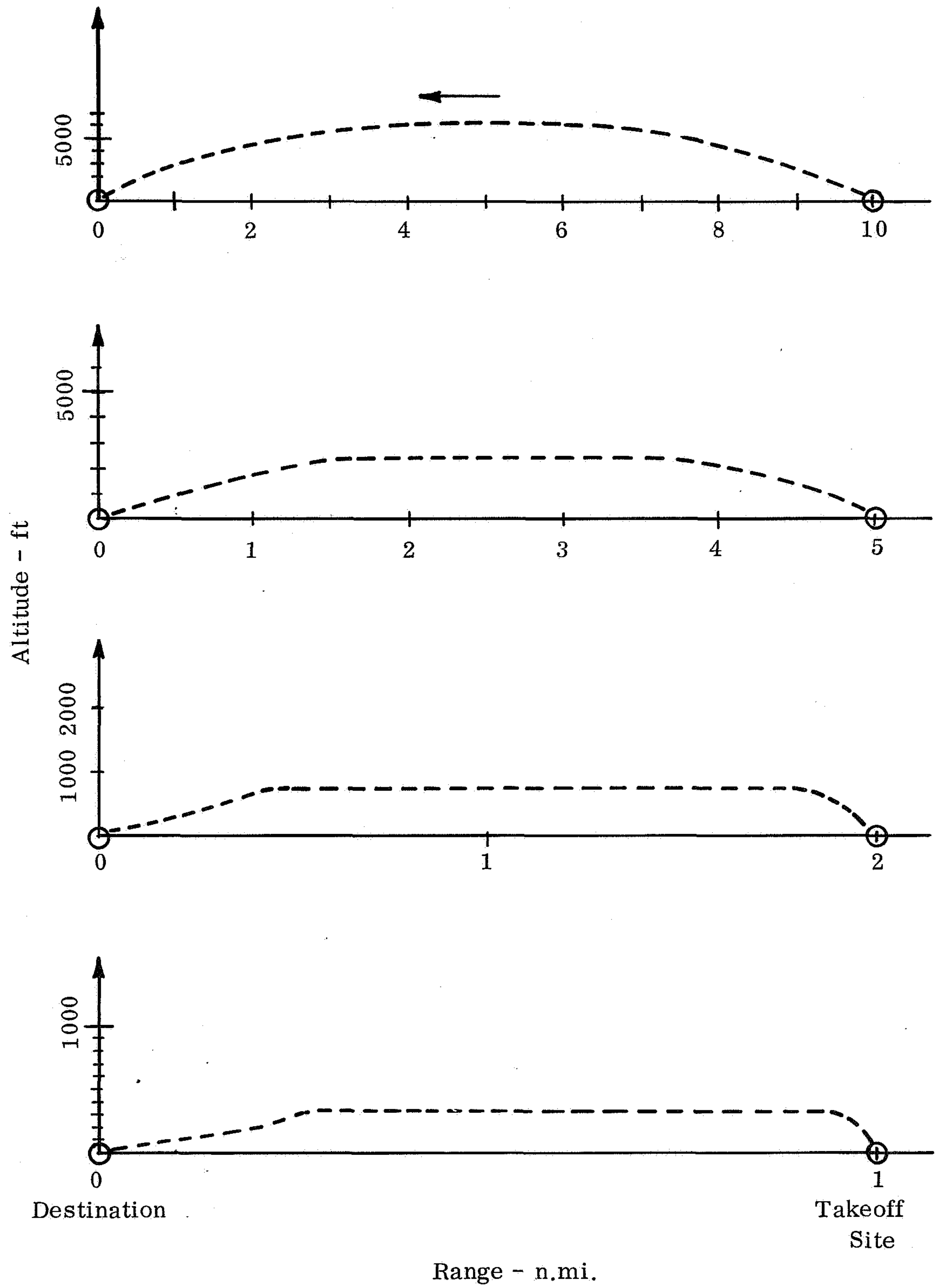


Figure 10.3. Nominal Flight Profiles

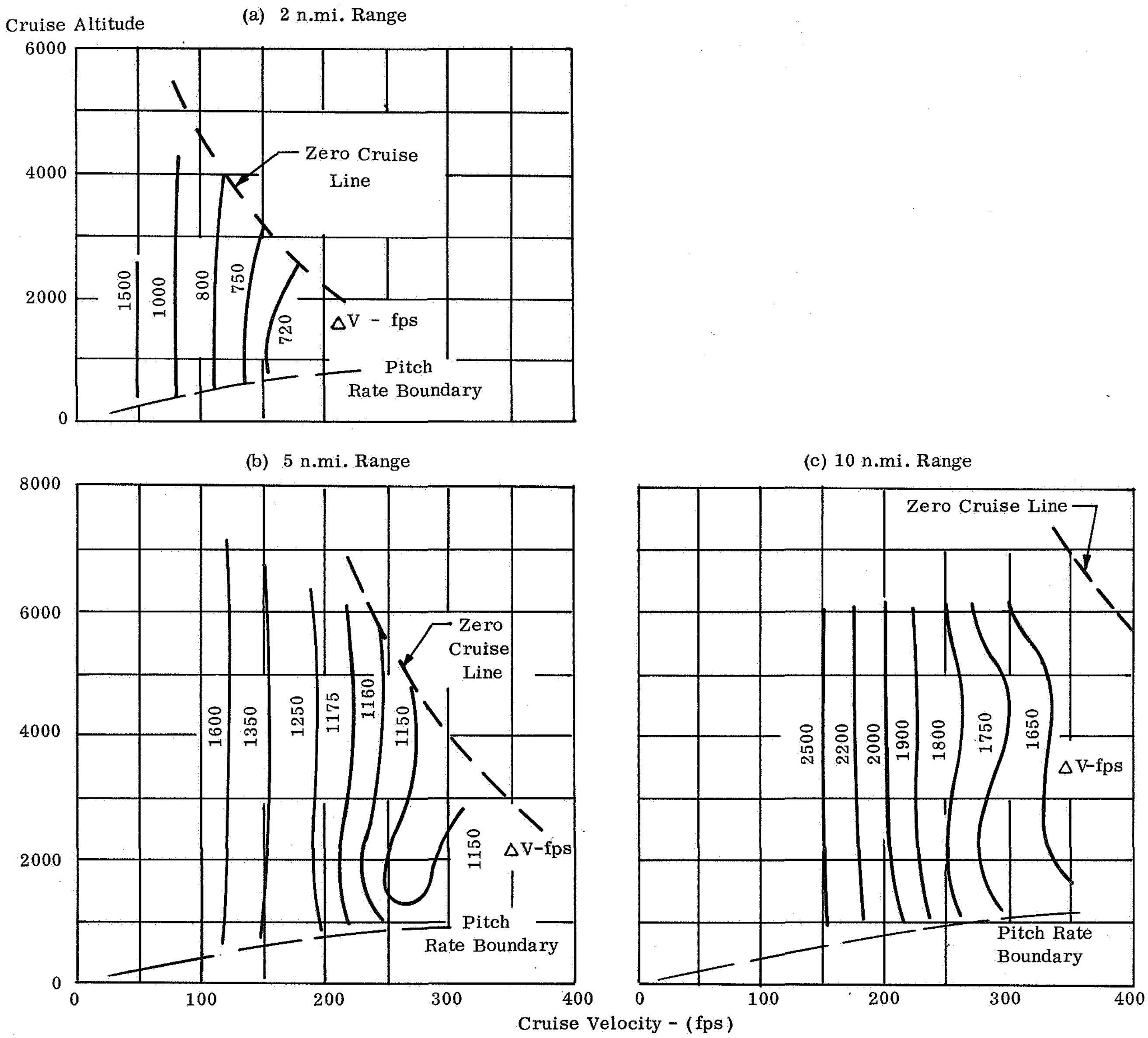


Figure 10.4 OMLFV Performance Characteristics as a Function of Cruise Altitude and Velocity - Zero Payload

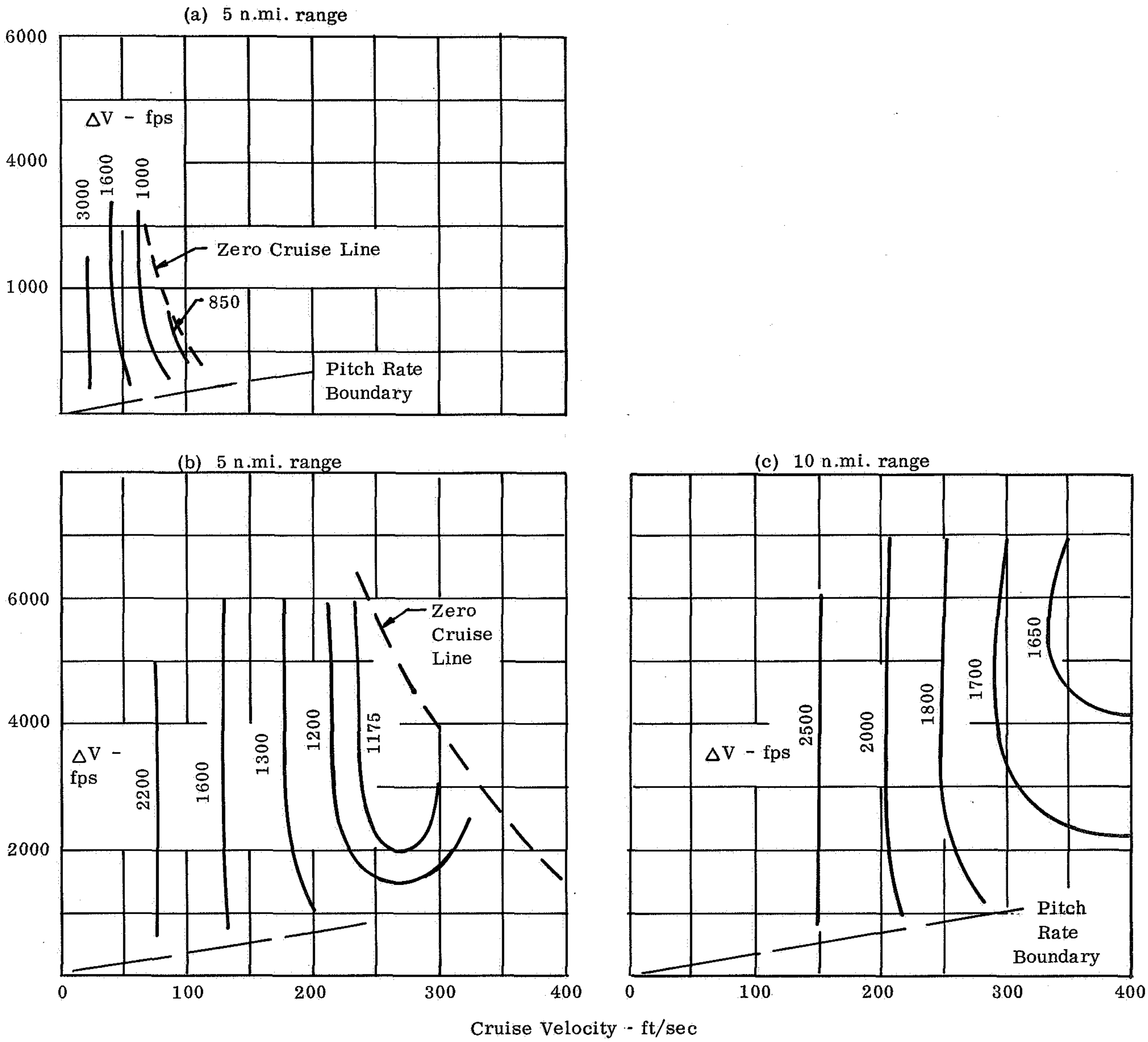


Figure 10.5. OMLFV Performance Characteristics as Function of Cruise Velocity - 100 lb Payload

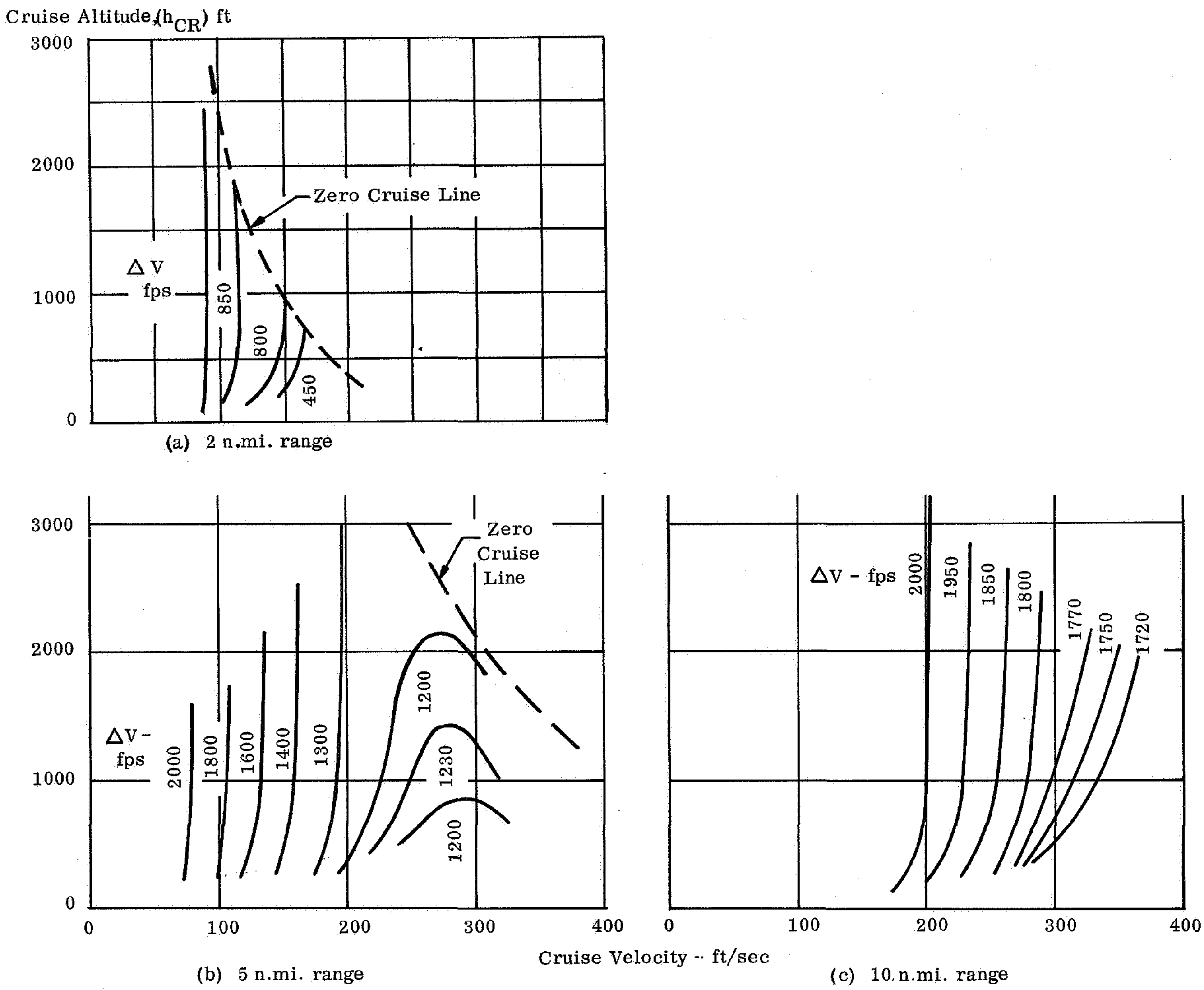


Figure 10.6. OMLFV Performance Characteristics as a Function of Cruise Altitude and Velocity - 370 pound Payload

the amount of propellant which is saved if various sets of navigation aids are carried onboard. A performance payoff is realized by providing a set of navigation aids which reduces the system errors and permits a more efficient nominal trajectory to be flown.

The procedure used to determine the effects of navigation aids on propellant requirements is as follows: (1) for each navigation aid considered, define a suitable guidance system and a corresponding set of errors, (2) using a Monte Carlo method, find a nominal trajectory, and (3) determine the system hardware weight and the nominal propellant weight requirements for each system.

To evaluate navigation aids, five navigation and guidance (N and G) systems were postulated. The systems were chosen so that the effects of better information about the key N and G signals could be determined. Table 10.2 lists the five systems and identifies the equipment used in each system and its weight. The weights presented are total system increments which include the structure required to support the N and G equipment, displays and a battery. Where available, weights were obtained from manufacturer's literature; in other cases, the weights data were estimated.

Table 10.3 lists the milestones which are key points in the flight and the guidance parameters which identify the occurrence of these milestones. Vehicle attitude is also presented since it is a critical signal for flight path control. The columns to the right present the source of the guidance parameter for each of the five systems.

Since the evaluation of navigation aids accounts for the sensitivity of the system to errors as well as the efficiency of propellant consumption, it is important that the error sources and sizes be known. Hardware errors (radar, gyros, thrusters, etc.) can be defined with a relatively high confidence level. However, those systems relying to a large extent on the capabilities of the astronaut are more difficult to assign errors to. A literature search has indicated that very little test data exists regarding the ability of the astronaut to estimate distances and rates, particularly in the lunar environment. Estimates of errors in these quantities were made.

The perceived value of altitude, altitude rate, range and range rate is assumed to be linearly porportional to the actual value of the variable. The constant of proportionality is given by a normal distribution having a mean value of unity and a standard deviation of 0.1 (i.e., perceived information is in error by 10% on a 1σ basis). In addition, both altitude rate and range rate data, estimated by the astronaut, contain an added error which is proportional to altitude. Again, the constant of proportionality is given by a normal distribution, but one having a mean value of zero and a standard deviation of 0.01 (i.e., estimated rates are in error by an additional 10 fps per 1000 ft on a 1σ basis). Table 10.4 lists the guidance parameter errors which were used in the evaluation.

TABLE 10.2

NAVIGATION AIDS

System	Navigation and Guidance Equipment	Typical Manufacturing Model	Weight lbm
I	Timer		1
II	Timer, Simple Sight		3
III	Velocity Meter, Rate Gyros/Integrators, Battery, Displays	Honeywell GG177 U. S. Time CD-040	7
IV	Velocity Meter, Rate Gyros/Integrators, Radar Altimeter Battery, Displays	Honeywell GG177 U. S. Time CD-040 Westinghouse	23
V	Distance Measuring Equipment Rate Integrating Gyros, Radar Altimeter Battery, Displays	RCA VHF ranging sup. Honeywell GG334 Westinghouse	33

TABLE 10.3

GUIDANCE PARAMETER USAGE AND SIGNAL SOURCE

	Source of Guidance Parameter				
Flight Milestone/Guidance Parameter	System I	System II	System III	System IV	System V
1. Attitude (Pitch, Roll and Yaw)	Astro Judgement	Simple Sight	Rate Gyro/Integ	Rate Gyro/Integ	Rate Gyro/Integ
2. Boost Phase Termination (Range Rate)	Time	Time	Velocity Meter	Velocity Meter	Dist Measure Equip.
3. Cruise Phase Initiation (Altitude Rate)	Astro Judgement	Astro Judgement	Astro Judgement	Radar Altimeter	Radar Altimeter
4. Cruise Phase Thrust (Altitude and Altitude Rate)	Astro Judgement	Astro Judgement	Astro Judgement	Radar Altimeter	Radar Altimeter
5. Cruise Phase Termination (Range)	Map/Time or Astro Judgement	Map/Time or Astro Judgement	Map/Time or Astro Judgement	Map/Time or Astro Judgement	Distance Measure Equipment

TABLE 10.4

GUIDANCE PARAMETER ERRORS

Guidance Parameter	1 σ Error Value				
	Sys. I	Sys. II	Sys. III	Sys. IV	Sys. V
1. Pitch Attitude (Degree)					
● Controllability	2	2	0.5	0.5	0.5
● Reference Accuracy	6	2	1.5	1.5	0.5
2. Altitude (h) (Percent)	10	10	10	2	2
3. Altitude Rate ($\dot{h}$) (Percent)					
● Proportional to $\dot{h}$	10	10	10	2	2
● Proportional to h	1	1	1	0	0
4. Range (R) (Percent)	10	10	10	10	1
5. Range Rate ($\dot{R}$) (Percent)					
● Proportional to $\dot{R}$	10	10	2	2	1
● Proportional to h	1	1	0	0	0

System I, the simplest system, provides the astronaut with only a time signal which correlates directly with range rate and range, assuming a known T/W and pitch attitude. This time signal could be provided by either a timer or by an audio signal to relieve the astronaut from reading a display. All other N and G data must be obtained via astronaut judgement. In the case of range data, however, maps and recognizable landmarks could provide good information.

System II includes a simple sighting device in addition to a timer. The sight provides pitch, roll and yaw information by aligning crosshairs with horizon and/or celestial features. This improves the astronaut's ability to orient the vehicle to the proper attitude by a factor of two. A comparison of the propellant requirements for Systems I and II provides a means of evaluating the effects of improved attitude information.

The effects of better attitude and better range rate information can be obtained by comparing the performance of Systems II and III. System III includes a velocity meter with its sensitive axis aligned with the thrust vector and a set of rate gyros/ integrators. The velocity meter provides a better boost phase cutoff signal than time, since it accounts for uncertainties in takeoff mass and thrust level. The rate gyro/ integrator package provides relatively good attitude information early in the flight, before gyro errors can integrate into large angle uncertainties. The error value presented in the above table is based on 100 sec of flight. This is sufficient time to complete the boost phase even for the longer range flights. During the terminal approach and landing, the flight is flown VFR in which the astronaut modulates thrust and attitude, as required, to reach the destination.

System IV duplicates System III and, in addition, contains a radar altimeter (Reference 26). An evaluation of the effect of good altitude/altitude rate information on performance can be made by comparing Systems III and IV.

System V provides better range/range rate information by using VHF ranging equipment and uses three rate integrating gyros to provide good attitude information. The gyros are used in a strapdown manner and are coupled to digital counters and displays.

The error characteristics for each of these systems were inputted into the Monte Carlo digital computer program described above. With these characteristics, new nominal trajectories were determined which yielded a 95% probability of successfully meeting flight terminal conditions specified previously. Using this value in each evaluation provides a valid comparison of one aid with another.

Table 10.5 shows the propellant requirements for flying one five-mile flight using each of the N and G systems. As the capability of the system increases, the propellant weight decreases and the hardware weight increases. It can be seen that for a round trip having a 5 n.mi. radius, a total propellant saving of 38 lbm could be realized using System V.

TABLE 10.5

PROPELLANT REQUIREMENTS FOR VARIOUS NAVIGATION AIDS FOR A FIVE N.MI. ROUND TRIP

System	ΔV fps	Prop. Wt lbm	Δ Prop. Wt Compared to Sys. I lbm	Additional Sys. Weight lbm
I	2340	251	-	-
II	2290	246	5	2
III	2195	237	14	6
IV	2030	222	29	22
V	1950	213	38	32

NOTE: Propellant weights are based on carrying a 100 lb payload. Takeoff weight is therefore 1064 lbm.

From Table 10.5 it can be seen that there is a significant increase in performance of Systems IV and V compared to the others. This suggests that altitude/altitude rate information provided by the radar altimeter can have a large performance payoff in addition to safety of flight considerations. Further study is required to determine the extent to which some or all of the navigation aids are necessary or desirable. The training program discussed in Section 13.0 describes such a flight research plan. Also, factors such as cost, reliability and development time, must be considered. It should be noted, however, that even in the absence of navigation aids, except a time signal (System I), considerable utility can be made of the OMLFV in early missions where long flights are less frequently required. Because of this utility and the underlying philosophy of system reliability, System I performance characteristics and design features have been selected.

10.3 NAP-OF-THE MOON PERFORMANCE

10.3.1 General

Early exploration flights will be flown in the same "nap-of-the earth" manner used on earth rocket vehicles and helicopters. They will be flown with conservative (low and slow) nap-of-the moon flight rules in which the flight path follows the lurain. These flights are characterized by small pitch attitudes (less than 20°), low altitudes (less than 100 ft), and low speeds (less than 150 fps).

The advantages of this type of flight are: (1) Astronaut judgement of altitude, speed, and sink rate is much better at low altitudes, so the flight will be safer. The onset of dangerous situations can be avoided because of the better cues and because the low speeds offer the astronaut more reaction time. (2) Flight experience on earth vehicles has been with this type of trajectory. The primary disadvantage of nap-of-the moon flying is its higher propellant requirements, particularly as the ranges extend beyond 2 or 3 miles.

10.3.2 Analytical Procedure

Flight rules for nap-of-the moon flying are:

Liftoff: increase throttle to $T \cong W$ and control attitude as required to null out translational velocities introduced because of locally sloping terrain. Climb to an altitude of approximately 50 feet.

Acceleration: Pitch to a nosedown attitude, while simultaneously increasing thrust so that no sink or climb rate is developed. Hold this attitude and throttle position until the midpoint of the initial range is reached or until a suitable cruise velocity is attained. (The cruise velocity is a parameter in the performance study.)

Cruise: (This phase is required only if cruise velocity is reached before the halfway point.)
Pitch to a vertical attitude and position the throttle so that the cruise altitude

above the local lurain is constant. Maintain these cruise conditions until a range is reached such that a nose up attitude will reduce the cruise velocity to zero by the time the destination is reached.

Terminal Approach: Pitch the vehicle nose up and increase thrust so that altitude is maintained. Hold these conditions until the cruise velocity is reduced to zero at the destination.

Touchdown: Control attitude to zero the horizontal velocity and modulate thrust to descend at an acceptable sink rate to touchdown.

The simplifying assumptions made and the equations used to compute the propellant requirement for nap-of-the moon flights are:

(1) The astronaut changes pitch attitude in a stepwise fashion.

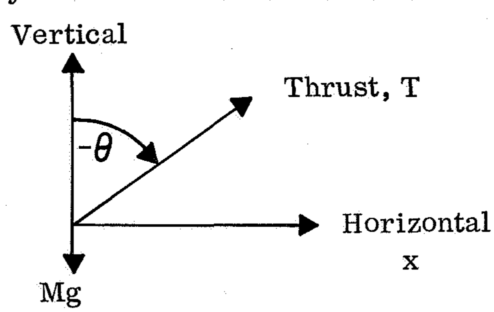
(2) There are no errors in vehicle attitude or the ability of the astronaut to judge distance and velocity.

(3) I_{sp} is constant.

(4) There is a negligible time and amount of propellant consumed during liftoff and touchdown. It is assumed that these small actual requirements will be assigned to the propellant reserve requirements.

(5) No out-of-plane maneuvering is required.

The equations are: (The coordinate system shown in the sketch is used.)



Conditions at the end of the acceleration phase are given by:

$h = 50$ ft = altitude

$$x_a = \frac{V_c^2}{2\, g \tan \theta} = \text{range covered}$$

$$\dot{x}_a = V_c = \text{cruise velocity}$$

$$t_a = \frac{V_c}{g \tan \theta} = \text{time required to attain } V_c$$

$$\Delta V_a = \frac{V_c}{\sin \theta} = \text{characteristic velocity}$$

Changes in the above conditions made during the cruise phase are:

$$\Delta x_c = x_o - \frac{V_c^2}{g \tan \theta} = \text{incremental range covered during cruise}$$

where x_o = initial range-to-go

$$t_c = \frac{x_o}{V_c} - \frac{V_c}{g \tan \theta} = \text{duration of cruise phase}$$

$$\Delta V_c = \frac{g x_o}{V_c} - \frac{V_c}{\tan \theta} = \text{characteristic velocity required for cruise}$$

Changes made during the terminal approach phase are:

$$h_{ta} = -50 \text{ ft}$$

$$\Delta x_{ta} = \frac{V_c^2}{2 g \tan \theta}$$

$$\Delta \dot{x}_{ta} = - V_c$$

$$t_a = \frac{V_c}{g \tan \theta}$$

$$\Delta V_{ta} = \frac{V_c}{\sin \theta}$$

The total propellant requirement in terms of characteristic velocity is given by ΔV_T.

$$\Delta V_T = \Delta V_a + \Delta V_c + \Delta V_{ta}$$

$$= V_c \left(\frac{2 - \cos \theta}{\sin \theta} \right) + \frac{g x_o}{V_c}$$

The amount of propellant expressed in pounds mass required to travel a range given by x_o is:

$$W_p = \frac{m_o V_c}{I_{sp}} \left(\frac{2 - \cos \theta}{\sin \theta} \right) + \frac{m_o g_m x_o}{I_{sp} V_c}$$

where m_o = mass at takeoff

Because of these flight rules, there is a maximum velocity which can be attained for any given range-to-go. This velocity is that given by extending the acceleration phase to the mid-range point of the flight. An expression for this velocity is:

$$V_{max} = \sqrt{g x_o \tan \theta}$$

10.3.3 Performance Data

Using the above expressions, performance data were computed for nap-of-the moon type flights. Data are presented for a pitch attitude of 20° during the acceleration and terminal approach phases and for cruise velocities of 50, 100 and 150 fps. A 20° pitch attitude was chosen because it yields propellant requirements which agree quite well with short range piloted simulation results and because it represents a modest pitch attitude which is in keeping with a conservative philosophy which underlies nap-of-the moon flying. It is interesting to note that for the flight rules used, a pitch attitude of 60° yields optimum performance. Using a 60° pitch attitude and the best cruise velocity, a 30% reduction in propellant requirements could be realized over the $\theta = 20°$ flights.

Table 10.6 shows the vehicle weight breakdown used in the nap-of-the moon performance calculations.

TABLE 10.6

WEIGHT BREAKDOWN

	lb
Vehicle dry weight	235
25% dry weight growth allowance	59
Usable propellants less reserves	270
Reserve propellants	30
Astronaut weight	370
Maximum payload	370
Maximum gross weight	1334

A value for I_{sp} was obtained by considering the engine duty cycle average for payloads of zero and 370 lb payload and for tanks full and tanks empty. This information is shown in the accompanying table:

Condition	Thrust per Engine-lb	I_{sp} - sec
Max payload, full tanks, T = W	111	277.6
Max payload, full tanks, T = W sec θ	118	278.2
No payload, full tanks, T = W	80	268.8
No payload, full tanks, T = W sec θ	85	271.0
Max. payload, empty tanks, T = W	89	272.4
Max. payload, empty tanks, T = W sec θ	95	274.0
No payload, empty tanks, T = W	58	257.7
No payload, empty tanks, T = W sec θ	62	260.0
	Average I_{sp} =	270.0

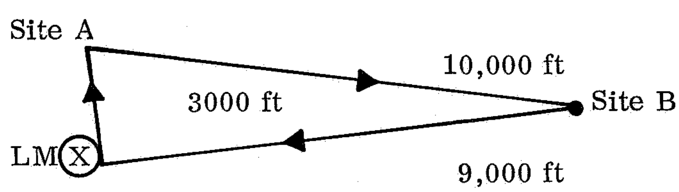
Figure 10.7 shows the propellant requirements, in terms of characteristic velocity, ΔV, as a function of initial range-to-go. The dashed line shows the propellant requirements for flights which have no cruise phase. The maximum velocity attained during these flights is proportional to the square root of range.

Using the ΔV requirements of Figure 10.7, the maximum radius of operation for various payloads can be computed. Figure 10.8 presents this information for cruise velocities of 50, 100 and 150 fps.

Data showing the weight of the consumed propellant as a function of range and takeoff weight is presented in Figure 10.9. These curves can be used by mission analysts in planning multi-legged sorties. The following example illustrates the procedure to be followed.

EXAMPLE:

Taking off from the LM, fly first to site A then to site B and finally back to the LM (see the accompanying sketch). A total payload of 330 pounds is required for this exploration sortie. Of this, 100 pounds is to be dropped off at site A, another 100 pounds is dropped off at site B and the remainder is delivered back to the LM. The empty weight, without payload is 664 pounds. Velocities are not to exceed 100 feet/second. Determine whether this sortie can be flown with a 300 pound propellant capacity.



First Leg (LM to Site A):

1. Known Information: Takeoff weight = 1294 lb (empty wt = 664, Payload = 330, propellant = 300)
   Range = 3000 ft
2. Figure 10.9 is entered to find the minimum propellant weight required for the known information. From this figure 70 pounds of propellant is required for this leg. Thus, the weight decrement for the first leg, including propellant and payload weight, is 170 pounds.

Second Leg (Site A to Site B)

1. Known Information: Takeoff weight = 1294-170 = 1124 lb
   Range = 10,000 ft

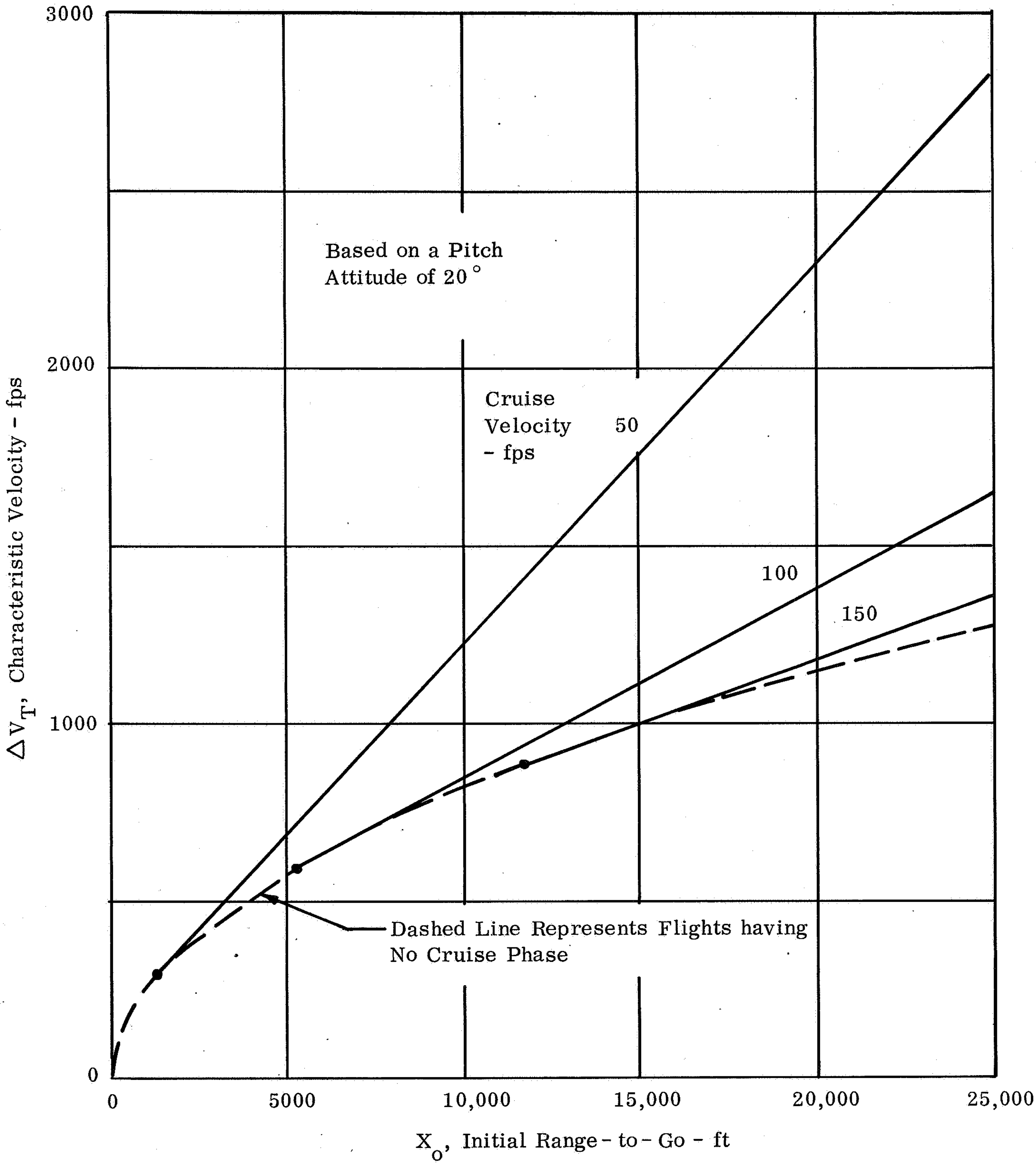


Figure 10.7. Characteristic Velocity Requirements for Nap-of-the Moon Flight Rules

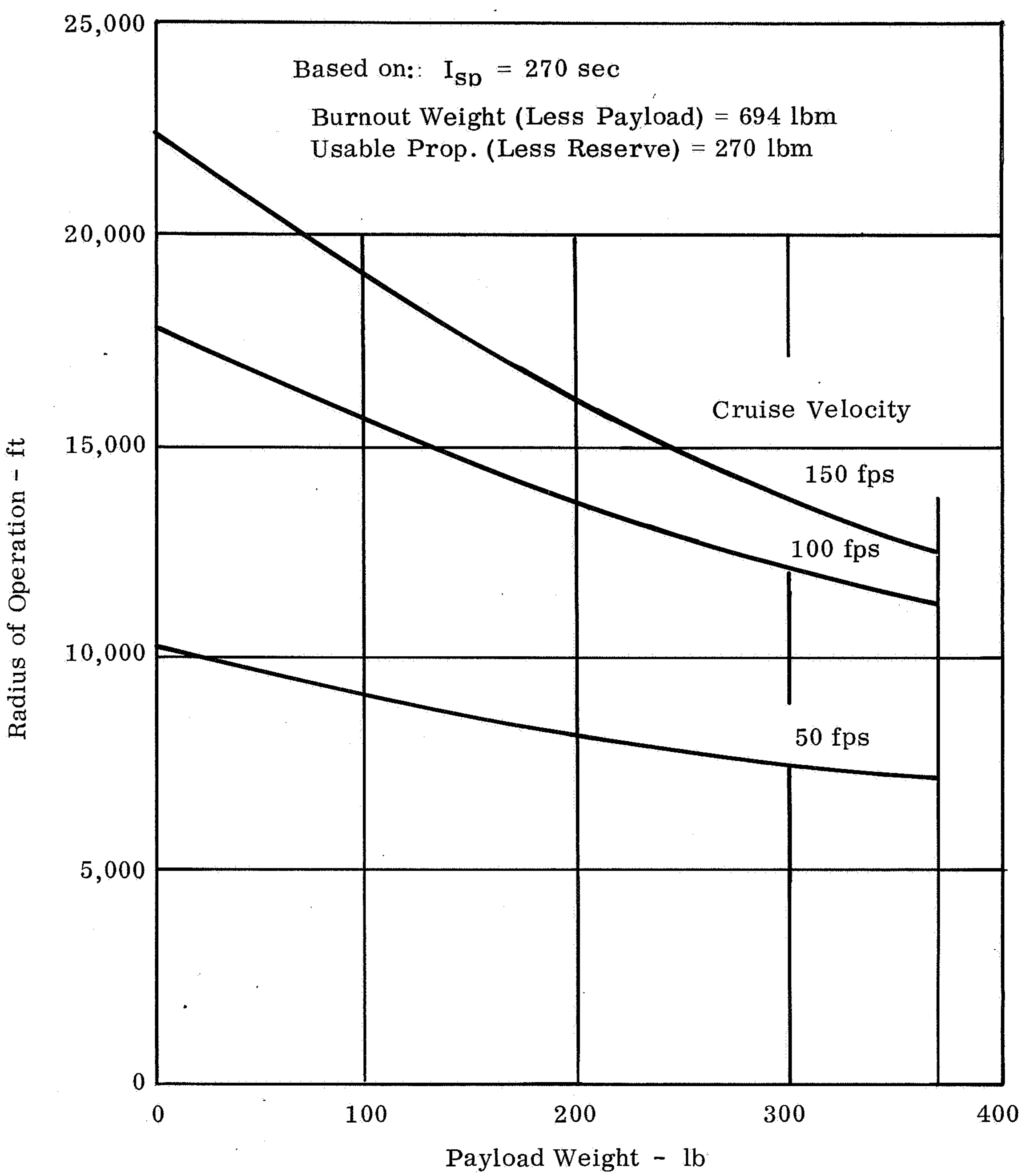


Figure 10.8. Maximum Radius of Operation as a Function of Payload for Nap-of-the Moon Flight Rules

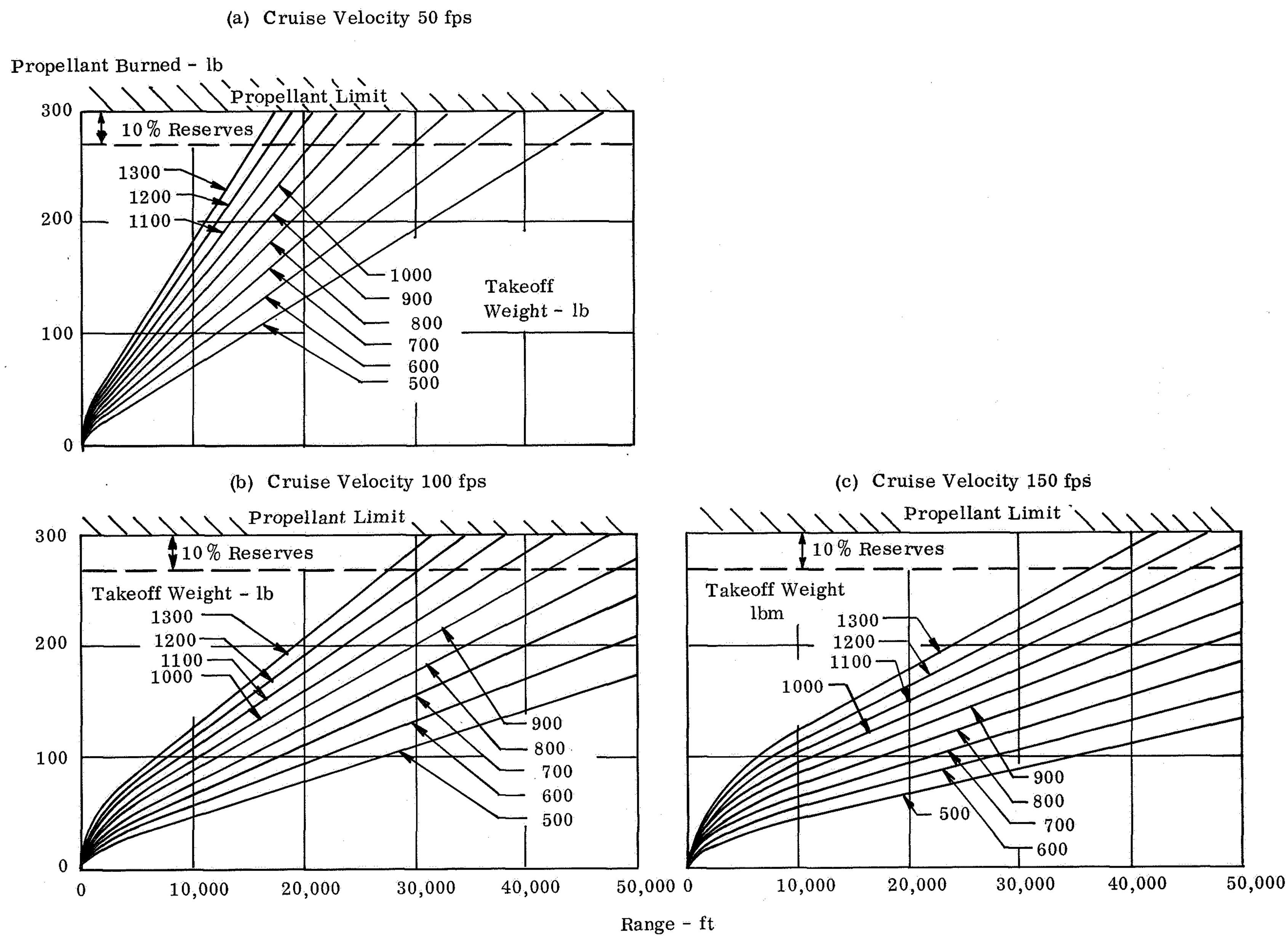


Figure 10.9. Propellant Requirements as a Function of Range and Takeoff Weight for Nap-of-the Moon Flight Rules

2. From Figure 10.9, it is seen that 110 pounds of propellant is burned on this leg. This weight together with the payload drop results in a weight decrement of 210 pounds.

Third Leg (Site B to LM)

1. Known Information: Takeoff weight = 1124-210 = 914 lb
   Range = 10,000 ft

2. From Figure 10.9, 80 pounds of propellant is required. The total propellant required for this sortie is thus 70 + 110 + 80 = 260 pounds. This indicates that the sortie can be flown, with a 15% propellant reserve.

10.4 COMPARISON OF ANALYTICAL AND SIMULATION PERFORMANCE

Thus far, two sets of performance data have been presented. Both sets were established from analytical tools which modeled the guidance laws used by the astronaut and, in one case, included provisions for errors and uncertainties in the vehicle characteristics and flight data. A piloted simulation, described in Appendix C, was also used to obtain performance data. This data provides a basis for verification of the analytical tools. Figure 10.10 shows a comparison of the three sets of performance data.

The solid line on Figure 10.10 presents the nominal ΔV requirements for the basic vehicle as described in Paragraph 10.2.2. The nap-of-the moon data is based on flying at a cruise velocity proportional to range, up to a range of 1 n.mi., at which point the cruise velocity is limited to 100 fps for all longer ranges. Simulation data points presented in the figure include all final evaluation flights flown by the four pilots used in the simulation program, and hence contain pilot-to-pilot as well as run-to-run variations. Table 6.5-6 presents more information on these flights.

There is relatively good agreement between the two analytical curves for the shorter ranges. For ranges greater than about two miles, there is an ever increasing spread between the nap-of-the moon and nominal curves. This spread is due to the 100 fps cruise velocity limit used in the nap-of-the moon analysis. The analytical data indicates generally lower ΔV requirements than the simulator results, particularly at the 1600 ft range. However, the bulk of the spread in simulator data is attributable to pilot-to-pilot variations; the analytical data is in best agreement with Pilot A, where data is consistently at the lower end of the ΔV spread.

Figure 10.11 shows the flight range capability as a function of payload for the analytically derived ΔV data.

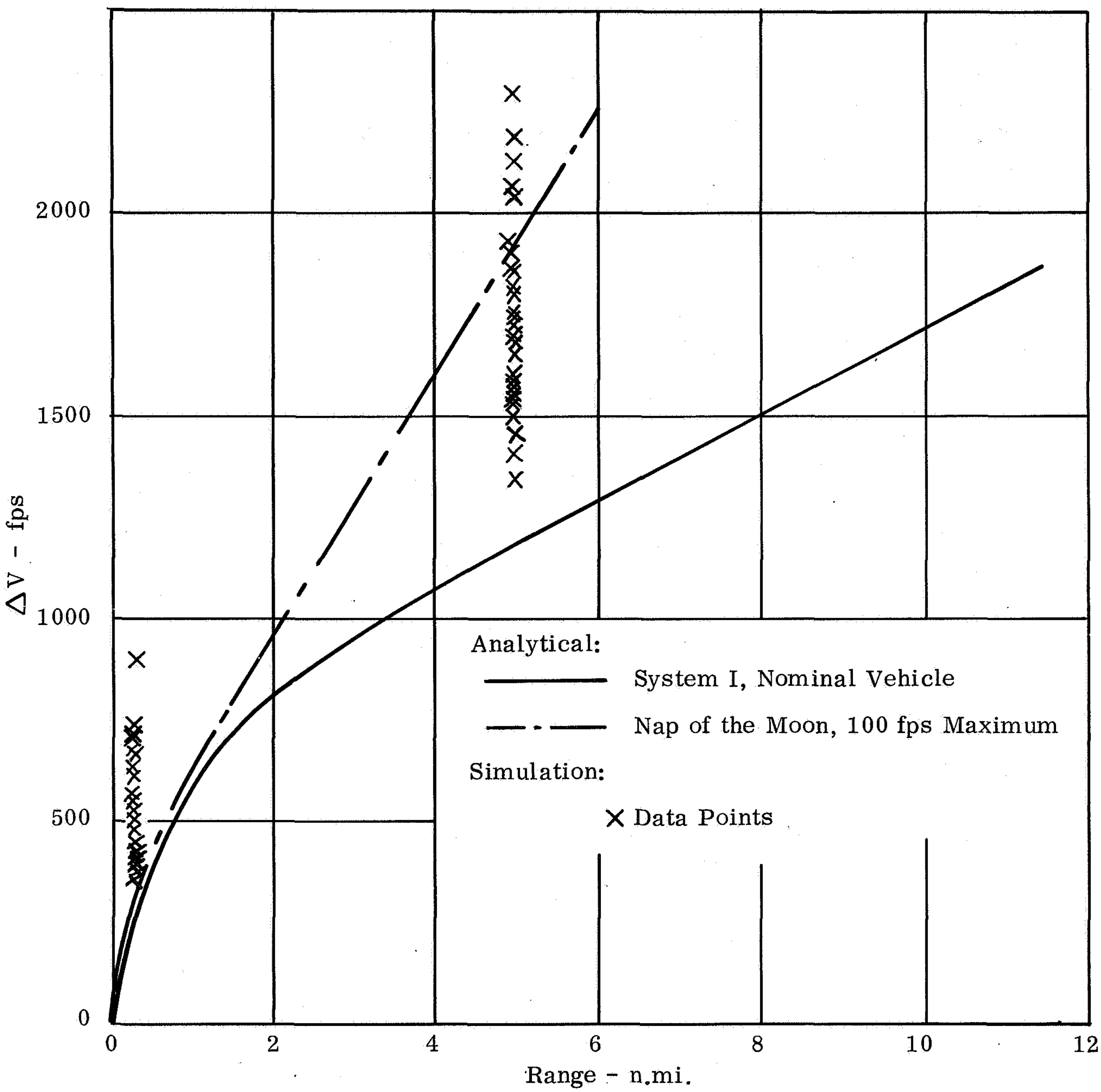


Figure 10.10. A Comparison of Analytical and Simulation Obtained Performance

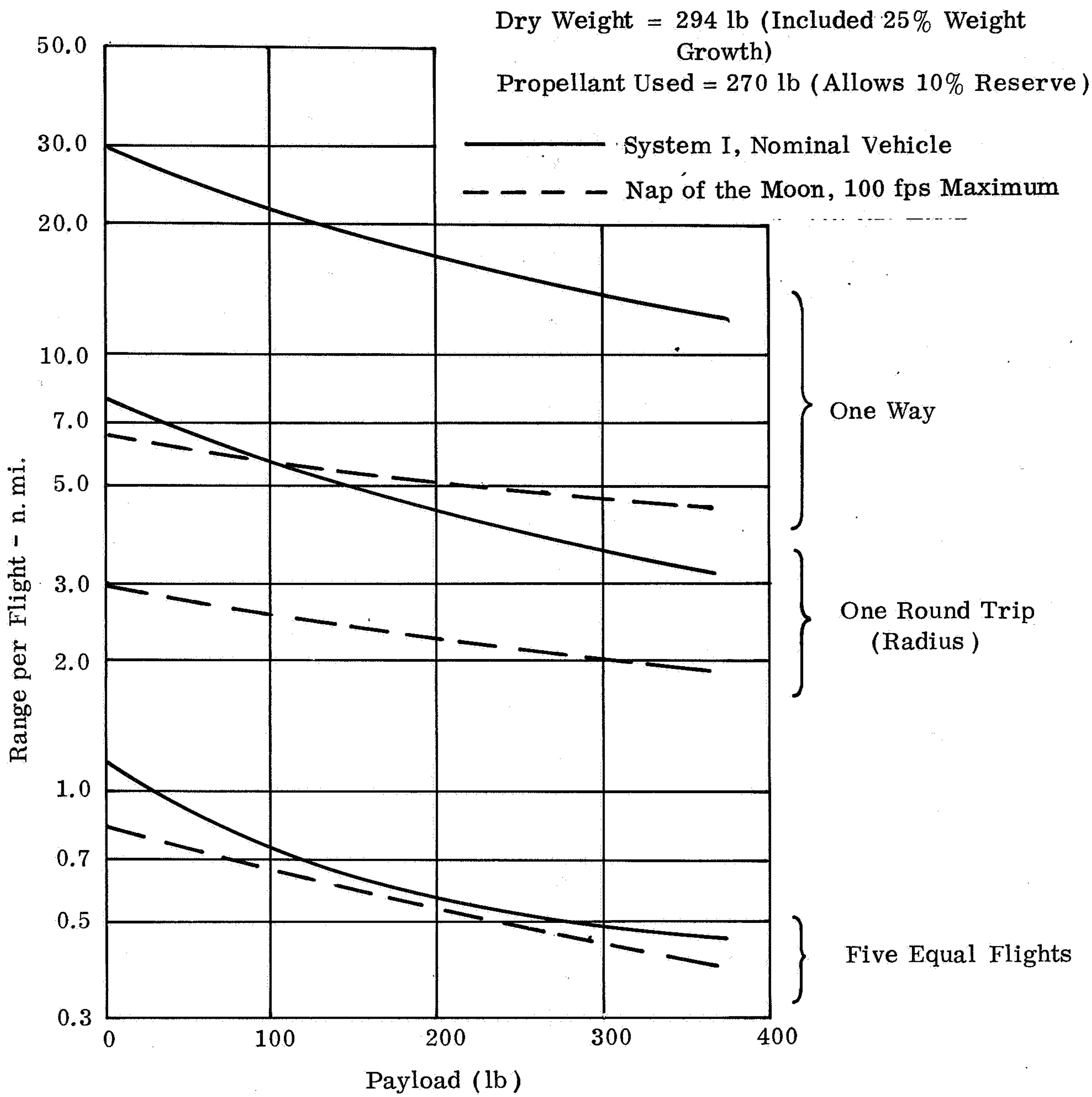


Figure 10.11. Vehicle Performance

11.0 LUNAR MODULE INTEGRATION

The integration of the lunar flyer with the LM requires investigation in the areas of:

1. Stowage space volume and shape
2. System installation and deployment technique
3. Refueling system
4. LM cg effects
5. LM RCS System/Deck Extension Effects

Maximum use was made of information developed by Grumman in its LM Utilization Studies and reported in References 28, 29 and 24.

11.1 STOWAGE SPACE VOLUME AND SHAPE

The information presented in References 28 and 30 has been used as a guide in defining the LM/Flyer interfaces. These documents provided suggested stowage locations, available envelopes, and hard points as well as post landing attitude conditions. The suggested stowage location was in the descent stage quadrants I and IV between X stations 133.6 and 193.5.

During the conceptual design and comparison phase of the study, one of the evaluation figures of merit was the area of LM thermal shielding required to protect the various flying vehicle concepts. The area of thermal shielding (shown in Table 2.3) was obtained for each concept by establishing stowage and shielding layouts. It was found that all of the early concepts could be stowed with varying degrees of flyer folding or disassembly and varying thermal shielding area.

Another early investigation was conducted to determine the feasibility of stowing two flyers in one quadrant. Figure 11.1 shows the nesting arrangement used to accomplish this for configuration 8.2 (described in Section 2.0). While stowage of two flyers in one quadrant is possible, it is not recommended due to the difficulty of deployment and complexity of structural tie in of the vehicles to the Lunar Module.

11.2 SYSTEM INSTALLATION AND DEPLOYMENT TECHNIQUE

The selected configuration is shown installed in the Lunar Module in Figure 11.2. The vehicle, with landing gear legs folded and secured, is stowed erect in the quadrant. Two latching fittings, mounted on the vehicle platform, engage mating fittings on the lower support beam attached to LM. These fittings react vertical and lateral loads. The top of the vehicle is supported by a fitting in the vehicle top deck that is pinned to a mating fitting on the upper beam assembly. This upper fitting reacts lateral loads.

The refueling hoses are stowed in Quandrant I and the helium tank assemblies in Quandrant IV.

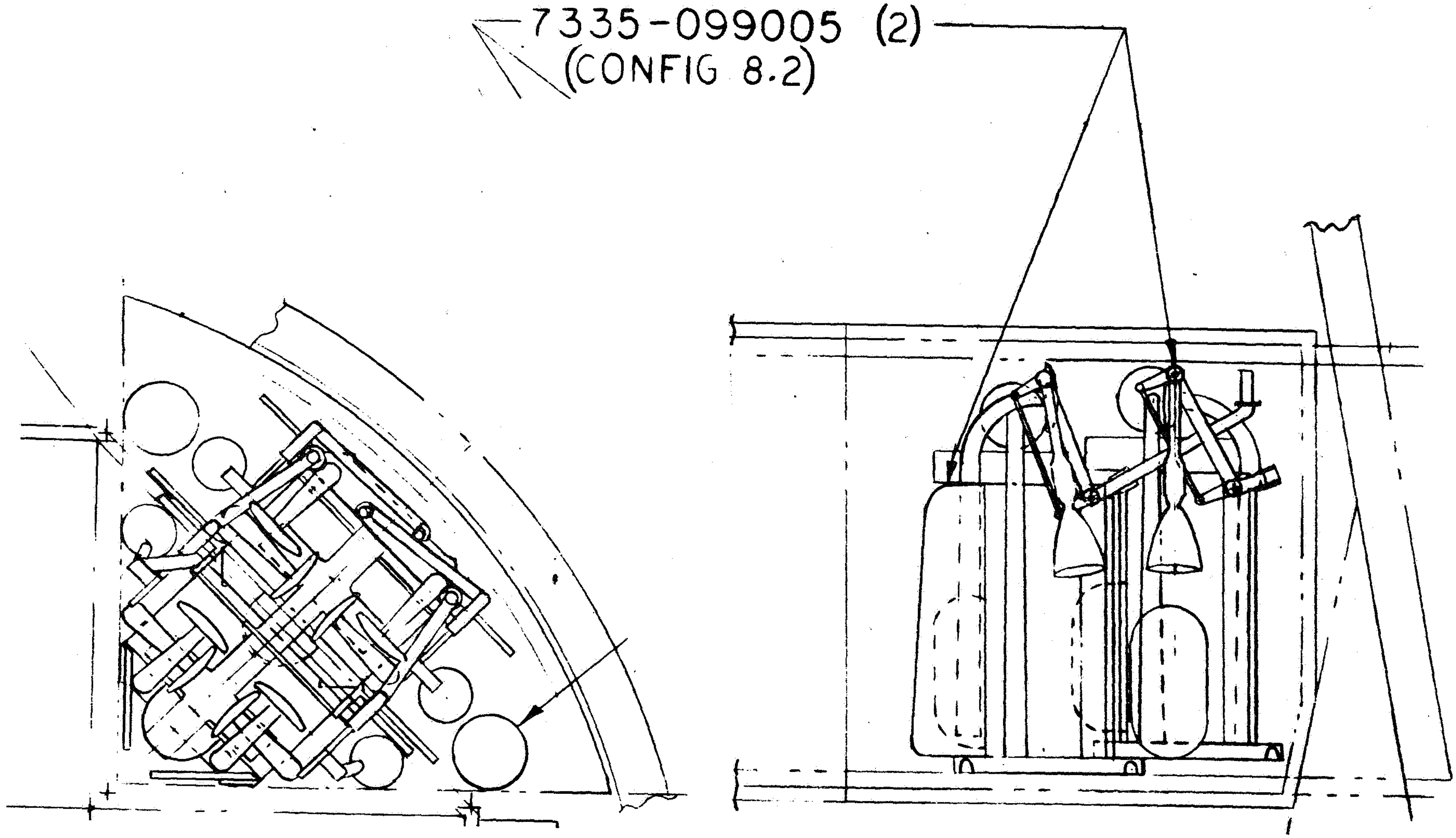
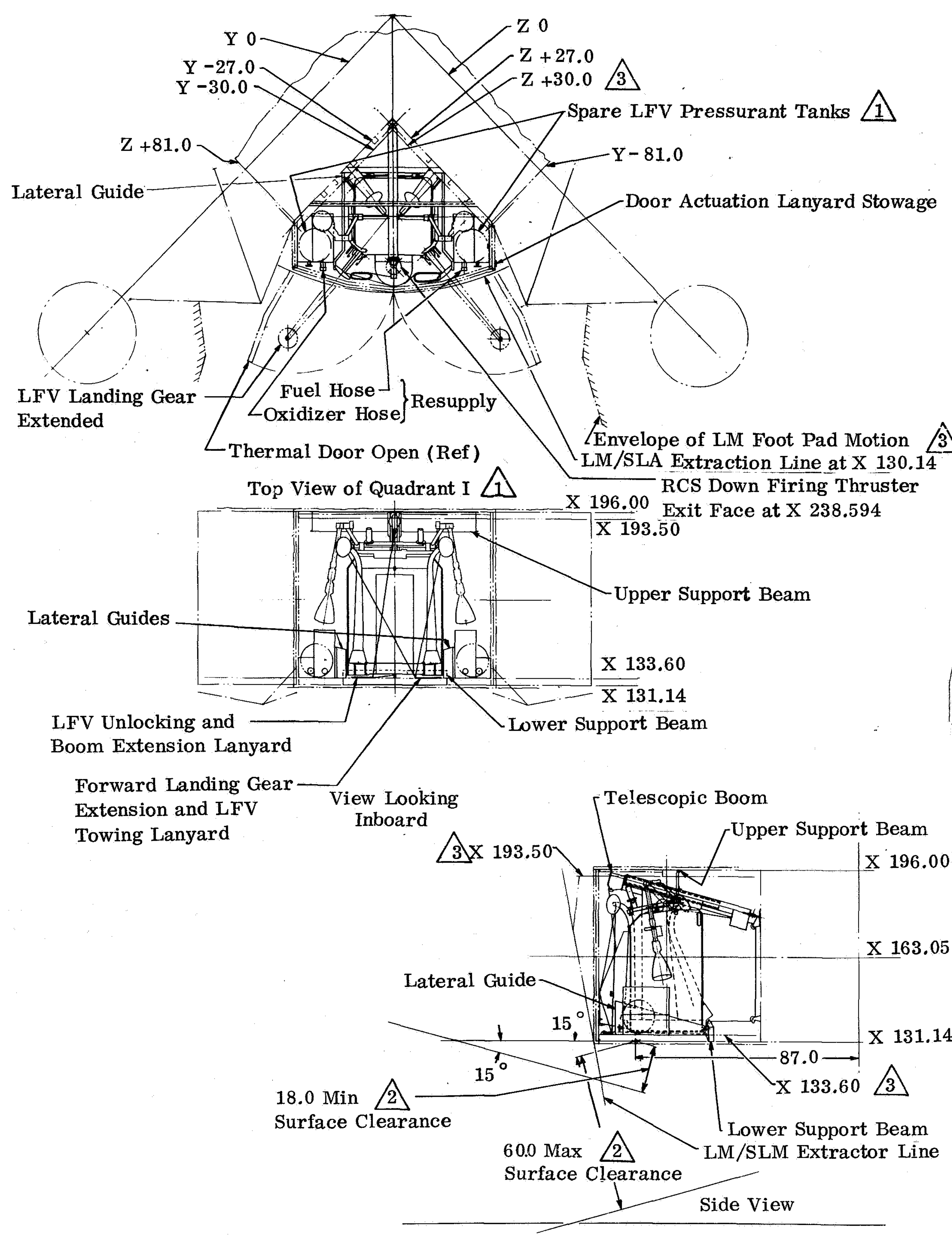
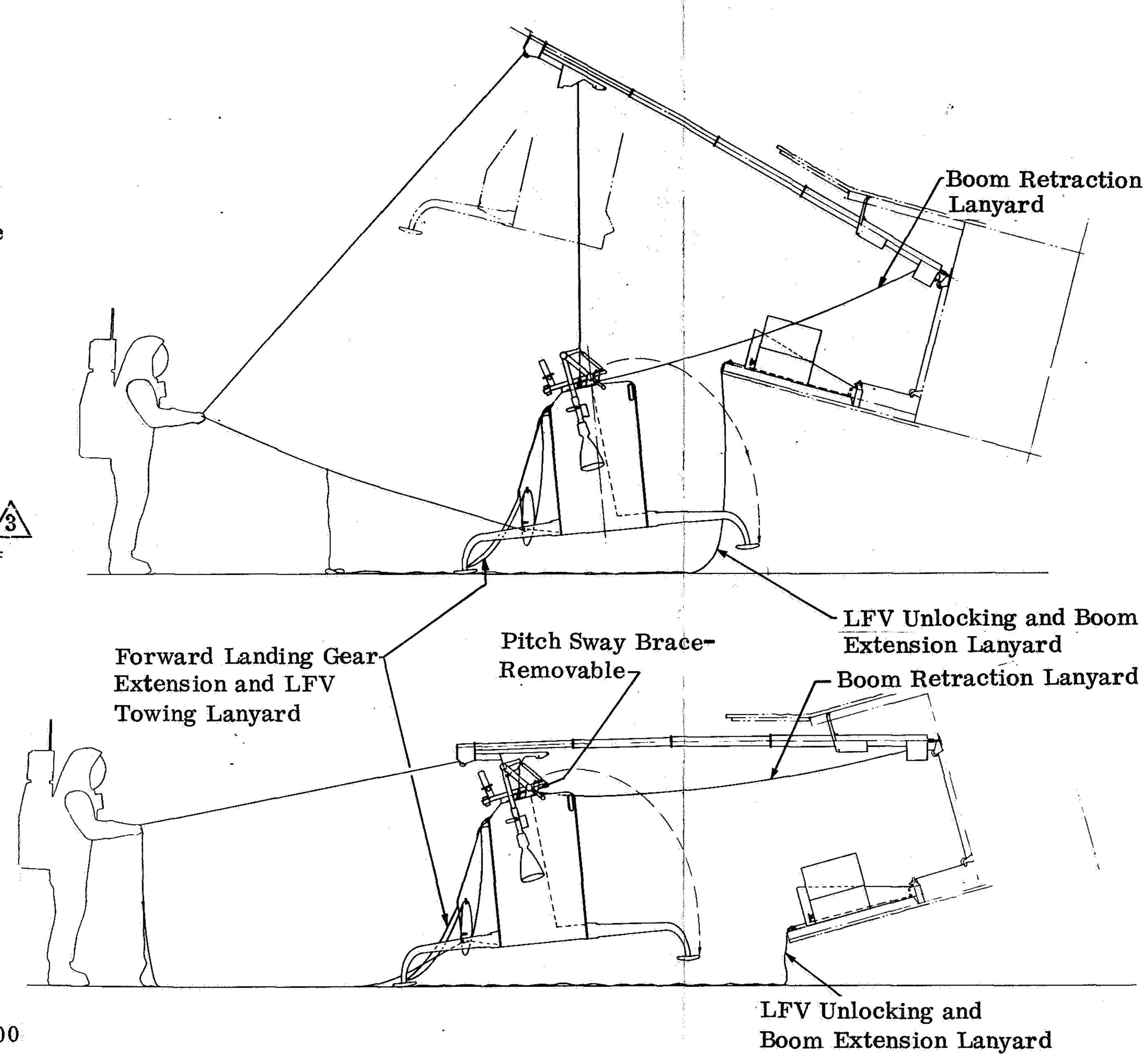


Figure 11.1 Nesting Arrangement





NOTES:

1. Installation in Quadrant IV same as in Quadrant I except Pressurant Tanks (2) Stowed in Place of Fuel and Oxidizer Resupply Hoses
2. Per Dwg LID 360-22812 Grumman Aircraft Engineering Corporation
3. Interface Definition per Report No. ARP 325-7B Grumman Aircraft Engineering Corporation

Figure 11.2. OMLFV Stowage and Deployment

Other LM modifications include the installation of flyer support beams, deployment system (consisting of an extendible boom and lanyards) and extension of the thermal shields of Quadrants I and IV. (See summary in Table 11.1) The flyer support beams and the root of the boom are attached to fittings added to existing LM structure. Additional boom support is provided by the upper vehicle support beam. The boom has four telescoping sections that extend the flyer out of the quadrant to a position from which the vehicle can be lowered to the surface without interfering with LM structure.

TABLE 11.1

LM MODIFICATIONS WEIGHT SUMMARY

Item	Weight lbm One Flyer System	Weight lbm Two Flyer System
Refueling System	30.0	30.0
Flyer Support Beams and Attachments	57.0	114.0
Deployment System	27.1	55.2
Additional Thermal Shielding*	11.0	22.0
Total	125.5	221.2

* Based on 0.15 lb/ft^2.

The boom is extended and retracted by an extendable element within the deployment boom. This extendable element is mechanically operated by a cable drum, cables and an endless lanyard.

The original envelope of the Descent Stage Quadrants I and IV must be extended to protect the flyers. The front panels are hinged to provide access to the flyers and the helium tank assemblies.

The deployment sequence and timing is discussed in Section 9.0.

11.3 REFUELING SYSTEM

The modifications to the LM necessary to accomplish transfer of propellant from the LM descent stage tanks to the flyers are discussed in Reference 24. Several alternative tapping points were investigated prior to the selection of the

pressure port tap system which incorporates liquid level sensors and solenoid actuated shut-off valves to maximize propellant transferred to the flyer.

This permits extraction of all but 23 to 78 pounds of the available propellant depending upon the direction of the vehicle worst case tilt angle of 15 deg. More can be extracted at lesser tilt angles. Additional lines are installed on the LM from the tapping point to Quadrant I where the flexible refueling lines are attached.

11.4 LM CG EFFECTS

The effect of the flyer and associated equipment weight on the LM center of gravity is illustrated in Figure 11.3. This plot, (Reference 28), shows the distribution of 750 pounds of payload between Quadrants I, II and IV. The plot indicates the combinations which fall within boundaries of one and two degrees of descent engine tilt to maintain vehicle trim. The symbol indicates a representative combination of 282.5 pounds in Quadrant I and 289.5 pounds in Quadrant IV associated with 178 pounds in Quadrant II and shows that the cg falls well within the allowable 2 deg boundary. The weight of equipment indicated for Quadrants I and IV is representative of that for two flyers and associated refueling equipment, deployment equipment and helium bottles. It should be noted that this plot is applicable for a LM payload of 750 pounds. Similar plots can be constructed for other payload weights. No significant problems are anticipated regarding payload distribution, which includes flying vehicle systems, since the flying system equipment can be distributed among quadrants.

11.5 LM RCS SYSTEM/DECK EXTENSION EFFECTS

Stowage of the flyers in Quadrants I and IV of the lunar module descent stage will require extensions of the descent stage upper deck beyond the existing LM structure. These extensions will be subject to RCS exhaust plume impingement; resulting in heating of the exposed surface and a reduction of RCS engine effectiveness due to pressure forces. Proper design of the thermal shielding surrounding the flyer systems, as discussed in Section 7.0 Thermal Analysis, will account for the heating effects.

An analysis conducted by Grumman (Reference 29), indicates the loss of RCS engine effectiveness. The penalty associated with extensions on two quadrants of $\approx$ 10 inches beyond the existing LM quadrant faces, is a 14 pound increase in the amount of RCS propellant used for the total descent. For the flyer stowage arrangement currently designed (illustrated in Figure 11.2) a 26 inch extension is required. This will result in an RCS propellant increase estimated not to exceed 40 pounds. Additional analysis can establish the exact penalty, and whether existing RCS propellant reserves are adequate. The referenced document pointed out that if plume deflectors were employed on the lunar module (a modification under consideration at that time) the propellant penalty for such extensions would be negligible.

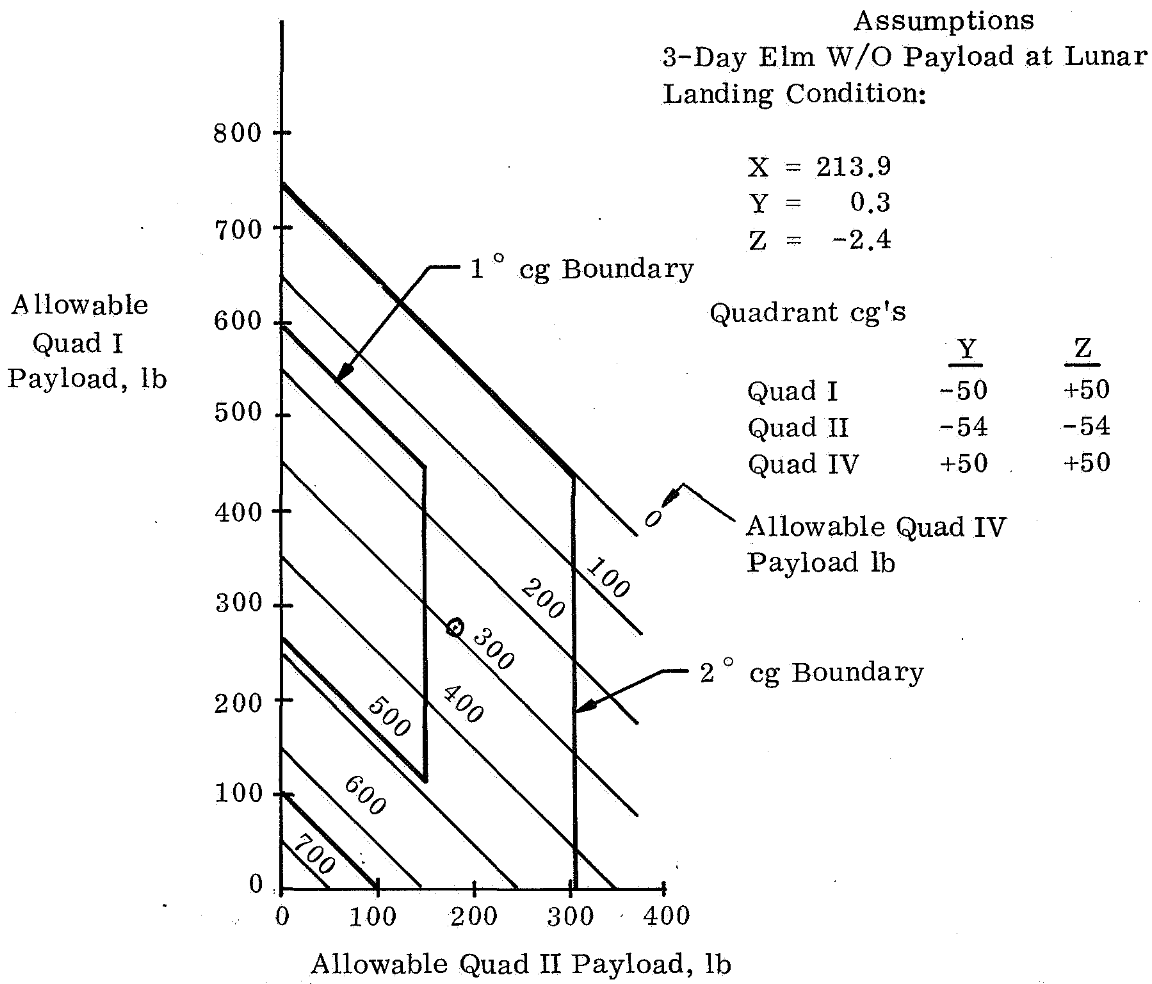


Figure 11.3. Payload Distribution

12.0 FLIGHT SUIT STUDIES

12.1 INTRODUCTION

Since the standard Apollo pressure suit is worn by the astronaut while servicing and flying the OMLFV, studies and mockup tests were conducted to ensure compatibility between the vehicle and the pressure suit. These investigations included suited subject/mockup tests and propellant/suit chemical compatibility tests discussed in this section thermal effects analysis discussed in Section 7.0 and suit/flight control simulation tests discussed in Section 6.0.

Previous lunar flyer studies have shown that only preliminary evaluation of vehicle concepts can be made, using pressure suit dimensional and mobility data. The data required for realistic vehicle preliminary design must be obtained by man/suit testing with suitable mockups. Since mockup testing was accomplished in the MFS program (Figure 12.1 and 12.2) for seated configurations, in the current study, flight/suit mockup studies were conducted for stand-up configurations to investigate the man/suit/vehicle interface. These studies provided a basis for evaluating the compatibility of candidate vehicle configurations with the capabilities and limitations of a pressure suited astronaut.

The objectives of the flight suit mockup tests were: (1) develop design-data to ensure operational compatibility between the suited astronaut and vehicle design; (2) provide information to assist concept selection; and (3) provide a basis for task/time-line data.

To assist in accomplishing these objectives, an Apollo A6L flight-suit and a backpack PLSS were delivered by NASA, and wood mockups (Figure 12.3) which could be modified and reconfigured to investigate various ingress/egress approaches and different controller configurations were constructed.

The approach to performing the tests, conducted with the suit pressurized, involved an initial listing of tasks to be performed, and the mockup modifications required, for each day's activity. Scheduled tasks were reviewed with the subjects prior to donning the suit and as necessary, dry runs in an unpressurized suit were conducted.

During those tests where a pressurized suit was used, photographs were taken and measurements recorded. The technique used involved having the subject pause in the execution of a task while photographs were taken using a yardstick to indicate the required dimension. In addition to these measurements, instructions to the participating subjects and their comments were tape recorded. Subsequent review of the notes, photos and tapes for each day's mockup activity were used for updating design.

Figure 12.1. MFS Suit/Mockup Tests - Termination of Sitting Procedure

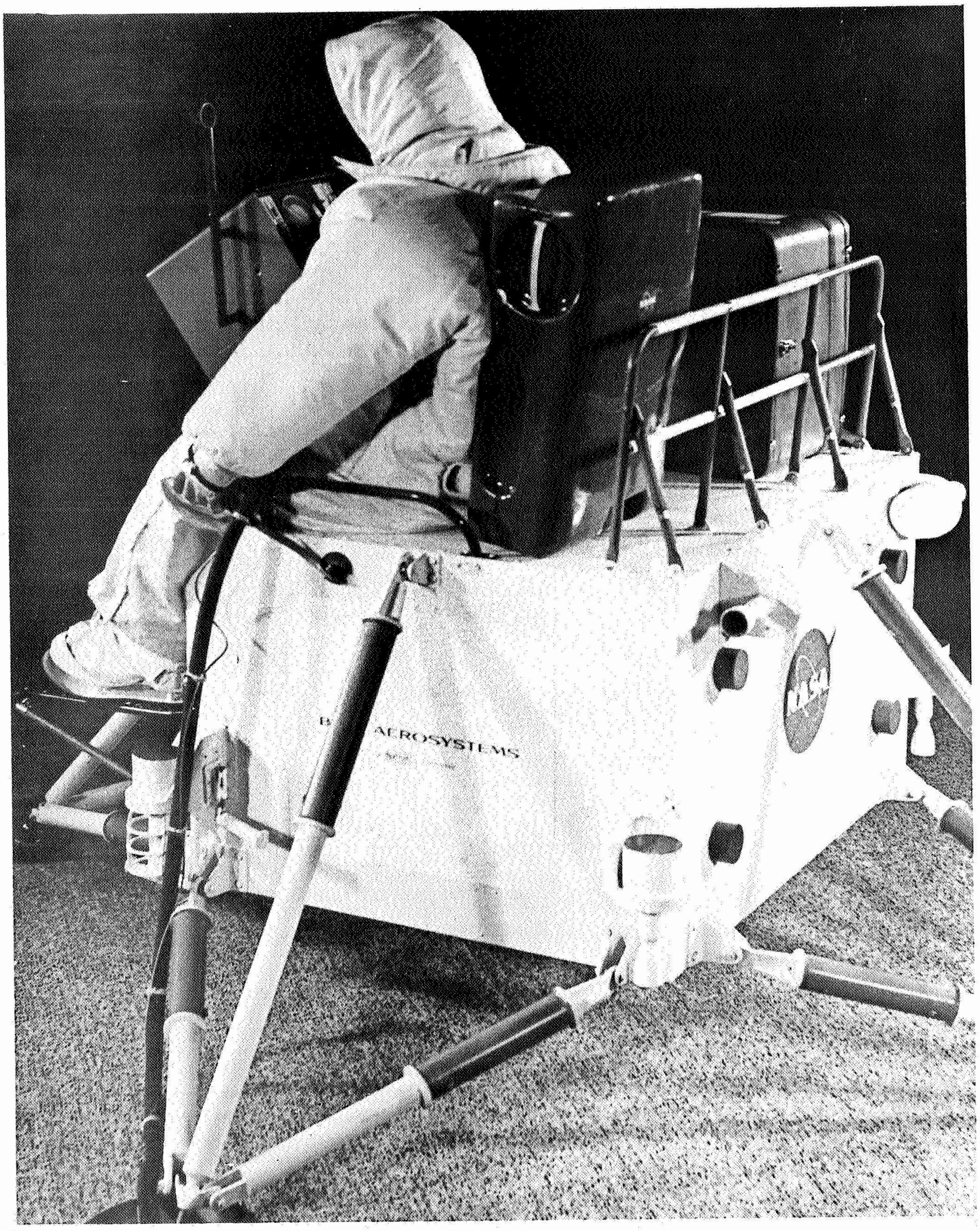


Figure 12.2. MFS Suit/Mockup Tests - Rear View

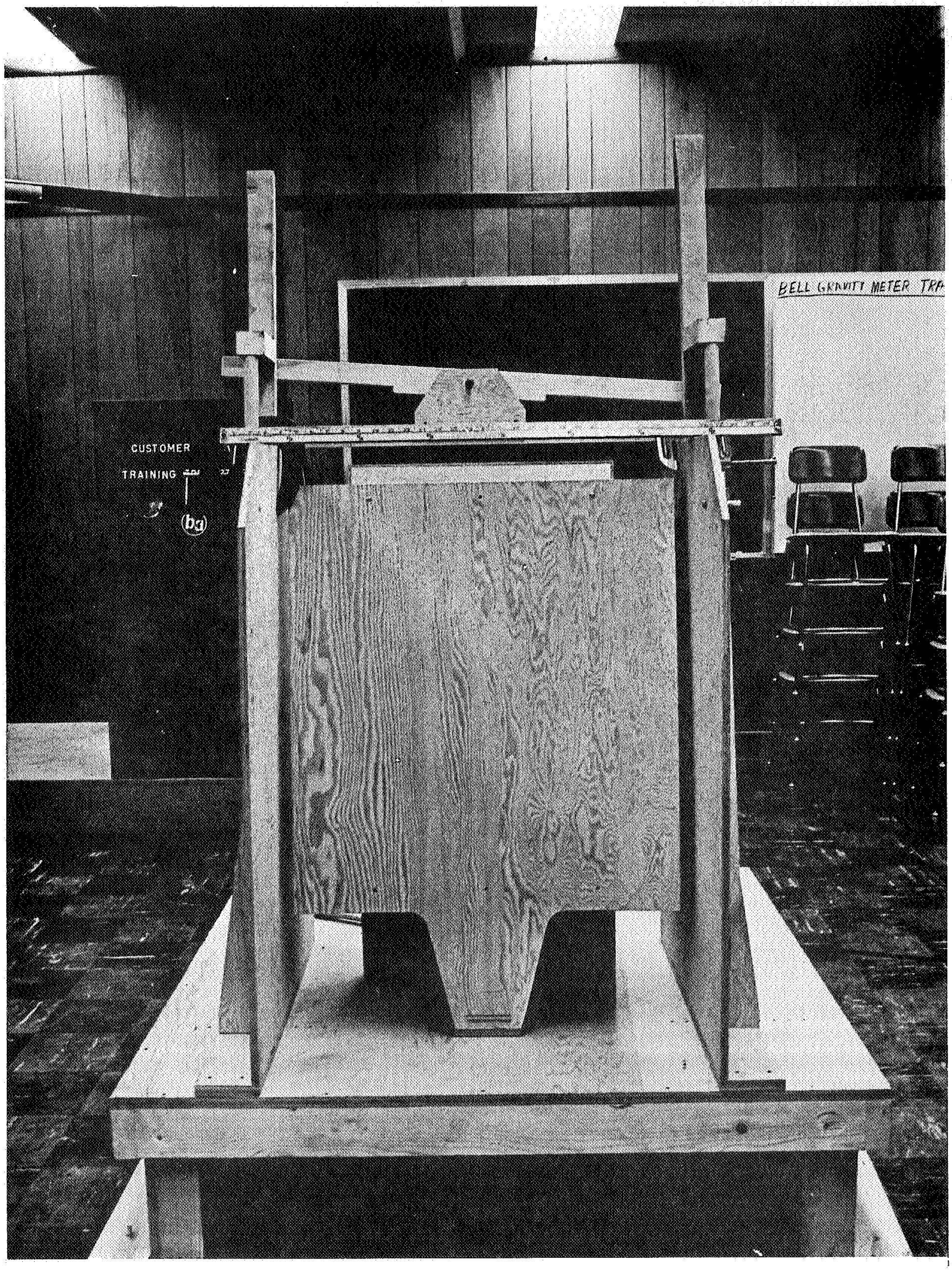


Figure 12.3. OMLFV Mockup

12.2 INGRESS/EGRESS

The first objective of the mockup tests was to obtain space/volume data for various ingress/egress approaches. Approaches considered were: forward or front entry, rear entry, and side entry.

Initially, the mockup platform was set up 13 inches high with 40 inch sides. The subject was required to mount and walk repeatedly across the platform while the sides were adjusted to determine the minimum width required for a simple walk-through without side interference. This distance was established as being 31 inches with the arms in a normal position at the sides (Ref Figure 12.4). If the subject held his arms in or together while walking through, 29 inches was acceptable.

The ability of a suited operator to enter and execute a 90 degree turn was next investigated. The subject was required to mount the platform, step to the middle and execute a turn. The first efforts resulted in interference between the PLSS and the rear wall during the turn. However with continued execution of the task, this interference was successfully controlled, and the subject could turn adequately within the 31 inch dimension. In part, this was accomplished by adopting a position closer to the front side during the turn.

Also established was the platform space required for a 90 degree turn without PLSS interference. The rear side was replaced with a 30 inch high side which did not interfere with the PLSS during a turn. For this configuration, the subject was able to mount and execute a turn within 21 inches (Ref Figure 12.5).

The platform dimensions for a side-entry vehicle were established as part of the initial ingress/egress tests. Thirty-one inches was identified as the minimum width for ingressing and executing a 90 degree turn within 40 inch sides (Ref Figure 12.6). It was found that if the rear side is lower than the PLSS, the width requirements reduced to 21 inches.

It was also discovered that a visual reference to aid location of the front side facilitated entry, since the subject cannot readily see the side in turning or facing forward because of the location of the RCU on the front of the suit.

The minimum platform-space required for rear-entry vehicles was established as follows: the sides were positioned to the edge of the platform 31 inches apart, with a moveable partition located between (Ref Figure 12.3). Handholds were appropriately located and the subject required to mount and stand at the edge of the platform. The moveable partition (with cutouts for feet) was then moved rearward, in close proximity to the front of the suit.

A platform dimension of 9.5 inches, measured from the bottom center of the partition to the platform edge, was established as the minimum required to allow the subject to mount and dismount without interference from the front partition of the vehicle.

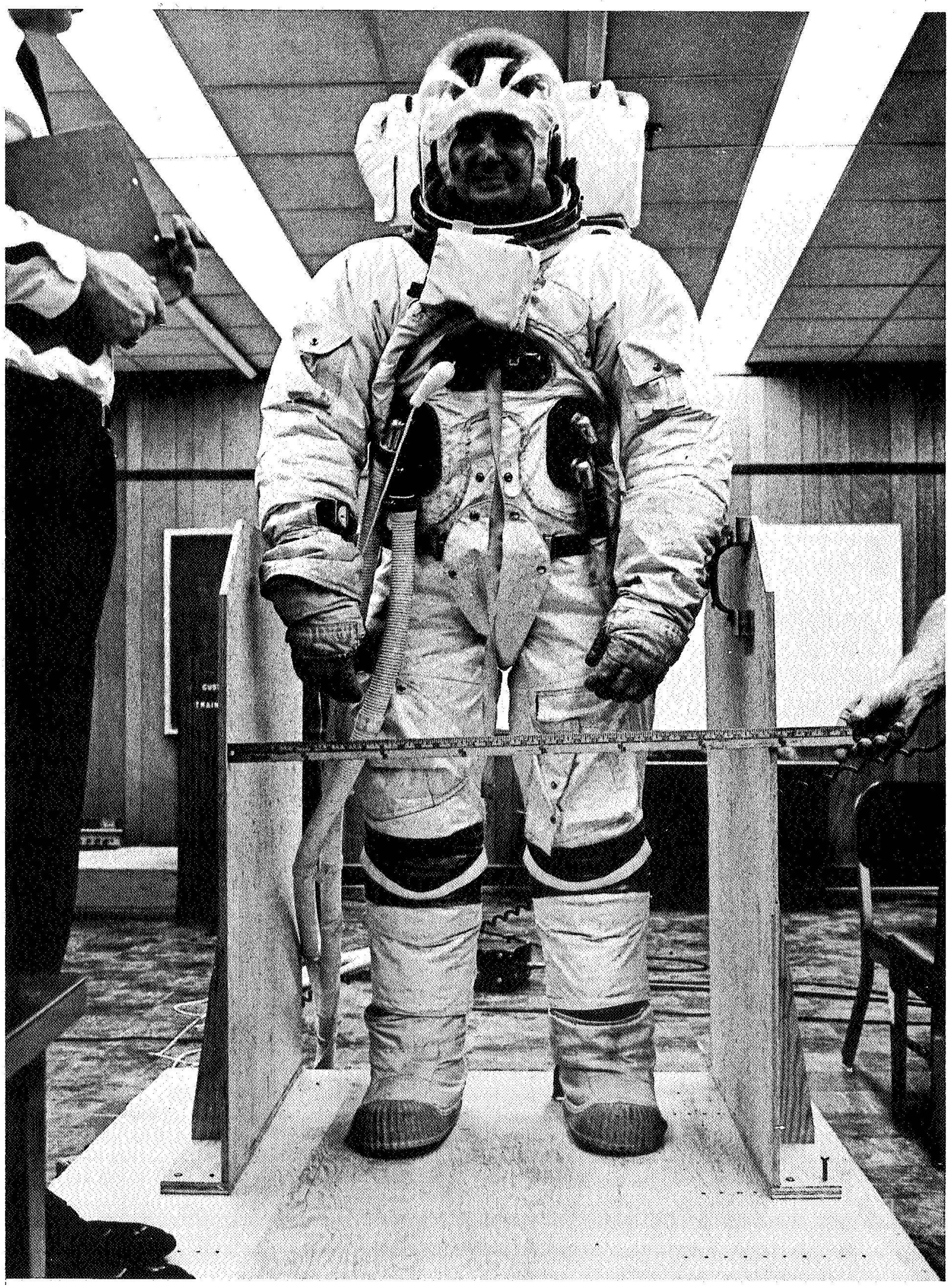


Figure 12.4. Minimum Walk-Through Distance

Figure 12.5. Turn Execution Distance

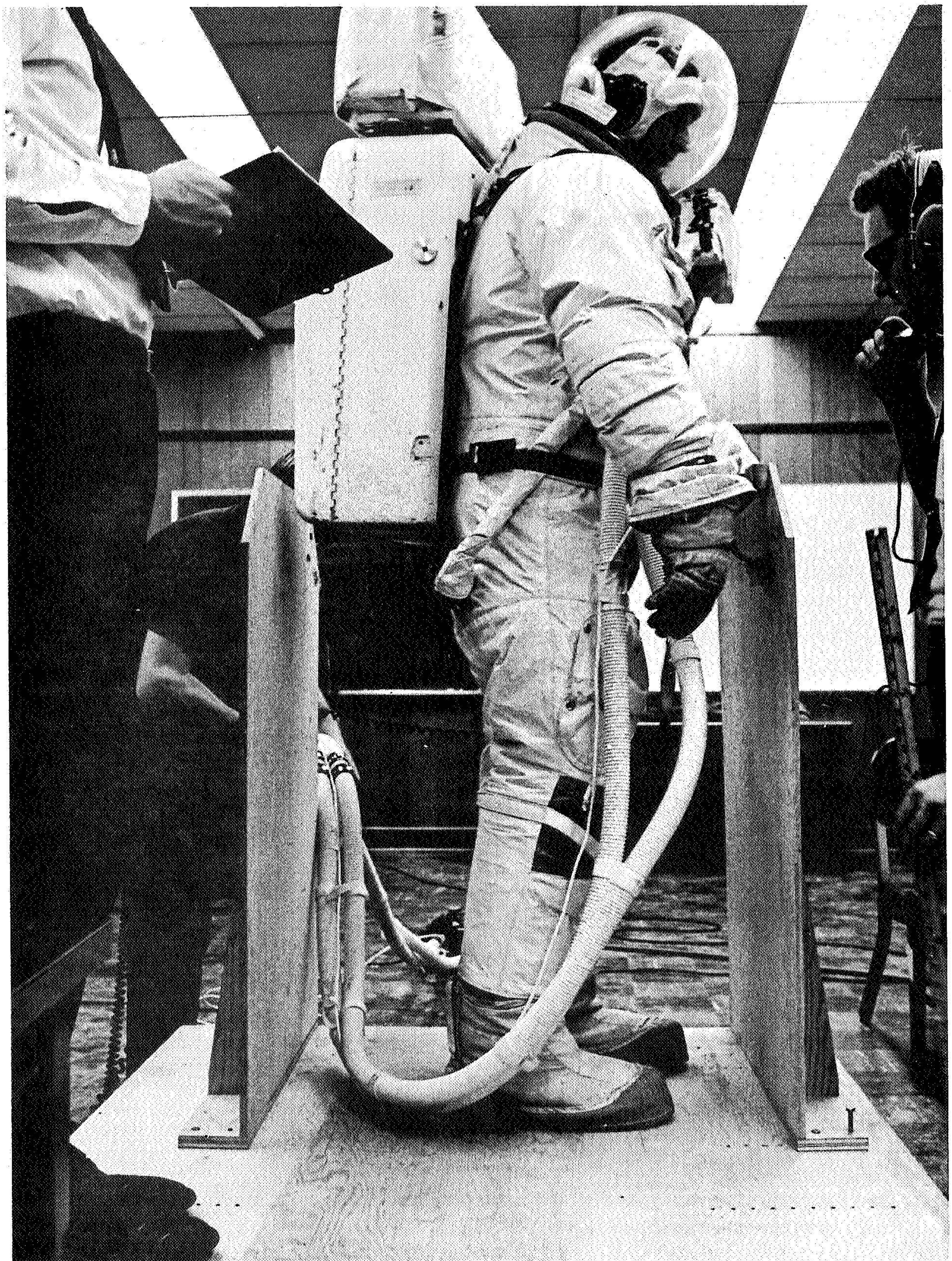


Figure 12.6. Side Entry Dimensions

Thus a platform measuring 31 inches x 9.5 inches and 13 inches high was shown adequate for entry and exit. Also noted, was the importance of the effect of arm-position; for 31 inches represents the minimum distance required for arms held within the sides of the vehicle. Less room was required if the subject keeps his arms above the sides. Therefore, an evaluation was conducted with the sides set at 28 inches, the minimum width required for mounting the propellant tanks. This reduction of clearance did not impede the subject, and the test was repeated using 14.5 inch and 17 inch platform heights. At both heights the subject initially had difficulty positioning his left foot on the platform (Ref Figure 12.7). This was due in part to the width constriction which limited body swing, and also to the height increase. Difficulty was extreme at 17 inches. However, ingress improved somewhat with practice, and for both heights, egress was not difficult.

In executing rear entry, subjects used the sides of the vehicle for handholds. External rear-mounted handles or cutouts in the sides were both found convenient for ingress and egress, and became more critical with increases in platform height. **For the higher platforms, handholds for the subject to pull on were found necessary. In evaluating handholds it was found that different subjects had different preferences for handhold location. A good compromise location was found to be above the midpoint of the sides.**

An investigation of required space-dimensions for front-entry vehicles was next conducted. This configuration requires that the astronaut position himself on the front of the vehicle facing forward (Ref Figure 12.8). To accomplish this, it is necessary for the astronaut either to step up facing backwards or to step up facing the vehicle and turn 180°. In order to conduct the necessary evaluations, the mockup was modified by placing the sides to the edge of the 13 inch platform and locating a moveable partition, simulating the contour of the rear wall, between them. The two approaches to entering this configuration were evaluated as a function of platform size.

a. The Back-On Approach

The subject was required to walk to the platform, execute a 180 degree turn and attempt to mount backwards. This exercise was tried with and without sides. The following observations were made:

(1) A subject can mount a 13 inch high platform with no sides, by swinging his body laterally to provide momentum for the step-up. The addition of sides 31 inches apart caused extreme difficulty in effecting entry.

(2) A variety of handhold locations to assist backward entry did not yield significant improvement.

(3) Should an off-balance situation develop during back-on, the effect of the cg insensitivity inherent in suited operations is more serious, since falling is more probable. The problem is somewhat alleviated in front entry vehicles since cg sensitivity and control is improved by the freedom to lean forward.

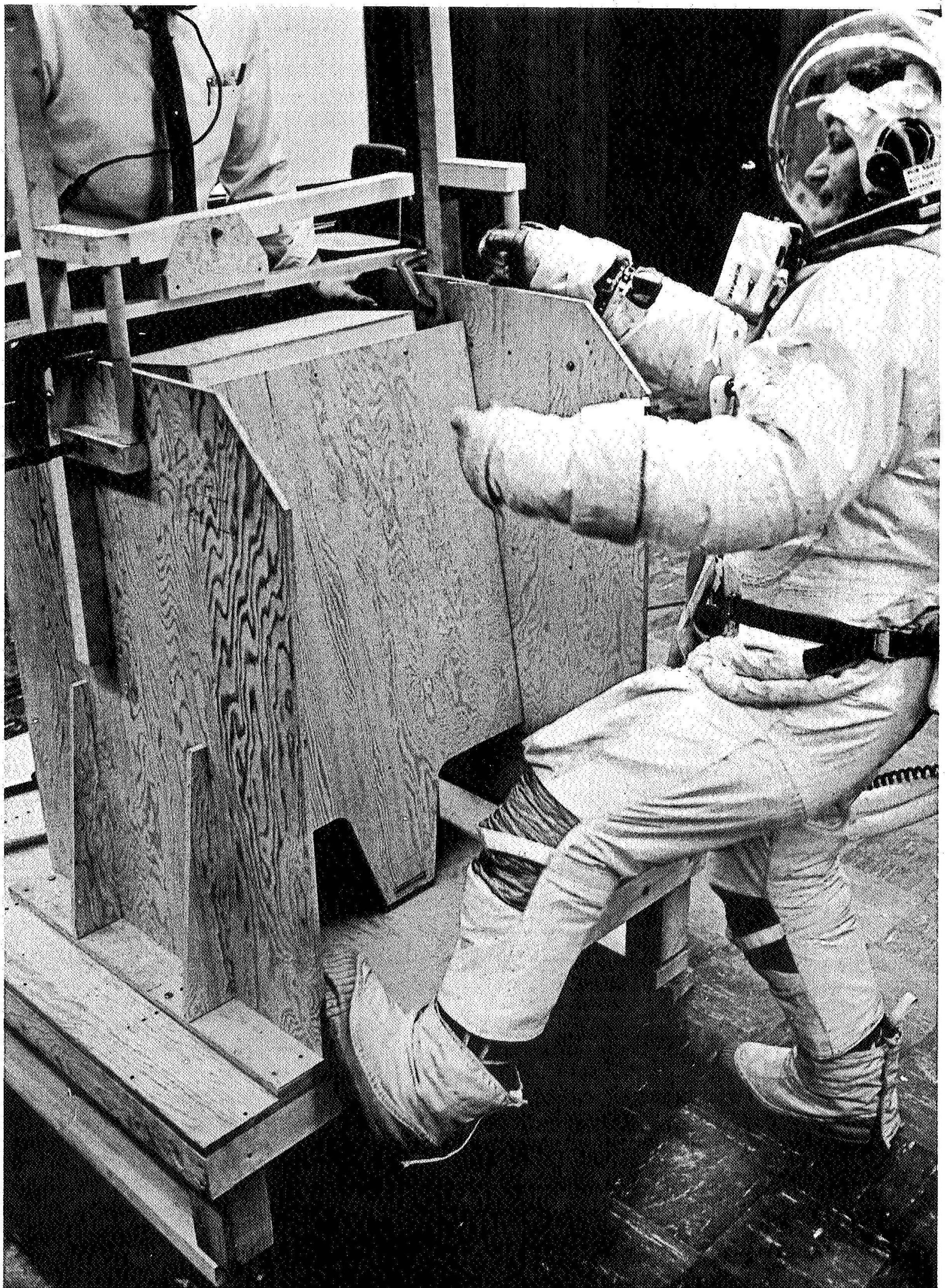


Figure 12.7. Foot Positioning for Step-Up

Figure 12.8. Front Crew Station

The effect of twin-handled controllers was investigated by using parallel side bars 31 inches apart and 49 inches high, extending in front of the mockup to simulate the pivot. The bars, which extended under the subject's arms, were found difficult to grasp and impeded entry.

It was concluded that backing on is not an acceptable entry technique.

b. The Walk-On Approach

The walk-on approach involves a walk to the platform and a 45 degree turn to the left, executed by placing the right foot on the platform and stepping up while turning into the forward position (Ref Figure 12.9). The 31 inch platform width again proved to be the minimum for acceptable PLSS clearance. The minimum platform depth required to execute this maneuver was found to be 14 inches.

During the ingress/egress tests it was observed that PLSS height varies on the same subject as well as among subjects. The distance from the bottom of the PLSS to the floor varied among subjects, after pressurization by as much as 3 inches. For the same subject a two inch variance was found between pressurizations.

As a result of these investigations, the following design recommendations were made:

(1) Of the three approaches evaluated (front, rear, and side entry), the rear entry vehicle proved best in terms of ease of entry while the front entry vehicle was the least satisfactory. A rear entry design is therefore recommended.

(2) Minimum acceptable platform dimensions for a rear-entry vehicle are 31 inches x 9.5 inches assuming arms below the sides, and 28 inches x 9.5 inches with arms above the sides.

(3) A ridge on the edge of the platform is recommended to provide positioning information to the astronaut. This facilitates positioning the feet relative to the platform edge.

(4) PLSS support, if required, should be adjustable, since PLSS height was found to vary on the same as well as among subjects.

(5) Handholds on the sides, or slightly above the top of the sides in the 34 to 43 inch range, were found useful to all subjects and are recommended.

12.3 CONTROLLERS

The second objective of the mockup tests was to evaluate and obtain design-data for various controller configurations. The following configurations were mocked-up and evaluated:

Figure 12.9. Stepping and Turning

a. Twin handled
b. Apollo
c. Joystick
d. Side bars
e. Throttle types and locations

The twin-handled controller configuration evaluated in the mockup tests (Ref Figure 12.10) would operate in the obvious senses for both pitch and roll; yaw and throttle control would be achieved by twisting the right and left handles, respectively.

The recommended distance between controllers requiring twist motions is 19 to 21 inches. Beyond this range wrist movements were found increasingly difficult due to suit constraints.

The recommended vertical height for this configuration is 39 to 45 inches from the flyer platform to the base of the handles. If 5 inch handles are included, this distance would be 44 to 50 inches.

The right hand 3-Axis Apollo type grip which was mocked-up and evaluated (Ref Figure 12.11) operates to give pitch, roll and yaw, by movement in the obvious sense.

It was found that subjects preferred to grip this type of controller from above rather than from the side for maximum comfort and freedom of movement.

Tests indicated the optimum location for such a controller to be approximately under the subjects right forearm. An envelope approximately 14 to 17 inches from the suit face and 9 to 14 inches off the suit center line defines approximately the acceptable area.

The joystick controller operates to give pitch, roll and yaw motion by operation in the required direction. Subjects found the most satisfactory handgrip to be atop the stick (see Figure 12.12) and experienced difficulty making yaw inputs due to the suit's limitation of wrist movement.

The recommended location for a joystick type controller is under the right hand with the arm roughly parallel to the floor. It was also found that the top of the stick should be within the astronaut's cone of vision, thereby allowing visual feedback of stick-position information.

The lateral side-bar controller concept (Ref Figure 12.13) was designed for use solely with kinesthetic vehicles; one side bar being twisted for throttle, the other for yaw.

Figure 12.10. Twin Handled Controller

Figure 12.11. Three Axis Controller

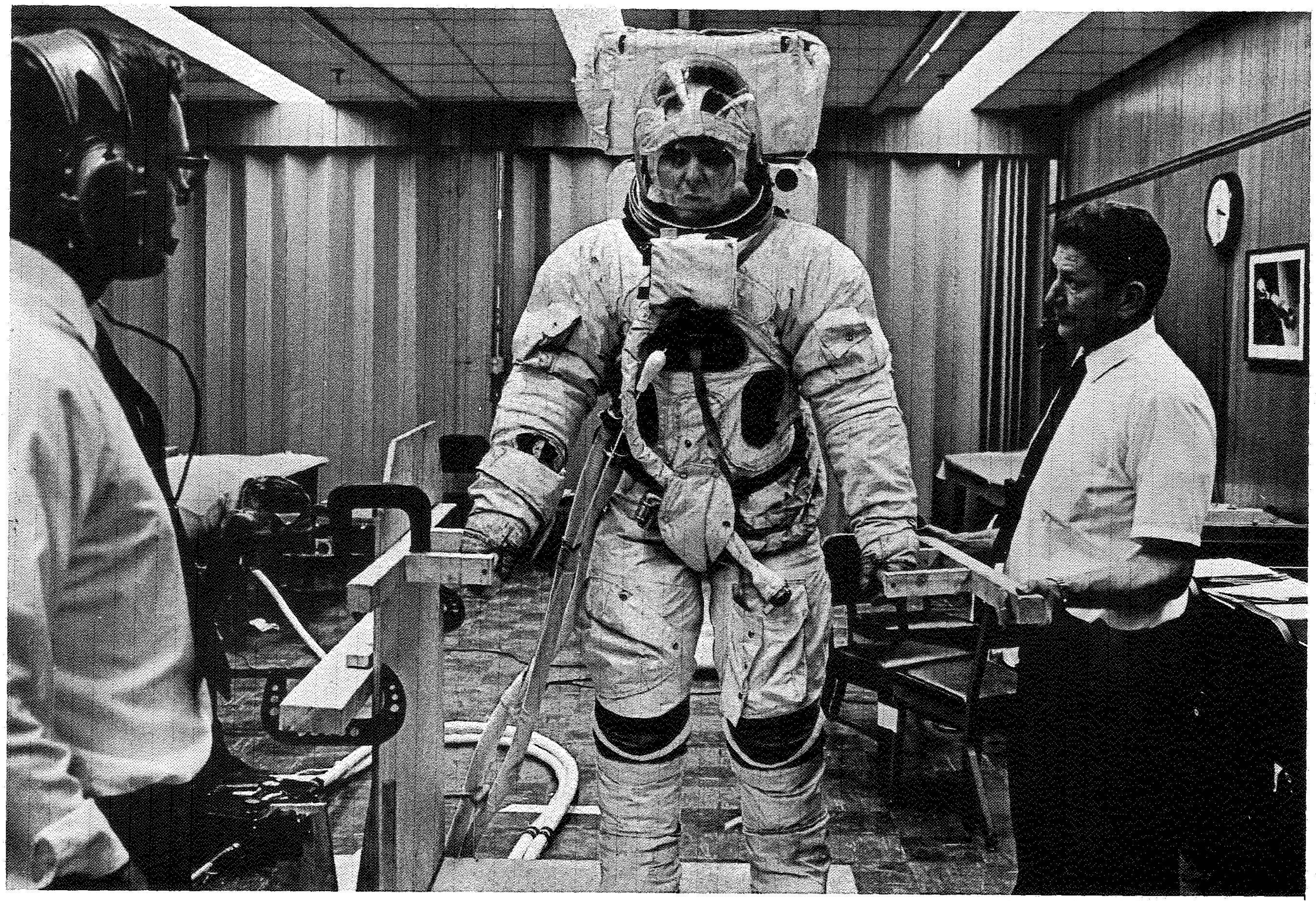


Figure 12.12. Joystick Controller

Figure 12.13. Side Bar Controller

It was found that these rotational movements tend to cause a cg displacement, and are also difficult to perform accurately in a flight-suit. If used, the optimal location would be approximately 32 inches high and 14.5 inches off-center.

An attempt was made to assess left hand throttle location independent of configuration. Whereas the twin handled and side bar controllers dictate the location of the throttle, more flexibility in location is available in the Apollo and joystick configurations.

A T-handled throttle requiring vertical and/or horizontal movements about a pivot was evaluated (Ref Figure 12.14) and it was found that, in fore and aft movement, the weight of the arm is supported by the throttle which tends to slide within the grip. Sensitive throttle movements were therefore difficult. For this reason a throttle operating in the vertical plane was preferred, and ease of grip found optimal with an L rather than T shaped handle.

Throttles requiring wrist movements were also examined in a variety of locations. Wrist movements within the suit were extremely difficult in all cases, and no acceptable location was found.

12.4 REACH ENVELOPES

In order to increase the information available on the capabilities of the suited operator in unloading equipment from the LM, the ability of suited subjects in retrieving objects from a 62 inch high shelf was investigated (see Figure 12.15). This height was selected for the LM since it is the maximum assumed to be required for unstowing operations.

With the subject standing as close as possible before a 62 inch high opening (4 inch from partition to suit face, Reference Figure 12.15) the maximum reach incursion into the opening was found to be 6 to 8 inches. Stepping back to a position 12 inches from the partition resulted in approximately the same reach envelope.

To simulate the loading of scientific gear, subjects were next required to remove a box, turn, and place it on the 13 inch high platform. It was found that the subject could reach to within 9 to 11 inches of the platform, or 21 to 24 inches of the ground, while grasping and lowering the box without knee bending.

12.5 VISIBILITY

The Line-of-sight (LOS) of the standing subject, over the Remote Control Unit (RCU) was established. The subject, in full lunar gear, was required to stand erect and sight over his RCU to the nearest visible point on the floor. For a subject with an eye-height of 65 inches, this point was 34 inches from his feet. If the subject was permitted to tilt his head, he could see a point 27 inches away. The LOS downward angle for eye movement alone was approximately 62.5 degrees from horizontal; for the tilted condition it was approximately 67.5 degrees.

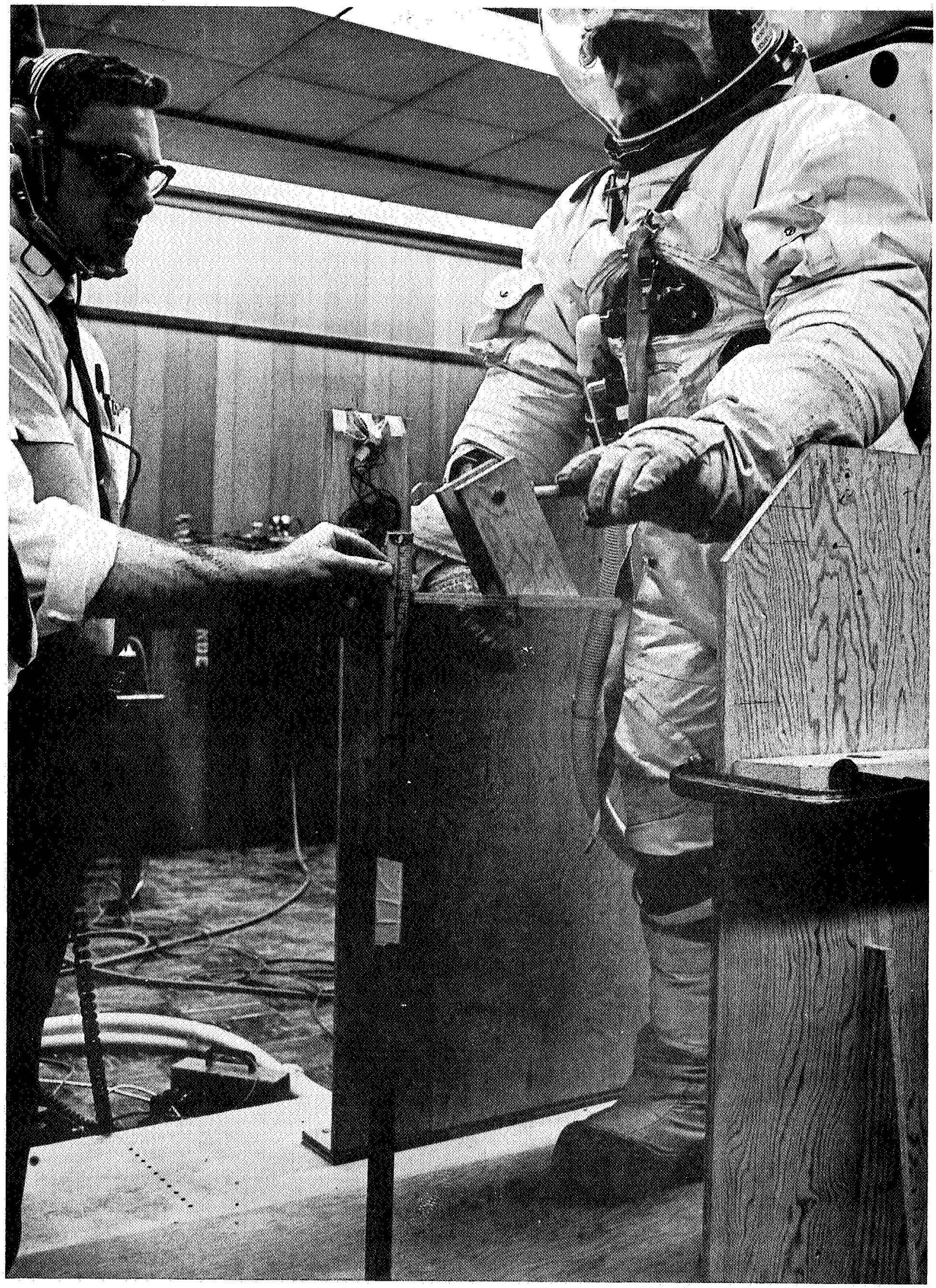


Figure 12.14. T-Handled Throttle

Figure 12.15. Reach Envelope

Additional tests sighting to a display panel were conducted. The display panel was located at various distances from the front of the subject. With the display in a plane 12 inches from the suit face and 47 inches from the floor it could be seen. At positions closer to the floor the view was blocked by the RCU. In a plane 16 inches from the suit face the corresponding dimension was 38 inches.

12.6 SUIT/PROPELLANT INVESTIGATIONS

Another suit/flyer interface area investigated was the effect of propellant upon suit materials. It is highly unlikely that either of the propellants would come in contact with, and stay on, the suit. During the fueling operations, discussed in Section 9.0, Operations Study, small quantities (on the order of 0.01 to 0.03 cc) of fuel and oxidizer will be exposed to the lunar vacuum at the time when the fueling line/ vehicle disconnects are opened.

Tests were conducted at Bell to determine the effects when small quantities of fuel and oxidizer are instantaneously exposed to high vacuum under lunar ambient temperature conditions. The results of these tests are reported in Reference 31. One tenth cc of 50/50 fuel blend was exposed to 1×10^{-4} torr vacuum with the result that the propellant evaporated completely in 23 seconds. The same quantity of N_2O_4 oxidizer froze in 1.5 seconds and then completely sublimed in 15 seconds. These results indicate that the small quantities involved with the lunar disconnects will evaporate quickly and multi-directionally. A small portion may arrive at the suit in the gaseous state and bounce off into the surrounding space. Furthermore, tests conducted at MSC (Ref 32) indicated that the suit would not be adversely affected if either propellant (in the liquid state) comes in contact with it. This is due to the inability of the chemicals to react with, or penetrate, the outer Beta glass layer. The report recommended that the astronaut wear a loose fitting, Beta fabric, disposable outer garment while performing the fueling operations. Before entering the spacecraft, the garment could be discarded, reducing the chance of introducing traces of the propellants into the cabin. The need for such a precaution is open to question at this time in view of the vacuum test results. In any event, this does not appear to be difficult problem.

13.0 TRAINING

13.1 GENERAL INTRODUCTION

The experience of Bell Aerosystems in the design, manufacture, operation and training pilots in the operation of rocket/jet lifting vehicles provides considerable data to show that safe lunar operation of such a vehicle is feasible. Moreover, the training program for operation of a One Man Lunar Flying Vehicle (OMLFV) can be achieved in a very straightforward manner, using principles already established, and equipment either presently or readily available.

This confidence does, however, pre-suppose that,at least initially in the lunar environment, the OMLFV will be operated conservatively and cautiously; that is, to say low and slow. Data currently available from Bell test programs, demonstrate that safe VFR flight can be accomplished in a flight regime approximately bounded by 75 ft altitude, and speeds of 100 ft/sec. For efficient flights at ranges of five miles or more, somewhat greater speeds and higher altitudes are desirable. The question arises: "At what speeds, altitudes and ranges does it become necessary to add instrumentation and what kind of instrumentation is required for safe and efficient flight in the lunar environment?"

With these considerations in mind, the program defined on succeeding pages will accomplish the following:

1. Define training requirements for deployment, servicing, flying and navigating the OMLFV.
2. Explore the limits of the flight-envelope of the OMLFV into those regions where most efficient use of the vehicle is made, and develop safe operating procedures.
3. Integrate the findings of the research program (2 above) into later training programs to provide safe increments in the operational envelope, as they become demonstrably safe, to achieve more efficient vehicle use.

The point must be made that conservative operation in the early lunar exploration phases is not a disadvantage, since no matter what the capabilities of the vehicle, cautious initial operation would be the rule for other reasons, e.g., the effect of the unique aspects of the lunar environment, on operation. At later phases of exploration, more will be known about operation in the lunar environment, and about the operational limits of the OMLFV. In the latter connection, it may be confidently predicted that significant increases in the operational envelope will be demonstrated as feasible for relatively small increases in system complexity, e.g., the addition of vertical and horizontal velocity information.

The approach is summarized in Figure 13.1. Astronaut operators will be thoroughly trained for conservative operation in the early lunar exploration phases,

using procedures based on current earth experience. Meanwhile, feedback from lunar missions and the proposed earth research program will have defined safe extensions to the operational envelope with or without increased system complexity. These modifications will be reflected in training program modifications and in turn, further lunar experience will suggest new modifications.

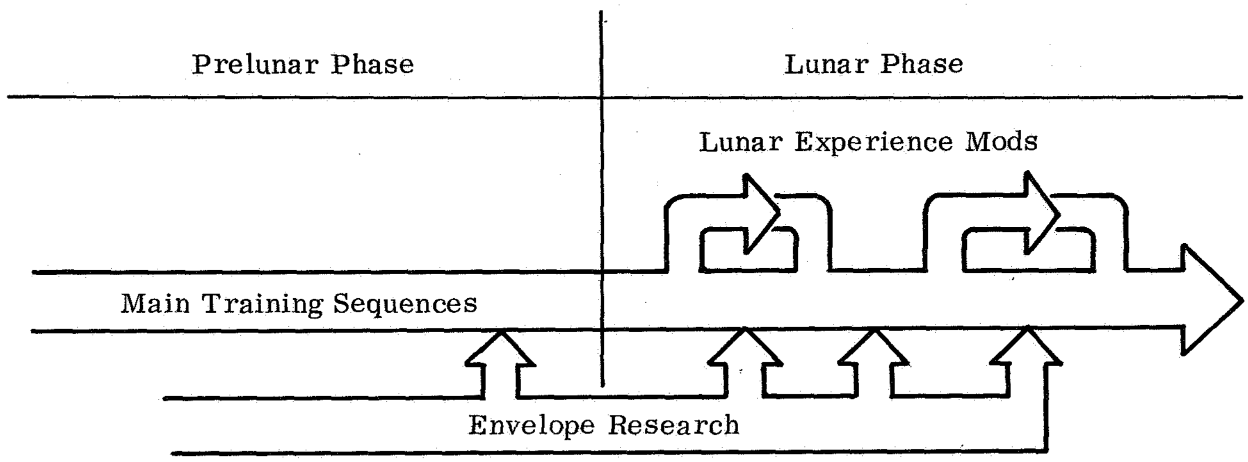


Figure 13.1. Summary of Training and Research Program

13.2 TRAINING PROGRAM APPROACH

Training for flight in rocket/jet lift type vehicles to high degrees of proficiency, has been demonstrated many times. However, all operations, from the deployment of the LFV to fueling and flying it, are critical to mission safety/success.

It should be recognized that all aspects of mission training interact beneficially to improve performance. This is recognized in military flight training by making a thorough understanding of the aircraft, its servicing and the principles of its operation, a necessary precursor to flight training. A pilot who thoroughly understands his vehicle, flies it more confidently and competently. The program outlined reflects this view.

The discussion which follows, is divided, consistent with Figure 13.1, into three parts.

Part 1: Training program for early lunar missions

Part 2: Research into optimal OMLFV flight envelope

Part 3: Training program modification and development

13.2.1 Part 1, Training Program for Early Lunar Missions

13.2.1.1 Flight Training Program

The objectives of the flight training program are:

(a) To provide astronauts thoroughly familiar with the salient aspects of rocket/jet vehicles.

(b) To establish capability of confident, competent control of the vehicle throughout the currently established safe VFR flight envelope, in earth gravity.

(c) To achieve complete familiarity with changes in control characteristics wrought by lunar gravity.

(d) To assure that astronauts are thoroughly briefed on the effects of variables unique to the lunar environment and other than gravity, on vehicle navigation and control.

(e) To assure that astronauts are able to perform feats of control and navigation at least as complex as those to be encountered during lunar missions, under the best available earth free-flight simulation of lunar conditions.

The following training equipment is required to meet these objectives:

(a) Lunar Flyer Simulator(Visual)- This simulator should be a moving base device giving angular cues in three degrees of freedom coupled with a visual display of lunar terrain to provide visual cues in the three translational degrees of freedom. An early version of such a device is currently in use for research and training at Bell.

(b) 1 g Jet Pogo - This vehicle is a version of the flyer, powered by a jet engine. It is to be used in free earth flight, by pilots in shirtsleeves, where there is a requirement in the training and research programs for flight durations and ranges, greater than those feasible using a rocket powered system. It will be composed basically of the same hardware as the Bell Jet Belt, currently undergoing flight test, but modified to provide a structural post on which to mount foot-pans and landing gear. The vehicle will provide flight durations of up to 10 minutes.

(c) 1 g Rocket Pogo - The Rocket Pogo is similar to the lunar vehicle except that hydrogen peroxide monopropellant is used and a total of 1,000 lb thrust is generated to permit earth free flight by a pressure-suited operator for durations of up to 1 minute.

(d) 1/6 g Jet Pogo Trainer - This vehicle will be powered by two small turbojet engines which will be installed in approximately the same position as the flyer thrusters. The vehicle will be flown on a 1/6g tether and will simulate the dynamics of lunar operation. Within the confines of the tether support rail and drive system, up to 5 minutes of flight will be possible.

The main guiding features of the Flight Training Program are:

(a) Training steps will be in small increments, to provide maximum confidence/proficiency, with minimum risk.

(b) An insistence on repeated demonstrations of proficiency in all critical skills will be maintained throughout, and nothing will be proposed for inclusion in lunar mission definition that is not performed safely, easily, confidently and repeatedly on earth.

(c) Every aspect of lunar operation that can be, will be simulated with maximum possible fidelity. In this way, the transition increment from last earth flight to first lunar flight will be kept as small as possible.

In addition to the ground school session with which the flight training program commences, each subsequent training step will be accompained by a requisite period of ground school instruction which will outline the practical exercise, and deal with areas related to making the best use of it. Table 13.1 shows the flight training sequence and Figure 13.2 shows the overall timing and integration with the servicing training described below. The astronaut will average approximately 1.5 hours per day on flight and servicing training during the 21-week program. This will involve as much as a full day during ground school periods and as little as 10 to 20 minutes per day during shirtsleeve flight training periods.

13.2.1.2 System Deployment/Servicing Training

The objective of this part of the training program is to train astronauts to that point where they are capable of single-handedly deploying, checking, fueling and performing all tasks related to maintaining and operating the OMLFV in the lunar environment.

The following training equipment is required:

(a) Lunar Flyer Servicing Trainer - This will be a non-flying vehicle, containing the basic features of the OMLFV. It will be used for deploying, fueling and refueling exercises.

(b) LM Training Mockup - This mockup will contain all the working systems necessary to simulate OMLFV deployment and servicing sequences, and will be used in conjunction with the Lunar Flyer Servicing Trainer, and the 1 g Rocket Vehicle during field training.

(c) KC-135 Airborne 1/6 g Simulation - This vehicle will be equiped for practice of critical servicing operations under 1/6 g conditions.

The guiding features of the flight training program in so far as they are relevant, will be applied to the remaining training task, which will in any event, parallel flight training and join with it where appropriate.

TABLE 13.1

FLIGHT TRAINING ACTIVITIES AND SEQUENCE

Stage No.	Activity	Duration	Equipment	Description of Activity
1	Initial Ground School	1 day		Activity: Classroom briefing on: (a) Rocket/Jet Lift Vehicles; Principles of Operation. (b) Briefing on Objects of and Syllabus for Program. (c) Detailed Familiarization with Training Vehicles.
2	Preflight Familiarization	2 days (1 day simulation 1 day 1/6 g rig)	LFV Simulator; 1/6 g Rocket Pogo in tether	Demonstration of control layout, control operations and effects of controls in isolation and interaction, using simulator; and demonstration of noise and wind generated by vehicle, using 1/6 g pogo.
3	Tethered Flight	20 days for total of 55 flights, scheduled. 2 flights/day during week 1 and 3 flights/day during weeks 2, 3 and 4.	1 g Rocket Pogo, in tether.	Introduction to flight and training for shift-sleeve operation to that standard of proficiency allowing free-flight in a similar vehicle. First free flight conducted on same day as last tethered flight.
4	Free-Flight	15 days for total of 40 flights, scheduled. 1 flight on first day, 2 flights/day for remainder of week, and 3 flights/day during weeks 2 and 3.	1 g Rocket Vehicle removed from tether	Introduction to free flight and training for shirt-sleeve operation to that standard of proficiency showing complete confidence and competence throughout the flight envelope of the vehicle.
5	Vehicle Operation in Pressure-Suit	10 days for total of 20 flights, scheduled 2 flights per day throughout.	1 g Rocket Pogo in tether	Training for pressure-suited operation to that standard of proficiency allowing free-flight in a pressure suit, using a similar vehicle.
6	Free-Flight in Pressure-Suit	10 days for total of 20 flights, scheduled 2 flights per day, throughout.	1 g Rocket Pogo removed from tether	Training for pressure-suited free-flight operation to that standard of proficiency showing complete confidence and competence throughout the flight-envelope of the vehicle.

TABLE 13.1 (CONT)

Stage No.	Activity	Duration	Equipment	Description of Activity
7	Introduction to Flight in Lunar Gravity	5 days for total of 10 flights, scheduled. 2 flights per day, throughout.	1/6 g tethered Jet Pogo	Introduction to and familiarization with control dynamics of LFV lunar operations, in pressure-suit.
8	Free-Flight Lunar Mission Simulation	20 days for total of 40 flights, scheduled. 2 flights per day, throughout.	1 g Jet Pogo and 1 g Rocket Pogo	Introduction to and training in flying in Lunar type terrain. Emphasis will be a navigation and control while performing missions of increasing range and durations. Rocket and Jet vehicles will be used as range and pressure-suit requirements dictate.
9	Lunar Navigation Simulation	10 days for total of 10 hours training scheduled at 1 hour per day.	Lunar Flyer Simulator (Visual)	Use of specially constructed lunar terrain models to familiarize trainees with area of upcoming lunar mission. Trans-Lunar flights will be simulated to provide navigation practice in conditions of varied lighting.
10	Flight in Lunar Gravity, Continued	10 days for total of 20 flights, scheduled. 2 flights per day, throughout.	1/6 g Tethered Jet Pogo	Achievement of complete confidence and competence, flying in pressure-suit under 1/6 g, throughout the envelope of the simulation facility.

Note: At this point, the training program, per se, is complete. The astronaut maintains currency by flying regularly, a minimum number of 1 g, 1/6 g and simulated missions. The initial lunar mission flights will provide verification of training procedures and highlight areas requiring further examination. Cautious expansion of the flight envelope during lunar missions will also provide important feedback for future training programs.

So far as is possible, by careful design, the operations will be of the type which if performed at all are performed perfectly. The remaining requirement will be to perform in correct sequence and within the required time. Although time-lines have been established analytically, these will be refined, and time-standards set, in practice.

The training sequence is defined in Table 13.2. The nature of the parallel with the flight training program and the interface with it, are defined by the dual flow chart shown in Figure 13.2.

13.2.2 Part 2, Research Into Optimal LFV Flight Envelope

The previous paragraphs defined a training approach which would lead to a very high probability of success in accomplishing nap-of-the-moon exploration. However, the characteristics of the vehicle are such, that nap-of-the-moon is not the most efficient way to fly at longer ranges.

The research program is designed to establish requirements for vehicle handling characteristics, instrumentation, and automation which would increase the efficiency of longer range flights.

Some of the problems that could arise are, for example: How do navigational errors increase with speed? How can they be minimized? How fast can one fly VFR at various altitudes? etc.

It is therefore required to conduct studies separate from but parallel with vehicle development and astronaut training, which will indicate more precisely the VFR limits of the OMLFV and the extent to which efficiency can be improved by the addition of instrumentation.

In order to obtain the flight ranges and durations necessary for the type of research envisioned, a jet powered vehicle will be used; ideally in an earth environment presenting the closest available simulation of lunar surface characteristics. The traditional approach to mapping the vehicle envelope can be used here, with the addition of objective data on control activity. Various speed/altitude combinations would be flown by at least three test pilots to map Cooper-Rating boundaries. In addition the vehicle would be instrumented to record rates and positions for each control.

Also necessary is an examination of methods of control in descent to a landing, which becomes progressively more difficult, the greater the altitude at commencement. Here the objective data will be of great importance, since it should be possible to demonstrate a relationship between throttle handling and altitude at commencement of descent. In turn, this relationship can be optimized by the addition of instrumentation. Best use of the data will be made by organizing the flight test program such that it conforms to an experimental design amenable to statistical analysis.

TABLE 13.2

SYSTEM DEPLOYMENT/SERVICING TRAINING ACTIVITIES AND SEQUENCE

Stage No.	Activity	Duration	Equipment	Description of Activity
1	Initial Briefings	1 day		Definition of scope of training program, outline of basic training tasks, with special attention to critical areas e.g., fuel/refueling operations.
2	Deployment Training	2 days for total of 8 sequences, scheduled 4 per day.	OMLFV Servicing Trainer LM Mock-up	Pressure-suit training in accessing, and deploying LFV from LM mock-up.
3	Fueling/Refueling Training	Sequences, scheduled 2 per day, to interface with flight training.	OMLFV Servicing Trainer and LM Mock-up	Fueling/refueling sequences practiced dry in pressure suit.
4	Fuel/Refueling Training	2 sequences scheduled as time available (see Figure 13.2)	OMLFV Servicing Trainer and LM Mockup	Fuel/refueling sequences practiced wet in pressure-suit.
5	Critical Operations in Lunar Gravity	Total of 3 flights scheduled as time is available (see Figure 13.2)	KC 135 Airborne simulation	Practice of initial servicing operations under lunar gravity e.g., unstowing, walking with and stowing pressurized fueling lines.
6	Total Operation Training	4 sequences per trainee 2 in mid-course, 2 at end of course as time available (see Figure 13.2)	OMLFV Servicing Trainer and LM Mockup	Practice of all servicing operations in chronological order as specified in the EVA sequences outlined in the Operations Analysis Section.

Note: At this point the training program per se, is complete. The astronaut maintains currency by reviewing the operations regularly.

Initial lunar deployment/servicing activity will provide verification of training procedures and highlight areas needing further scrutiny. Continuing modifications to procedures can be made on a feedback basis, as mission experience builds up.

Weeks From Commencement

Week	M	T	W	T	F
1					
2	2	2	2	2	3
3	3	3	3	3	3
4	3	3	3	3	3
5	3	3	3	3	2*
6	3	3	3	3	3
7	3	3	3	3	3
8	3	3	3	3	3
9	2	2	2	2	2
10	2	2	2	2	2
11	2	2	2	2	2

* Including first free flt

No. of Flights per Trainee (weeks 2–11)

Flight Training

Briefing

Simulator

1/6 Tethered Rocket

Stage	1 and 2	3	4	5	6
Activity	Ground School and Pre-Flight	Tethered Flight	Free Flight	Operation in P.S.	Free Flight in P.S.
Equipment	LFV Sim. and 1/6 g Rocket Pogo	1 g Tethered Rocket Pogo	1 g Rocket Pogo	1 g Tethered Rocket Pogo	1 g Rocket Pogo

Deployment/Servicint Training

Deployment Training		2 Sequences per trainee per day, to total of 10 sequences, scheduled to interface with flight training.	Total of 2 wet sequences per trainee, scheduled as time available.	1 Airborne simulation run/practice per day per trainee for total of 3 runs per trainee, scheduled as time available.	2 Complete deployment/servicing sequences per trainee, scheduled as time available.
Stage	1 and 2	3	4	5	6
Activity	Briefing and Deployment	Dry Fuel/Refuel Training	Wet Fuel/Refuel Training	Critical Operations in Lunar Gravity	Total Operation
Equipment	OMLFV Servicing Trainer and LM Mockup			KC 135 Suitably Equipped	Servicing Trainer and LM Mockup

Weeks From Commencement

Week	M	T	W	T	F
12	2	2	2	2	2
13	2	2	2	2	2
14	2	2	2	2	2
15	2	2	2	2	2
16	2	2	2	2	2
17	2	2	2	2	2
18	1	1	1	1	1
19	1	1	1	1	1
20	2	2	2	2	2
21	2	2	2	2	2

Weeks 12–17: No. of Flights per Trainee; Weeks 18–19: 1 Hour per Trainee; Weeks 20–21: No. of Flights per Trainee

Stage	6	7	8	9	10
Activity	Free Flight in P.S.	Lunar g	Free Flight Lunar Mission Simulator	Lunar Navigation Simulation	Lunar g
Equipment	1 g Rocket Pogo	1/6 g Jet Pogo	1 g Rocket Pogo and 1 g Jet Pogo	OMLFV 3 Axis Moving Base VFR Simulator	1/6 Tethered Jet Pogo
					2 Complete deployment/servicing sequences per trainee, scheduled as time is available.
Stage					6
Activity					Total Operation
Equipment					Servicing Trainer and LM Mockup

Figure 13.2. Flying/Servicing Training Schedule

13.2.3 Part 3, Training Program Modifications and Development

The assumption has been made that it is possible to train astronauts to fly VFR in an envelope bounded by approximately 75 ft altitude above the lunar surface and speeds of 100 fps. However, it is a very reasonable assumption, based on both Bell's rocket vehicle experience and on helicopter experience. Moreover, there is every reason for confidence that the envelope can be enlarged to make even better use of the vehicle's efficiency characteristics. This will be done when an earth research program to explore further VFR/IFR flight envelopes has been completed, and when data from early lunar operation of the vehicle is available. Improvements will in all probability be affected by modifications to the training program and/or the addition of instrumentation.

On the basis of presently available information, tests should be conducted using the equivalent of an altimeter, forward speed indicator and vertical speed indicator, to facilitate operation in a more efficient flight regime. The reasons for the speculative prediction as to the possible requirement for these instruments are as follows:

(a) Altimeter - Visual estimates of absolute altitude and altitude changes deteriorate as altitude increases.

(b) Forward Speed Indicator - In lunar operation, the only information on horizontal speed would be obtained either by mental integration using acceleration and time cues, or by judging the rate of travel across the terrain. The former and the latter are both inaccurate, the latter especially so as altitude increases.

(c) Vertical Speed Indicator - Judgments of vertical speed become progressively less accurate as altitude increases. Thus operation at heights where cues are poor could lead to the build-up of rates of descent from which recovery is impossible.

In addition to modifications resulting from the research program outlined in Part 2, early lunar operation will provide information which will take into account those aspects of operation unique to lunar flying eg. horizon curvature, high contrast lighting, opaque shadows, visual washout, absence of air velocity damping, blackness of sky, etc.

Training program modifications will be made on the basis of such information, as it becomes available.

14.0 RESOURCES PLAN

14.1 INTRODUCTION

In order to achieve a sound and economical development program, good basic planning must be established early and maintained during the entire program cycle. This planning is an orderly establishment of detailed tasks and associated costs, both concurrent and sequential, and of necessity must consider the total system in order to plan for systems cost and timely operational availability. However, the total system development can be broken into major contributing areas and cost estimated accordingly; these areas are: Management including documentation, Engineering, Reliability, Manufacturing, Quality Control, Test, and Logistics - which for the purposes of this report includes Training, Technical Manuals, Field Support, Spares and Maintainability. These areas can, and must, be individually planned and controlled with well defined interfaces and adequate milestone objectives to assure orderly progress of concurrent and series tasks. Only in this manner can a realistic total schedule be established and maintained, and therby, lead to an economical program and timely operational availability.

Presented in this section is the basic resources plan for the One Man Lunar Flying Vehicle (OMLFV) based upon a total system concept, and based upon application of the philosophy previously described. Preliminary basic scheduling requirements for major tasks in each area of concern are presented as well as composite scheduling, to reflect the total system cycle with major interfaces and milestones identified.

The resources plan is divided into Phases B, C and D based on the guidelines presented in NASA Phased Project Planning Guidelines (Document NHB 7121.2, August, 1968).

The resources plan presented herein is similar in philosophy and approach to that presented as part of the Lunar Flying Vehicle and Manned Flying Systems Studies. This plan reflects the same minimum risk program established in these studies. The resources plan includes the following subplans:

(a) Management Plan

(b) Engineering Plan

(c) Test Plan

(d) Reliability Plan

(e) Manufacturing Plan

(f) Quality Control Plan

(g) Logistics Plan

Program cost estimates are included and included as appendices are: E, One Man Lunar Flying Vehicle Preliminary Parts Breakdown, and D, Quantity of Hardware Included in Cost Estimates.

14.2 PROGRAM CONCEPT AND SUMMARY

Basic planning for the OMLFV follows the phased project concept outlined in NASA document NHB 7121.2, August 1968, "Phased Project Planning Guidelines", to provide a progressive buildup of the program and minimize technological, scheduling and resource risk and uncertainty.

The phased plan is presented in Figure 14.1. The phases are shown with no elapsed time from the end of one phase to the start of the next phase to permit delivery of an operational system in thirty-two months. The plan will provide safe and reliable systems through a comprehensive and rigorous early engineering effort to define the system in detail followed by extensive component, subsystem, and system level testing. "Off the shelf" qualified components will be used wherever possible.

Phase B, the first phase of the program, is a four month definition phase in which the flyer concept that has been developed in past studies is refined as required to define the follow-on Phase C. During Phase B, conceptual designs of earth flight training vehicles are developed, preliminary system and interface specifications are outlined, and lunar and earth support equipment requirements are determined. Systems analysis is accomplished in the areas of propulsion, controls, and structures to define requirements. Detailed resources planning for Phase C and preliminary planning for Phase D are accomplished in Phase B.

Phase C is an eight month period wherein preliminary designs are developed and final specifications are written for the lunar flyer and earth flight trainer concepts recommended from Phase B as well as for lunar and earth support equipment. Systems and supporting analysis is conducted in greater depth and simulation studies and pressure suit/vehicle mockup tests are conducted. Long lead items, such as rocket engine, tanks, and throttle valve are designed, fabricated and subjected to development tests. Detail Phase D resources plans are developed.

In Phase D, the detail design of all hardware is completed; test hardware is manufactured; development, PFRT, and qualification testing is accomplished; operational hardware is manufactured, acceptance tested and delivered; and contractor support is provided for astronaut training and mission operations.

The phased plan provides for delivery of six operational vehicles in forty-two months and the delivery of two of each of three types of earth flight trainers in time to provide a minimum of six months of astronaut training prior to the delivery of the first operational system.

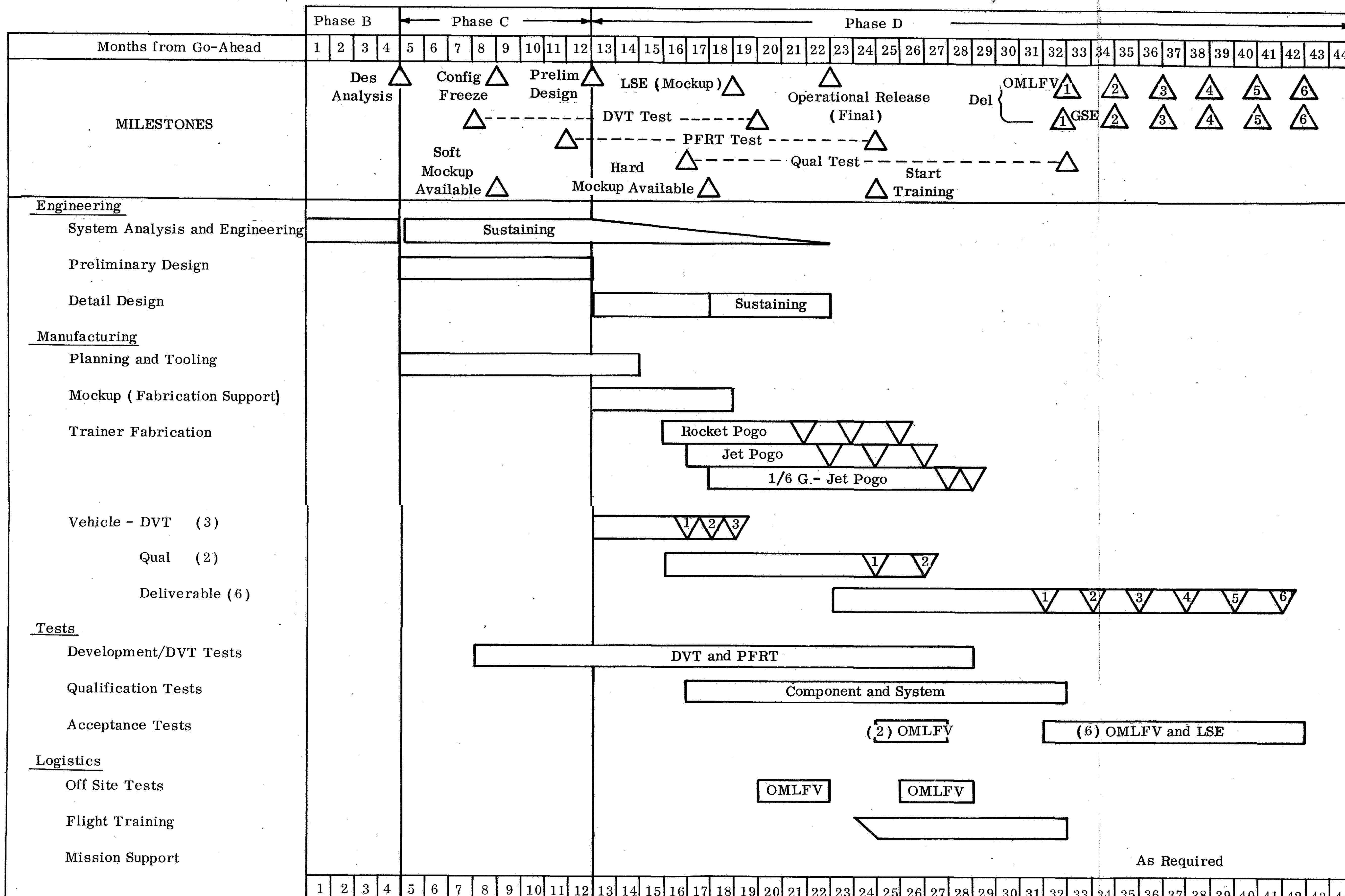


Figure 14.1. One Man Lunar Flying Vehicle Summary Schedule

14.3 MANAGEMENT PLAN

14.3.1 Project Organization

A contractor project management team accomplishes all planning, coordinating, controlling, and directing of the development, production, and test cycle.

The OMLFV Project Organization consists of three categories; Project Management, Project Direction, and Functional Activities. Each category has defined responsibilities and interface requirements within the OMLFV contractor's organization, and between the contractor's organization and associate contractors, integrating contractor, and NASA. The project organization is shown in Figure 14.2 and Manpower loading for the management function is shown in Figure 14.3.

Project Management is headed by a corporate vice president and includes, in addition, a project manager, assistant project manager, and the functions of project administration and contract administration. The project manager, together with the assistant project manager, has complete responsibility for the overall program effort. In the absence of the project manager, the assistant project manager acts in his behalf and has the same responsibilities.

Project direction includes a technical director and directors of manufacturing, product assurance, project control, logistics and test. The technical director has the responsibility of all technical direction associated with the OMLFV and its integration requirements. Functional activities under the technical director include structures, propulsion, flight controls, engineering test, integrated systems, and supporting systems.

The manufacturing director is responsible for the functional activities of procurement, manufacturing engineering, manufacturing control, manufacturing, and manufacturing management.

The product assurance director is responsible for quality assurance, quality control, reliability, and value analysis.

The project control director is responsible for master scheduling, configuration management, and data mangement.

The logistics director is responsible for field service, training support, spares and modifications, and technical manuals and publications.

The test director is responsible for vehicle qualification and acceptance testing.

14.3.2 Documentation Plan

The documentation plan to be implemented on the OMLFV program utilizes procedures which will provide: (a) uniformity in determining and acquiring data; (b) continuity of data flow throughout the development and use life cycle; (c) positive

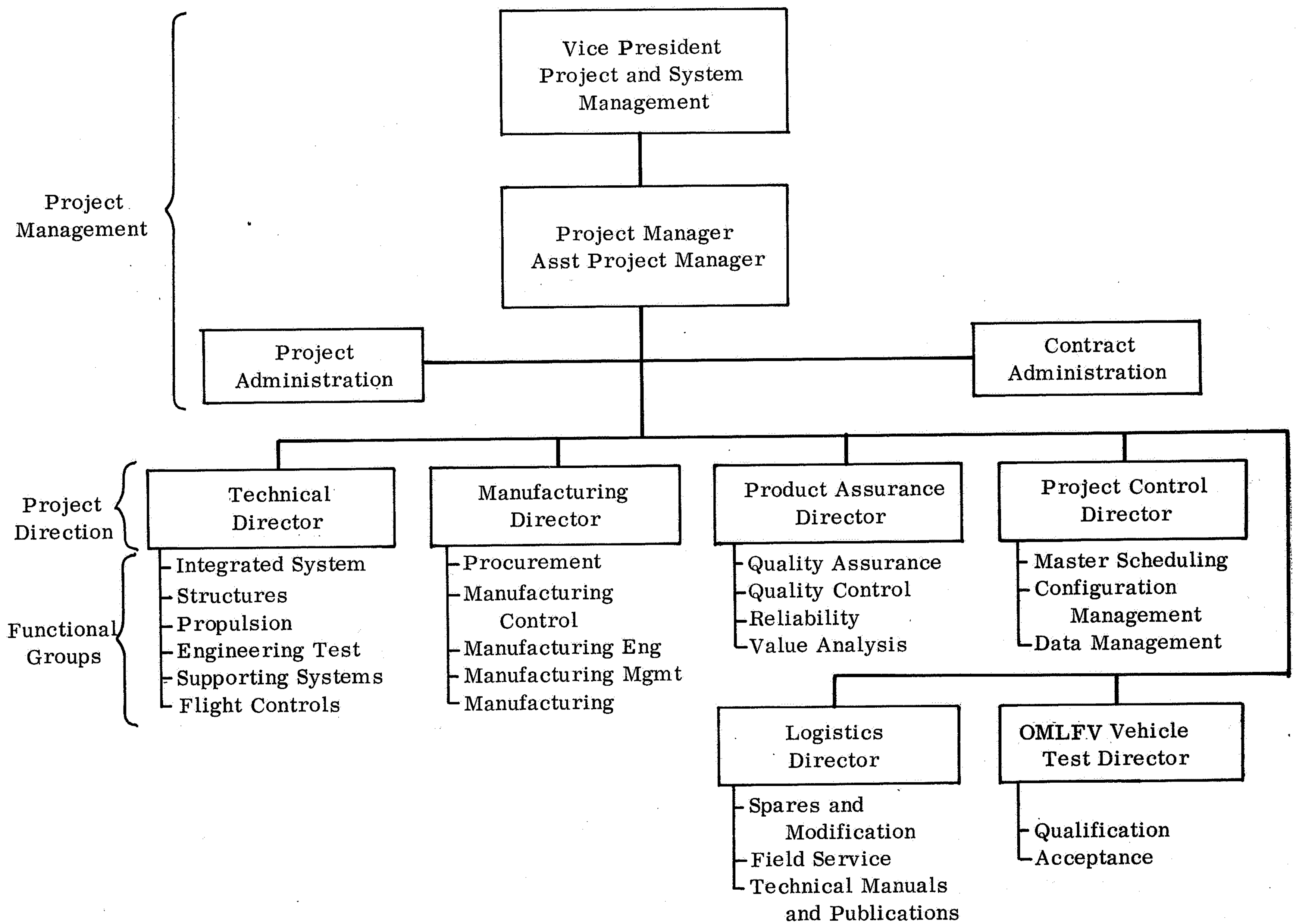


Figure 14.2. Project Organization

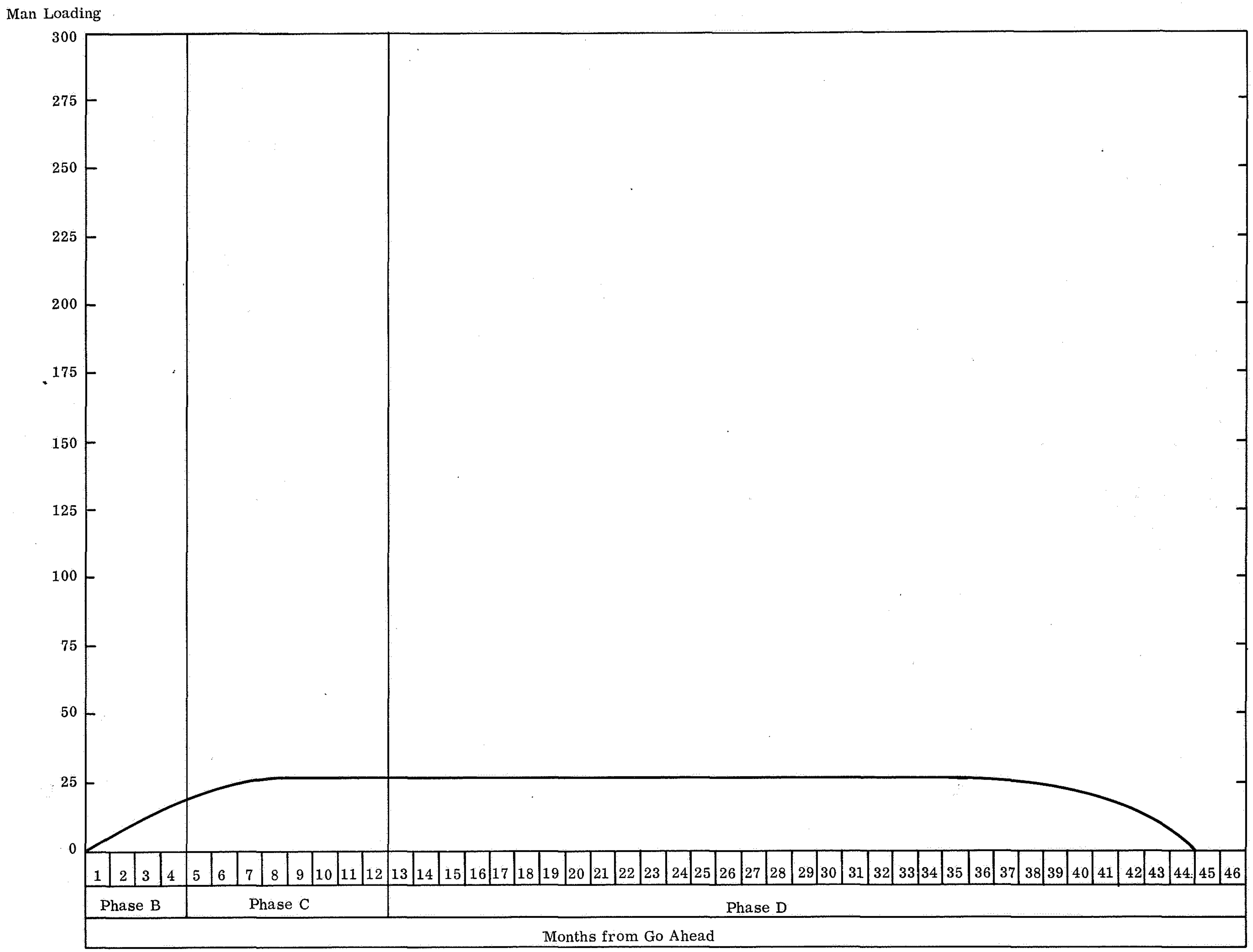


Figure 14.3. Project Management/Documentation Manpower Loading

control of data generation by continuous review of data requirements; and (d) clear and relevant information necessary for program visibility and management.

Table 14.1 lists the documents in the following categories for purposes of classification and control. The categories are those outlined in NPC 500-6. The documents listed are those required for the OMLFV program.

(a) Program Management

(b) Program Scheduling

(c) Procurement and Contracting

(d) Documentation

(e) Configuration Management

(f) Logistics/Support

(g) Facilities

(h) Manning and Financial

(i) Technical Description and System Engineering

(j) Reliability and Quality Assurance

(k) Safety

(l) Test/Manufacturing

(m) Site Activation for Launch

(n) Mission Operations

(o) Mission Oriented Training

(p) Related Program Interfaces

(q) Advanced Missions

14.3.3 Integration Plan

14.3.3.1 Introduction

The integration effort controls the interfaces between the OMLFV flight equipment, OMLFV lunar support equipment, aerospace ground equipment (including both operational and maintenance equipment), factory/field/checkout test equipment, and Apollo test/launch/mission facilities.

The basic requirements to achieve this objective are the responsibility of the OMLFV prime contractor, and include:

TABLE 14.1

DOCUMENT REQUIREMENTS LIST

Report	Submittal
A. Program Management	
1. Program Plan	Semiannually
2. Final Report	6 mo. after technical completion
3. Progress Report	Semimonthly
4. Work Breakdown Structure	First month each phase
5. Red Flag Item Report	As required
B. Program Scheduling	
1. Master Schedule	First month each phase
2. Milestone Chart	
C. Procurement and Contracting	
1. Contract Management Plan	4th month
2. Procurement Standards	4th month
3. Contract Change Notices	As required
D. Documentation	
1. Documentation Management Plan	6th month
2. Document Requirements List	6th month
3. Documentation Matrix	6th month
4. Subcontractor Document Requirements	8th month
E. Configuration Management	
1. Configuration Management Plan	12th month
2. Specification Tree	12th month
3. Specifications	Revise pages as required
a. Lunar Vehicle Detail Specifications	7th month
(1) Component Detail Specifications	15th month
b. Lunar Support Equipment Detail Specifications	9th month
(1) Component Detail Specification	15th month
c. Ground Support Equipment Detail Specifications	9th month
(1) Component Detail Specifications	15th month
d. Trainer Detail Specifications	9th month
(1) Component Detail Specifications	15th month

TABLE 14.1 (cont)

Report	Submittal
F. Logistics/Support	
1. Spares Provisioning Plan	12th month
2. Ground Support Equipment List	12th month
3. Lunar Support Equipment List	12th month
4. Maintenance Plan	12th month
5. Maintainability Analysis Report	9th month
G. Facilities	
1. Facitlities Plan	12th month
H. Manning and Financial	
1. Manpower Resources Report	Quarterly
2. Financial Management Report	Quarterly
I. Technical Description and System Engineering	
1. Controls Analysis Report	10th month
2. Propulsion System Analysis Report	10th month
3. Structural Analysis Report	10th month
4. Landing Dynamics Analysis Report	8th month
5. Environmental Analysis Report	9th month
6. Performance Analysis Report	7th month
7. Handling Qualities Analysis Report	10th month
8. Mission Analysis Report	
9. Design Plan	12th month
10. Mass Properties Report	10th month
11. Thermal Analysis Report	10th month
12. Failure Mode Analysis	12th month
13. Drawings	90 days prior to vehicle delivery
a. Installation	
b. Detail	
c. General Arrangement	
d. Three-view	
e. Interface	
14. Electrical Load Analysis Report	12th month
J. Reliability and Quality Assurance	
1. Reliability	
a. Reliability Plan	12th month
b. Reliability Estimate Report	12th month
c. Reliability Apportionment Report	12th month
d. Reliability Status Report	Quarterly

TABLE 14.1 (cont)

Report	Submittal
2. Quality Assurance	
a. Quality Control Plan	12th month
b. Acceptance Test Plan	45 days prior to acceptance test
c. Acceptance Test Procedure	30 days prior to test
d. Acceptance Test Data Sheets	15 days after test
e. Special Test and Handling Equipment List	30 days prior to fabrication
f. Quality Control Performance Audits	Quarterly
g. End Item Narrative Reports	30 days after test
h. Manufacturing Process Control Plan	30 days prior to fabrication
K. Safety	
1. Safety Program Plan	16th month
2. Ground Operation Safety Procedures	18th month
3. Flight Operation Safety Procedures	20th month
4. Lunar Mission Safety Procedures	20th month
L. Test/Manufacturing	
1. Test	
a. General Test Plan	
b. Development and Design Verification Test Plan	45 days prior to test
c. PFRT Plan	45 days prior to test
d. Qualification Test Plan	45 days prior to test
e. Development and Design Verification Test Procedures	30 days prior to test
f. PFRT Procedure	30 days prior to test
g. Qualification Test Procedure	30 days prior to test
h. Development and Design Verification Test Report	Preliminary 15 days after test Final 60 days after test
i. PFRT Report	Preliminary 15 days after test Final 60 days after test
j. Qualification Test Report	Preliminary 15 days after test Final 60 days after test
2. Manufacturing	
a. Manufacturing Plan	12th month
b. Inspection Procedures	30 days prior to fabrication
c. Process Control Procedures	30 days prior to fabrication

TABLE 14.1 (cont)

Report	Submittal
d. Special Tooling List	30 days prior to fabrication
e. Material Specifications	30 days prior to fabrication
f. Parts Specifications	30 days prior to fabrication
g. Standards	30 days prior to fabrication
M. Site Activation for Lanuch	
1. Lunar Vehicle Checkout Procedure	30 days prior to test
2. Lunar Support Equipment Checkout Procedure	30 days prior to test
3. Launch Site Requirements Report	30 days prior to test
N. Mission Operations	
1. Mission Operations Analysis Report	With vehicle delivery
2. Pre-flight Procedure	With vehicle delivery
3. Flight Operation Procedure	With vehicle delivery
4. Post-Flight Procedure	With vehicle delivery
5. Technical Manuals	With vehicle delivery
O. Mission Oriented Training	
1. Training Plan	Semiannually
P. Related Program Interfaces	
Q. Advanced Missions	
1. Advanced Mission Analysis Report	At program completion

(1) Plan and utilize control documentation and specifications.

(2) Establish and maintain design control documents to provide control of all OMLFV system, subsystem, and interface requirements.

(3) Integrate associate contractor, integrating contractor and program assessed reliability requirements.

(4) Participate in establishing program reliability goals.

(5) Establish and maintain selected control activities planned by the integrating contractor and associate contractors to ensure consistent design and test criteria and objectives.

(6) Establish and maintain effective communications to monitor program activities that shall include associate contractor environmental and functional tests and reliability activities.

(7) Recognize and effectively identify existing and potential incompatibilities among the integrating and associate contractors' hardware, software, systems, plans and schedules.

(8) Provide closed loop requirements for all corrective action associated with the OMLFV/AAP.

(9) Maintain files, document results, and submit required documentation to the associate contractors, integrating contractor, and the customer.

14.3.3.2 Contractual and Integration Communications

Lines of contractual/integrating communications are clearly defined to ensure proper coordination of systems/program efforts. Figure 14.4 illustrates the inner relationships of communication requirements. All communication and/or systems integration are direct among all organizations.

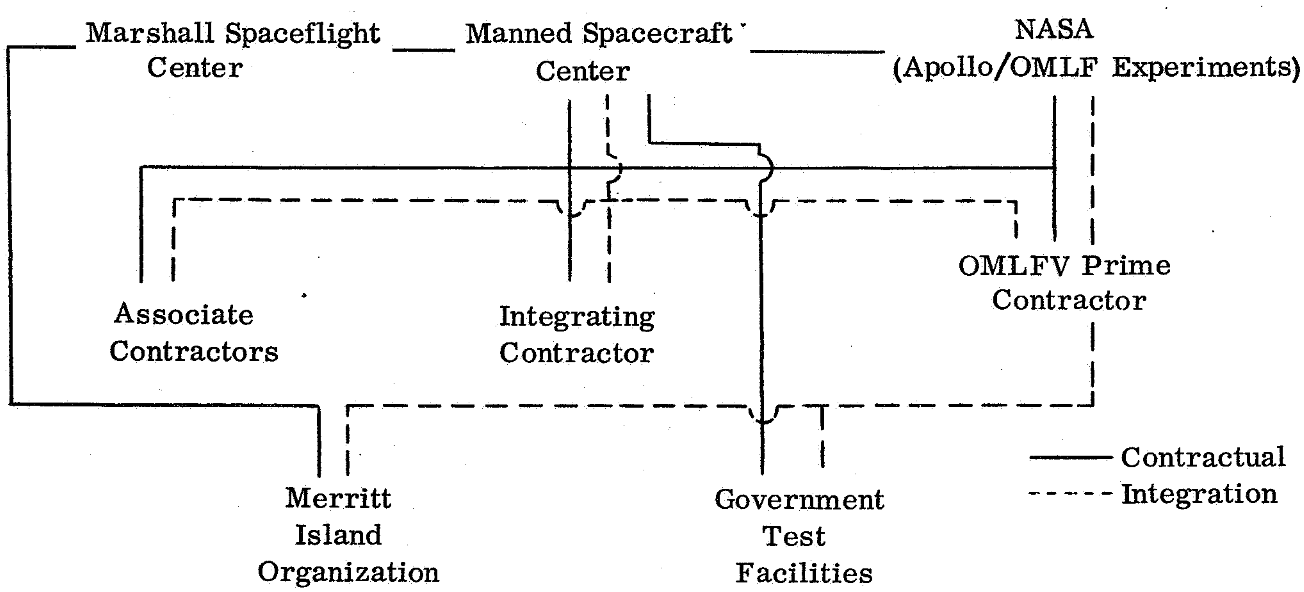


Figure 14.4 OMLFV Program - OMLFV Contractor Oriented

14.3.3.3 Integration Responsibilities

The OMLFV prime contractor's organization is responsible for the OMLFV integration requirements in three defined areas: overall OMLFV program integration, overall OMLFV program technical integration, and OMLFV contractor technical integration.

The overall program integration is the responsibility of the Project Manager and Assistant Project Manager, Technical Director, and Contractor's Program Coordinator. This responsibility includes program accomplishment within prescribed schedule, budget, quality requirements, and all contractual relationships and approvals.

The overall program technical integration is the responsibility of the Project Director and includes all project engineers. The Project Engineers direct and evaluate program technical tasks, performance, progress, and requirements. In addition, their responsibility includes the resolution and approval of technical reports and documents, and maintaining close coordination with their integration/associate contractor and NASA counterparts.

The OMLFV prime contractor's System Engineering area is responsible for all OMLFV contractor in-house system integration and, in addition, support the project department in overall AAP/OMLFV system integration. The System Engineering department includes systems integration, engineering reliability, test integration, and design and development.

The systems engineers, and their associated prime groups are responsible for: all engineering changes; technical problem solving as affected by integration requirements; technical adequacy of OMLFV design in support of integration requirements; systems and subsystems specifications; systems, subsystems, and design requirements, technical integration analysis to establish and confirm OMLFV system performance.

Engineering reliability is responsible for: monitoring all MFS/associate contractor's reliability programs; establishing and assisting reliability programs; establishing and assisting in failure reviews to determine need for corrective action; establishing requirements for vendor and subcontractor reliability programs; provide and review integrated inputs/outputs to reliability models; provide integrated reliability, human factors and maintainability requirements and data to support all OMLFV program design and testing.

Test integration is responsible for: providing field test, flight test, and training planning in close coordination with logistics and, in addition, include all instrumentation, test data, and launch and mission support requirements; approve all test requests, conduct and/or monitor all tests both on-site and off-site; assist in evaluating and reporting field, flight, training, launch, and mission test results; integrate all test requirements into a unified test program; coordinate the establishment of all test parameters and test procedures.

Design and development is responsible for: designing and developing all OMLFV, OMLFV lunar support, and GSE equipment to meet the integrated design requirements to provide maximum support of the mission objectives; initiate requests for test required to develop and evaluate all design; and perform and integrate stress and weight analysis.

14.3.3.4 Interface Specifications and Documentation (Figure 14.5)

Interface specifications define both functional and physical interfaces between subsystems and systems for all affected associate contractors. The definition is applicable to the OMLFV vehicle, lunar support equipment, ground support equipment, and test, launch, and mission facilities.

During the preparation of interface specifications, the concurrence of affected associate contractors, the integrating contractor, and the approval of the customer are obtained on an iterative milestone program. Upon approval, interface specifications become a part of the detail systems specifications and are used as the controlling documents for affected interfaces.

Changes to interface specifications are accomplished by specification change notices and/or change proposals dependent on the baseline configuration freeze milestone. The procedures for distribution, incorporation, and approval of changes to these specifications are established and controlled through the program Joint Operation Plan.

End item specifications, system specifications, and detailed system specifications together with interface specifications are utilized in preparing baseline design requirements. Once the design requirments for configurations are final released, signifying the beginning of the baseline configuration freeze, any revision is accomplished through a change control system.

During release cycles, engineering drawings and specifications are distributed between all associate contractors, integrating contractor, and the customer. These drawings and specifications are identified as "preliminary" and "final". The preliminary documents are distributed prior to release and the final documents after release. Change documentation follows the same procedure. All associate contractors, integrating contractor, and the customer keep an up-to-date file on preliminary and final documentation. The OMLFV contractor requires a minimum of four copies of all documentation, and up-to-date files are located in the program and project offices.

14.3.3.5 Joint Operating Plans

Joint Operating Plans are initiated by the integrating contractor and formulated with all associate contractors. These plans contain policy requirements, basic agreements, designated authority associated with the relationship of all contractors, and are changed utilizing supplemental agreements between associate contractors.

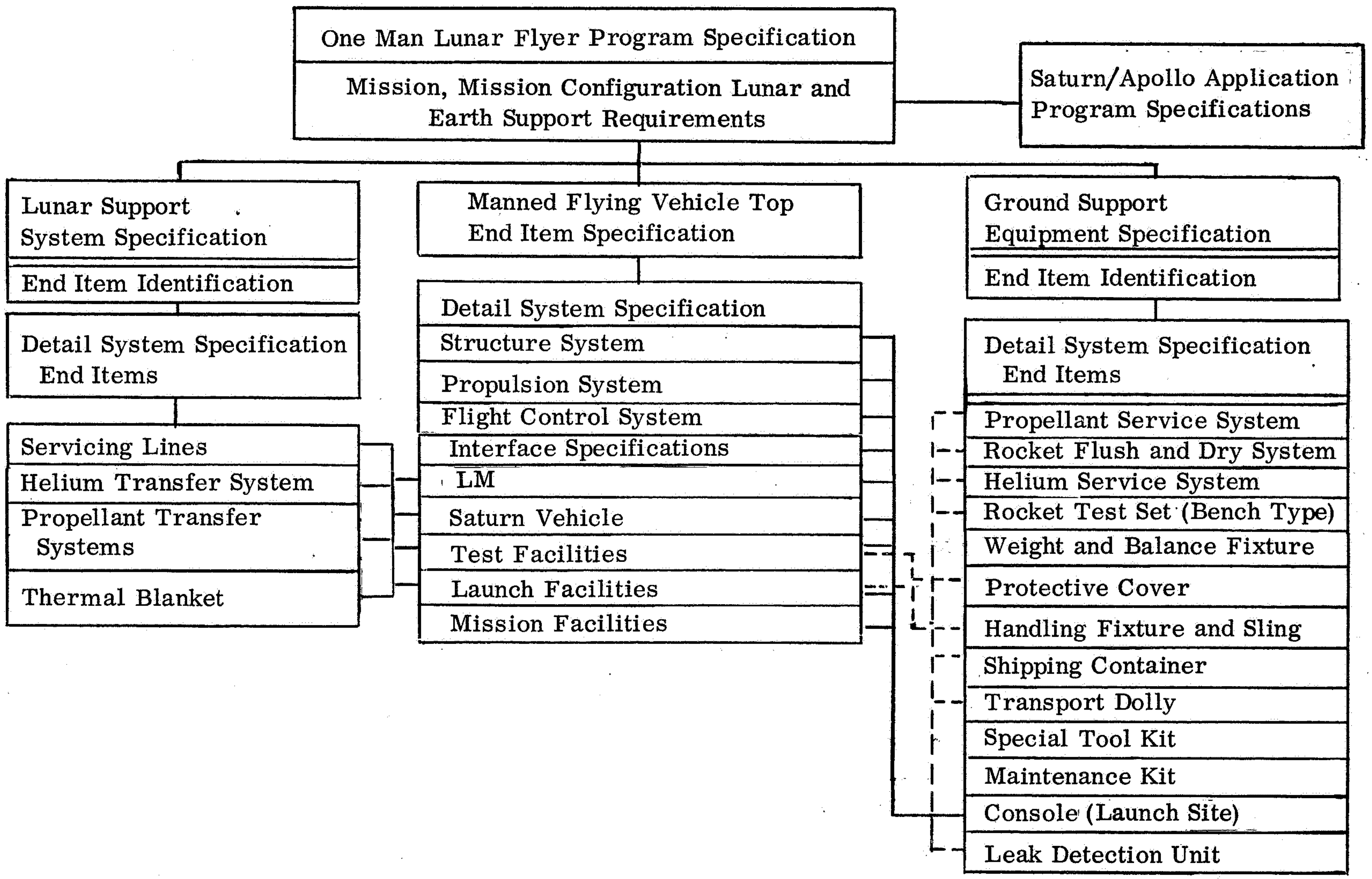


Figure 14.5. Interface Specification Requirements

Concurrence of all associate contractors is obtained prior to final issuance and for all changes. Final approval is the responsibility of the customer Government agency.

The Joint Operating Plans are developed in two phases:

Phase I

(1) Identify specific personnel who have the responsibility to negotiate/approve: technical and contractual agreements, launch and test operations, mission operations, reliability integration details, test integration details, etc.

(2) Establish space, facility, and supply allocations and document as required.

(3) Define and establish meeting schedules and places.

(4) Define and establish methods of correspondence, procedures, forms, and problem solving techniques.

(5) Integrate the program method of change control.

(6) Establish procedures to be by reliability and test integration.

(7) Establish test requirements affecting OMLFV vehicles and support equipment.

(8) Any other specific area of control.

Phase II

Feedback requirements to contain detail methods and procedures arrived at by associate contractors and subject to approval by the customer government agency.

14.3.3.6 Problem Resolution and Failure Analysis

Evaluation of all referred unsolved problems that affect the OMLFV vehicle and its support equipment, including associate contractor problems, require support of the reliability function. Associate contractor problems are reported to the integrating contractor and the customer with an evaluation of the significance of the problem to the overall OMLFV system, and in addition, a recommended priority.

Failure analysis is made on hardware and supporting equipment. The reliability function reviews and approves all resolution for corrective action. Associate contractor failure analysis reports are reviewed to provide integrated data on associate contractor's reliability problems. This ensures that comparable criteria are available for review and classification before using the data in the integrated reliability models.

14.3.3.7 Test Integration

The contractor's functional test section responsibility includes the integration of and support equipment tests with the associate contractor's test programs. This ensures optimum system and subsystem integrated performance requirements. To accomplish this effort the OMLFV contractor's functional test section establishes and maintains test requirements, parameters, and criteria as packaged data associated with development, design, qualification, and acceptance tests for both the OMLFV system and associate contractor's systems.

The contractor reviews, where applicable, associate contractor's tests, including procedures and specifications. These reviews result in evaluations of test criteria affecting the system, identification of problem areas that may develop, and an integrated understanding of test philosophy. The contractor prepares a review report(s) and submits to the integrating contractor and customer as directed by the policies in the Joint Operations Plan.

14.3.3.8 Launch Operations Integration

To establish an organization for integrating requirements at the launch site, the contractor utilizes a nucleus of personnel from the functional project group responsible for reliability, test, mockups, and training. Organizational control is the responsibility of logistics and support requirements are established from areas in system engineering. A Launch Support Project Manager is responsible for all contractor activities at the KSC area. He has responsibility interfaces with the associate contractor, integrating contractor, customer, and OMLFV Contractor Program and Project Offices.

Prior to first delivery of the operational OMLFV, acceptance reviews are conducted between all contractors and results implemented into the Joint Operations Plan under a Launch Plan subsection. Areas to be covered in these reviews include facilities available and required; personnel and equipment assignments; activation requirements; integrated checkout, precount, and countdown; support requirements; schedules and procedures.

14.3.3.9 Flight and Mission Requirements (Figure 14.6)

The same functional organization for launch support is utilized for flight and mission requirements and evaluation. The former Launch Support Project Manager is in charge of flight evaluation and reviews from earth to moon. During the lunar mission the Launch Support Project Manager becomes an assistant to the OMLFV Mission Project Manager at the Manned Spacecraft Center.

Flight evaluations result from reviews and reports prepared by an assigned OMLFV Test Evaluation function.

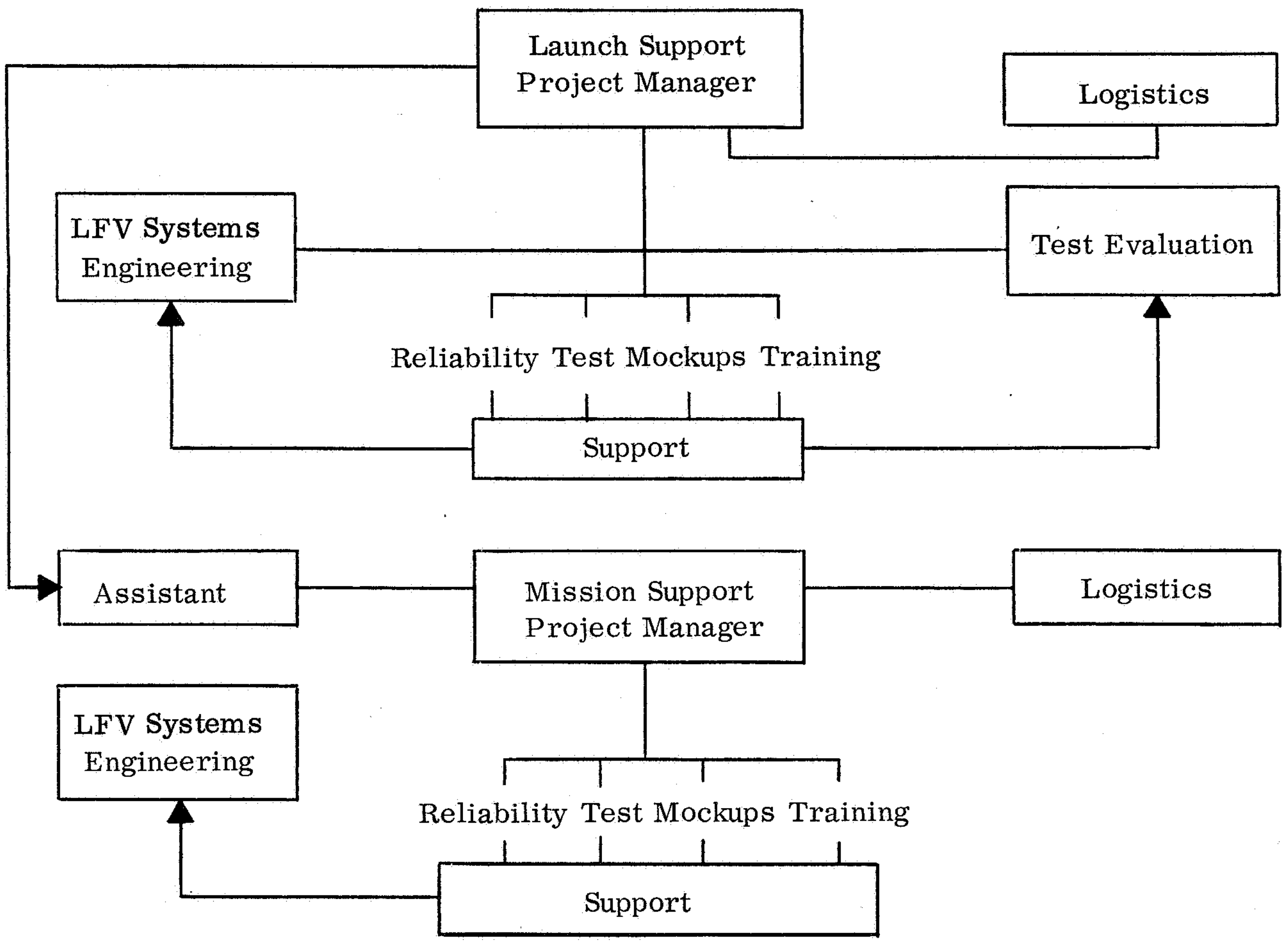


Figure 14.6. Flight and Mission Requirements

14.4 ENGINEERING PLAN

14.4.1 Introduction

The OMLFV Engineering Plan includes analysis and design. This analysis and design covers all aspects of engineering from program inception to post delivery engineering support. The activity starts with Phase B and continues through Phases C and D.

The primary design task is to analyze, define and substantiate technical requirements for the OMLFV program and translate these requirements into the specific operational hardware.

Tradeoff studies, design reviews, mockups, vehicle simulation studies, and development and integrated testing are utilized in support of all design activities.

In pursuing these objectives, in the phased project planning concept, the design and analysis tasks are as shown in Figure 14.7. The manloading to implement these tasks is shown in Figure 14.8.

Phase B

Phase B is a conceptual design phase during which sufficient design and analysis is accomplished on alternate approaches to flyer design problems to recommend a single approach for Phase C. The present study has recommended a flyer configuration for follow on programs. During Phase B, analysis and design are directed toward the refinement of this configuration approach and the development of its logistic support equipment, including earth flight trainers.

During this phase, contract end item preliminary specifications are written for the lunar vehicle and trainers, and the earth and lunar support equipment requirements are determined.

Phase C

Phase C is an eight month period during which all preliminary design is completed, final contract end item specifications are written, and the detail design and prototype development of critical long-lead items is initiated. Systems analysis is conducted to evaluate tradeoffs and verify vehicle design.

Critical hardware for the flyer has already been identified as the rocket engines, propellant tanks and bipropellant throttle valve. During Phase C, engine prototypes are designed, fabricated, tested and modifications resulting from the prototype tests are designed. Oxidizer and fuel development tanks are designed and tested.

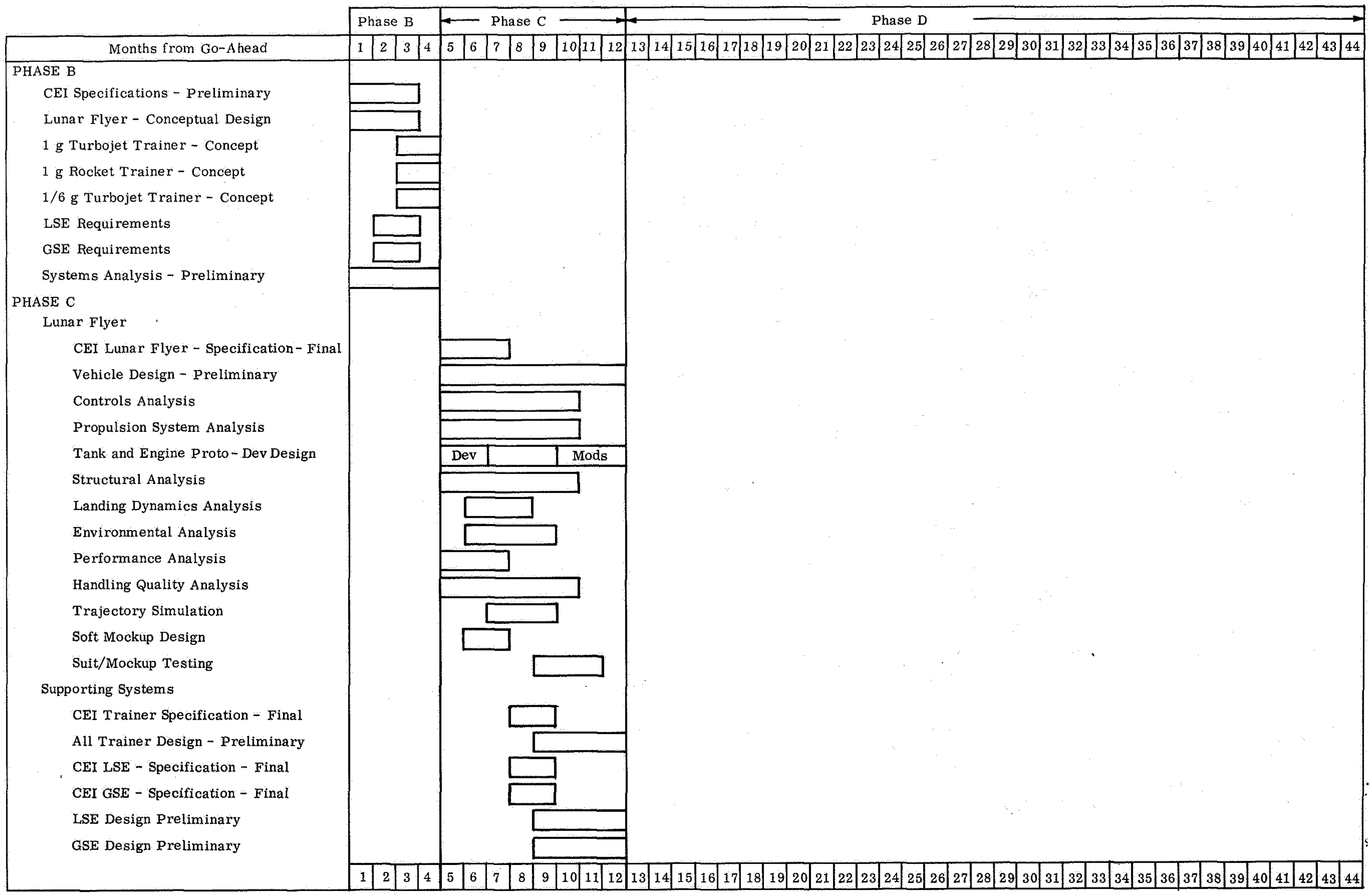


Figure 14.7. One Man Lunar Flying Vehicle Engineering Plan (Sheet 1 of 2)

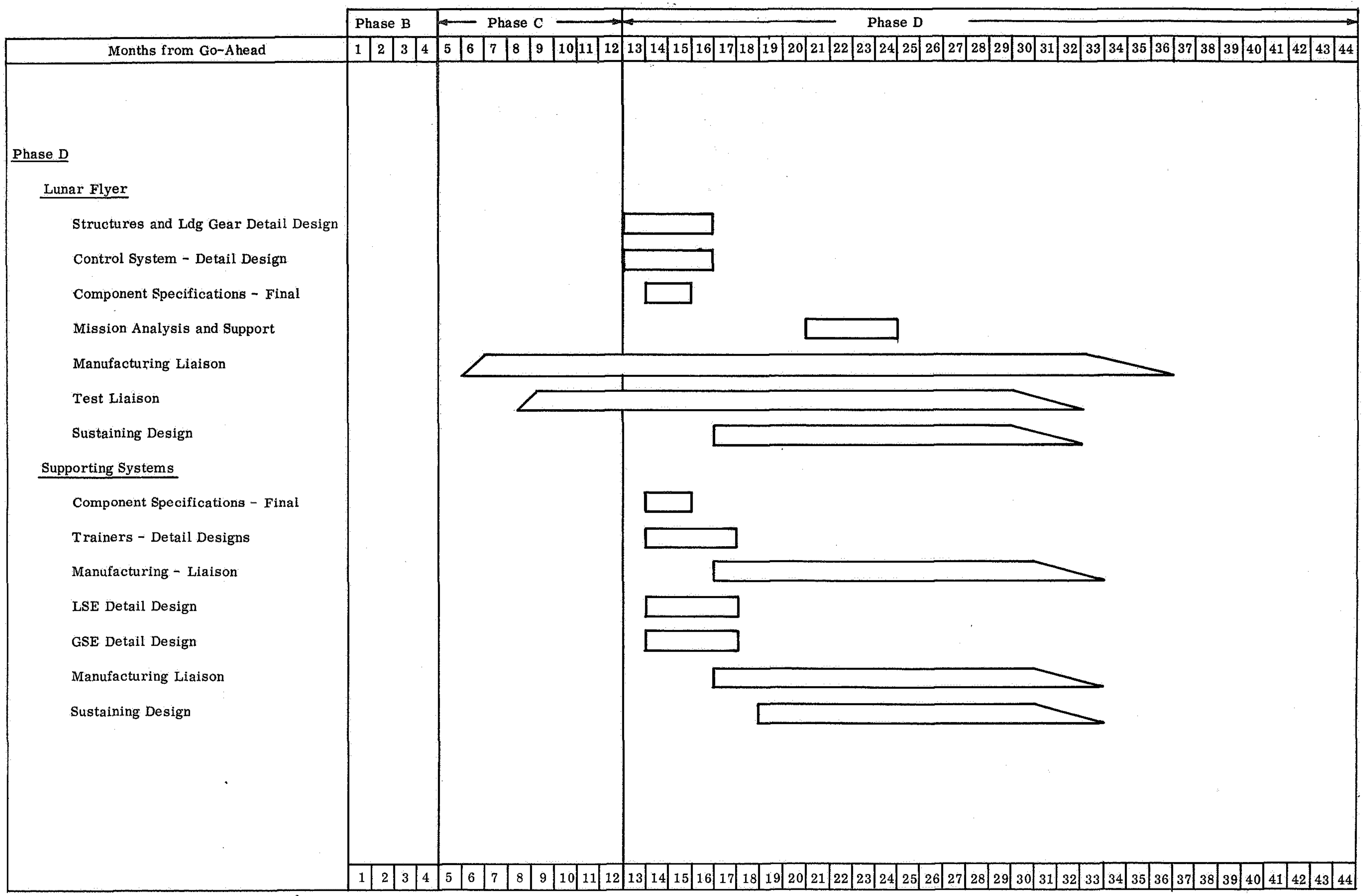


Figure 14.7. One Man Lunar Flying Vehicle Engineering Plan (Sheet 2 of 2)

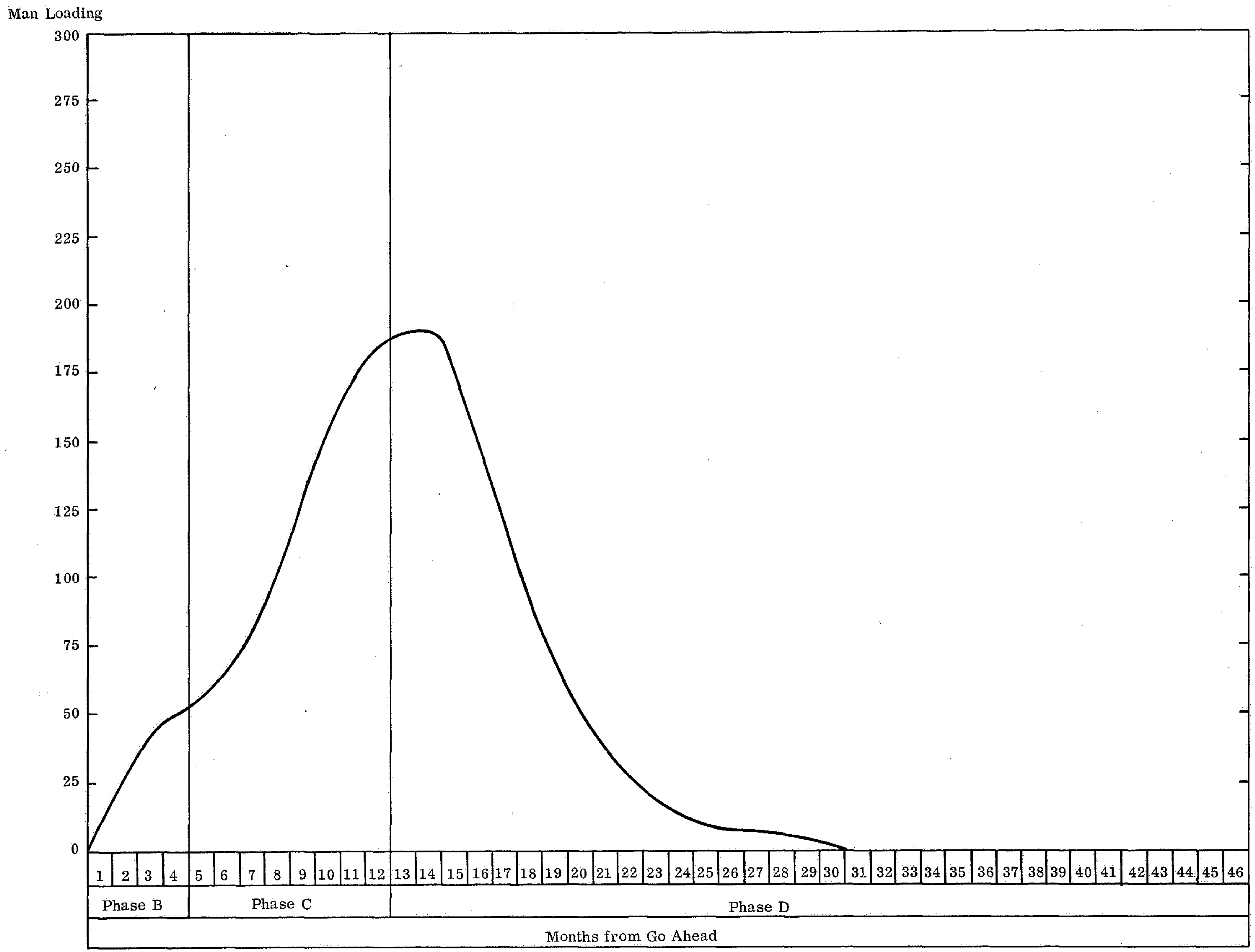


Figure 14.8. Engineering Manpower Loading

The analyses to be conducted are the following:

(a) Controls analysis, including simulation studies to optimize the lunar flyer handling qualities and to determine what control characteristics the earth flight trainers should have.

(b) Propulsion system analysis to define and optimize the complete propulsion system while maintaining an "off the shelf" philosophy wherever practicable to minimize development cost and maximize reliability.

(c) Structural analysis to develop loads criteria and optimize structural concepts for the lunar flyer and trainers.

(d) Environmental analysis to select the optimum method of maintaining acceptable thermal regimes in all temperature sensitive components and systems during a complete lunar mission.

(e) Performance analysis to determine lunar flyer capabilities, tradeoffs and off-nominal performance. Similar analysis is made for training vehicles.

Phase C also includes trajectory simulation, preferably by free flight to investigate pilot capabilities and limits without instrumentation. Suit/mockup tests are conducted for interface compatibility.

Phase D

Phase D is a 24 month period for hardware detail design and test and the fabrication and delivery of operational hardware. The detail design of all systems and subsystems is completed for the lunar vehicle, trainers, lunar and ground support equipment. Mission analysis and support are provided by engineering. Sustaining design activity is carried out as needed.

14.5 TEST PLAN

14.5.1 General

The test operational plans of Figure 14.9 show all effort associated with the initiation, performance, and completion of testing required of the OMLFV and its supporting equipment. Related documentation is included and time phased to the conduct of the testing. All test planning is completed in Phase C. Test procedures for all tests except qualification are written in Phase B. Testing conducted in Phase C is limited to engineering, development and design verification (DVT) component testing on long lead items. These tests are conducted on rocket engines, plexiglass fuel tanks for baffle development, and on bipropellant throttle valves. Limited test buildup is initiated in Phase C in areas where DVT tests are scheduled to start at the beginning of Phase D. All other test activity is conducted in Phase D. Man loading for the testing is shown in Figure 14.10.

Testing is subdivided into four basic categories:

a. Engineering and Development/DVT Testing
b. Preliminary Flight Rating Tests
c. Qualification Tests
d. Acceptance Tests

Each of the above categories is further segregated into levels of complexity identified as components or parts, major subsystems or installations, and complete systems. Significant hardware items are identified within each complexity level for the four test categories. Quantities and types of hardware are assigned to each of the categories that will provide the required level of confidence.

14.5.2 Test Matrices

Test matrices are presented in Table 14.2. These matrices are predicated on the test plans given in Figure 14.9. Periodic updating of these generalized matrices is required during the conduct of the OMLFV program.

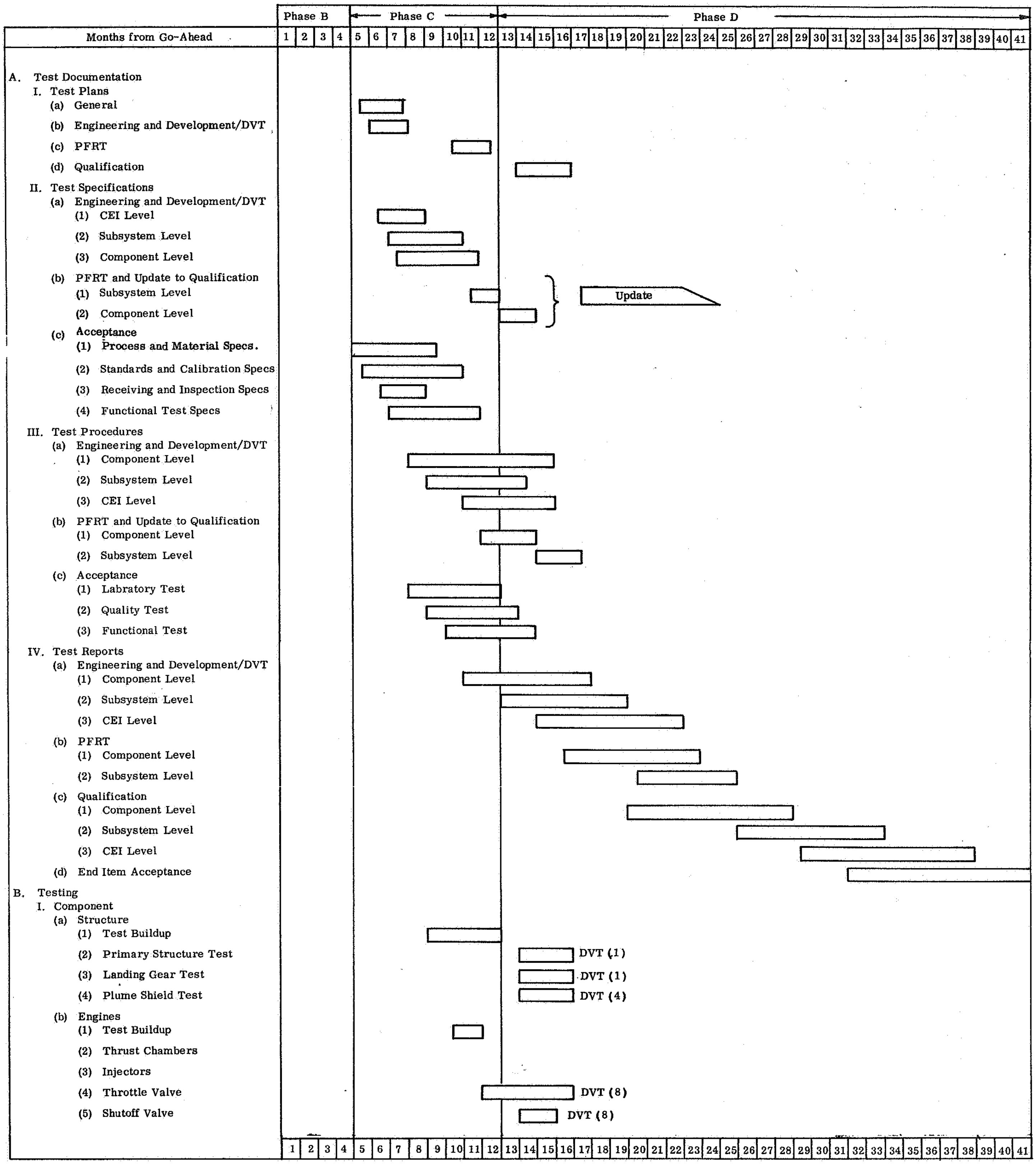


Figure 14.9. One Man Lunar Flyer Vehicle Test Operations Plan (Sheet 1 of 2)

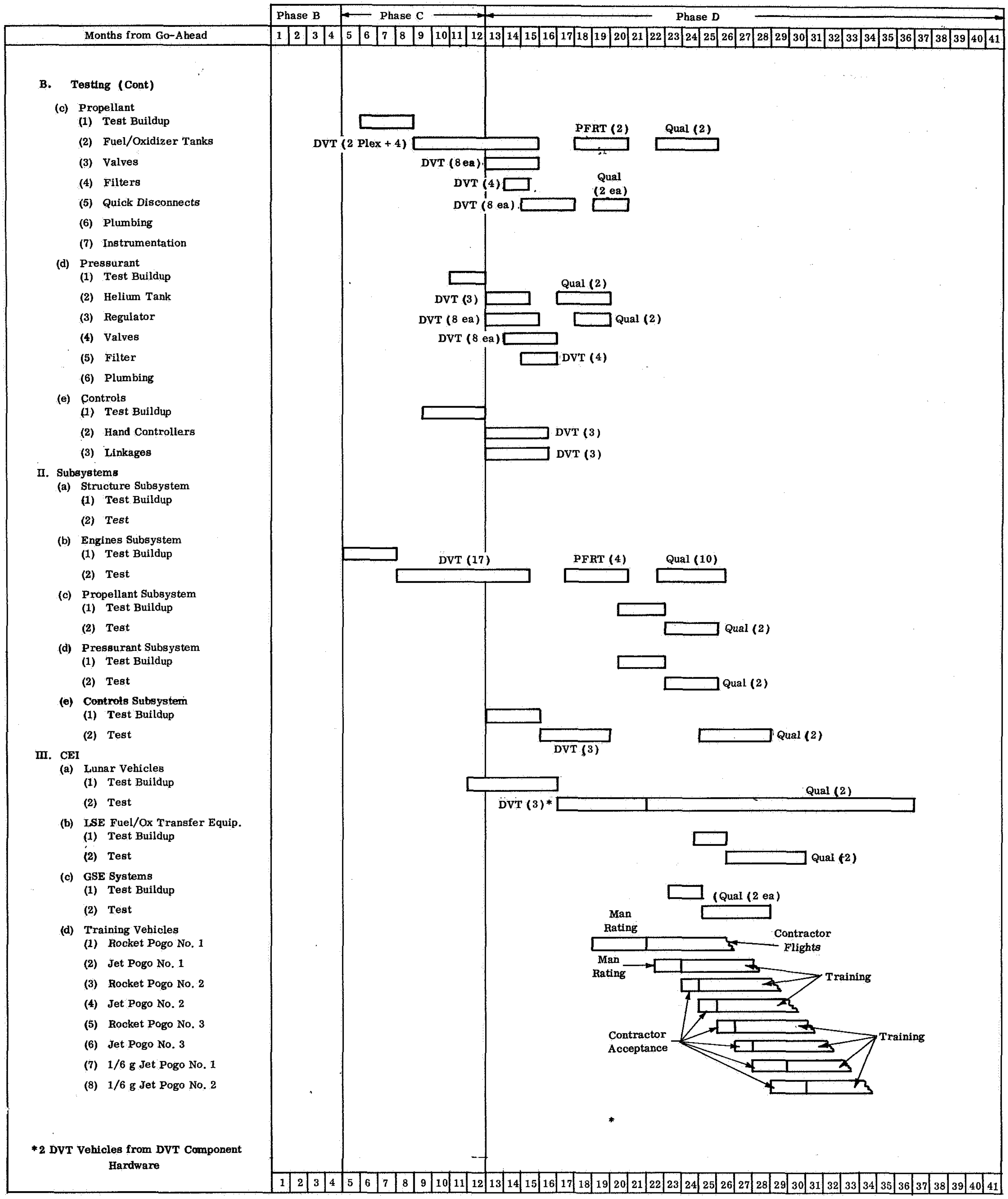


Figure 14.9. One Man Lunar Flyer Vehicle Test Operations Plan (Sheet 2 of 2)

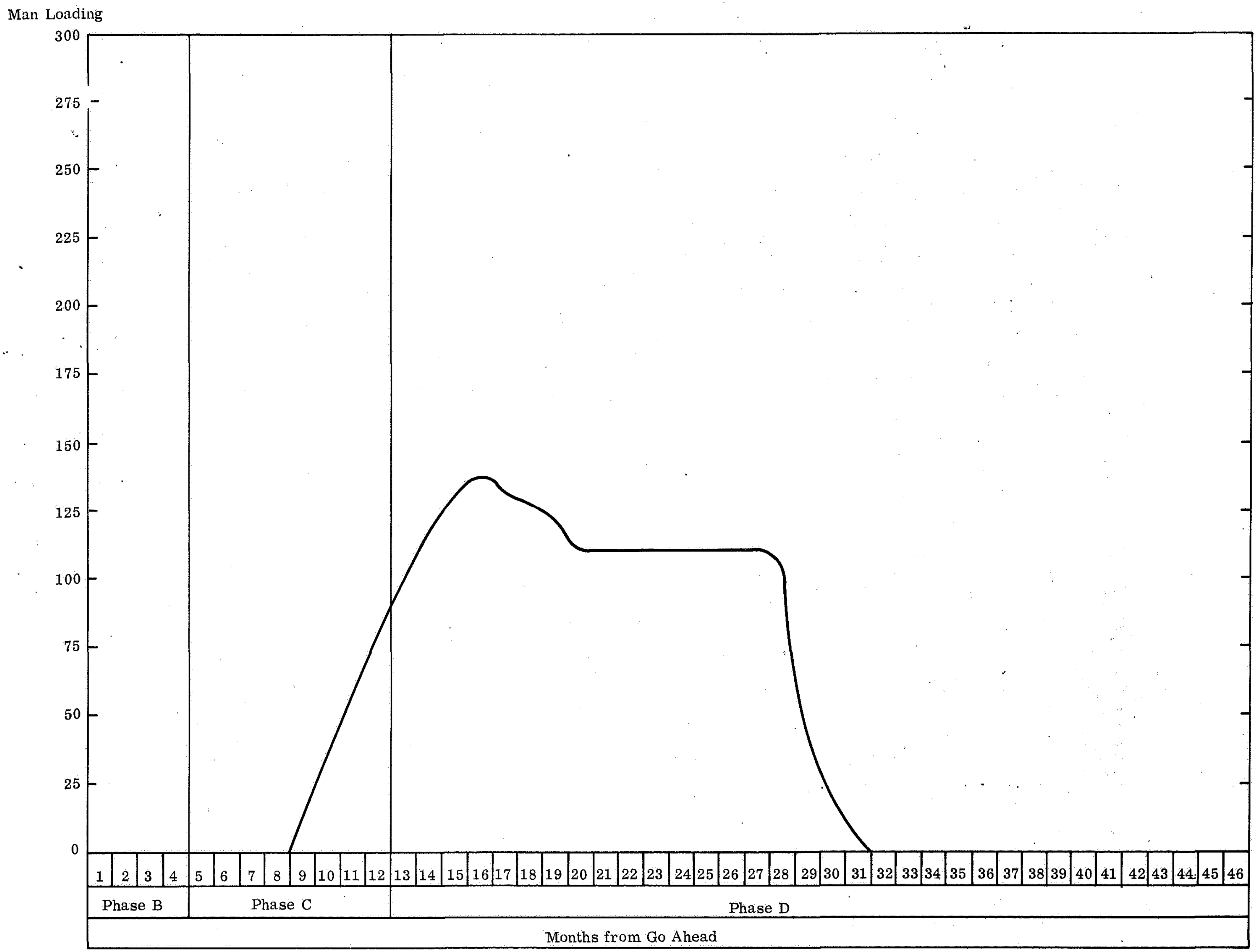


Figure 14.10. Test Manpower Loading

TABLE 14.2

TEST MATRIX

Test Category: Eng Dev/DVT

N - Nonoperating
O - Operating

Test Level	1 - Functional	2 - Static Leakage	3 - Proof Pressure	4 - Burst Pressure	5 - Expulsion	6 - Pressure Drop	7 - Calibration	8 - Variable Thrust	9 - Safety Limit	10 - Drainage	11 - High Temperature	12 - Low Temperature	13 - Engine Transients	14 - Ignition Spike	15 - Combustor Stability	16 - Maximum Disolved Gas	17 - Chamber Pressure Survey	18 - Mixture Survey	19 - Spec Limit Propellants	20 - Thermal Conductivity	21 - Thermal Vacuum	22 - Pressure Cycle	23 - Emissivity	24 - Fluid Compatibility	25 - Radiation	26 - EMI	27 - Plume/Shield	28 - Weight	29 - Mass Properties	30 - Static Tests	31 - Drop Tests	32 - Thermal Cycle	33 - Run to Failure	34 - Design Margin	35 - Fungus	36 - Sand and Dust	37 - Humidity	38 - Shock, Accel, Random Vibration	39 - Sinusoidal Vibration and Slosh	40 - Mag. Fluor. Insp.	41 - Metallurgical Inspection	42 - Disassemble and Inspect	43 - Spring Tests	44 - Tensile Specimen	45 - Compression Set- Hardness	46 - Stress Relaxation	47 - Dissect and Evaluate
CEI																																															
OMLFV	O	N	N		O	O				O	O	O									O							N	N	N			O			O	N	N	N		N	N	N	N	N	N	
LSE	O	N		N		O				O														N					N							O	N					N	N	N	N	N	N
GSE																																															
Subsystem																																															
Structure																																															
Engines	O	N				O		O	O	O			O	O	O	O	O	O	O			O						N	N	N			O			O	N	N		N	N	N		N	N		N
Propellant		N			O	O			O	O																		N		N				O		O	N	N			N	N		N	N	N	N
Pressurant		N							O																			N		N						O	N	N			N	N		N	N	N	N
Controls	O																													N						O		N			N	N		N	N	N	N
Component																																															
Structure																																															
Primary Structure																				N			N					N		N										N	N	N		N		N	N
Landing Gear																							N					N		N	N									N	N	N		N		N	N
Plume Gear																				N			N					N		N		N	O				N				N	N		N		N	N
Engines																																															
Thrust Chambers			N	N																N				N				N		N		N						N		N	N	N		N			N
Injectors				N																				N				N				N									N	N		N			N
Throttle Valve	O	N	N	N																		O		N				N		N		N						N			N	N	N		N	N	N
Shutoff Valve	O	N	N	N																				N				N				N						N			N	N			N	N	N
Propellant																																															
Fuel/Ox Tanks			N	N	O															N		N		N				N		N		N						N	N	N	N	N		N	N	N	N
Valves	O	N																				O		N				N										N			N	N	N		N	N	N
Filters																								N				N										N			N	N			N		N
Quick Disconnects				N																				N				N				N				O	N				N	N	N		N	N	N
Plumbing			N																					N				N													N	N		N	N	N	N
Instrumentation	O																											N								O		N				N					N
Pressurant																																															
Helium Tank			N	N																		N						N		N		N								N	N	N		N	N	N	N
Regulator	O	N																				O						N				N						N			N	N	N		N	N	N
Valves	O	N																				O		N				N													N	N	N		N	N	N
Filter																												N													N	N			N		N
Plumbing			N																									N													N	N		N	N	N	N
Controls																																															
Hand Controllers	O																											N	N	N			O				N	N			N	N		N	N	N	N
Linkages	O																											N	N	N			O				N	N			N	N		N	N	N	N

TABLE 14.2

TEST MATRIX (CONTINUED)

Test Category: PFRT

N - Nonoperating
O - Operating

Test Level	1 - Functional	2 - Static Leakage	3 - Proof Pressure	4 - Burst Pressure	5 - Expulsion	6 - Pressure Drop	7 - Calibration	8 - Variable Thrust	9 - Safety Limit	10 - Drainage	11 - High Temperature	12 - Low Temperature	13 - Engine Transients	14 - Ignition Spike	15 - Combustor Stability	16 - Maximum Disolved Gas	17 - Chamber Pressure Survey	18 - Mixture Survey	19 - Spec Limit Propellants	20 - Thermal Conductivity	21 - Thermal Vacuum	22 - Pressure Cycle	23 - Emissivity	24 - Fluid Compatibility	25 - Radiation	26 - EMI	27 - Plume/Shield	28 - Weight	29 - Mass Properties	30 - Static Tests	31 - Drop Tests	32 - Thermal Cycle	33 - Run to Failure	34 - Design Margin	35 - Fungus	36 - Sand and Dust	37 - Humidity	38 - Shock, Accel, Random Vibration	39 - Sinusoidal Vibration and Slosh	40 - Mag. Fluor. Insp.	41 - Metallurgical Inspection	42 - Disassemble and Inspect	43 - Spring Tests	44 - Tensile Speciman	45 - Compression Set - Hardness	46 - Stress Relaxation	47 - Dissect and Evaluate
CEI																																															
OMLFV																																															
LSE																																															
GSE																																															
Subsystem																																															
Structure																																															
Engines	O	N	N	N		O	O	O	O	N	O	N			O	O	O	O	O			O						N				N	O				N		N	N	N	N		N	N	N	
Propellant	O	N	N	N	N	O			O	N												O						N				N					N		N	N	N	N		N	N	N	
Pressurant	O	N	N	N		O			O	N												O						N				N					N		N	N	N	N		N	N	N	
Controls																																															
Component																																															
Structure																																															
Primary Structure																																															
Landing Gear																																															
Plume Gear																																															
Engines																																															
Thrust Chambers																																															
Injectors																																															
Throttle Valve																																															
Shutoff Valve																																															
Propellant																																															
Fuel/Ox Tanks		N	N																			N						N				N							N	N	N	N		N	N	N	
Valves																																															
Filters																																															
Quick Disconnects																																															
Plumbing																																															
Instrumentation																																															
Pressurant																																															
Helium Tank		N	N																																					N	N	N		N	N	N	
Regulator																																															
Valves																																															
Filter																																															
Plumbing																																															
Controls																																															
Hand Controllers																																															
Linkages																																															

TABLE 14.2 (Continued)

Test Category: Qual

N - Nonoperating
O - Operating

Test Level	1 - Functional	2 - Static Leakage	3 - Proof Pressure	4 - Burst Pressure	5 - Expulsion	6 - Pressure Drop	7 - Calibration	8 - Variable Thrust	9 - Safety Limit	10 - Drainage	11- High Temperature	12 - Low Temperature	13 - Engine Transients	14 - Ignition Spike	15 - Combustor Stability	16 - Maximum Disolved Gas	17 - Chamber Pressure Survey	18 - Mixture Survey	19 - Spec Limit Propellants	20 - Thermal Conductivity	21 - Thermal Vacuum	22 - Pressure Cycle	23 - Emissivity	24 - Fluid Compatibility	25 - Radiation	26 - EMI	27 - Plume/Shield	28 - Weight	29 - Mass Properties	30 - Static Tests	31 - Drop Tests	32 - Thermal Cycle	33 - Run to Failure	34 - Design Margin	35 - Fungus	36 - Sand and Dust	37 - Humidity	38 - Shock, Accel, Random Vibration	39 - Sinusoidal Vibration and Slosh	40 - Mag. Fluor. Insp.	41 - Metallurgical Inspection	42 - Disassemble and Inspect	43 - Spring Tests	44 - Tensile Speciman	45 - Compression Set - Hardness	46 - Stress Relaxation	47 - Dissect and Evaluate
CEI																																															
OMLFV	O	N			O	O		O																				N	N			N			N	O		N	O	N	N	N	N		N	N	
LSE	O	N	N			O																						N				N			N	O		N			N	N			N	N	
GSE	O	N																										N														N					
Subsystem																																															
Structure																																															
Engines	O	N				O		O						O	O	O	O	O	O					N	O	O		N	N			N	O		N	O		N	O	N		N					
Propellant		N			O	O																O						N							N	O		N	O	N		N	N				
Pressurant		N				O																						N							N	O		N	O	N		N					
Controls	O																											N	N						N	O		N		N		N			N	N	
Component																																															
Structure																																															
Primary Structure																																															
Landing Gear																																															
Plume Gear																																															
Engines																																															
Thrust Chambers																																															
Injectors																																															
Throttle Valve																																															
Shutoff Valve																																															
Propellant																																															
Fuel/Ox Tanks		N				O														N								N	N			N						N	N	N		N	N				
Valves																												N				N						N	N		N		N		N	N	
Filters																																															
Quick Disconnects		N																																													
Plumbing																																															
Instrumentation																																															
Pressurant																																															
Helium Tank		N				O														N								N	N			N						N	N	N		N	N				
Regulator	O	N				O																						N				N						N	N	N	N	N	N		N	N	
Valves																																															
Filter																																															
Plumbing																																															
Controls																																															
Hand Controllers																																															
Linkages																																															

The matrices indicate specific tests for hardware items within the levels of component, subsystem, and system. The emphasis is directed more to testing on the subsystem and system level and less to the component level as the program advances from the Design Verification to its Qualification stage. This approach is in accord with the intent to employ components whose validity for use in the lunar environment has been established.

An operating or nonoperating mode has been assigned to each hardware item for each type of test that it must sustain. The test mode is indicated as follows:

a. Operating - symbol "O": While in test, the unit will be required to function or be periodically activated.

b. Nonoperating - symbol "N": While in test, the unit will not be operated.

The actual mode of in-test operation, together with the fact that it be continuous or interrupted is identified as part of the Detailed Test Procedure for a given item. Similarly, where a particular unit is assigned as nonoperating in test, any need for pre- and post-test performance checks are stipulated in the Detailed Test Procedure.

Every test can be related to one of the following classifications:

a. Performance tests wherein equipment operational characteristics are demonstrated, modifications are evaluated, or performance limitations are established.

b. Chemical/Physical tests such as those wherein corrosive effects, materials properties, weights, moments of inertia, and centers of gravity are determined.

c. Environmental tests inclusive of those that simulate or approximate the natural environments such as temperature, vacuum, and radiation as well as induced environments such as vibration, shock, acceleration, and slosh.

The four mission phases that are the basis for conducting the tests are:

a. Earth conditions in conjunction with the ground support equipment adjunct to conducting a system pre-launch checkout.

b. Translunar flight with the OMLFV and lunar support equipment in a stowed configuration aboard the LM.

c. Lunar surface conditions, exclusive of lunar flight, where inactive storage, propellant servicing operations, and manipulations involved in the handling of the OMLFV and its lunar support equipment are critical functions.

d. Lunar flight

Test matrices will be refined in Phase C. The test sequence for a given unit must be established.

Time durations must be allocated for each type of test in each category. When assembled in the desired sequence for a specific hardware item, the result defines the time required to complete testing of that item. Turn-around times between tests of a particular unit is identified, particularly when tests must be conducted at widely separated facilities.

Test facility and equipment usage must be scheduled, and for those facilities that the contractor is unable to furnish, subcontractors, or alternate contractors identified.

Similar test matrices must be generated for the training vehicles as their designs become defined.

14.5.3 Test Descriptions

The tests listed in the test matrices of Table 14.2 are described as follows:

1. Functional. Activation of valves and controls prior to and after other tests and at designated periods during protracted tests. Limited to components which can be checked without decreasing the overall reliability by excessive cycling.

2. Static leakage. All fluid systems, as specified in the model specification, tested for leakage. The test pressure starts at a low differential pressure and then increases at a rate consistent with normal use to the specified static leakage pressure for the complete system. The maximum test pressure is maintained for a minimum of 2 minutes. External or internal allowable leakage of major subassemblies fluids shall not impair or endanger proper functioning of the system or vehicle. Design goal is zero leakage. Leakage rates and provisions for disposition shall be specified in the model specification.

3. Proof pressure. Proof pressure tests conducted on applicable portions of the systems which contain fluid pressure. Hydrostatic pressure equal to proof pressure for PFRT or Qualification test, is imposed and held for a minimum of 2 minutes. Evidence of leakage, deformation, or detrimental permanent set while under pressure is cause for rejection.

4. Burst pressure. A burst test is conducted on portions of the systems which contain pressure. Pressure causing initial yield of the material and the minimum bursting pressure for PFRT or Qualification test shall be as specified in the model specification.

5. Expulsion. Performance tests conducted on the propulsion assembly at vehicle, subsystem, and component levels to demonstrate the controlled release of propellants and expulsion efficiency.

6. Pressure drop. Tests conducted on all fluid systems to determine losses and fluid flow characteristics.

7. Calibration. Calibration tests are conducted and performance shall be within the limits specified in the model specification. Direct thrust determinations are made and used in calculating performance. For engines operating primarily at altitudes other than the test altitude, the altitude thrust is correlated to the thrust under test conditions. This correlation is experimentally determined and approved by the procuring activity. The attitude of the engine and the sequence of its operation shall be as specified in the model specification. A minimum of 3 (one for PFRT) runs are made at maximum, minimum, and each appropriate intermediate thrust level for reusable engines and one run at each condition for single duty cycle engines. The duration of these runs is equivalent to full duration at the respective thrust levels.

8. Variable thrust. Engines are subjected to 5 duty cycles (2 for PFRT) in accordance with the schedule specified in the model specification for PFRT or Qualification test.

9. Safety limits. At least one test is conducted to demonstrate that the engine, when supplied electrical and fluid inputs specified in the model specification shall, under any specified single condition of malfunction, start and operate in a safe manner, or shutdown without presenting a hazardous condition that could cause damage to the vehicle or crew. An analysis of all single malfunction conditions and pertinent second malfunction conditions anticipated in service usage is prepared as a separate report. Each malfunction is specified, the resulting sequence of events determined, the degree of hazard established, and the corrective action described. This analysis is made for engine operation during start, steady state, shutdown and restart (if applicable). This analysis is prepared upon completion of initial engine design and is kept current through the Qualification Tests. A list of recommended critical conditions under which safety should be demonstrated during PFRT and Qualification, is included in the analysis. For manned applications, the analysis includes a definition of additional sensing and control devices required to insure crew safety. These devices may be external to the engine system. For engines requiring multi-starts during a duty cycle, a restart attempt is made after each malfunction test unless analysis indicates a hazardous condition to the engine or facility resulting from each restart.

A minimum of 50 start-shutdown tests are conducted, 10 for single duty cycle engines.

10. Drainage. With the engine in a normal servicing attitude, the fluid systems are filled as in normal use, then drained to the maximum extent possible without firing. The fluids remaining are determined. Fluids remaining, in excess of the amount specified in the model specification, are cause for rejection.

11. High temperature. The engine is subjected to a high ambient storage condition temperature for 24 hours and then stabilized at the high temperature starting condition specified in the model specification. The rocket engine is then be supplied with propellants at the high temperature starting condition and run at a rated thrust for rated duration to demonstrate satisfactory operation.

12. Low temperature. As soon as practicable after the humidity test specified, test No. 37, the engine is subjected to a low ambient storage condition temperature for 24 hours and then stabilized at the low temperature starting condition specified in the model specification. The engine is then supplied with propellants at the low temperature starting condition and run at a rated thrust for a rated duration to demonstrate satisfactory operation.

13. Simulated Engine Transients. A series of tests, to be specified in the model specification, is conducted in which input and output transient conditions simulate those encountered during engine operation.

14. Ignition Spike Tests. The engines are started at simulated high altitude and low temperature to demonstrate the existence or non-existence of ignition spikes and, if present, their effect on engine operation.

15. Combustion stability. The engines are operated under simulated lunar thermal vacuum conditions to demonstrate that the combustion stability meets the requirements of the model specification.

16. Maximum dissolved gas. Tests to determine the effects of gas saturated propellants on the performance of engines.

17. Chamber pressure survey. A systematic survey is made of the engine operating characteristics as a function of chamber pressure. The limits of engine throttling are determined in DVT and demonstrated in PFRT and Qualification.

18. Mixture survey. Performance tests are conducted to determine the effects of propellant mixture ratio on the engines and the limits in mixture ratio for satisfactory engine operation.

19. Specification limit propellants. Performance tests are conducted to determine the propulsion system operating characteristics with known contamination (quantities/sizes, solids/water).

20. Thermal conductivity. The thermal conductivity is measured of critical hardware such as propellant tanks, insulation, tank mounts, thruster mounts, etc., for use in analysis and development.

21. Thermal vacuum. A complete lunar vehicle is subjected to a simulated lunar day/night cycle with associated thermal/solar radiation while the temperatures of critical components are monitored for design verification.

22. Pressure cycle. Propulsion system components are subjected to cycling operating pressures until a minimum number of cycles, as prescribed by the model specification, are exceeded or until operating limits are exceeded.

23. Emissivity. The emissivity characteristics of typical coating materials is determined by measuring radiation, conductance, etc., to determine suitability for application to temperature critical components. The tests are indicated for structure in Table 14.2 since coatings are not specifically listed.

24. Fluid compatibility. The effects on the fluid systems of the chemical action of the fluids within the specification limits of the fluids, as well as the effects of aging with the fluids, drying in air, contact with vapors, or the worst combination thereof, or both, are determined. The portions of the systems subjected to the fluids and phase of the fluid used, simulate as closely as practical, the conditions encountered in the actual application in performing the following tests.

    The compatibility of the engine with the fluids is normally demonstrated by means of satisfactory operation of the engine during other Qualification tests and inspection. Additional tests applicable to service usage, such as long term storage after cleaning, purging, neutralizing, etc., are specified in the model specification.

    The appropriate portion of the systems, as specified in the model specification, mounted to simulate actual installations, are placed in a chamber and allowed to stabilize at the chamber's ambient temperature, after which they are subjected to a spray of the liquid (propellant, coolant, pressurant, etc.), until all exposed surfaces are wetted. They are then air dried for one day and inspected for corrosion or incompatibility.

    The appropriate portion of the systems, as specified in the model specification, are mounted in a chamber to simulate actual installation at room ambient temperature with the humidity such that the wet bulb temperature is not less than minus 4°F (15.6°C) below dry bulb temperature. Then the gas at room ambient temperature, or resulting vapor temperature if a vaporizing liquid, shall be introduced into the systems for an exposure time of 60 plus or minus one minute. After air drying at 80°F (26.7°C) for one day at a relative humidity not greater than 80 percent, they are inspected for corrosion or incompatibility.

25. Radiation. The engine is subjected to the radiation environment specified in the model specification and then run at a rated thrust for a rated duration to demonstrate stable, safe, and reliable operation.

26. Electrical and Electronic Interference. Test per MIL-R-5149B.

27. Plume Shield. The complete vehicle is tested under simulated lunar vacuum conditions. The engines are operated and the plume characteristics and its effects on vehicle components are measured.

28. Weight. Weighing components, subsystems and vehicles, as applicable, as evidence that they do not exceed the weights specified in the model specification.

29. Mass properties. The determination, by measurement, of the moments of inertia and centers of gravity of items as required for controls analysis.

30. Static test. Structural load tests conducted on critical components to verify limit and ultimate loads.

31. Drop tests. Design verification tests conducted on the landing gear wherein dynamic loads are applied corresponding to the design limit loads.

32. Thermal cycle. Tests conducted on selected systems and components by exposing them to the prelaunch temperature environment followed by demonstration of their operation.

33. Run-to-failure. Life tests on operating components or subsystems wherein useful life is determined by operating/cycling/etc., a number of times specified in the model specification, or until failure. Includes engine firings, valve actuations, control actuations, etc.

34. Design margin. Propulsion system tests are conducted to determine how far beyond specification limits the engines can be operated without failure. It includes propellant mixture ratio at engine start, chamber pressure and high propellant feed pressure.

35. Fungus. Fungus tests are conducted on applicable portions of the vehicle as specified in the model specification, prior to salt spray test. Applicable portions selected are inoculated with a mixture of fungus spores containing at least those types listed below:

(a) Group I	Chaetomium globosum USDA 1042.4 or Myrothecium verrucaria USDA 1334.2.
(b) Group II	Aspergillus niger USDA Tc215-4247.
(c) Group III	Aspergillus terreus PQMD 82J.
(d) Group IV	Penicillium citrinum ATCC 9849.
(e) Group V	Fusarium moniliforme USDA 1004.1.

Evidence that all materials used do not support fungus growth constitutes grounds for waiver of this test. The ambient atmosphere is then maintained at a temperature of 85° plus or minus 5°F and a relative humidity of 95 plus or minus 5 percent for a 28-day period. The parts are inspected and operated that the fungus had no adverse effect on the life or function of the systems.

36. Sand and dust. The hardware is subjected to sand and dust as follows: They are placed in a test chamber equal to MIL-C-9436 and the sand and dust density raised and maintained at 0.1 to 0.5 grams per cubic foot within the test space. The test chamber is vented to the atmosphere. The relative humidity does not exceed 30 percent at any time during the test. Sand and dust used in the test is of angular structure and has characteristics as follows:

(a) 100 percent of the sand and dust passes through a 100-mesh screen, U. S. Standard Sieve Series.

(b) 98 plus or minus 2 percent of the sand and dust passes through a 140-mesh screen, U. S. Standard Sieve Series.

(c) 90 plus or minus 2 percent of the sand and dust passes through a 200-mesh screen, U. S. Standard Sieve Series.

(d) 75 plus or minus 2 percent of the sand and dust passes through a 325-mesh screen, U. S. Standard Sieve Series.

(e) Chemical analysis of the dust is as follows:

Substance	Percent by weight
$Si\,0_2$	97 to 99
$Fe_2\,0_3$	0 to 2
$Al_2\,0_3$	0 to 1
$Ti\,0_2$	0 to 2
Mg 0	0 to 1
Ign Losses	0 to 2

The internal temperature of the test chamber is maintained at 77°F (25°C) for a period of 6 hours with sand and dust velocity through the test chamber at 2500 plus or minus 500 feet per minute. After completion of the sand and dust test the systems are operated at a rated thrust to demonstrate stable, safe, and reliable operation.

37. Humidity. The systems are prepared as specified in the model specification and subjected, for at least 5 days, to an atmosphere of air having the high pre-launch temperature and relative humidity specified in the model specification. The systems are then inspected for indications of harmful external corrosion, evidence of which is cause for rejection.

38. Shock, acceleration and random vibration. Vehicle, subsystems, or components, mounted to simulate vehicle installation or transportation and handling, are subjected to the magnitude and duration of shock specified in the model specification. While mounted to simulate vehicle installation, they are subjected to the applied acceleration along the three major axes of the test specimens and random vibrations at frequencies, levels and duration specified in the model specification.

    Any resonant frequencies that are noted during testing are recorded. When specified in the model specification, non-firing functional tests of the engine are performed when simulating flight conditions to determine that the engine shall perform satisfactorily. Following the dynamic tests, the systems are examined for evidence of malfunction or failure which is cause for rejection. After inspection, the engine is operated for one rated duration test at rated thrust.

39. Sinusoidal vibration and slosh. The engine, mounted to simulate vehicle installation, is subjected to a sinusoidal vibration test at the acceleration level and frequency ranges and for the number of vibration cycles (duration) specified in the model specification for PFRT or Qualification test. See test 38 for test results.

    During engineering test, slosh tests are conducted on propellant tanks for baffle design and rocket engines with propellant tanks, mounted to simulate vehicle installation, are subjected to a sinusoidal motion under the conditions and applied frequencies, amplitudes, and duration specified in the model specification.

40. Magnetic and flourescent inspection. Following tests in which parts become highly stressed, such parts made of magnetic material are subjected to magnetic particle inspection in accordance with MIL-I-6868 or AMS 2640. Fluorescent penetrant inspection may be substituted when this method is more readily applied to the part and provides adequate means for detecting detrimental defects or when interpretation of indications can be more positively controlled.

    All highly stressed parts made of non-magnetic material are subjected to fluorescent penetrant inspection in accordance with MIL-I-6866 or AMS 2645.

41. Metallurgical examination. Metallurgical test of specimens removed from components after tests have been conducted. The results are compared with tests of similar specimens of "as delivered" material.

42. Disassemble and inspect. After completion of tests, the major subassemblies are disassembled for examination of parts as specified in the model specification. Measurements and photographs are taken as necessary to disclose excessively worn, distorted, or weakened parts. Further disassembly may be made at the option of the procuring activity. Calibration of controls and components may be required prior to disassembly at the option of the procuring activity.

43. Spring tests. After completion of specific tests, all springs are removed from vehicle systems and tested for free length and load/deflection characteristics to determine that no degradation has occurred.

44. Tensile specimen. Tensile tests performed on specimens, or coupons, that have been removed from components after the completion of other tests to determine any degradation in properties.

45. Compression set-hardness. Tests of seals, rings, flexible sleeves, etc., to determine the extent of permanent set resulting from their use during tests of higher level items.

46. Stress relaxation. Tests of seals, rings, flexible sleeves, etc., to determine remaining flexibility following their use in parent articles.

47. Dissect and evaluate. Following some tests, the systems are reduced to their elementary parts and cut up for tests to the extent that reassembly is prohibited.

14.6 RELIABILITY PROGRAM PLAN

14.6.1 Reliability Program

The Contractor should establish and maintain a reliability program for the achievement and assessment of the reliability of the OMLFV in general agreement with the requirements of NASA - NPC 250-1.

A reliability group cooperates with the functional design groups to establish reliability goals, a reliability budget, requirements for redundancy, evaluation of reliability based upon similar components used on other programs and perform a failure mode and effects analysis. The reliability program is required to incorporate reliability into the basic design; to ensure the maintenance of high reliability goals during the manufacturing and testing phase of the overall program, and to provide a method for the collection and analysis of reliability data to be fed back into the engineering design process. This overall effort includes the establishment of requirements for additional reliability testing if other tests do not provide sufficient data. An important task of the reliability function is the follow-up during testing and flight training vehicle operation to assure that the reliability objectives have been achieved.

The manpower loading to implement the reliability program is shown in Figure 14.11.

14.6.2 Reliability Integration

A reliability operation should be maintained which is responsible for the integrating of all reliability functions of the associated contractors to assure that data, procedures, plans and specifications are compatible. Scheduled assessments, reviews and audits determine progress and current status of integrated programs in the above disciplines. Test plans permit integration of reliability and quality data obtained from associated contractors, subcontractors and suppliers with similar data obtained from the OMLFV contractor's own test program. Activities in the disciplines of human factors engineering and maintainability avoid overlapping or duplication of effort among associated contractors.

14.6.3 Reliability Assessments and Estimates

Reliability Engineers establish reliability requirements for major subsystems down to major components. The math model is updated at program milestones with development, qualification, and integration requirements which are used to develop relative failure rates. Initial estimates of system reliability are based on life test data on similar systems collected through tests, APIC and other aerospace companies.

The goals for the system are apportioned to the major subsystems and distributed to designers for design consideration. The initial estimates of the major subsystems and system are revised as the design progresses through the development phase and when development and qualification test data become available.

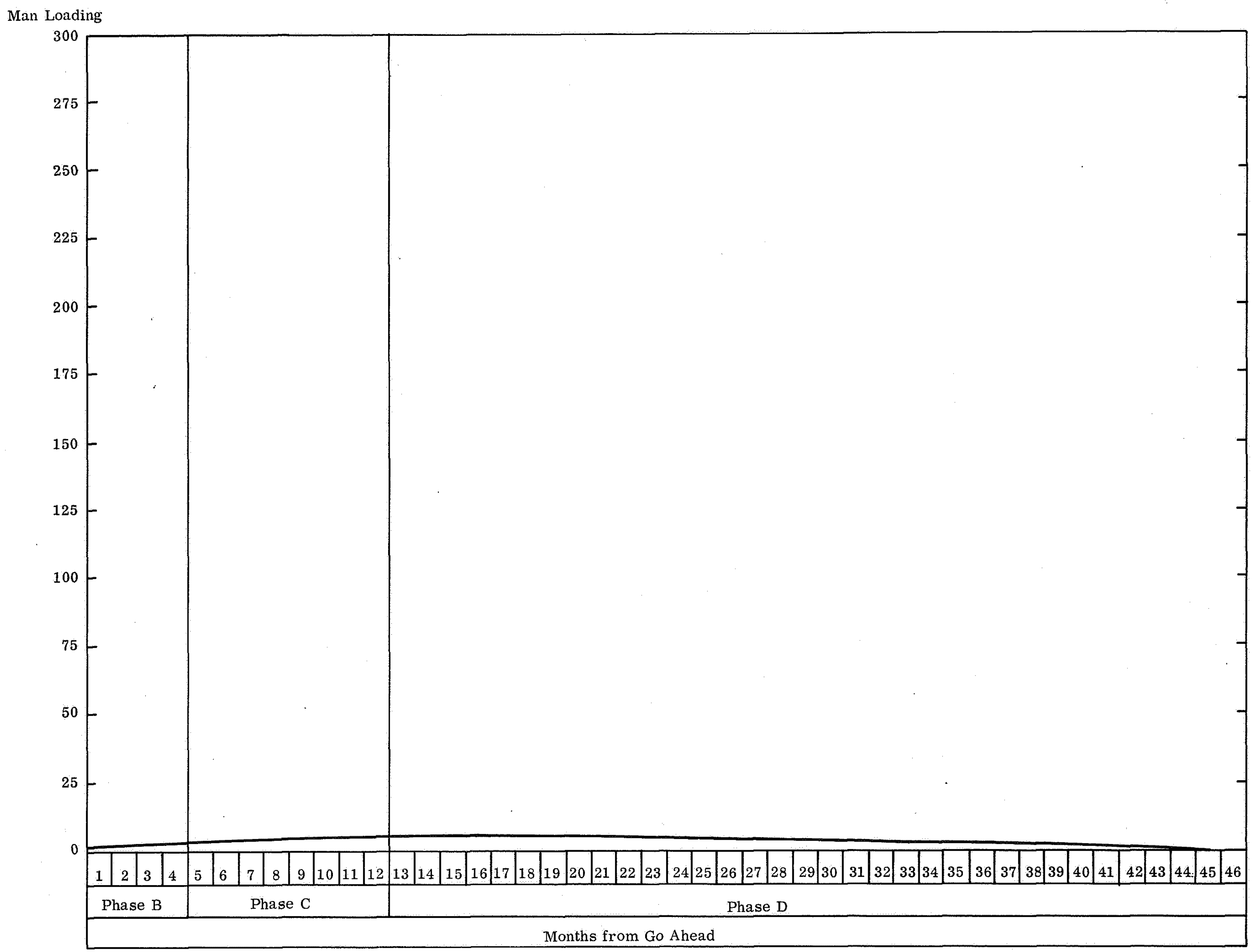


Figure 14.11. Reliability Manpower Loading

Integration of these goals with associate contractors and NASA - MSC are considered to be major milestones during the OMLFV Program.

14.6.4 Reliability Management Control

The system for management control and audit of the reliability program, accomplishes the following:

(a) Identifies each reliability task with the organizational element responsible for its execution, detailed time-phasing data and reliability milestone identification.

(b) Provides, for each reliability task, a detailed listing of manhours, materials, facilities, services and support, with associated costs by time-phase.

A detailed overall plan for management control and scheduling of the reliability effort is included in the Reliability Program Plan. Progress in implementation of the reliability control system are reported periodically as a separate section of the financial and management reports.

14.6.5 Reliability Program Review

Formal reviews of the reliability program assess its progress and effectiveness and determines with NASA the need for adjustments or changes. These reviews are scheduled at major milestones in the program and also periodically as prescribed by NASA - MSC or as requested by the contractor. These reviews are documented by the contractor and resultant revisions to the OMLFV Reliability Program Plan are submitted to NASA - MSC for approval within 30 days following the review. Reliability program changes within the scope of work of the contract are implemented within time periods agreed upon at the review meeting.

14.6.6 Reliability Test Support

The integrated test program will evaluate all aspects of the performance capability of the OMLFV system and its elements. It includes evaluation of reliability throughout the tests at system level and lower levels of assembly.

Evaluation, compatibility, PFRT, experimental, qualification, receiving inspection, and production functional tests on site are monitored by the reliability group to determine the reliability impact of equipment failure. Test operations are monitored and reviewed to determine overstress conditions that could degrade hardware reliability. Test plans and procedures are reviewed to ensure that tests are statistically designed for retrieval of valid reliability data.

14.6.7 Supplier Reliability Program Control

Subcontractors and suppliers are required to meet the reliability requirements of the overall system. All subcontracts include provision for review and evaluation of the subcontractor's reliability effort by NASA - MSC or its representative.

The reliability of all components obtained from suppliers who are not required to maintain a formal reliability program are controlled by specifications similar to those prescribed in NASA -NPC 250-1 for parts. A listing of these items and the control provisions are included in the formal reliability program plan, and pertinent qualification test and inspection requirements are prescribed in the quality program plan.

Reliability program requirements are imposed on the vendor or subcontractor by use of the Source Control Drawing and the Design Procurement Specification to ensure that his reliability program is compatible with the contractor's. Periodic surveys of the supplier's facilities are made with Quality Control, to assure the intent of the reliability and quality control requirements are understood and being complied with. If a new vendor or subcontractor is to be considered, his capability is surveyed prior to request for proposal. Reliability control capability and corrective action policy are included in the survey criteria.

14.6.8 Reliability Demonstration Program

Demonstration of the system reliability is achieved by analytical methods utilizing test data derived from all phases of the program. Reliability engineers support the test organization and witness all tests on hardware to assure maximum utilization of test data for reliability demonstration purposes.

14.6.9 Failure Recurrence Prevention

Failure analyses are conducted on all critical, major, and repetitive minor failures. These analyses include description of failure, failure mode, failure mechanism and failure cause. Cause of failure is identified as due to design, materials, production processes or controls, environment, or human error. This information is used as a basis for implementing or recommending corrective action to the design or documentation as applicable.

A central data center is maintained for reliability and quality control data, defect reports, etc. and assures compatibility of subcontractor data sources for the accumulation and dissemination of reliability and quality control data. This effort includes the collection, processing, storage and retrieval of applicable data extracted from records established and maintained from incoming inspection through to final acceptance test and field use data.

The Failure Recurrence Prevention system in conjunction with configuration control should provide traceability from the highest assembly to the lowest serialized

component and show serialized assembly effectivity for all design changes. In addition the system should be capable of providing tabulations, trends and summaries of historical data relating to component, assembly and end item configuration including drawings and specifications. This information is updated and maintained current throughout the life of the contract.

Major components are evaluated to determine the failure effects on subsystem and system performance. This analysis identifies the failure modes of each major component as well as determines the critical items and limited life items. The critical and limited life items lists are distributed to design, process control engineering, and quality engineering for proper action and follow-up throughout the production cycle.

All tests are considered to be valid reliability tests and are closely monitored with reduction of test results to discover failure modes. The test data are utilized to update system reliability estimates used in the math model on a continuing basis. The failure reporting system provides feedback of test, manufacturing, and field data that supplies information for the basis of design changes and corrective action follow-up. Reliability consideration is given to all design changes based on the failure history. This data is provided to Quality, Engineering, Manufacturing, Reliability, and Systems Project, and Project Management personnel as well as the customer and associate contractors.

Failure analyses are conducted of each critical and major failure encountered in testing, manufacturing, storage, and field use. Reliability Engineers receive copies of all failure reports and are responsible for follow-up of all failure analysis. Results of these failure analyses are published in the monthly status reports and summarized in the quarterly report to NASA-MSC.

14.6.10 Reliability Reports

Periodic progress reports are submitted to NASA - MSC and include ground test results, critical items list, limited life items list, math model status, failure summaries, description of reliability problem areas, and overall program status. A task and problem Logbook is maintained to inform reliability management of program progress and to assist in customer reviews.

14.6.11 Parts Reliability

The reliability organization participates in the selection and application of components to the OMLFV. Such participation includes studies consistent with subassembly failure rate apportionments and application data.

14.6.12 Failure Mode and Effect Analysis

Analysis of conceivable failure modes and their effects on the system are conducted during the design phase to identify potential system weaknesses.

14.6.13 Maintainability, Safety and Elimination of Human-Induced Failure

System designs are evaluated to insure the inclusion of maintainability, safety and the elimination of human-induced failures.

14.6.14 Reliability Indoctrination and Training

A reliability indoctrination and training program is conducted for the personnel specifically assigned to the contract work. In particular it applies to manufacturing personnel to assure the level of workmanship required to avoid degradation of the inherent reliability designed into the hardware. It also applies to personnel handling the hardware at all stages of construction, including the packing and packaging for shipment.

14.7 MANUFACTURING PLAN

14.7.1 Manufacturing Schedule

The manufacturing schedule for the OMLFV is shown in Figure 14.12 and the manpower loading to accomplish the program is shown in Figure 14.13. The Phase C activity includes only that required for long lead test items and the planning necessary for the start of Phase D where the major manufacturing activity is accomplished.

In Phase C, test hardware for development and design verification testing is manufactured for the rocket engines, propellant tanks, and bipropellant throttle valves. Test fixture design requirements are determined and the design and fabrication of tools for DVT fixtures for long lead items is accomplished as well as that design and fabrication of PFRT test fixture tools necessary for the initiation of long lead item PFRT at the beginning of Phase D.

In Phase D, the lunar vehicle and earth flight trainer vehicles are scheduled as parallel efforts to contain the entire program within an approximate 3-1/2 year span. All trainer vehicles are delivered prior to the time that the first of the six deliverable lunar vehicles is completed. The early months of the program require procurement of raw materials and outside purchased parts as soon as design confidence is established. Concurrently, planning sheets are processed to initiate tool design and fabrication activity. Design, procurement, and the beginning of the fabrication activity overlap. A priority list of design and procurement items is compiled to provide for a proper sequence of parts availability.

The fabrication of subsystems test components for development/DVT test starts in the sixth month with final deliveries in the eighteenth month. Complete test vehicles are scheduled for the 16th, 17th and 18th months.

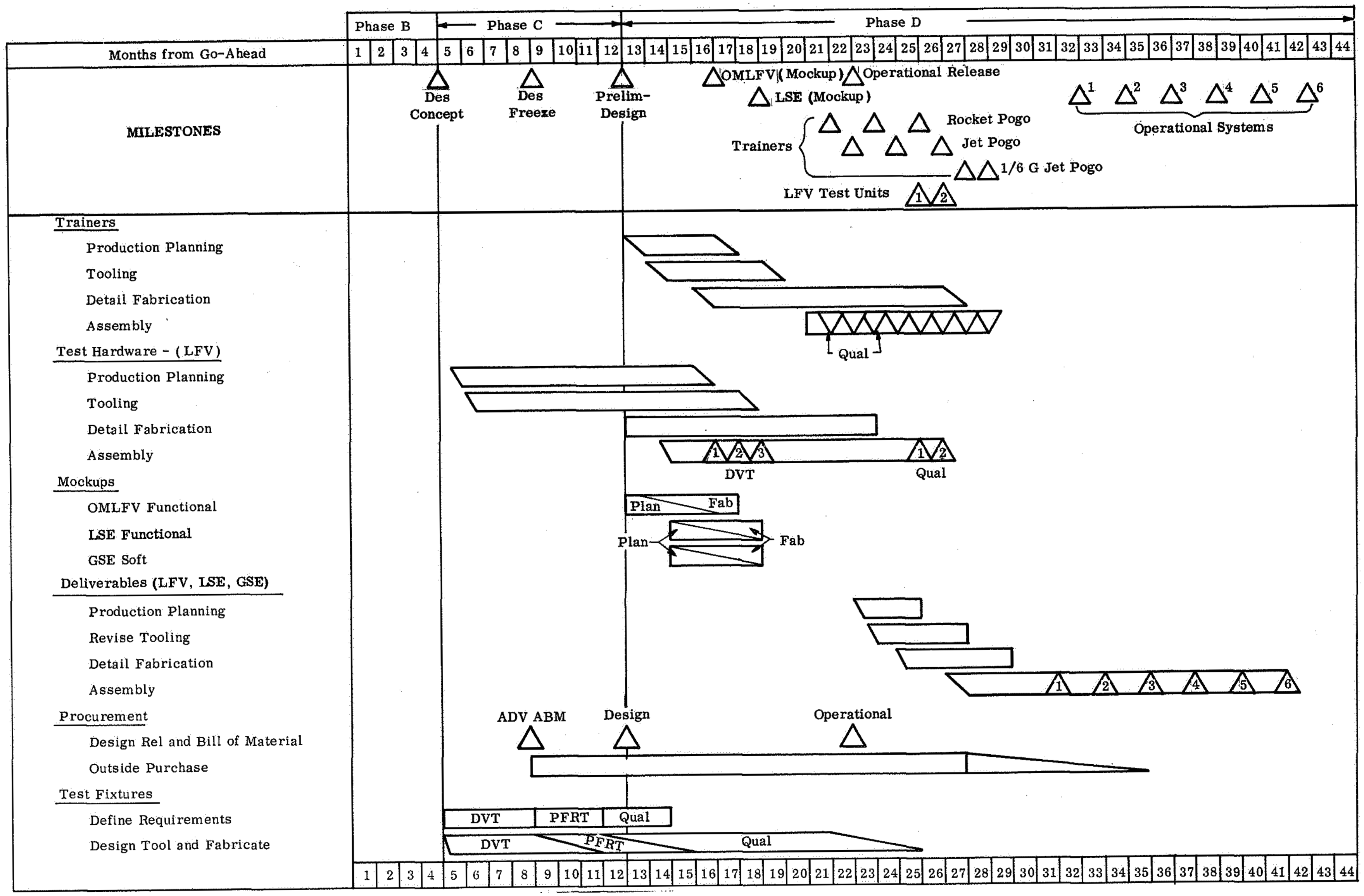


Figure 14.12. One Man Lunar Flying Vehicle Manufacturing Plan

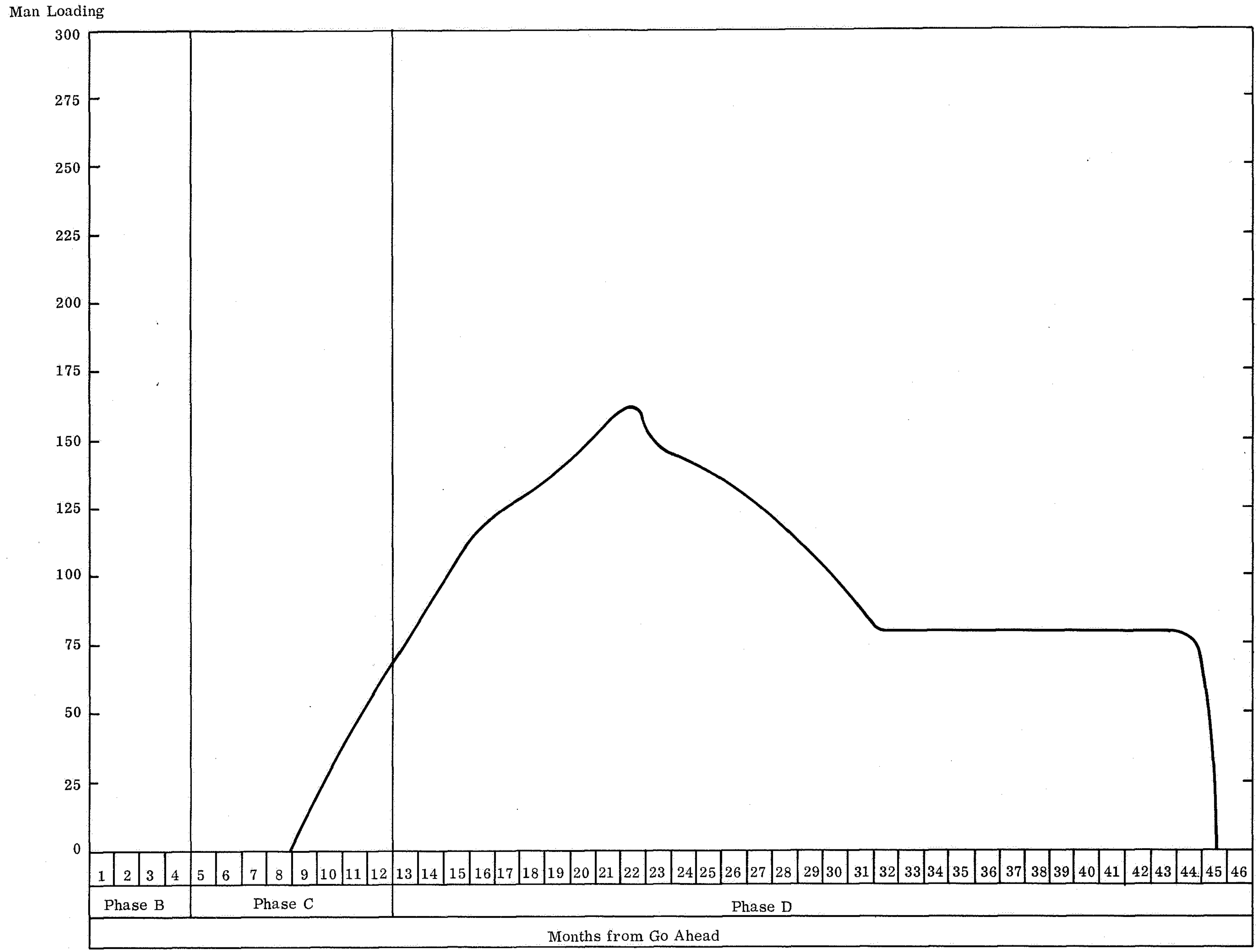


Figure 14.13. Manufacturing Manpower Loading

Subsystems hardware for PFRT tests is scheduled for delivery completion in the 18th month.

Complete vehicles for qualification tests are scheduled to be delivered in the 25th and 26th months.

Fabrication and assembly of the six deliverable systems is scheduled to start in the 23rd month with the completion of the first vehicle in the 31st month. All six deliverable vehicles are scheduled for completion by the 42nd month.

Spares requirements are factored into the program so that they are fabricated in-line with other program hardware.

The first rocket trainer for free flight is scheduled for delivery in the 21st month with the last jet powered trainer scheduled for delivery in the 28th month.

14.7.2 Manufacturing Responsibilities

14.7.2.1 Manufacturing Management

The manufacturing department for the OMLFV program acts as the primary control for all production operation and has the responsibility of producing products of a satisfactory quality and cost such that schedule requirements are reasonably adhered to.

The project organization for the OMLFV project is assigned a Manufacturing Director. Reporting to the Manufacturing Director are a Manufacturing Control Supervisor and Manufacturing Engineering Supervisor in functional departments.

The Manufacturing Directors responsibilities include: (1) ensuring efficient operation and integration of all functional departments of OMLFV manufacturing; (2) meeting delivery schedules; (3) budget control of costs pertaining to manufacturing; (4) coordination of procurement of raw materials; (5) shipment of customer hardware; and (6) off-site support requirements and coordination. In addition, he shall have coordination control with functional departments for assurance of conformance with technical requirements and all supplies, materials, and products.

The authority of manufacturing control, under the direct control of the OMLFV Manufacturing Director crosses functional areas which include production control, cost control analysis, and manufacturing standards. In this capacity, manufacturing control is responsible for: (1) control of scheduling of tooling for completion requirements; (2) release of material requirements; (3) issuing of shop orders; (4) control of parts during fabrication assembly and stock; (5) manufacturing hardware details and assemblies in the most efficient manner; (6) continuous monitoring of the engineering released drawings for purposes of cost control analyses; and (7) assignment of standards to all manufacturing operations performed at the contractor's facility.

In addition, coordination and integration requirements include:

(1) Inspection on receipt to assure conformance with technical requirements, all supplies, materials and products.

(2) Assuring that raw material conforms to applicable physical, chemical, and other technical requirements.

(3) Assuring that all machining, wiring, batching, shaping, and all basic production operations of any type, together with all processing and fabricating of any type, is accomplished under controlled conditions.

(4) Providing requirements for production documented work instructions, adequate production equipment, and any special work environment.

(5) Determining and providing physical examination for measurement or test of material or products associated with work operations and controlled conditions.

(6) Establishing and providing an inspection and monitoring process to include a system for final inspection and test of completed products.

Manufacturing engineering, under the direct control of the Project Manager, will be responsible in the areas of production engineering manufacturing processes, manufacturing techniques, and manufacturing R&D. Tasks specifically included are: (1) preparation of the manufacturing and tool plan; (2) design and fabrication of tooling for the OMLFV project; (3) review of engineering design concepts for producibility; (4) preparation of all manufacturing processes and material specifications; (5) development of new manufacturing techniques and capabilities; (6) institution of new machinery and equipment into the manufacturing capabilities of the contractor; (7) fabrication of development hardware; and (8) research and development of manufacturing engineering processes necessary for the successful completion of the project.

14.7.2.2 Production Control

Production control begins with the detail planning initiated by the limited budget Departmental Work Instructions (DWI) in which the Project Manager requests work package preparations. Work package development in the functional departments is concurrent with the overall manufacturing schedule, the first article flow plan, and the manpower loading plan. Individual detailed plans and schedules are prepared with the work package.

The complete manufacturing effort is portrayed in the production master schedule with each element of manufacturing plotted against a calender corresponding to the program master schedule maintained by Program Planning and Integration. The schedule identifies required dates for specific items of tooling, procurements, fabrication, etc., but is not used as a detailed operating schedule. The program work packages as approved are manloaded in all departments in compliance with this

schedule. Departmental budgets and load charts are developed to show the accomplishment of the work within the indicated time span. This is performed by the use of a line of balance chart on critical items and milestones.

Manufacturing costs and schedules are controlled by the use of: (1) daily manpower controls; (2) weekly manufacturing performance analyses; and (3) weekly contract status reports. Weekly tabulations reflecting (1) total parts completed, (2) the actual hours used for the manufacturing of these parts with associated work orders, and (3) the standard hour value of the part are furnished. From this information a weekly manufacturing performance analysis by the responsible foreman is furnished to the Factory Manager, Manufacturing Director and foremen for review. Excessive variances are investigated to determine the cause and corrective action to be taken. Estimates of hardware completion are also computed weekly and furnished to the Manufacturing Director for review. A weekly contract status report showing expenditures to date, including commitments, is furnished to the Manufacturing Director. Review of these reports identifies the cause of any delays and corrective action can then be initiated.

14.7.2.3 Operations Control (Refer to Flow Diagram, Figure 14.14)

Prior to release of engineering design, Production Engineering reviews preliminary layouts for manufacturing capability and determination of make-or-buy policy. Upon release, manufacturing provides a follow-up review for producibility defining:

(a) Make-or-buy
(b) Applicable manufacturing practices
(c) New production methods and processes
(d) Equipment and facility requirements
(e) Anticipated problem areas and remedies

After review and coordination by the Production Engineering Department, the design package moves to Production Control. This department schedules and budgets the project and releases production orders to Production Planning as required to conform to the established scheduled commitments. In addition, it is the responsibility of Production Control to acquire as necessary additional equipment or facilities to accomplish the production goals.

From advanced bills of material or drawing bills of material approved by Production Engineering, the Material Control Group procures and stocks the necessary materials to meet the demands of the project on a forecast basis. When the production order is released by Production Control, Material Control receives a material requirement order which releases the material to the floor.

Tool requirements initiated by Production Planning take the form of tool orders. These orders contain the part number, tool code, tool name, estimated manhours for tooling, scheduled completion date and explanation of the tool. These tool orders are

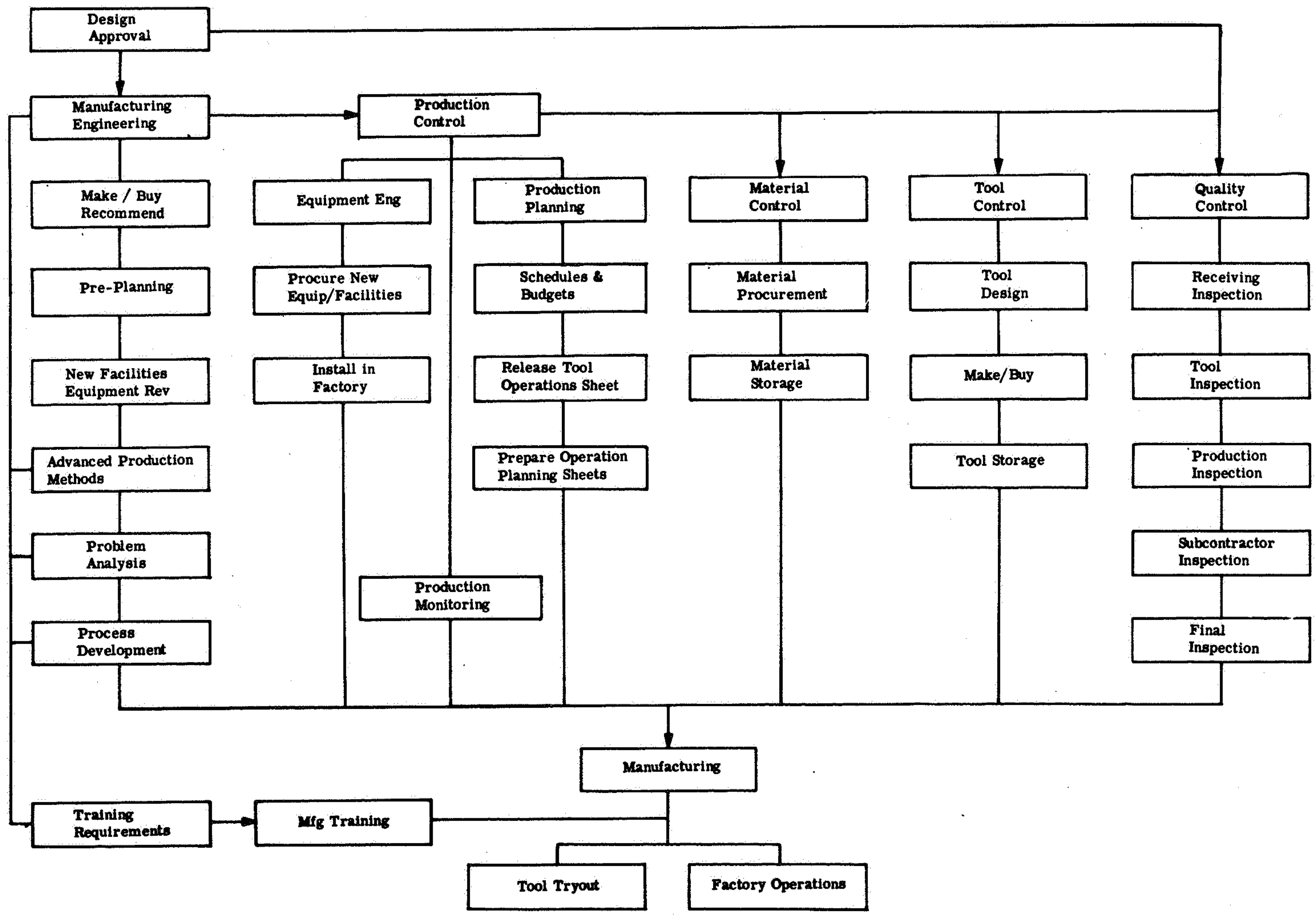


Figure 14.14. Manufacturing Program Control Flow Plan for One Man Lunar Flying Vehicle

released by Production Control. The order is sent to Tool Control for design, and make-or-buy. Completed tools are inspected and approved by the Quality Control Department, identified, and stored until required.

Production Planning prepares operation sheets for the factory for the fabrication and assembly of hardware. These sheets include specific fabricating and detailed processing operations. They not only include the individual operations properly sequenced to produce the desired end product, but also reflect the complete manufacturing philosophy upon which the program is based. Tools and machines are called out against individual operations as are standard setup and flow times. Engineering changes are included as revisions to the operation sheets controlled by revision number. Changes are initiated on an operating sheet change form controlled by Production Control.

Completed operation sheets are sent to the Manufacturing Industrial Engineering Department for the application of standard times. The sheets are then forwarded to Manufacturing Planning Control for mating with the shop orders and filed until the raw material and tooling are available, at which time they are released to the shop according to the predetermined release plan. Prior to release by Manufacturing Planning Control, the individual orders are scheduled according to their requirement as called out on a master manufacturing schedule.

Once released, the order becomes the responsibility of Production Control to monitor its progress through all phases of manufacturing to the completion of the end item. Production Control follows the physical hardware reporting status by way of a line of balance chart and supporting documentation. It is also the responsibility of Production Control to assure the foreman of each functional department that he has the latest revision of all operation sheets, process specifications, engineering drawings, and other documents he may require. It shall also supply floor inspectors with the proper paperwork to enable Quality Control to perform their functions. The responsibility of Production Control ends with the acceptance by the Quality Control Department of the final operation and the detail or assembly is delivered to the customer.

14.7.2.4 Manufacturing and Quality Control Coordination

Reliability practices within the manufacturing operations are implemented through quality control by the establishments of approved documentation and control procedures for manufacturing processes such as process specifications, standards, etc., using applicable standards and specification MIL-R-27542A as guides. Operating procedures are written, giving detailed instructions for proper operation of the process. Daily assurance of compliance is accomplished by the process operator through a system of checks which are documented in a process log. Before he can proceed with the process, at regular intervals and dependent upon the nature of the process, the Quality Control representative audits these logs, verifies their accuracy, documents his findings, and prepares samples for additional laboratory analysis. Thus, he sets up preventative guards against process deviations which might degrade the reliability of the hardware being produced.

Process failures are documented through the failure reporting system fed into the Reliability Group for analysis to provide a measure of parts and components rejection rates and achieved reliability.

Process Standards and Specifications are written by the Engineering Laboratory personnel in coordination with responsible engineering groups. They are approved by all affected departments and the Quality Department. The documents are called out, as required, on drawings and are available in the Manufacturing, Inspection and Test areas. Changes are controlled by the Configuration Control Board and must receive necessary approval (in-house and customer) prior to implementation. Upon approval, effectivity is assigned and the changes are issued. By exercising full control in this manner, there can be no degradation of reliability due to manufacturing processes.

Corrective action related to manufacturing deficiencies and nonconforming material is documented in material review procedures. However, critical and major defects (those which may affect reliability) are evaluated before and after changes to determine the impacts of any proposed changes.

14.7.2.5 Tooling Policy and Plan

The manufacturing plan is predicated on the basis of a single set of minimum type, general and special tooling because of the limited number of articles planned for delivery. This single set tooling concept, while stretching out complete system delivery schedules, reduces manufacturing costs.

A determination of the type and scope of tooling necessary is based on component subsystem, system and assembly complexity, tolerance, interchangeability, reliability, spares, and the economics of production.

14.7.3 Facility Requirements

Complete manufacturing facilities, including tooling, machining, heat treat, plasma spray, cleaning, and plating are needed to meet the requirements necessary to fulfill the manufacturing parameters pertaining to the OMLFV system.

The basic machine tools required for the vehicle are common to the Aerospace Industry. In addition a minimum level "C" clean room is required in the assembly area. All propulsion systems including, tanks, propellant lines, nozzles, injectors and other components are designated as clean room level assemblies together with specified electrical systems. Past experience indicates this requirement must be rigidly adhered to.

It is estimated that 40,000 sq ft of plant space is required for component fabrication, subassembly, subsystem assembly, final assembly and acceptance inspection and checkout area, for the OMLFV and associated GSE and LSE. This does not include an estimated 14,000 square foot increase which is an assumed normal subcontract approximation. Following is a breakdown of the manufacturing area requirements estimate:

		Area sq ft
1.	Machine Shop Requirements	10,000
2.	Fabrication - Sheet Metal Bench and Equipment Area	8,000
3.	Bench Subassembly	3,000
4.	Weld Assembly Area	1,000
5.	Subsystems Assembly (Clean Room)	3,000
6.	Final Assembly	8,000
7.	Acceptance Test and FACI (Hot Firing is Additional)	3,000
8.	Shipping Area	1,000
9.	Processing (Clean, Heat Treat, Bond)	3,000
	Total	40,000

14.8 QUALITY CONTROL PLAN

14.8.1 Introduction

A Quality Program Plan for the OMLFV should be in consonance with the requirements of NASA Quality Publication NPC 200-2 and NPC 200-3 to the extent shown herein. Quality Assurance procedures for inspection planning, data collection, corrective action, measurements, etc. will incorporate reliability requirements for evaluating progress toward the OMLFV program reliability objectives. Reliability requirements and integration are defined in the Reliability portion of this plan (Section 14.6).

14.8.2 Basic Requirements

Product Assurance maintains an effective and timely program plan in accordance with Quality procedures such as Quality Procedures Manual, Inspection Bulletins, and Inspection Instructions encompassing such functions as are noted in the ensuing paragraphs. Quality and Reliability will review drawings, specifications, procedures, purchase requisitions/orders, planning sheets, etc. to assure that the requirements which are established are adequate.

Compliance is assured through an integrated system of inspection and test under the constant surveillance of Quality Assurance personnel and under the direction of a Product Assurance Director. Immediately upon noting any inconsistencies between documentation, tooling, and the hardware itself, such problems are reported by inspection and test activities to Quality Assurance and corrective action initiated. Records showing the results of such investigations are maintained by a central data center.

Other procedures instrumental in maintaining an effective Quality System include process specifications, detailed inspection and test procedures, and a corrective action follow-up system which provides for early and prompt detection of actual or potential deficiencies.

14.8.3 Change Control

Design changes are referred to Quality for review and approval prior to release. All released design changes are transmitted to Quality Assurance. Quality Assurance reviews and approves changes to all specifications and technical documents and ensure that the changed characteristics are effected in the applicable quality assurance activities.

Quality is responsible for the audit of the drawing/specification change distribution and for determining compliance to quality control procedures and instructions.

Configuration accountability is maintained throughout the fabrication and test cycles to the point of delivery. Final configuration review is accomplished by the use of inspection records and current configuration requirements.

14.8.4 Quality Control Management

Figure 14.15 shows the general plan for the OMLFV Program, including the various Quality functions throughout the cycle from contract review to completion and shipping of the hardware. Figure 14.16 presents the manloading required to implement the quality control plan.

The Product Assurance Director has direct unimpeded access to higher management. He is responsible for assuring the performance of all elements of the Product Assurance Quality and Reliability Plans on time and within cost. He directs Program Quality and Reliability efforts and coordinates with Customer Representatives on Quality and Reliability matters. The Product Assurance Director draws support from the Product Assurance functional organizations as necessary to meet the Quality Program Plan.

14.8.5 Design and Development Control

The objective is to ensure inherent quality of design and deliverable material and inclusion of workmanship requirements during the design and development phases. The quality assurance effort begins with the review of preliminary design and continues until final design approval to anticipate and prevent potential quality problem areas that may arise during manufacturing, assembly, testing and use of the deliverable material.

All drawings and specifications are subject to Quality Assurance review and concurrence prior to release. The assigned Quality Assurance engineer assists personnel directly responsible for the control of design to avoid quality problems. The purpose of this activity is to:

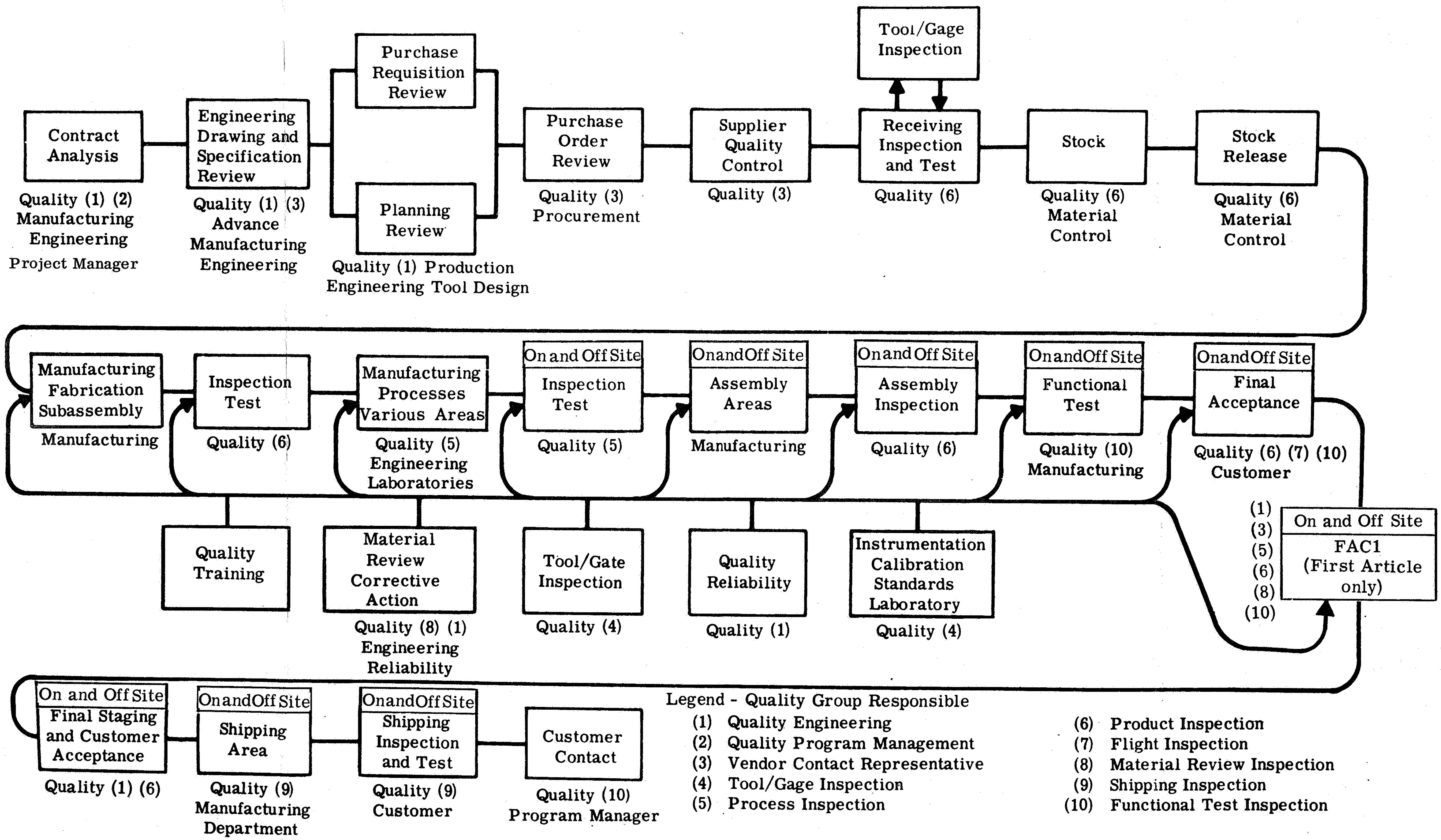


Figure 14.15. Quality Program Flow Plan

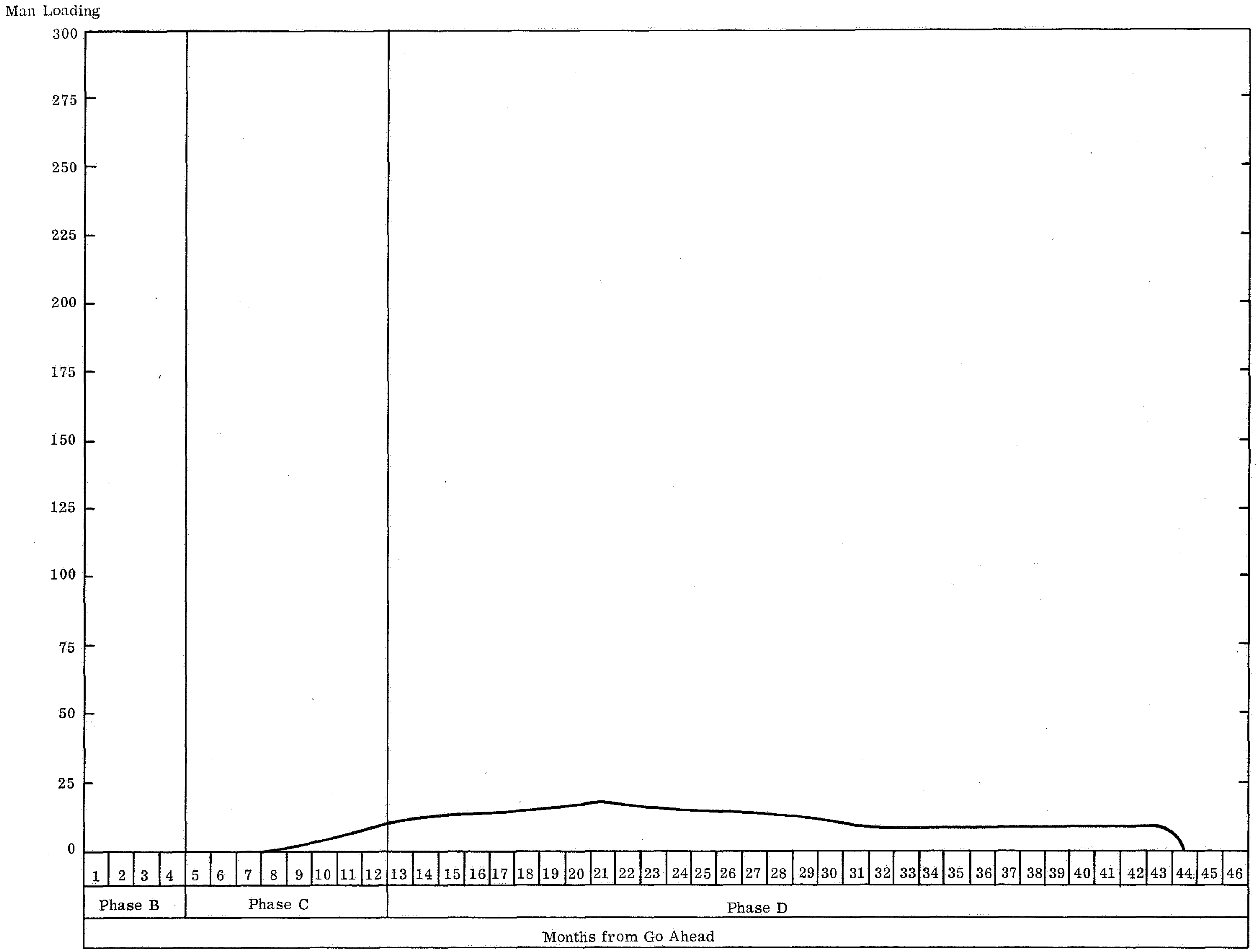


Figure 14.16. Quality Manpower Loading

(a) Evaluate components and processes to determine characteristics that could affect product quality.

(b) Perform tolerance analysis on drawings while considering machine and operator capability and product end-item function.

(c) Determine areas in which nondestructive test requirements should be made.

(d) Determine areas in which inspection tooling should be employed; check the product during manufacturing operations.

(e) Ascertain that specifications and specification callouts on drawings are provided to denote information necessary for special processes and applications.

(f) Determine adequacy and clarity of drawing and specification information for interpretation by inspection personnel.

(g) Provide ample lead time for establishment of inspection points in shop planning.

(h) Ensure that the product being designed is capable of being inspected.

(i) Determine product quality characteristics that must be controlled and coordinated prior to manufacture of fabrication tooling.

14.8.6 Qualification Tests

Qualification Test Procedures and changes thereto prepared by the Engineering Department are reviewed and approved by Quality to assure that all requirements have been included. Qualification tests are performed by the Engineering Department under the complete surveillance of the Quality Department to approved procedures. Inspection controls during the complete qualification test cycle are in accordance with a Quality procedure which, in summary, provides the general instructions for the conduct of the Qualification Test Program. Quality Assurance personnel monitor the complete operation to assure that procedures are available and are maintained current with the latest changes and that all parties concerned are familiar with the procedures. Discrepancy investigation, corrective action, corrective action follow-up and reporting is conducted by Quality Assurance. Final reports for submittal to the customer are prepared by Engineering and approved by Quality.

A cognizant government inspector/engineer and NASA Quality Representative are notified prior to the start of a qualification test program, and also prior to any rework to a system, subsystem or component. Final test acceptance by the government inspector/engineer is required.

A Qualification Status List is prepared showing the qualification status of each part, component, subassembly and higher level of assembly.

14.8.7 Identification

Major fabricated components, subassemblies, and end items are identified and controlled by serial number identification. Related records bear the same serial number as the part. Records are maintained such that all inspections/test may be traced to the inspector/tester accomplishing the operation.

14.8.8 Control of Contractor Procured Material

14.8.8.1 Selection of Procurement Sources

The selection of suppliers and the placing of purchase orders is the responsibility of the Procurement Department. The compilation of a register of approved suppliers, however, is a joint responsibility of Product Assurance and Procurement. Supplier evaluation is based on two criteria: (1) Acceptable Quality Control Systems, and (2) performance which is demonstrated by records of quality history compiled at Receiving Inspection and instances of deficiency detected at the supplier's facility by company Supplier Product Assurance Representatives.

14.8.8.2 Procurement Documents

Basic Technical Requirements

Purchase Requisitions are reviewed by Quality Engineering to ensure that all specific quality and reliability requirements, such as compliance to applicable specifications, qualification test data, technical data, process control, source inspection, identification, handling and storage of materials, are made a requirement of the Purchase Order.

Government Source Inspection Requirements

Government Source Inspection Requirements are incorporated into the purchase order as determined necessary by the cognizant government organization.

Contractor Source Inspection

Contractor source inspection requirements are included in the purchase order as determined necessary during the purchase order review and depending upon the type and complexity of the material being procured.

Subcontractor Quality Programs

Subcontractors are required to comply with the following:

(a) Supply LFV contractor with a copy of their inspection and test procedures.

(b) Maintain quantitative test log records showing the results of all tests performed as compared with the test requirements.

(c) Supply information relative to problems encountered which may affect parts delivered.

(d) Provide complete corrective action reports covering reported discrepancies, corrective action taken and effectiveness of corrective action.

(e) Supplier of major components are required to comply with the requirements of NPC 200-3 and/or parts of NPC 200-2.

Purchased Raw Materials

Suppliers are required to supply chemical and physical analysis test reports as determined to be a requirement during the purchase order review and depending upon the material being supplied.

Raw Materials Used in Purchased Articles

Suppliers of major components are required to forward certification of the raw materials' chemical and physical characteristics.

Evidence of Supplier Inspections Performance

Specific quality requirements are included in the purchase order to the suppliers requiring that quantitative data (inspection and test) be compiled showing the results of inspections and tests performed. This data to be identified with the unit serial number and inspection and date stamps showing inspection acceptance. The supplier is required to supply this data with each item shipped.

Identification, Preservation and Packaging

Suppliers are instructed to use good commercial practice for preservation, packaging and packing, properly and clearly identified with the article being shipped.

Age Control

Age control of materials and articles having definite characteristics of quality degradation or drift and/or use is maintained. Materials and/or components are marked accordingly.

Resubmission of Rejected Material

Material being resubmitted, following repair and/or rework, is clearly identified by the vendor's shipping document which cross references the contractor's purchase order which returned the part.

Article of Supplier Design

The supplier is advised by purchase order and by Specification Control Drawing that no changes are to be made to the material being supplied without prior approval of the contractor.

14.8.8.3 Contractor Source Inspection

During purchase order review, determination is made as to the need and extent of contractor source inspection, based upon the complexity of the article being procured, and whether in-process controls are of such a nature that the quality of the article cannot be determined solely by inspection or tests of the completed articles.

14.8.8.4 Receiving Inspection

The following inspection criteria is required upon receipt of procured articles:

(a) Review of all data including certifications, quantitative test data to assure compliance with specification requirements.

(b) Visual inspection to determine that articles have not been damaged.

(c) Dimensional inspection as required to verify vendors data.

(d) Functional test of the article of verify functional characteristics.

(e) Periodic disassembly of components as appropriate for verification by Quality Assurance that the details comply internally with the specified requirements and are of a high level of quality.

(f) Inspection and test equipment is available at the contractor's plant with the complete capability of performing the inspection and tests required for the program. Items requiring age control are marked and stored in a manner so as to assure complete control.

(g) Data provides acceptable verification of the physical and chemical properties. Physical and chemical analysis is performed by the contractor as required to verify supplier's data.

(h) Only accepted stock is allowed to enter the stock room. Rejected material is moved to a bond room for disposition.

14.8.8.5 Identification

As established by drawing or purchase order, components received from suppliers bear a serial number identification, associated records bear this serial number. Raw material received is identified.

14.8.8.6 Failure and Deficiency Feedback

Failures experienced at the contractor on vendor supplied materials is reported to the vendor on a corrective action form prepared by Quality and forwarded to the supplier with a copy to Engineering. The vendor performs an investigation into the cause of the discrepancy, and initiates the corrective action necessary to solve the problem.

14.8.8.7 Supplier Rating

Quality maintains a program whereby each supplier is evaluated and rated in accordance with his performance.

14.8.8.8 Coordination of Contractor-Supplier Measuring and Test Equipment and Standards

Coordination with the vendor to determine his inspection and test equipment capabilities and controls as compared with those of the contractor are provided during surveys of this facilities.

14.8.9 Control of Contractor-Fabricated Articles

14.8.9.1 Conformance Criteria

Quality Control of the fabrication process is initiated in the design review stage by determining specific need for inspection, inspection tooling, control of processes measuring and test equipment.

14.8.9.2 Inspection and Test Planning

Process sheets are prepared and contain the required operations in a sequential order necessary to process a detail part component, subsystem or system. Inspection/test operations are called out as determined necessary. Wherever special inspection/test procedures are pertinent they are referenced opposite the applicable inspection/test operation. Test procedures and inspection instructions provide information to the inspector/tester regarding the specific characteristics to be measured, the allowable tolerances, conditions under which the inspections and/or tests are to be accomplished and the criteria for acceptance or rejection. Quality Assurance verifies that the applicable inspections and tests have been performed in accordance with the requirements.

14.8.9.3 End-Item Tests and Final Inspection

Each end item is reviewed by inspection prior to test, to assure completion of all inspection operations, and that it is of the required configuration. Acceptance tests are performed to a procedure prepared by Engineering and approved Quality. Final inspection following test acceptance consists of:

(a) Review of acceptance test records and compilation of records for transmittal to the customer.

(b) Final configuration review to current engineering configuration list.

(c) Review of all inspection records.

(d) Final inspection and preparation for delivery by the Quality Inspector, customer, and government inspection.

(e) Preparation of all pertinent records for transmittal to NASA.

Modifications are handled in accordance with rework procedures, modification description, and additional test requirements resulting from the modification.

14.8.9.4 Fabrication Controls

Production Control process sheets are stamped by the stock room inspector verifying that the correct material is being supplied. Records of serial numbered parts are carried forward to next assemblies. Materials and articles having definite characteristics of quality degradation or drift with age/use are marked to indicate the date and test time, and the remaining life of the article.

Special clean room procedures are implemented on the OMLFV program for the control of critical components, with the capability of controlling the environment to a five micron level with 0.20 inch of H_2O pressurization and a temperature control of 70 $\pm 5^{\circ}$F.

A complete set of Process Control Specifications are provided for use in controlling all processes, per NASA Publication NPC 200-2. Periodic analysis/inspections are made as required by process specifications of materials, solution, and equipment used in the various processes. Quality control ensures that all processes employed in the fabrication of deliverable material are controlled in accordance with established procedures. To maintain process control, quality assurance monitors process operations to ensure conformance with predetermined techniques, procedures and standards and conducts certification programs for all personnel engaged in special processes.

14.8.10 Material Review/Approval of Contracting Officer/through Nonconforming Material

Quality Control System procedures provide for the review, control and disposition of nonconforming material. Each nonconformance is reviewed, and action taken to prevent recurrence of similar discrepancies. Written requests for approval by NASA are made if the nonconformance adversely affects safety, reliability, durability, performance, interchangeability of parts or assemblies, weights, or basic objectives of the contract. Review and dispositioning of material is accomplished by a Material Review Board consisting of a material review inspector, approved engineer(s) and government inspector.

Material, determined by preliminary review, to be suitable for "return for completion" or "rework to drawing" is not submitted for formal material review action. A permanent record of rework is maintained and accompanies the component.

14.8.11 Inspection Measuring and Test Equipment

Inspection measuring and test equipment is periodically calibrated to established procedures and at scheduled intervals based upon extent of usage and wear characteristics derived from records maintained by the calibration labs. Calibration facilities at the contractor's plant must be environmentally controlled to the level commensurate with the equipment under calibration. Standards used for the calibration of all equipment must be directly traceable to the National Bureau of Standards. Records are maintained showing the results of calibrations.

14.8.12 Inspection Stamps

Inspection stamps identifiable by serial number and characteristic of the function are provided, traceable to the individual to whom the stamp is assigned. All parts or tags accompanying parts processed are stamped depending upon the applicable operations.

14.8.13 Certification of Personnel

Personnel responsible for controlling special processes or for performing fabrication and inspection operations of a specialized nature having a significant effect upon quality (such as welding, soldering, wiring, radiography, magnetic particle, dye penetrant, and bonding) are certified.

14.8.14 Data Reporting and Corrective Action

Quality Engineering personnel ensure that corrective action is taken on each major repetitive or critical discrepancy:

(a) Critical Discrepancy - Any discrepancy which can cause the system to operate outside of the limits designated in governing specifications, create a safety hazard, or cause a mission abort.

(b) Major Discrepancy - A discrepancy that is not critical but which could degrade the reliability of the system because of cumulative tolerance build-up and/or significantly reduce the useability of the item for its intended purpose.

(c) Repetitive Discrepancy - Any discrepancy which occurs more than twice on any given item or a group of like items.

Data is reported in the form of inspection reports, failed equipment/replacement reports, component test reports, material disposition reports, and inspection records covering fabricated, assembled and procured items. A monthly quality

record status report is prepared providing narrative comment, recommendations, tabulations of pertinent data and summary or corrective action, etc.. End item test reports are submitted with each delivered item consisting of:

(a) Test log showing the quantitative results of all acceptance tests as compared with the specification/procedure.

(b) The configuration of the item.

(c) Copies of all MDR's associated with the item..

(d) A record of all approved deviations and authority for approval (TWX, wire, letter, etc.) if applicable.

(e) Contractually approved changes, if applicable.

(f) Total operating hours (time) for each system or subsystem.

(g) Contract End Item Specifications complete and approved.

14.9 LOGISTIC PLAN

14.9.1 Introduction

The logistics plan encompasses the ground support equipment, spares, field service, technical manuals, depot support, training support, and mission support for the OMLFV program.

The various logistics functions are covered under the following categories:

a. Spares

b. Field Service Support

   (1) Associate Contractor Test Sites

   (2) Training Sites

   (3) Launch Site

   (4) Lunar Mission Site

c. Training

d. Technical Manuals and Publications

e. Operational and GSE Support

f. Modifications

g. Depot Support

The manpower loading to implement the logistics plan is presented in Figure 14.17.

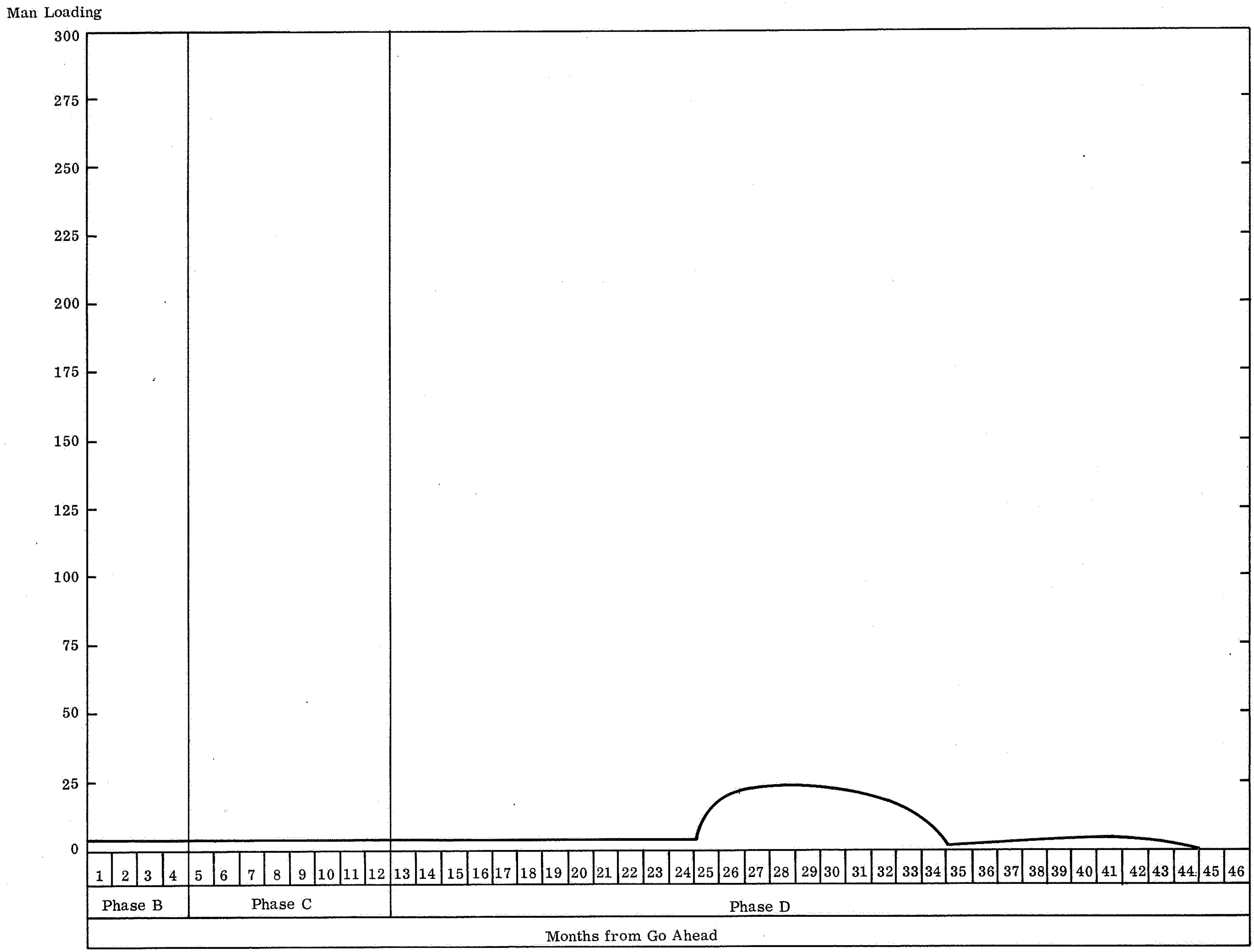


Figure 14.17. Logistics Manpower Loading

14.9.2 Contractor Logistics Integration and Management

The items comprising the logistics support fall in several different groups. A logistics manager, reporting to the project manager, through the program coordinator, is responsible for implementation and management of the program as follows:

a. Coordinates and monitors the spares program.

b. Provides liaison between field, engineering, and program management.

c. Supplies and coordinates flow of material and instructions to technical representatives and vehicles on site.

d. Coordinates in-plant support of site operations.

e. Assists in coordination of the training programs for both simulator and training vehicles.

f. Coordinates and supplies training manuals, technical publications, mating, prelaunch, preflight, in-flight, post-flight, countdown procedures, etc.

g. Provides status control of vehicles, i.e., E.O., modification incorporation and retrofit requirements.

h. Coordinates technical publications revisions and updates program requirements where applicable.

i. Collects, distributes, and integrates failure reports, and status reports in coordination with reliability and quality control.

j. Ensures proper coordination of availability of facilities, personnel, and equipment at various sites.

k. Coordinates closely with OMLFV contractor procurement functions on subcontracts in support of program logistics. This is particularly important in the use of spares lead times and rapid replacement of spares as required.

14.9.3 Logistics Test Support

Integrated OMLFV and LSE mockup prototype and interface tests require time schedules and facilities support coordination and integration with the associate contractors, integrating contractor, and NASA. To accomplish this planning, the following general requirements must be met:

a. Schedules must be planned and established four to six months prior to the activities. Planning and coordination with associate contractors and integrating contractors are finalized and approved two to three months prior to the activity. This ensures proper shipping times of equipment, arrival of personnel, etc., and the activity sites and the availability of the required facilities.

b. All support requirements are identified and approved for the test sites at Houston and the associate contractor's facilities. Final review and approval is made after coordination with associate contractors and NASA.

c. Training schedules and training planning are outlined during this period and these outlines are subject to approval by the integrating contractor and NASA. Technical manual and publication requirements are developed in two steps, preliminary and final. Both preliminary and final copies are subject to approval by the integrating contractor and NASA. Preliminary data for manuals and specifications are coordinated by the logistics department as required.

d. Technical training of off-site personnel begins during this period and the training program utilizes to the fullest extent all development tests and results being accomplished by the OMLFV contractor.

14.9.4 Test and Training Activities

Many of the activities in logistics test support are continued at the test and training sites to support feedback information that may affect the operational lunar flyer and LSE, launch support and mission support activities. The following planning supports the test and training activities:

a. Coordination and integration begins four months prior to qualification tests and the beginning of the training program.

b. To meet the need dates for the qualification tests and the beginning of the training program, detail schedules, support requirements, facility requirements are coordinated on a week by week basis one month prior to the activities.

c. Integration of test data results and reduction begins during the development test program and ends with the acceptance tests of the OMLFV and LSE operational systems.

d. Training of technical representatives for launch and mission support is finalized during the qualification and training periods. Maintenance manuals, support equipment manuals, operator manuals, spares tests, check lists, procedure guides, etc., are finalized and approved.

14.9.5 Astronaut Training

An astronaut training plan for deploying, servicing, flying and navigating the OMLFV is presented in Section 13.0 of this report. The plan describes the objectives, activities, training sequence, and the facilities and equipment required. The training schedule is integrated into the program schedule to fit the availability of equipment and facilities and to complete the first set of astronauts training, except for maintaining proficiency, prior to delivery of the first operational lunar vehicle. The milestone showing the initiation of training is shown on the summary schedule in Figure 14.1.

The training program is refined during each phase of the flyer program and the training is all conducted in Phase D.

14.9.6 Logistics Launch Site Support

This phase of the logistics plan requires close and continuous coordination and integration with associated contractors at the launch site, and with NASA.

Procedures developed during earlier phases require final review and approval, a minimum of three months prior to launch site activities. Mating procedures and check lists developed during the mockup and mating tests are refined and finalized during prototype and qualification testing and utilized during this period. Prelaunch checkout procedures are defined, approved, and published two months prior to launch site checkout. Countdown procedures are coordinated with the associate contractors and NASA and verified and approved three months prior to launch.

Flight and mission analysis policies and procedures are developed in coordination with NASA and the LM contractor. Logistics support during the launch and mission phases requires support of engineering, maintenance, quality, training, test equipment and checkout personnel, both GSE and special, and spares. Maintenance during this phase is anticipated to be on a remove and replace basis to the greatest extent possible.

14.9.7 Logistics Control

The logistics manager provides and supervises the facilities, equipment, shipping, coordination of schedules, training support, spares, etc. Through the logistics manager's administrative section and distribution center, all plans and work information is disseminated to all concerned personnel. Status charts are maintained to keep an up to date status of all testing, training support, work progress, etc. to reveal any slow downs or "bottlenecks" in the progression of the program and to coordinate schedules with program planning and integration.

14.10 COMPOSITE PLAN

The functions of Engineering, Manufacturing, Test and Logistics Support have been integrated in this composite plan. The resultant schedule reflects a minimum risk program with high emphasis on use of qualified space components, conventional materials and manufacturing techniques and extensive component, subsystem and system testing. Figures 14.18 and 14.19 present the Composite Man Loading Requirements and the Composite Plan, respectively.

Engineering development and analysis of the vehicle system as a whole as well as analysis of specific subsystem areas is required to determine design specification and performances. A four month Phase B study is conducted to refine the current flyer concept and to define follow-on Phase C activity. In Phase C a preliminary design phase is conducted in conjunction with the analysis effort so that the necessary

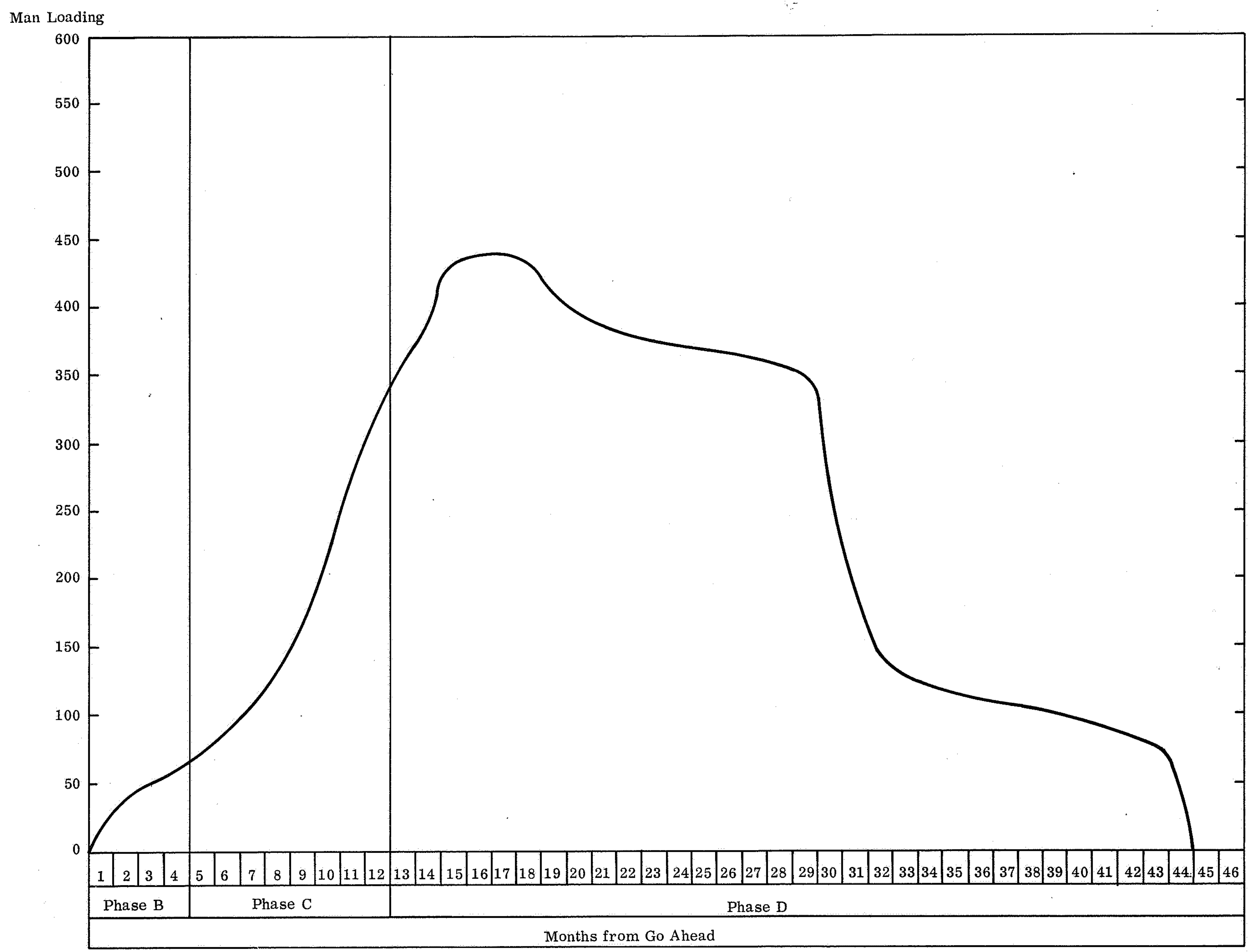


Figure 14.18. Total Manpower Loading

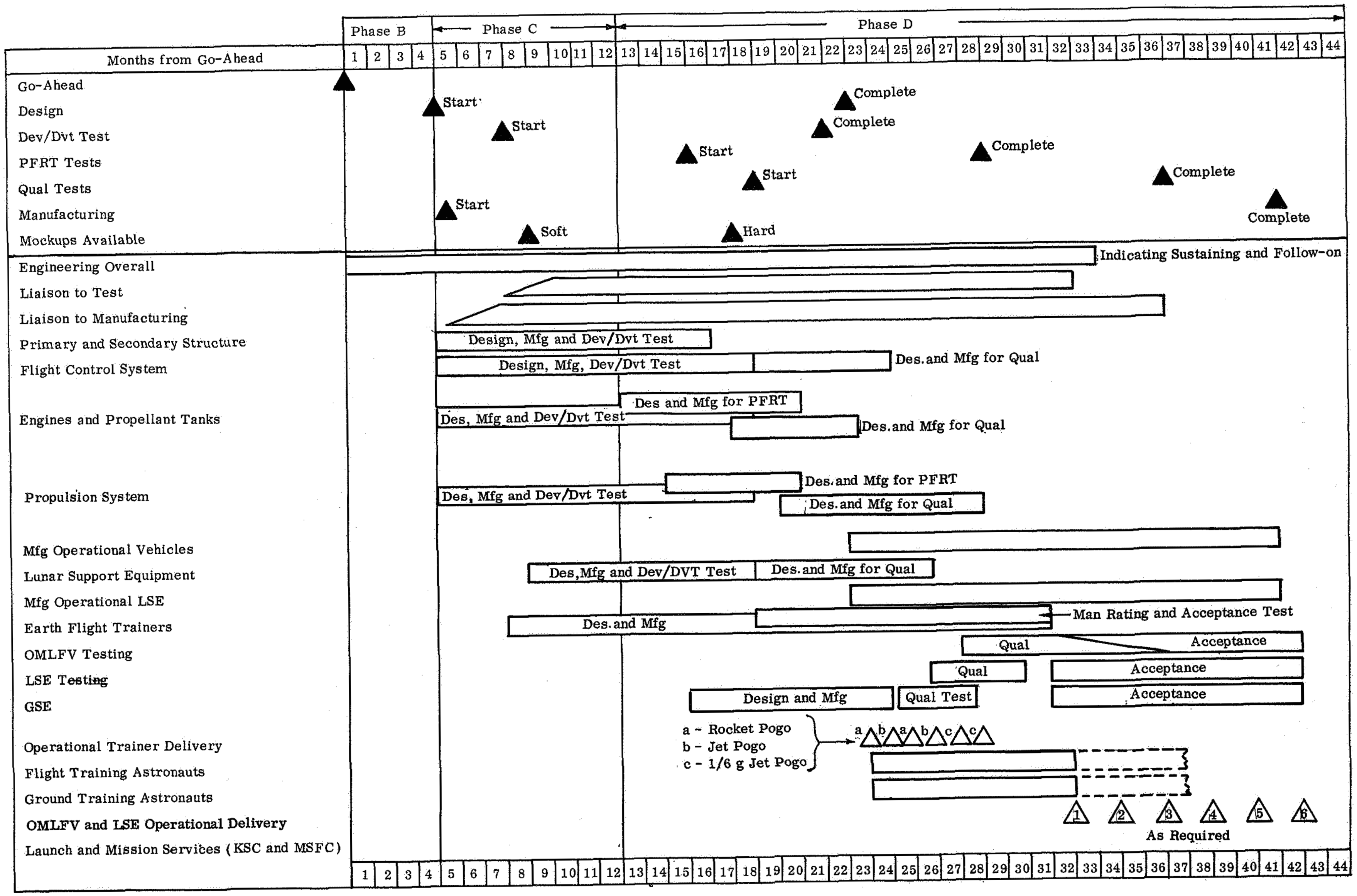


Figure 14.19. One Man Lunar Flying Vehicle Composite Plan

overall engineering information is available to start a coordinated detail design phase. During this phase, long lead-item hardware is designed and manufactured for development/design verification testing.

Phase D starts at the conclusion of the major portion of the development and analysis phase, Engineering detail designs are prepared and released for the fabrication of development hardware for both OMLFV and LSE. The hardware consists of structural elements, propulsion, and flight control subsystem components and LSE. Procurement effort is initiated for the purchase of those parts required for tests, as received, or for assembly with in-house fabricated components. Concurrently, manufacturing produces the necessary planning, tool design and tooling for the hardware fabrication. The hardware is subjected to tests to assure hardware design verification Any design changes found necessary as a result of these tests are incorporated into the component design prior to formal PFRT and qualification testing.

At this point in time, a complete release of the vehicle structure, landing gear, subsystems and components is accomplished for the fabrication of test vehicles, systems and test components. Additional tooling and assembly fixtures are fabricated as required for the buildup of the complete vehicles. The tooling prepared for the initial pieces of hardware will supplement new tooling so that a complete set is available for the OMLFV and LSE test vehicles and systems and associated GSE, and later for the deliverable OMLFV and LSE systems.

There is overlap in the types of testing of each system and several activities are conducted in parallel for timely completion of the program.

Qualified, operational system delivery occurs at the end of 32 months with subsequent deliveries at two month intervals.

14.11 COST PLAN

14.11.1 General Description of System Costs

The cost developed in this section is for a total program starting with an engineering analysis phase and carrying through to delivery of six operational vehicles and six sets of LSE fully qualified for lunar flight. Also included are costs for earth flight trainers, earth support equipment, manuals, mockups, use of government test facilities, and contractor support to the training, launch, and lunar mission operations.

14.11.2 Cost Approach

The OMLFV Program cost was estimated by breaking the vehicle down into hardware end items which are listed in hardware tree form in Appendix D. Costs were estimated for basic operations or functions required to provide each of the end items. Any costs for lower levels of assembly are included.

The estimates for ground support equipment were made by using actual costs for similar type equipment used on other programs. Cost for handbooks, simulators, trainers, and contractor operational support is also included. Costs for project management and documentation functions are shown.

14.11.3 Cost Estimate Guidelines

The considerations and assumptions upon which the accompanying cost estimate is based are summarized in this section. This section also defines certain items, the cost of which has been excluded.

14.11.3.1 General

The cost of integration of the vehicle with external hardware such as LM support equipment, other lunar payloads, and pressure suit is included. The cost of design integration of components into higher level assemblies is included in the basic design cost. Labor costs are presented in manhours and converted to dollars using the following rates which approximate rates in general use by the Aerospace industry. These rates include all overhead, burden, fee, and other indirect appendages.

(a)	Design, Engineering, Documentation Functions	$22/manhour
(b)	Testing Functions	$18/manhour
(c)	Manufacturing Functions	$14/manhour
(d)	Project Management	$22/manhour

For tasks which require travel, transportation and per diem is added to the labor cost.

The cost for manufacture of hardware is based on a program in which test hardware is built in multiple release. After completion of qualification testing and necessary redesign, operational hardware is built in follow-on release, using tooling reworked from the test programs.

Subcontract work is shown on cost sheets as if the prime contractor performed all the operations in order that costs be indicated in the proper functional columns.

No allowance is made for major contingencies such as total loss of a vehicle. Based on past manufacturing experience, a 15% increment has been included in the cost of purchased material.

14.11.3.2 Assumptions

(1) The OMLFV will be equivalent in size and complexity to the configuration shown in the frontispeice.

(2) The program will be accomplished in accordance with the defined schedule.

(3) The contractor will implement a quality program in accordance with NPC-200-2.

(4) The contractor will implement a reliability program in accordance with NASA document NPC-250-1.

(5) There will be no sterilization requirement for the OMLFV and LSE end items..

(6) No new contractor facilities will be required. The use of government test facilities will be available when required.

(7) Apollo program hardware will be available for integration tests when required at no cost to this program. This includes pressure suits, LM Shelter, Saturn adapter, and launch site equipment.

(8) Quantities of deliverable items will be as specified in Appendix E.

(9) Documentation required will be as specified in Section III and will conform to NPC-500-6 and NPC-500-1.

(10) The OMLFV will utilize the same propellants as LM.

(11) Operational hardware will be designed and qualified for conditions of acceleration, shock, vibration, pressure, temperature, electrical interference, acoustic noise, radiation and lunar environment in general conformance to applicable portions of the following standards and specifications.

   (a) MIL-STD-202, Test Methods for Electrical and Electronic Components

   (b) MIL-STD-704, Electrical Power - Aircraft, Characteristics and Utilization of

   (c) MIL-STD-810, Environmental Test Methods of Aerospace and Ground Equipment

   (d) MIL-E-5149, Engines, Rocket, Liquid Propellant General Specifications for

   (e) MIL-H-27894A (USAF), Human Engineering Requirements for Aerospace Systems and Equipment

   (f) MSFC ADL-258A, Space Vehicle and Supporting Equipment, Applicable Documents Listing for

(12) The OMLFV design and program plan will be based on the technology available as of July 1969.

14.11.3.3 Exclusions

The program costed herein does not include the costs of:

(1) Launch and lunar mission operations other than contractor support.

(2) Training operations and practice missions other than contractor support.

(3) Lunar or orbital tests.

(4) Development or procurement of space suits and environmental control systems.

(5) Standard items of GFE/GSE normally available at training and launch sites. However, all Lunar Support and Ground Support Equipment required to support the OMLFV contractors requirements are included.

(6) Rocket propellants, which are assumed to be government furnished at off-site operations.

(7) Cost of use of government test facilities.

(8) Communications Equipment

14.11.4 Detailed Cost

Table 14.3 presents cost by fiscal year. A summary of program costs by function and end-item is also presented in Table 14.4. Definitions of the functions listed in Table 14.4 are given in Reference 3.

TABLE 14.3
COST BY FISCAL YEAR
(Thousands of Dollars)

Fiscal Year	1	2	3	4	Total
Nonrecurring:					
Design and Integration	3,031	3,432	374		6,837
Manufacturing Test and Training Vehicles	237	3,563	2,138		5,938
Test	351	3,613	1,990		5,954
Launch GSE and Training Support	38	440	330		808
Subtotal	3,657	11,048	4,832		19,537
Recurring:					
Manf. Flight Vehicles			1,379	2,561	3,940
Lunar Support Equipment			430	200	630
Mission Support and Launch Service		84	738	290	1,112
Subtotal		84	2,547	3,051	5,682
Documentation and Program Management	631	1,675	1,431	200	3,937
Total	4,288	12,807	8,810	3,251	29,156

TABLE 14.4

OMLFV COST SUMMARY

	DESIGN							MANUFACTURE								TEST												
FUNCTION	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	END ITEM SUBTOTALS
END ITEM NONRECURRING	Design and Engineering	Analysis	Specifications	Reliability	Design Reviews	Data Processing	System Integration	Engineering Liaison	Tooling	Procurement Parts and Material	Fab and Processing	Qual Assur and Control	Final Assembly	Manf Planning	Manf Engineering	Development and Design Ver	Environ and Qualification	Reliability	Acceptance	Test Equip. Design and Fab	Engineering Liaison	Test Material	Data Processing	Integration Tests	Documentation	Project Management	Tech Reports	Spares	
Lunar Vehicle Soft and Hard Mockups	25,586	34,408	4,708	1,760	10,120			26,488	7,308	10,500	58,996	4,536	23,352	3,360	150									31,664	12,050	57,743			312,731
DVT PFRT and Qual	3,119,008	1,499,448	275,674	197,350	164,838	115,900	69,892	234,432	456,152	1,553,119	817,254	342,392	553,954	125,538	111,640	2,058,632	1,435,332	574,363	372,546	551,232	285,343	324,200	311,400		682,325	1,684,438			17,916,364
Rocket Trainer	182,512	155,144	35,376	25,960	18,920	12,700		43,780	31,080	128,750	174,468	41,440	42,560	15,200	4,800	144,000	54,000		26,180	18,000	26,224	30,000	22,000		61,192	140,819			1,435,105
1/6 g Jet Trainer	137,544	99,000	25,300	29,040	13,200	4,000		26,560	17,500	91,800	64,400	16,560	18,060	7,680	3,200				19,536		9,240				28,664	65,964			672,248
1 g Jet Trainer	182,668	135,080	35,860	20,680	16,060	1,000		29,260	23,240	48,600	103,320	25,640	24,640	10,400	3,680	108,000			42,768	18,000	49,896	24,000	22,000		48,512	111,778			1,139,142
Lunar Support Equip	34,408	32,692	9,856	21,384	3,360		1,940	21,032	4,354	14,000	30,800	17,984	18,424	4,636	250	120,240	36,940			3,500	13,948	300		9,000	33,559	77,230			787,037
RECURRING																													
Lunar Support Equipment								22,440		195,000	58,800	64,976	15,792						152,064		21,120				26,311	60,548			617,051
Lunar Operational Vehicles								386,398	14,000	645,563	925,904	373,504	812,374	31,876	15,600				563,486		167,596	15,000	7,500		200,425	461,231			4,700,457
Launch Services																									45,078	103,737	552,010	656,372	357,197
Mission Support	85,536																								10,612	24,421	128,304		248,873
Total	3,767,262	1,955,764	386,774	296,174	226,498	133,600	71,832	785,390	553,634	2,687,332	2,511,142	887,032	1,589,156	198,690	139,326	2,430,872	1,580,272	574,363	1,176,580	590,732	573,367	393,500	362,900	40,664	1,148,790	2,787,879	680,314	656,372	29,156,145

15.0 ESCAPE TO ORBIT

The feasibility of using the exploration vehicle for accomplishing an escape to orbit mission is reported in this section. The guidance technique, flight profile, ΔV requirements, and modifications which can be made to the exploration vehicle to adapt it to the escape to orbit mission are presented.

15.1 GUIDANCE AND ΔV REQUIREMENTS

15.1.1 General

This section presents (1) a guidance concept for the escape to orbit mission, (2) nominal trajectories for the concept, and (3) parameteric studies of the effects of the vehicle characteristics on the propellant required. Based on previous research, a simple four-phase guidance technique employing constant thrust and attitude during each phase, and requiring a minimum of additional hardware on the vehicle was selected. Nominal trajectories for this concept were then established using a digital computer program that minimizes the ΔV required to escape to orbit. The influence of the range angle traveled, the maximum initial thrust to weight ratio, and the engine throttling ratio on the ΔV requirements for these nominal trajectories were determined.

15.1.2 Guidance Concept and Implementation

Past research on guidance concepts for emergency escape from the lunar surface has ranged from considerations of operationally simple and hardware minimizing concepts requiring only an optical sight and timer, to more sophisticated propellant minimizing concepts requiring IMU's and airborne computation. These studies have shown that simple concepts requiring a minimum of hardware can be developed that are reasonably efficient in terms of propellant requirements.

Based on these results, a simple four-phase guidance concept utilizing constant thrust and attitude during each phase has been selected for the escape to orbit mission. The phases in this concept consist of: (1) a maximum thrust lift off in which the vehicle is boosted vertically until all nearby obstacles are cleared, (2) a maximum thrust boost during which most of the required orbital energy is imparted to the vehicle, (3) a minimum thrust cruise during which the vehicle ascends to the desired orbital altitude, and (4) a maximum thrust injection during which the vehicle is injected into the desired orbit. (A fifth phase, rendezvous and docking, is performed by the CSM.) During each phase, the vehicle thrust vector will be oriented at a fixed angle with respect to local vertical. This will be accomplished by using a simple side horizon viewing optical sight investigated by NASA, LRC, to obtain a reference for controlling the vehicle pitch and roll attitudes. Azimuth (yaw) is obtained by sighting on a star or the earth.

In addition to controlling the vehicle attitude, means must be provided to determine when to initiate the ascent and when to terminate each phase. The ascent begins when the OMLFV is a specified range angle ahead of the CSM. Each phase is terminated at a specified value of V'. This parameter, V', is obtained by integrating the output of a thrust-axis mounted accelerometer and is a measure of the energy imparted to the vehicle by the thrusters. This requires readily available but somewhat more complex hardware than a simple timer. However, studies have shown that more accurate orbital conditions are achieved than when time is used for phase termination. This is illustrated in Figure 15.1 which compares the distribution of terminal errors in range and range rate (measured relative to the CSM) for these two phase termination parameters.

After termination of the ascent, the OMLFV waits for the CSM to complete the rendezvous and docking maneuvers. The ascent vehicle will carry a lightweight transceiver and will work in conjunction with a VHF communication link ranging system aboard the CSM (similar to that developed by RCA for the Apollo program) and is used to provide range and range rate data to the CSM during rendezvous and docking. Line-of-sight angle and rate information is obtained from optical devices aboard the CSM.

If the CSM were carrying the rendezvous radar, the ascent could be flown more efficiently and more accurately by using a closed loop guidance law. In this instance, the OMLFV would carry a transponder which would work cooperatively with the rendezvous radar. This equipment would be useful for control during rendezvous and docking as well as the ascent phase.

15.1.3 Nominal Trajectories

For the guidance concept selected, there are six control parameters (the V' and pitch attitude during the boost, cruise and injection phases*) available for flying an in-plane trajectory which must be chosen so that they satisfy four terminal orbital conditions (two velocities and two positions) with respect to the CSM. As is generally the case when there are more controls available than there are boundary conditions, there is a multiplicity of sets of controls which result in trajectories that terminate at the desired orbital conditions.

A unique set can be found only if constraints are imposed or if a requirement that some parameter be maximized or minimized is specified. Both were done in this study. It was required that the trajectories terminate at a specified range angle and that they require minimum ΔV to obtain the desired orbital conditions. A typical trajectory profile meeting these requirements is shown in Figure 15.2 for an in-plane, 20° range angle, minimum ΔV flight.

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* The liftoff phase is assumed to be predetermined as that required for adequate obstacle clearance during the ascent.

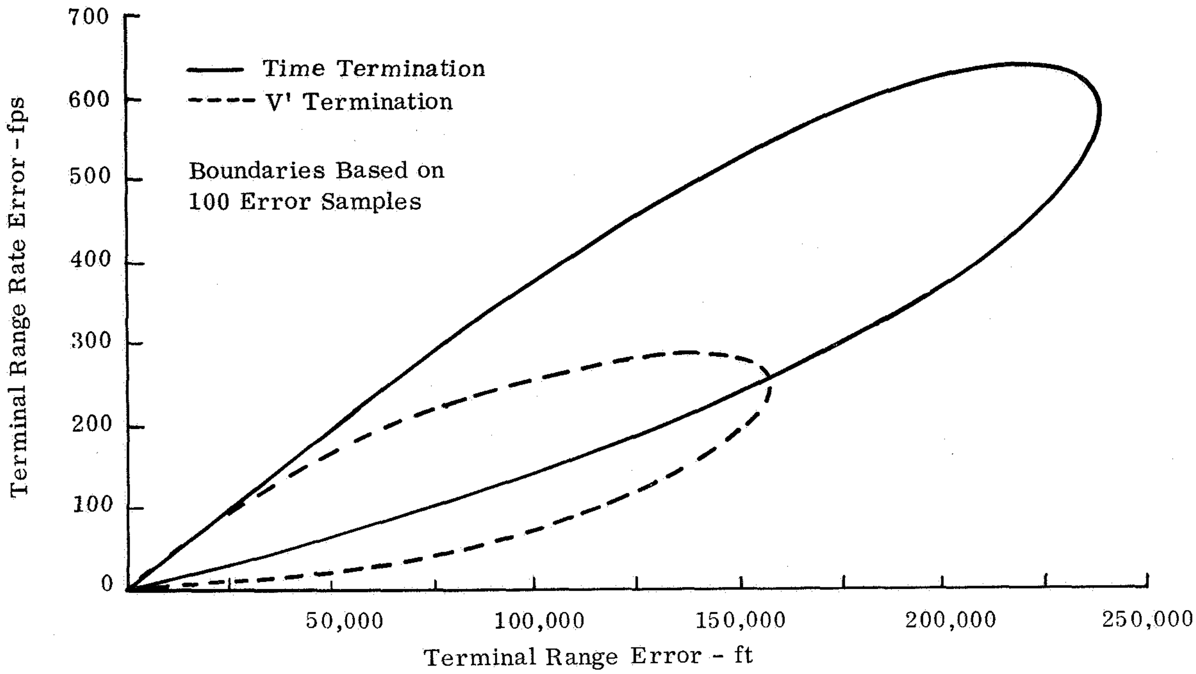

Figure 15.1. Effects of Phase Termination Parameter on Orbital Errors

Phase	Duration (Sec)	Thrust (lb_f)	Pitch (Degrees)
Lift Off	50.0	600.0	0.0
Boost	401.5	600.0	-52.7
Cruise	274.5	100.0	-49.5
Injection	103.9	600.0	-101.9

Notes: (1) Initial Mass = 73.6 Slugs
(2) I_{sp} = 285 Sec

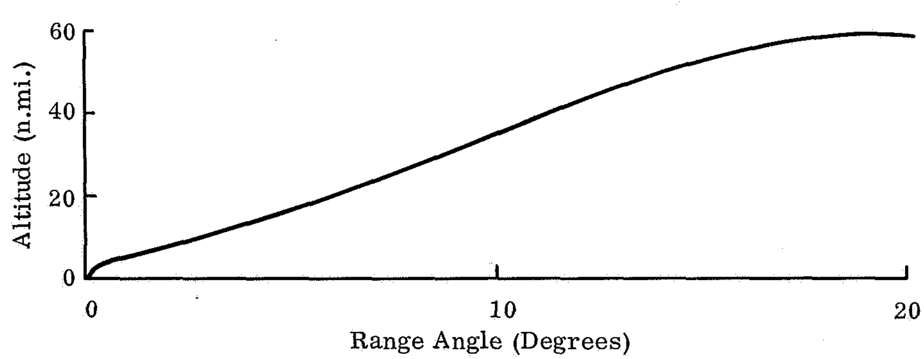

Figure 15.2. Typical Nominal Trajectory Profile

15.1.4 Parametric Studies

Parametric studies were made using a computer program that determines optimum ascents by a modified steepest ascent optimization technique. The influence of the following parameters on the propellant requirements for escape to orbit were determined: (1) range angle, (2) initial maximum thrust to weight ratio, and (3) throttling ratio.

The effect of range angle on the escape to orbit mission is shown in Figure 15.3. It can be seen that the ΔV required to escape to orbit with vehicles with maximum initial thrust-to-weight ratios, $(T/W^*)_{max}$ of 1.53 and 2.30 at first decreases with range angle, as is expected, but then starts to increase as the range angle becomes greater than about 50°. It was found that this increase is due to the relatively low (T/W)'s of these vehicles. For example, it can be seen that the ΔV required for a vehicle with a $(T/W)_{max}$ of 4.60 continues to decrease with range angle, as does the well known case of impulsive firings. However, it was also found that there are other reasons to operate at smaller range angles. As the range angle increases, the flights become increasingly sensitive to off nominal vehicle parameters (e.g. maximum thrust, vehicle weight, etc.), errors in measuring the necessary guidance information and pilot errors. This is illustrated by the upper curve in Figure 15.3, which shows the effect of a 3 percent error in maximum thrust on the orbital altitude at the end of injection. A range angle of 20° was selected as a nominal operating point with this guidance concept, because it results in near minimum ΔV and has low error sensitivity.

The effect of the initial $(T/W)_{max}$, ($(T/W)_{min}$ was taken as zero), on the ΔV required at this 20° range angle is illustrated in Figure 15.4.

Since it is not planned to shut the engine off to obtain a $(T/W)_{min}$ of zero during the cruise phase, a study was made to determine the effect of finite throttling ratios on the ΔV required for escape to orbit. As illustrated in Figure 15.5, it was found that the effect of throttling ratio was insignificant for the ranges of $(T/W)_{max}$ considered for a 20° range angle flights. Therefore, the 6:1 throttling ratios of the exploration vehicle engines is adequate for the escape to orbit mission.

15.2 Conceptual Designs

Three alternative configurations for accomplishing the escape to orbit mission are discussed in the following paragraphs.

Two One-man Escape Vehicles

Figure 15.6 shows modifications to a single exploration vehicle to make it suitable for accomplishing the escape to orbit mission with one man. The second

* Lunar Weight

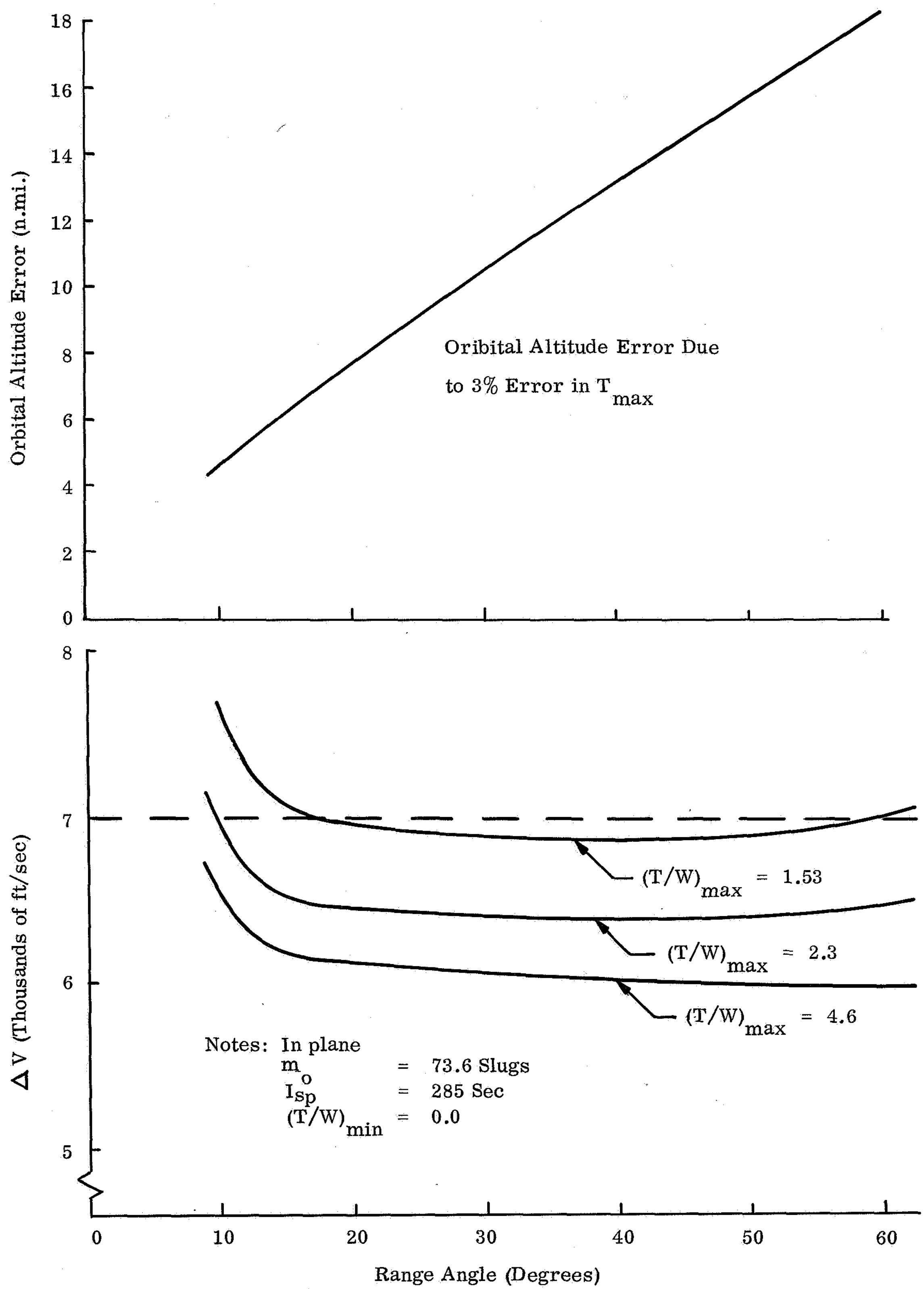

Figure 15.3. Influence of Range Angle on Escape to Orbit

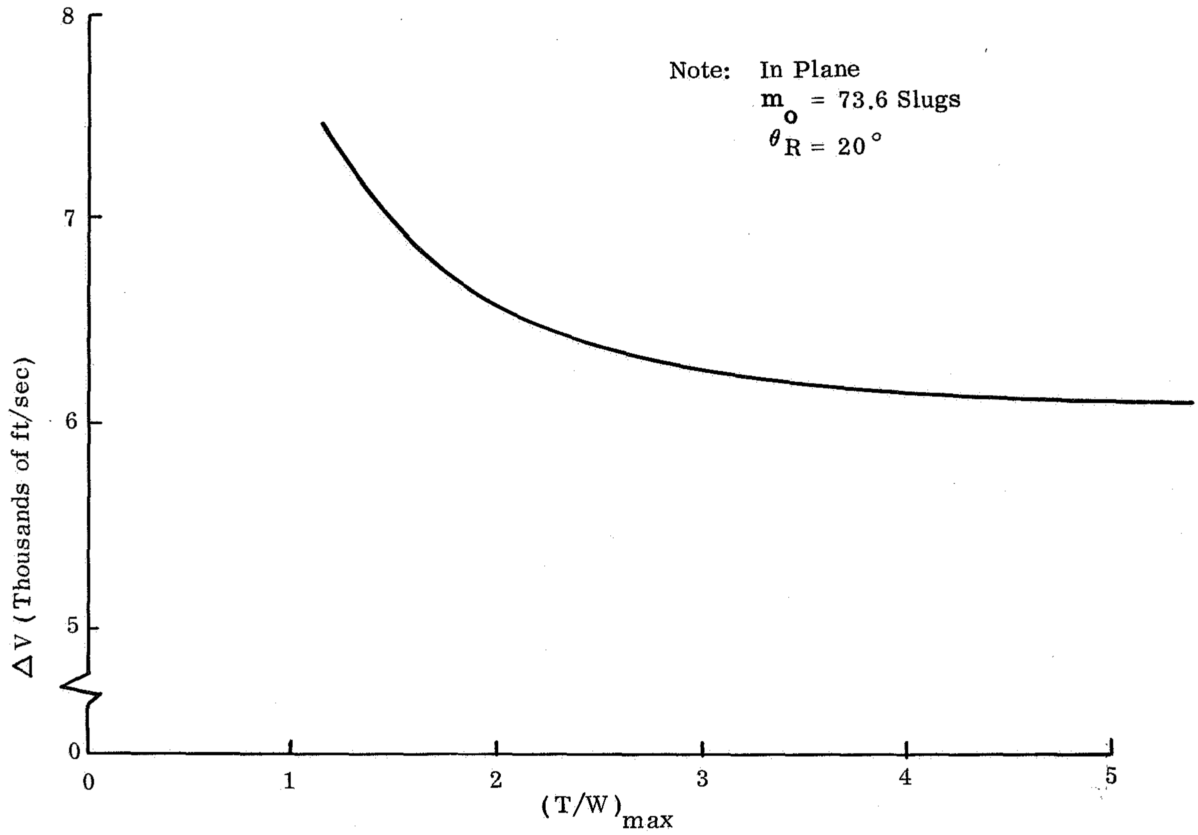

Figure 15.4. Influence of Thrust to Weight Ratio on ΔV Requirements for Escape to Orbit

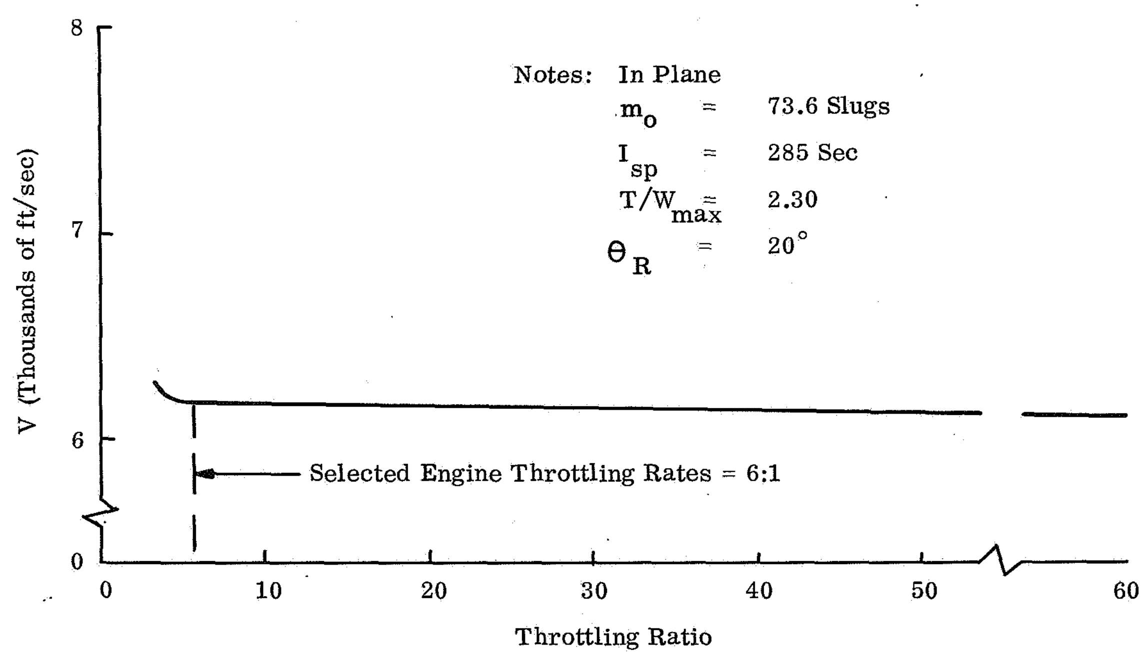

Figure 15.5 Influence of Throttling Ratio of ΔV Required to Escape to Orbit

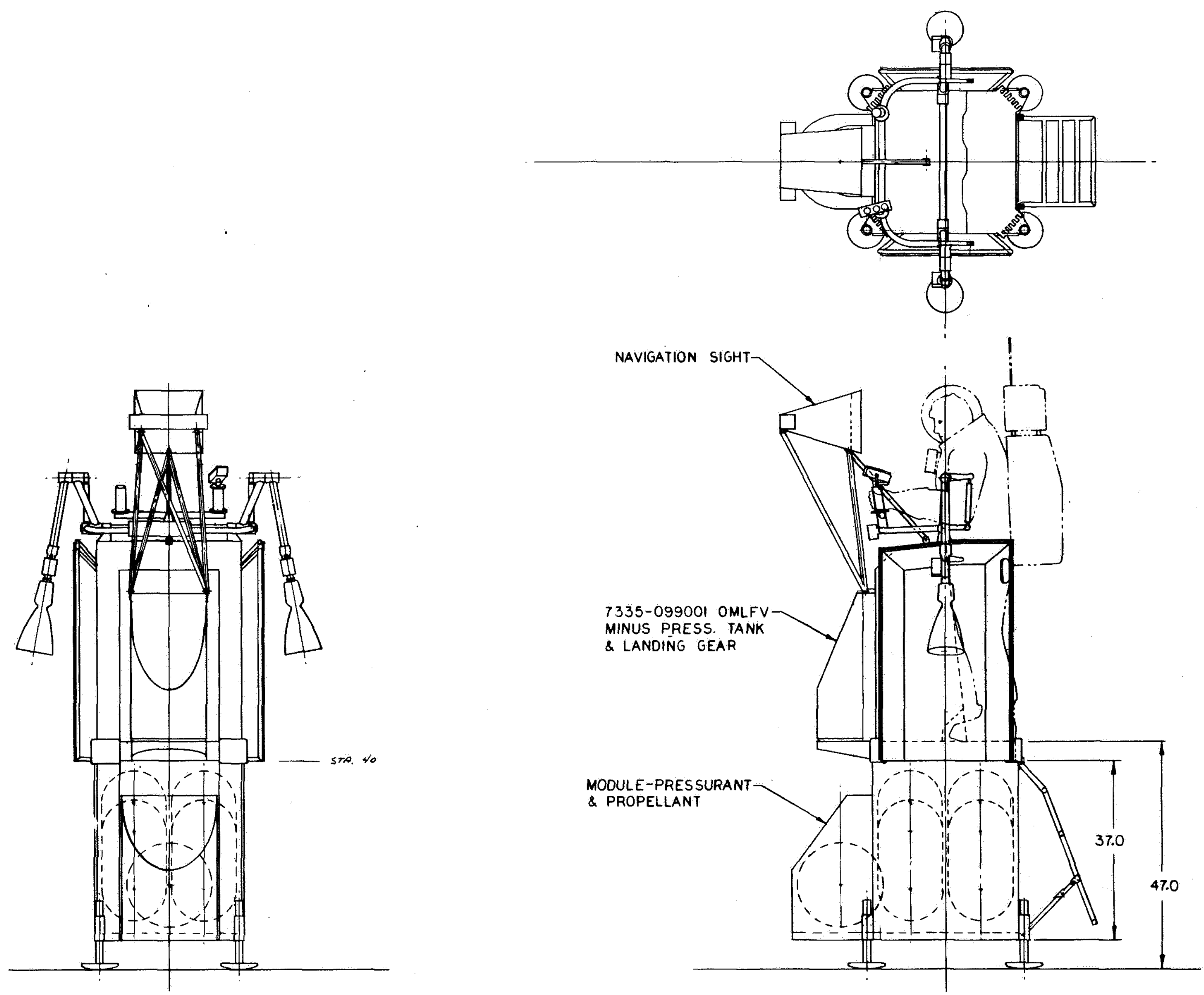

Figure 15.6. One Man Escape to Orbit Vehicle

TABLE 15.1

MASS PROPERTIES SUMMARY
ESCAPE TO ORBIT VEHICLES

		One Man Escape to Orbit Vehicle	Two Man Escape to Orbit Vehicle	Surface Rescue and Escape Vehicle
Burnout Condition				
	Weight Pounds	758.3	1,428.8	1,160.4
cg Locations	X	102.8	68.0	100.0
	Y	99.9	100.0	100.0
	Z	68.6	78.1	80.9
	I_x Slug Ft^2	182.0	174.0	150.0
	I_y Slug Ft^2	198.0	561.0	313.0
	I_z Slug Ft^2	37.0	434.0	219.0
Gross Weight Condition				
	Weight Pounds	1,543.3	2,928.8	2,670.4
cg Locations	X	100.1	68.0	100.0
	Y	99.6	99.5	100.0
	Z	53.3	70.6	73.3
	I_x Slug Ft^2	367.0	260.0	238.0
	I_y Slug Ft^2	384.0	721.0	402.0
	I_z Slug Ft^2	52.0	547.0	282.0

exploration vehicle will be similarly modified for the second man. The add-on module contains 485 pounds of propellant. A larger helium sphere is required to expel the total propellant load and connections between the exploration vehicle tanks and the module tanks are required. A mass properties summary is shown in Table 15.1.

The corresponding ΔV capability of the configuration is 6,350 ft/sec and the T/W at liftoff is 1.18. As indicated in the guidance and ΔV section (Figure 15.4) this combination is marginally unacceptable. Additional thrust, which can be provided by adding two more engines to the add-on propulsion module and $\approx$ 67 pounds more propellant are required to provide a 5% reserve ΔV (above the nominal required). These increases can be provided with little penalty in terms of additional volume or weight delivered to the moon.

Control power sensitivities for the configuration are shown in Table 15.2. The cg shift due to propellant usage causes a pitch trim angle of two degrees nose down at takeoff and two degrees nose up at propellant depletion.

For stowage on board the lunar module the exploration vehicles are carried in quandrants I and IV and the two sets of add-on tanks are carried in quadrant III. Fueling of the escape vehicle is accomplished from the ascent vehicle at the time when the need for escape vehicle usage is determined.

TABLE 15.2

CONTROL POWER SENSITIVITIES
ONE-MAN ESCAPE VEHICLE
($Deg/Sec^2/Deg$)

	Burnout Weight		Gross Weight	
	T_{Max}	T_{Min}	T_{Max}	T_{Min}
Pitch	3.79	0.63	2.96	0.49
Roll	1.23	1.23	0.66	0.66
Yaw	4.23	0.71	3.0	0.5

One Two-Man Escape Vehicle

A conceptual design of an escape to orbit vehicle which carries two men is shown in Figure 15.7. In this concept, two modified exploration vehicles are connected together and a propellant module containing the additional propellant is added. Linkages are added to permit the pilot to actuate both sets of engines for vehicle attitude and thrust control. The total propellant provided is 1,500 pounds. The mass summary for this concept is presented in Table 15.1, and the corresponding control sensitivity data is shown in Table 15.3.

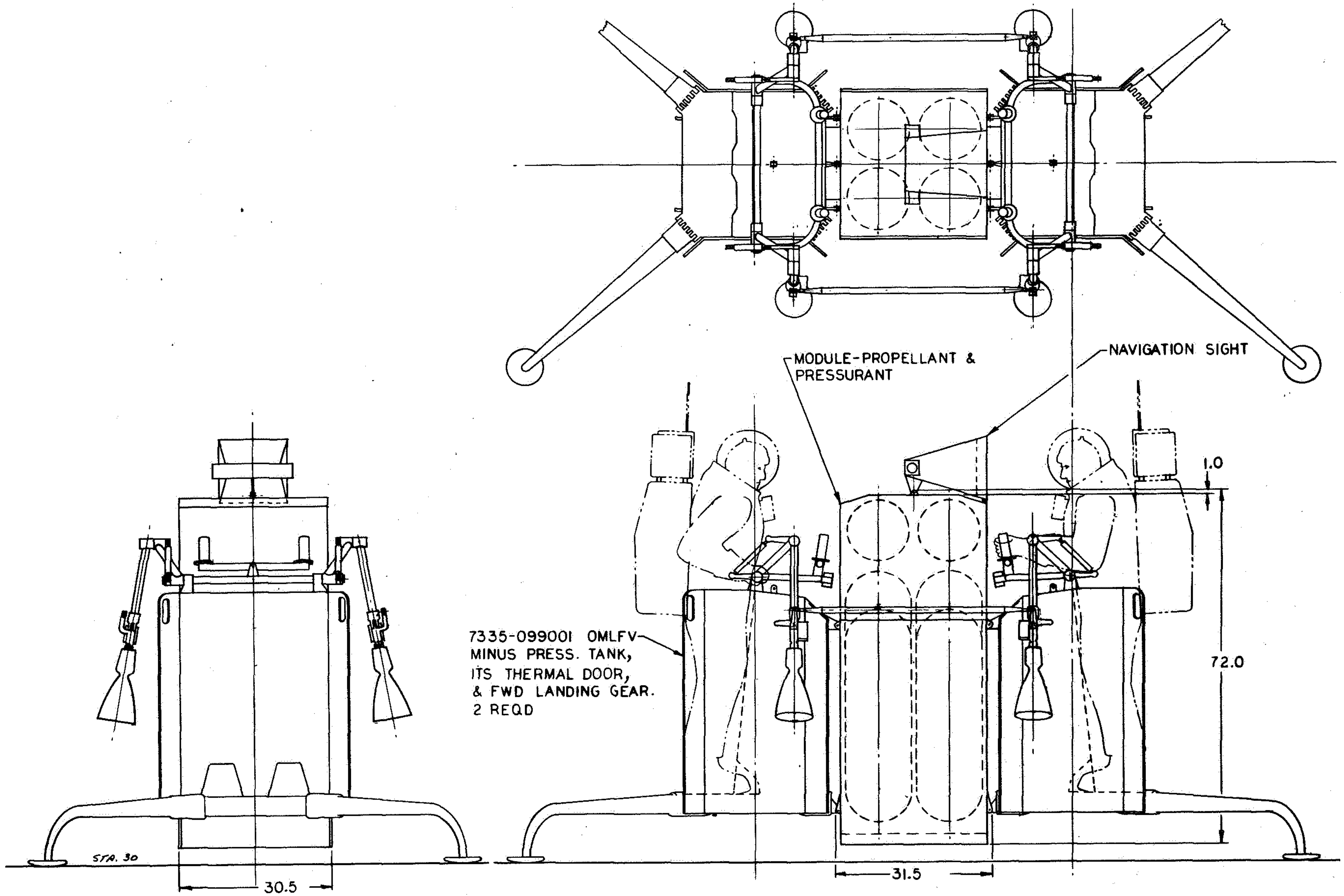

Figure 15.7. Two Man Escape to Orbit Vehicle

TABLE 15.3

CONTROL POWER SENSITIVITIES
TWO-MAN ESCAPE VEHICLE
($Deg/Sec^2/Deg$)

	Burnout Weight		Gross Weight	
	T_{Max}	T_{Min}	T_{Max}	T_{Min}
Pitch	1.32	0.22	1.59	0.27
Roll	2.28	2.28	1.61	1.61
Yaw	0.72	0.12	0.57	0.095

The thrust to weight ratio provided by the four engines is 1.24. This is inadequate and four more fixed engines must be provided on the add-on propulsion module to provide adequate thrust for the ascent mission.

Since the cg moves vertically with propellant usage there is no vehicle trim angle. Stowage of this concept is similar to that for two one-man vehicles except that the add-on module is stowed as one package instead of two.

Surface Rescue and Escape Vehicle

In this concept, a one-man exploration vehicle and a second dual purpose two-man vehicle are placed on the lunar surface. This dual purpose vehicle is capable of rescuing a stranded astronaut on the lunar surface or carrying both men to lunar orbit. A configuration concept for the dual purpose vehicle is illustrated in Figure 15.8. This concept utilizes the same engines as the exploration vehicle. It would be partially filled to accomplish the surface rescue mission at the beginning of lunar surface operations so that it is ready at all times to accomplish that mission. For escape to orbit the filling would be completed from the ascent stage tanks. The tanks are sized for 1,510 pounds of propellant.

Here again, the thrust provided by the four engines is insufficient for an efficient ascent. Two more engines of the same thrust can be provided on the vehicle at little penalty in stowed weight and volume. These engines can be bolt mounted to the frame or added to the cross members carrying the two engines shown.

A mass properties summary is presented in Table 15.1 and control sensitivity data is presented in Table 15.4.

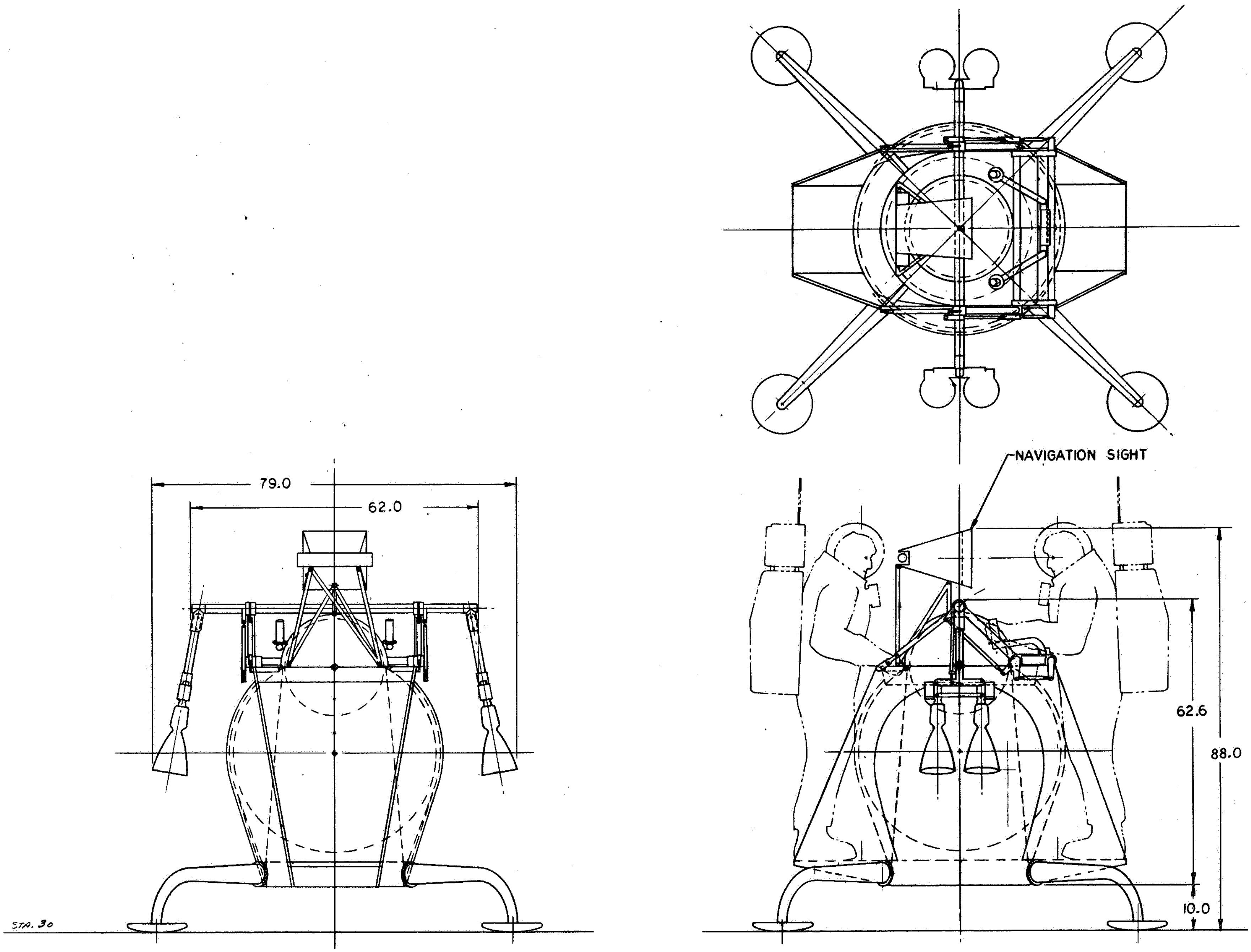

Figure 15.8. Surface Rescue and Escape Vehicle

This configuration can be stowed in one of the quadrants of the lunar module descent stage by modifying the upper deck (locally raising it by 5 to 7 inches). Alternatively, delivery to the lunar surface could be accomplished by one of the lunar logistic stages being studied by NASA.

TABLE 15.4

CONTROL POWER SENSITIVITIES
SURFACE RESCUE AND ESCAPE VEHICLE
($Deg/Sec^2/Deg$)

	Burnout Weight		Gross Weight	
	T_{Max}	T_{MIN}	T_{Max}	T_{Min}
Pitch	3.2	0.53	3.48	0.58
Roll	1.78	1.78	1.18	1.18
Yaw	1.6	0.27	1.24	0.21

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APPENDIX A

OMLFV LANDING DYNAMICS SIMULATION

INTRODUCTION

A digital computer simulation of the lunar landing dynamics was developed as a tool for the systematic design of the OMLFV landing gear. The development of this simulation represents an extension of previous efforts dealing with simulation of landing-dynamics. The list of References presents the documentation of the support materials.

The math-model used in this simulation has the following features:

1. Two-dimensional motion, vertical, horizontal, and pitch attitude motion of an unpowered OMLFV type vehicle is represented.
2. The vehicle is represented by a three-mass lumped parameter system.
3. The soil dynamic effects are represented by compressibility, momentum, and friction effects.
4. Landings that may be simulated with good approximation are the two types illustrated in Figure A.1.

The digital computer program was written in FORTRAN IV for use on the IBM System 360-50 Remote Access System at Bell Aerosystems (RAX).

DESCRIPTION OF THE SIMULATED SYSTEM

The class of vehicles considered here are simple rocket powered vehicles used for transportation in lunar environment. The landing gears for this type of vehicle consist of four legs or struts of either the cantilever or the inverted tripod type tipped by light landing pads.

Vehicle Math-Model

Figure A.2 illustrates the vehicle math-model and the coordinate systems used - the x-z coordinates coincide with the orthagonal principal axes of the center body; the orthagonal coordinates r and h are taken tangential and normal to the flat soil surface. The vehicle is represented by lumped-parameter system moving in two-dimensional space with 7 degrees of freedom. The center body mass can translate in the r and h direction as well as rotate in the r-h plane (three degrees of freedom) under the influence of a constant gravitational attraction and the forces and moments developed from the landing gear deflections. Each of the two masses representing the landing gears is free to translate in the r and h direction (two degrees of freedom each), under the influence of landing gear and soil reaction forces.

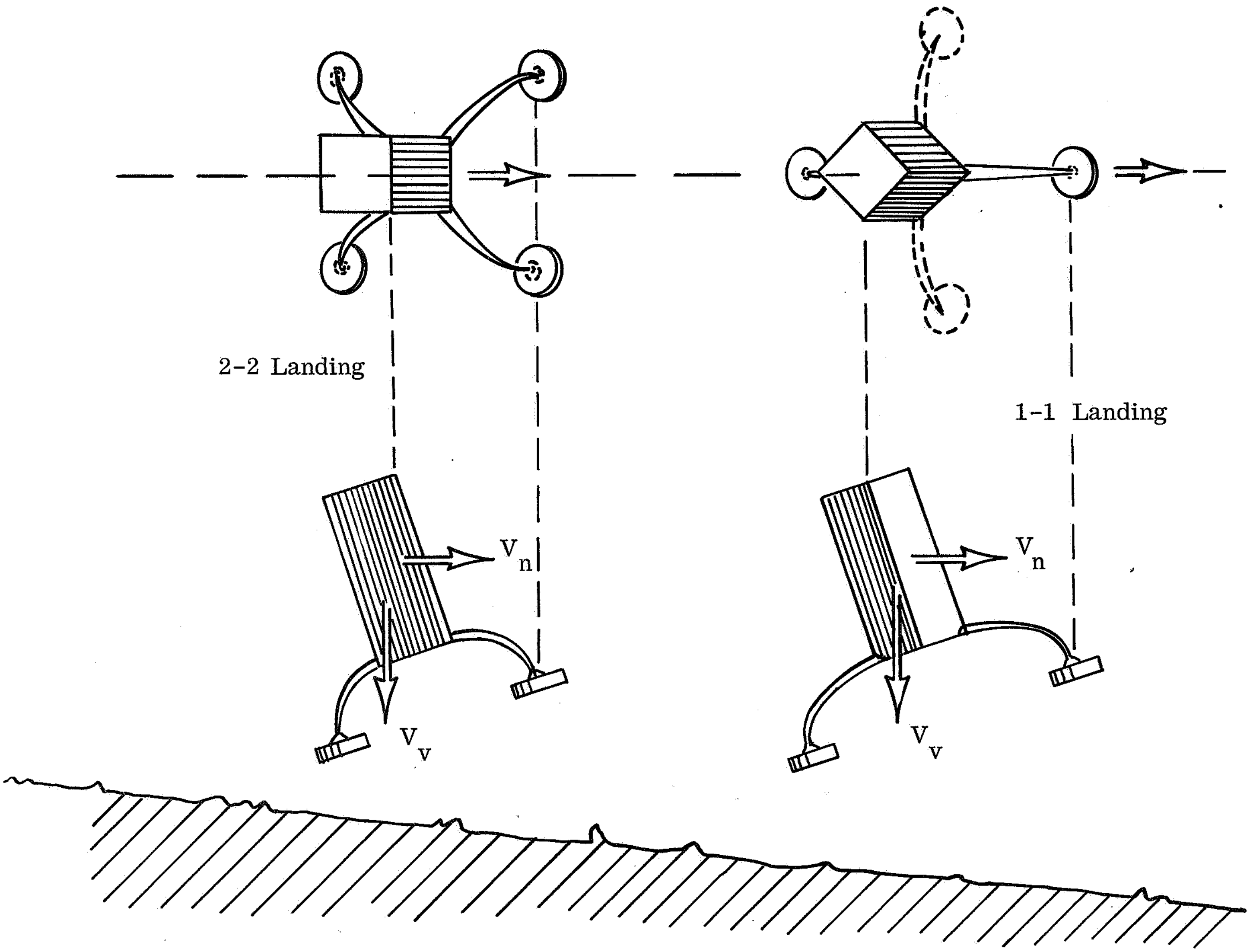


Figure A.1. Types of Two-Dimensional Landings

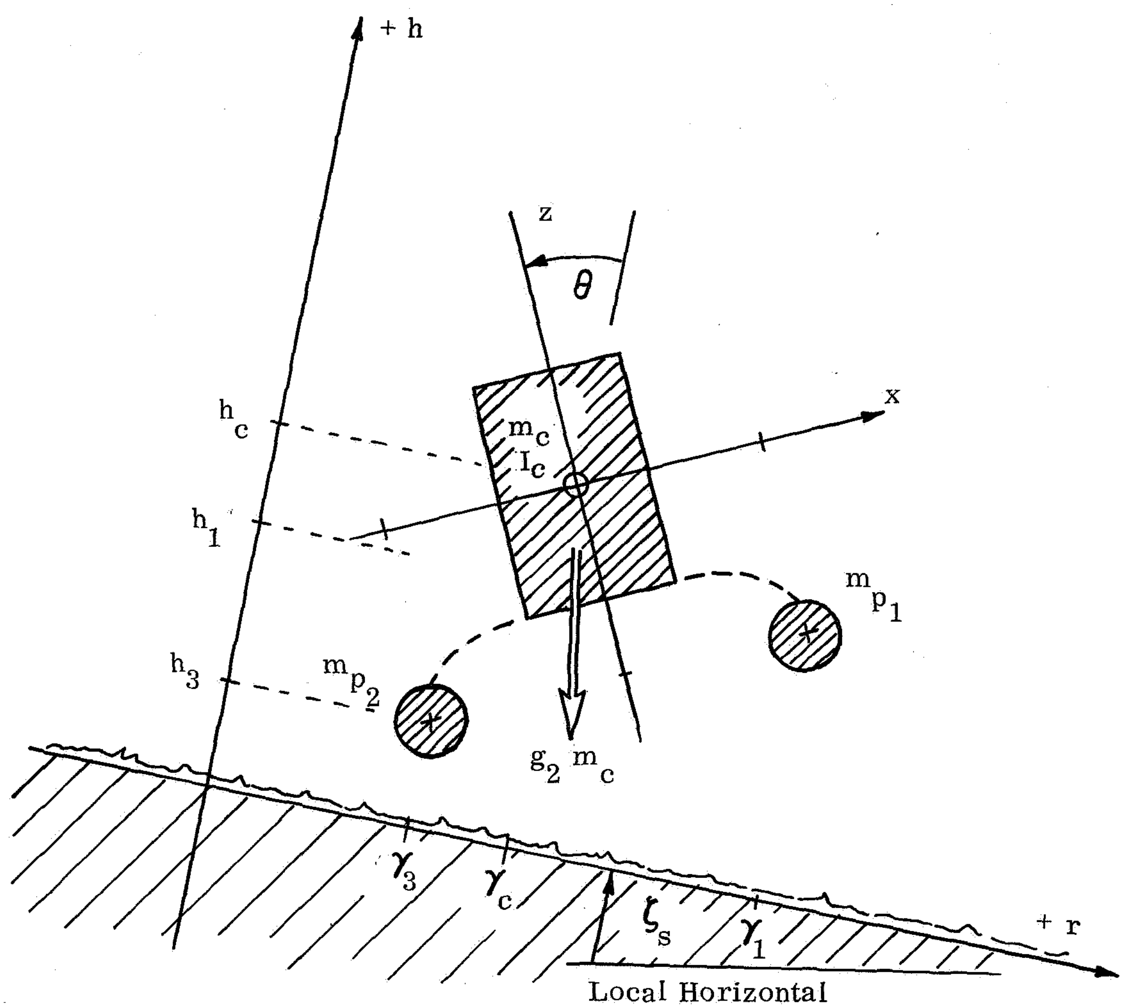


Figure A.2. Vehicle Math - Model

Landing Gear Math-Model

1. Cantilever

The cantilever landing gear is represented by a lightly damped linear spring element; the foot-pad deflection and deflection rates relative to the center body determine the landing gear forces. The forces are calculated first in the x-z coordinate system relative to the centerbody (Figure A.2) and are then transformed into the r-h coordinates relative to the surface. The equations governing these forces are given in Figure A.3.

The spring forces are obtained from a stiffness matrix and the cantilever tip deflections, these x and z forces are thus computed as linear combinations of the x and z deflections. The stiffness matrix coefficients describe the small deflection-force characteristics of the particular landing gear.

Two types of damping were assumed, linear viscous and simple coulomb damping. The viscous damping forces are calculated from a damping matrix and the cantilever tip deflection rates. The coulomb damping force is of constant magnitude with a line of action such that it opposes the x-z deflection rate; x and z components of this force are calculated based on the x and z components of the tip deflection rate. The various damping coefficients describe the landing structure's energy absorbing characteristics.

2. Inverted Tripod Landing Gear Math-Model

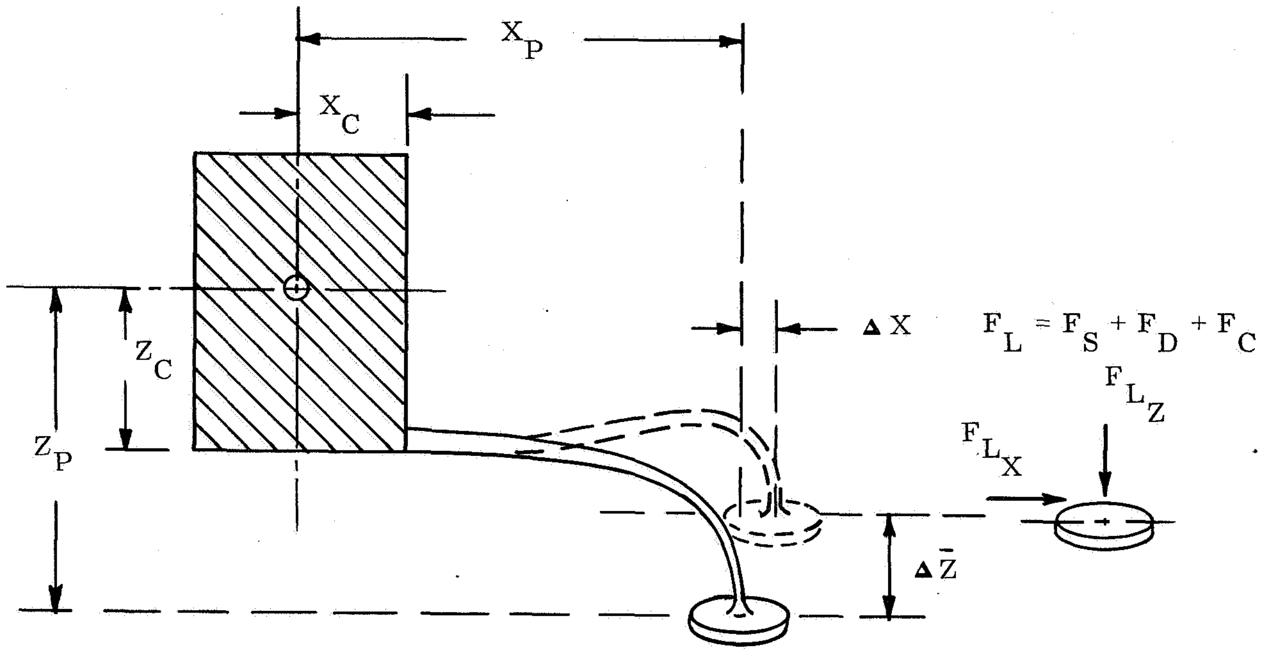
The math-model for the inverted tripod landing gear is illustrated in the diagrams of Figure A-4. The actual three-member structure is represented in this two-dimensional simulation by a two-member unit, where the lower member represents the two lower struts of the actual unit.

The primary force governing component for this type of structure is the compressible one-directional dry-friction element. A typical load-stroke diagram for such an element is presented in Figure A.4; indicated in the diagram are the constant force compression stroke characteristics, the no-load return, and the stiff elastic behavior of the unit at the stroke limits.

The reaction force along each strut is determined for a given footpad position and direction of motion relative to the center body; summing of the appropriate force components gives the landing gear forces in the x-z coordinates, and a simple transformation then gives the forces in the r-h coordinate system.

Soil Model

The soil model used in this simulation is based on work done by the Bendix Corporation — Reference 5 presents the pertinent details. Modifications to this "Bendix Soil Model" were made that deal with the "virgin soil" effects needed to account for footpad motion through previously deformed soil.



$$\begin{Bmatrix} F_{S_X} \\ F_{S_Z} \end{Bmatrix} = \begin{bmatrix} K_X & K_{XZ} \\ K_{ZX} & K_Z \end{bmatrix} \cdot \begin{Bmatrix} \Delta X \\ \Delta Z \end{Bmatrix}$$

Spring Forces Stiffness Matrix Strut Tip Deflections

$$\begin{Bmatrix} F_{DX} \\ F_{VZ} \end{Bmatrix} = \begin{bmatrix} D_X & D_{XZ} \\ D_{ZX} & D_Z \end{bmatrix} \cdot \begin{Bmatrix} \dot{X}_P \\ \dot{Z}_P \end{Bmatrix}$$

Viscous Damping Damping Matrix Tip Deflection Rates

$$\begin{Bmatrix} F_{CX} \\ F_{CZ} \end{Bmatrix} = \frac{C_D}{\sqrt{\dot{X}_1^2 + Z_P^2}} \cdot \begin{Bmatrix} \dot{X}_P \\ \dot{Z}_P \end{Bmatrix}$$

Coulomb Damping Damping Coeff. Tip Deflection Rates

Figure A.3. Cantilever Landing Gear Model

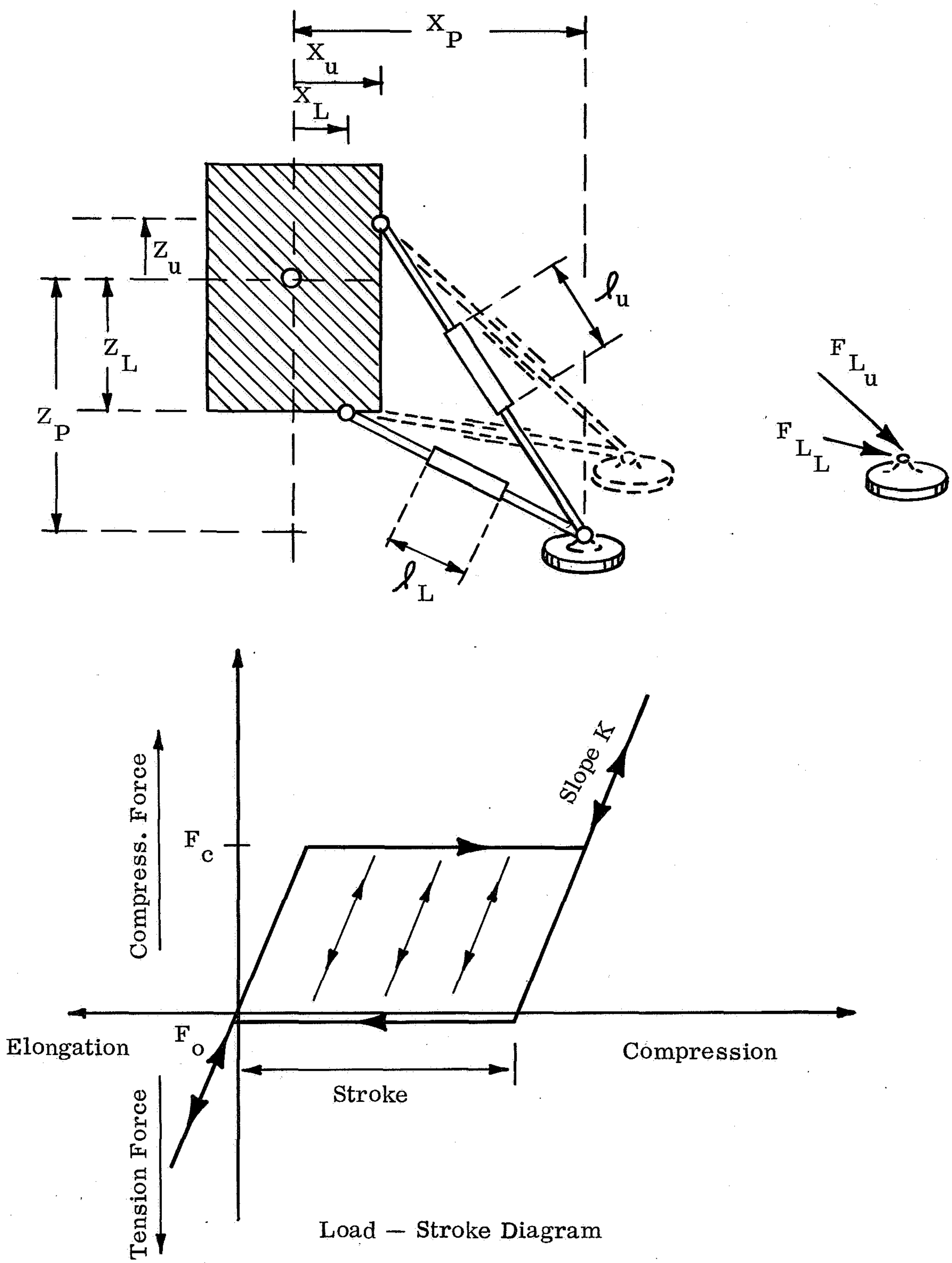


Figure A.4. Inverted Tripod Landing Gear Model

Verification of the Computer Simulation Program LULA

The digital computer simulation of the lunar landing problem (program LULA), based on the math-models outlined in the preceding paragraphs, was checked and verified by comparing results of this simulation with results generated by earlier proven computer programs.

The landing gear and vehicle dynamics parts of this simulation were checked against results from a landing simulation developed by the Bendix Corp.; this Bendix landing simulation — documentation of which is given in Reference 3 does not account for any soil dynamic effects. For this particular verification, the simulation (LULA) was modified to represent the landing on flat solid ground of a vehicle with inverted tripod landing gears. A set of vehicle parameter values corresponding to Bell's MFS-DTC configuration was taken from Reference 3. A schematic of the vehicle is given in Figure A.5. The parameters for the soil model were selected to simulate "solid ground" with large "stiction" to correspond to the simple ground assumed for the Bendix simulation.

Several comparison runs were made to generate stability data corresponding to those presented in the Bendix MFS Landing Gear Study (Reference 3). Figure A.5 presents a stability boundary curve taken from page 2-31 of Reference 3; indicated on this graph are the results of the comparison runs. The good agreement of the results is apparent from this figure.

The math-model for the soil dynamic effects used in the simulation program LULA is based on the soil model developed by the Bendix Corp.; documentation of the Bendix work is given in Reference 5. A substantial test and correlation effort has been done by Bendix to verify the accuracy of their soil model and to establish good sets of soil model parameter values; Reference 5 presents a summary of these test results. The validity of the Bendix soil model was demonstrated by comparing computer simulation results with telemetered data from the Surveyor I landing in lunar soil; Figure A.6 which is reproduced here from Reference 2 illustrates typical results of this comparison. Soil model parameter values that represent lunar soil characteristics are presented in Figure A.7 which is reproduced here from Reference 8 .

The soil dynamic portion of the lunar landing simulation program LULA was verified by comparing results obtained from LULA with corresponding results from a Bendix developed footpad-soil penetration simulation. The documentation of the applicable work by Bendix is presented in References 5 and 7 . The two-dimensional 7 degrees-of-freedom simulation LULA was suitably initialized and used to correlate results of the one-dimensional, three-degrees-of-freedom simulation developed by Bendix. The results of this comparison are presented in FigureA.8; the good agreement is apparent from these curves. Further evaluation of the soil model was done in a heuristic manner by showing that the simulated soil reactions displacements and forces for various landing conditions are in a general sense plausible and realistic.

Bell-MFS-DTC 2-2 Loading

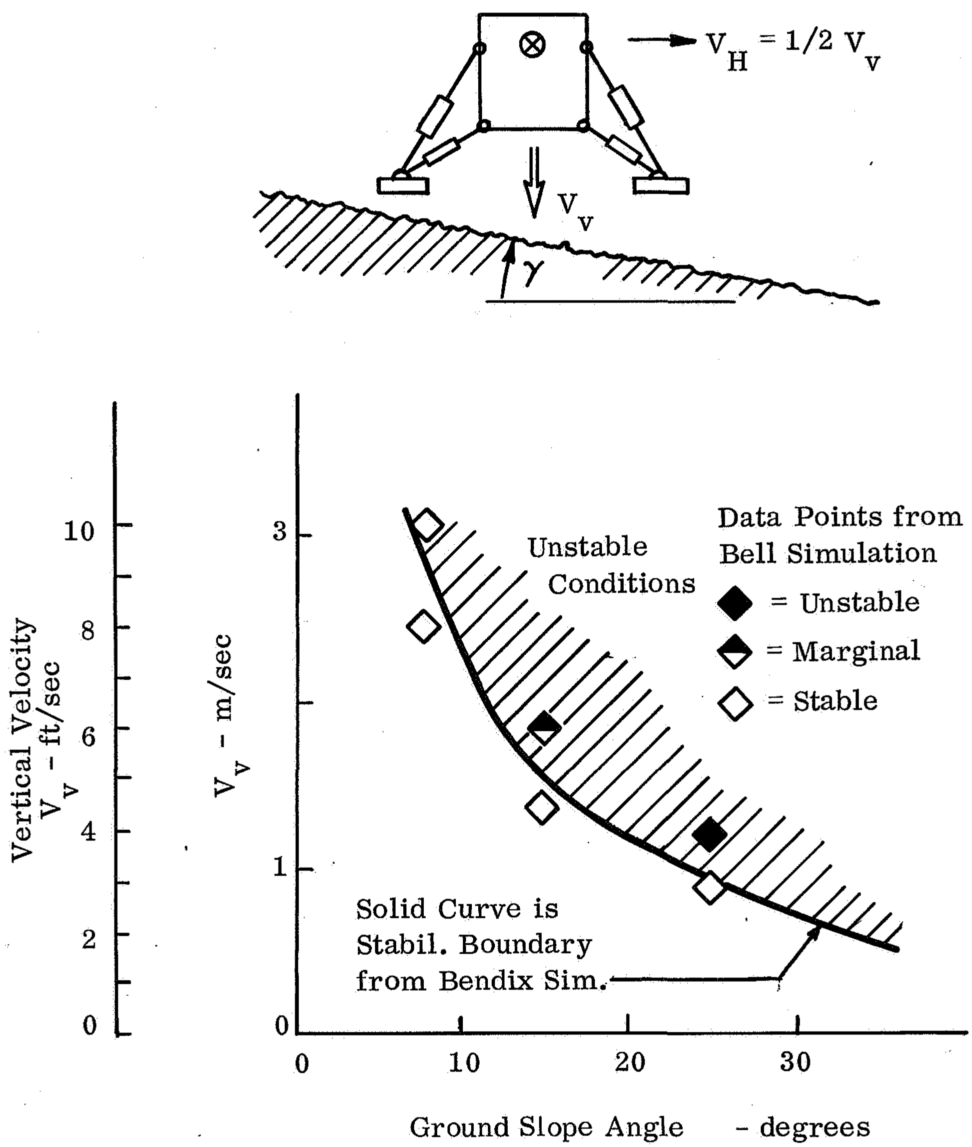


Figure A.5. Simulation Verification Data

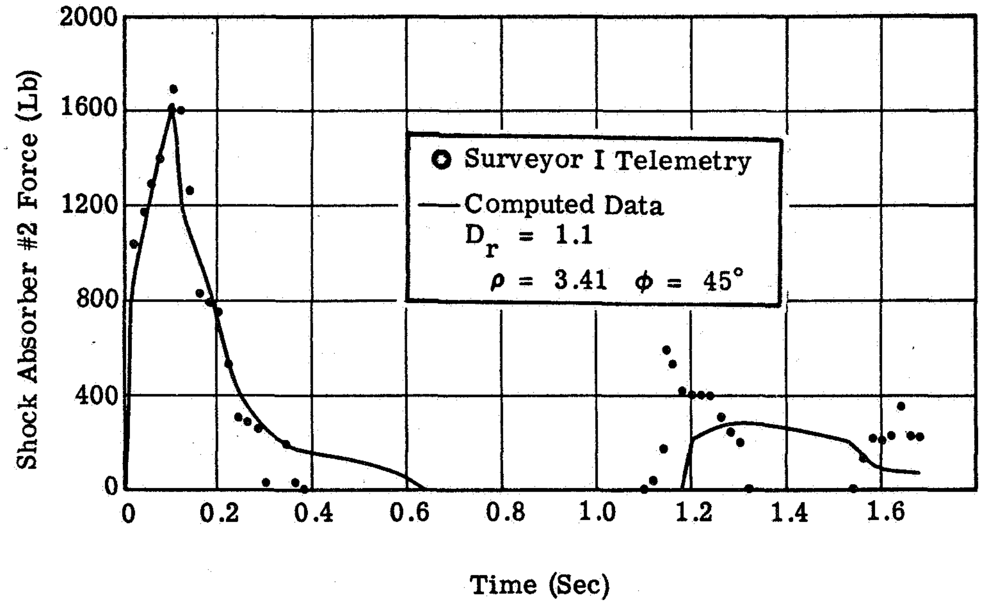


Figure A.6. Correlation between Telemetered and Calculated Shock Absorber Force for Surveyor I (Leg 2)

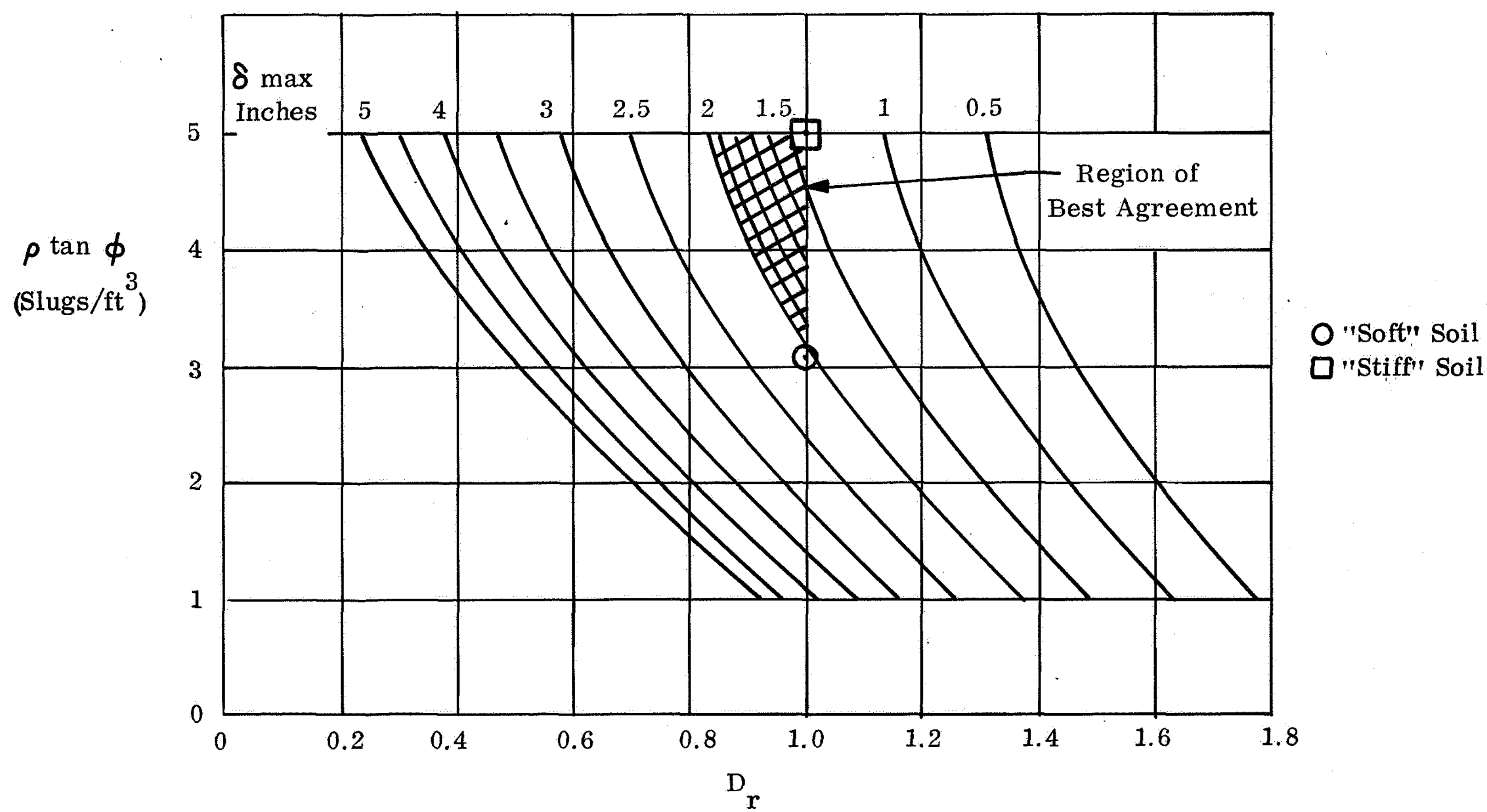


Figure A.7. Surveyor I Touchdown Conditions Maximum Computed Penetration of Footpad No. 2 as Function of Soil Descriptors

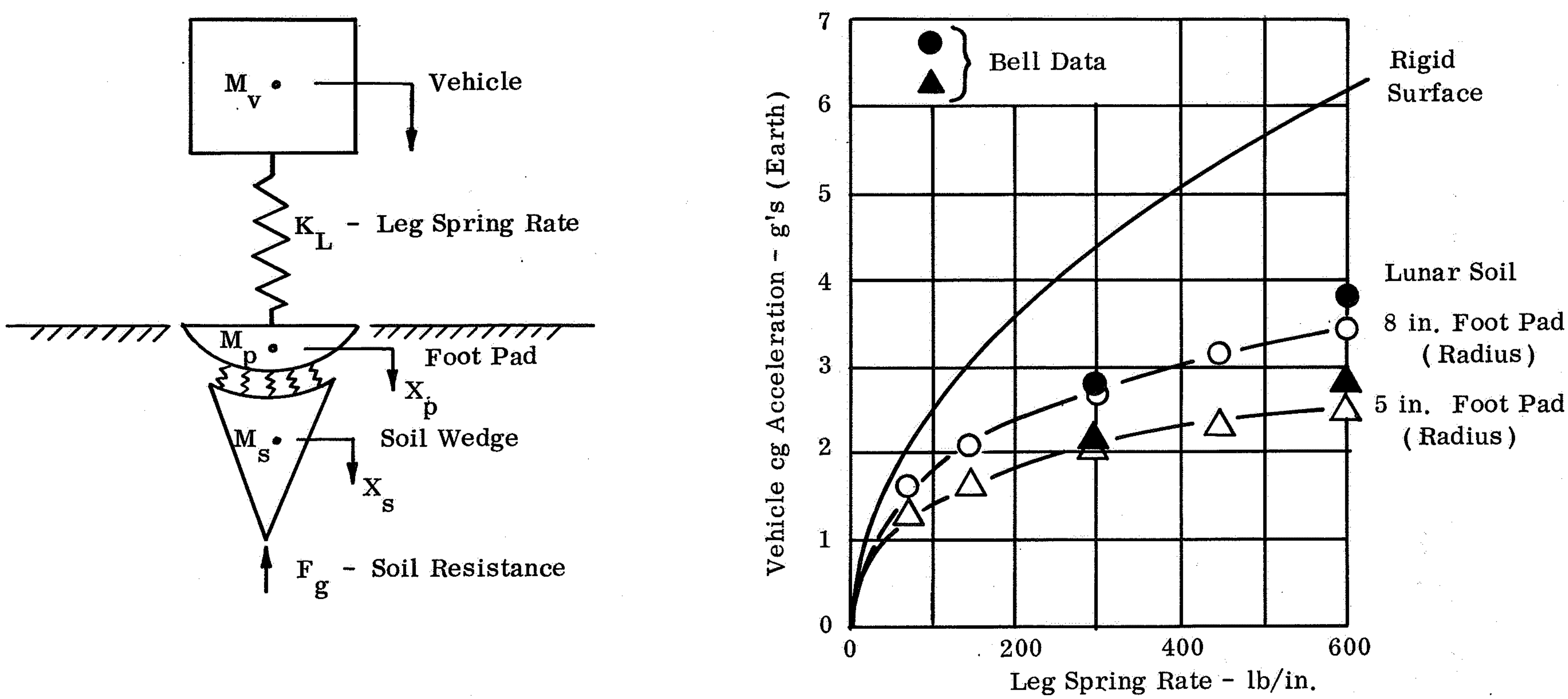


Figure A.8. Lunar Soil Model and Load Attenuation

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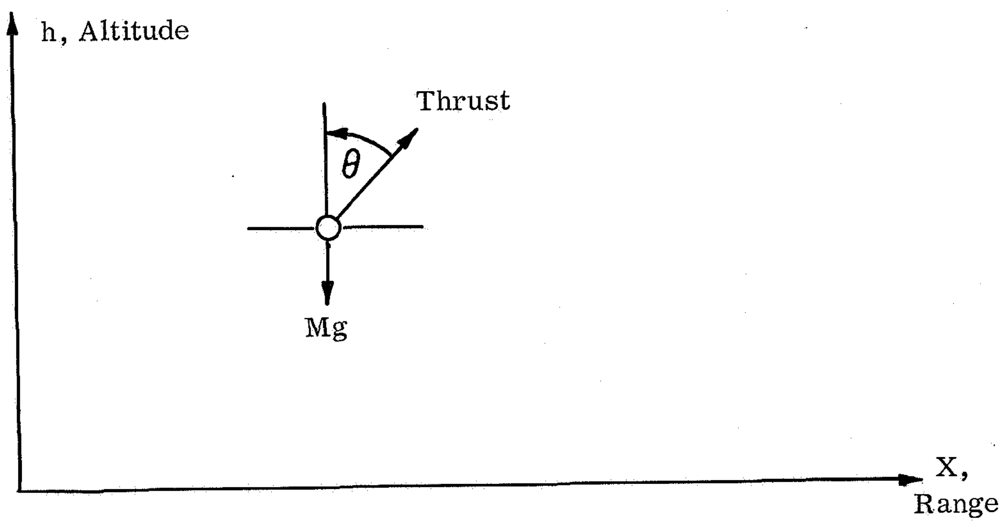
APPENDIX B

VEHICLE PERFORMANCE MATHEMATICAL MODEL

A description of the equations definining the variable mass, in-plane OMLFV flight path is presented in this Appendix. Constraints imposed on flight path variables and flight procedures and the manner in which error effects were modeled are also described.

1. BASIC EQUATIONS OF MOTION

The basic equations of motion are written with respect to the coordinate system shown in the sketch.



The equations of motion for the system are:

Vertical acceleration $\ddot{X} = \frac{T}{M} \cdot \sin\theta$

Horizontal acceleration $\ddot{h} = \frac{T}{M} \cdot \cos\theta - g_1$

Pitch rate $\dot{\theta} = \text{constant}$

Propellant Flow Rate $M = \frac{T}{g_e I_{sp}}$

ΔV rate $\Delta V = \frac{T}{M}$

The two command variables are thrust T and pitch angle θ, the selection or computation of these command variables is given in the following paragraphs for each of the six flight phases. Various symbols and subscripts are used in the following equations, most of these are self explanatory with the exception of the subscript P. Any variable with subscript P identifies a variable observed or measured by the pilot, differing from the actual value of the variable by some error term. That is, $X_P = X + \text{error}$

Commands for the various flight phases are:

Phase 1 - Lift-off and Boost

At time zero, maximum thrust is commanded together with a pitch angle, $\theta_{Boost} < 0°$; thus thrust is set to T_{MAX} while the pitch is changed from the initial zero position toward the commanded attitude.

$$T = T_{MAX} \text{ lb}$$

$$\theta: \ 0 \rightarrow \theta_{Boost}$$

$$\dot{\theta} = -5 \text{ deg/sec for } \theta_P > \theta_{Boost}$$

$$\dot{\theta} = 0 \text{ for } \theta_P \leq \theta_{Boost}$$

The pitch attitude rate limit of 5 deg/sec was selected based on manned simulation flights that demonstrated the tendency of pilots to pitch over at about this rate. This restriction on the boost phase tends to establish a relationship between speed and altitude for flights with short duration boost phase.

Phase 2

At time t_B, minimum thrust and zero pitch attitude are commanded.

$$T = T_{min}$$

$$\theta: \ \theta_{Boost} \rightarrow 0$$

$$\dot{\theta} = 10 \text{ deg/sec for } \theta_P < 0 \text{ deg}$$

$$\dot{\theta} = 0 \text{ deg/sec for } \theta_P = 0 \text{ deg}$$

Again the pitch rate limit was selected based on the observed trends in manned simulation flights.

Phase 3

When the observed vehicle altitude rate first reaches zero at an altitude h_{CR} the commanded thrust is modulated to maintain this constant altitude; the commanded pitch remains at zero.

When $\dot{h}_P = 0$

$$T = (0.36)\,(h_{CR} - h_P) - (0.69)\,(\dot{h}_P) + m \cdot g_L$$

constrained to $T_{min} \leq T \leq T_{max}$

$$\theta_P = 0$$

The feedback gains in the thrust-equation were also based on results from manned simulation flights.

Phase 4

When the observed range-to-go first reaches a selected value, X_{des}, minimum thrust is commanded; commanded pitch remains at zero.

When $X_P = X_{des}$

$$T = T_{min}$$

$$\theta_P = 0^\circ$$

Phase 5

The terminal approach phase is initiated when the computed thrust, T_T, is equal to or greater than 75% of T_{max}. The computed thrust is obtained by solving a set of closed loop guidance laws which are used to represent the astronaut's VFR flying technique. These laws are similar to those used in rendezvous and docking studies, modified only to account for the gravity bias terms which exist for surface operations.

$$\text{Commanded Thrust} = T_c = m\sqrt{(\ddot{R})^2 + (R\ddot{\sigma})^2}$$

$$\text{Commanded pitch attitude} = \theta_c = \frac{\pi}{2} - \sigma_P - \alpha$$

where,

$\ddot{R}$ = acceleration along the line-of-sight

$$\ddot{R} = A_{LOS} - K_1 \left[A_{LOS} + \frac{\dot{R}_P \left| \dot{R}_P \right|}{2R_P} \right] + g_L \sin(\sigma_P)$$

$R\ddot{\sigma}$ = acceleration normal to the line-of-sight

$$R\ddot{\sigma} = -K_2 \dot{\sigma}_P + g_L \cos(\sigma_P) + K_3 (\sigma_{RF} - \sigma_P)$$

α = thrust orientation relative to the line-of-sight

$$\alpha = \tan^{-1}\left(\frac{R\ddot{\sigma}}{\ddot{R}}\right)$$

The constants used in the acceleration equations were obtained from piloted simulation runs which were made to verify the appropriateness of these guidance laws. Values for these constants are:

$K_1 = 2$

$K_2 = 500 \text{ fps}^2/\frac{\text{rad}}{\text{sec}}$

$K_3 = 0.25$

$A_{LOS} = 2.5 \text{ fps}^2$

The geometric relationships and equations required to compute the commanded accelerations are:

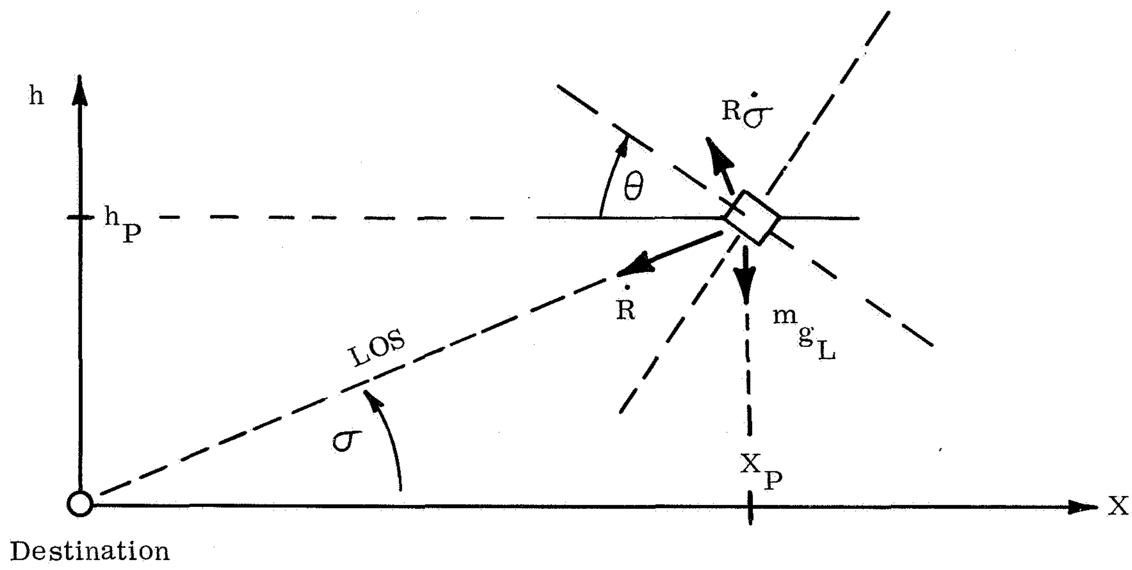
$$R_P = \sqrt{(X_P)^2 + (h_P)^2} = \text{perceived line-of-sight range}$$

$$\sigma_P = \tan^{-1}\left(\frac{h_P}{X_P}\right) = \text{perceived line-of-sight angle}$$

$$\dot{\sigma}_p = \frac{\dot{h}_P \cos(\sigma_P) - \dot{X}_P \sin(\sigma_P)}{R_P} \quad \text{perceived line-of-sight rotation rate}$$

$$\dot{R}_P = \dot{h}_P \sin(\sigma_P) + \dot{X}_P \cos(\sigma_P) = \text{perceived closure rate along the line-of-sight}$$

The relationship between the variables is illustrated in the sketch below.



Several constraints and guidance law modifications are included in the math model to approximate the actions of a pilot confronted with certain marginal operating conditions.

Changes in thrust from a given level to a newly computed command thrust will be executed only if the pitch attitude is within 5 degrees of the commanded pitch.

That is, $T = T_c \quad \text{if} \quad \left| \theta_P - \theta_c \right| \leq 5.0$

This guidance law modification represents the tendency of a pilot to first align the vehicle into the desired attitude before applying desired thrust.

Pitch-up attitudes with respect to the line-of-sight are limitted to values of 60 degrees.

$$\theta_c \leq 60.0 - \sigma_P$$

This constraint represents the tendency of the pilot to maintain visual contact with the target.

During the second half of the descent, possible requirements for rapid large angular maneuvers are eased by the introduction of an "apparent" target location. This modification in the command computation is to represent the flexibility of a pilot in tradingoff severe maneuvering requirements with off-nominal approaches and

touch-down points. The destination location used by the astronaut in his VFR approach is then given by:

$$\hat{X} = (0.8)\,(X) + (0.2)\,(h)\left(\frac{\dot{X}}{\dot{h}}\right)$$

Phase 6

The final touch-down phase is initiated when the vehicle is within 200 feet of the destination.

The commanded accelerations during this phase are

$$\ddot{h} = \frac{\dot{h}_P \left| \dot{h}_r \right|}{2h} + g_L$$

$$\ddot{X} = 2\,\frac{\dot{X}_P \ddot{h}}{\dot{h}_P} \cong \frac{-\dot{X}_P \left| \dot{h}_P \right|}{h}$$

$$T_c = m\sqrt{(\ddot{h})^2 + (\ddot{X})^2}$$

$$\theta_c = \tan^{-1}\left(\frac{\ddot{X}}{\ddot{h}}\right)$$

Touchdown is completed when either

or $\dot{h} \geq 0$
$h \leq 1$ ft

2. ERROR MODEL

A special feature of the math model is that the system variables and selected system parameters are treated as random variables. Normal distributions with specified means and standard deviations was assumed in all cases.

A first group of random variables represents the variations of the actual system parameter or initial condition values from their specified nominal values. Treated in this manner are:

X_o	the range-to-go (initial distance to the target),
m_o	the initial (lift-off) mass of the vehicle,
T_{max}, T_{min}	the maximum and minimum thrust levels
I_{sp}	the specific impulse

For each of these the mean is equal to the specified nominal value, and the standard deviation is indicative of the uncertainity in the actual value; Table B.1 lists the values of the statistics that were used.

A second group of random variables represents those system variables that are estimated or measured by the pilot and used in computing command signals. The general form of a variable in this group is;

observed value = actual value + error terms

The error terms are computed as products of system variables and random error percentages, where the error percentages are normally distributed with zero mean and some selected standard deviation.

Expressions for the measured system variables are given below, where E represents the random error percentage, subscripts P imply "observed by pilot", and non-subscripted variables signify actual values.

range	$X_P = X + E_{xx} \cdot X$
altitude	$h_P = h + E_{hh} \cdot h$
range rate	$\dot{X}_P = \dot{X} + E_{\dot{x}h} \cdot h + E_{\dot{x}\dot{x}} \cdot \dot{X}$
alt. rate	$\dot{h}_P = \dot{h} + E_{\dot{h}h} \cdot h + E_{\dot{h}\dot{h}} \cdot \dot{h}$
pitch attit.	$\theta_P = \theta + E_\theta$

Note that E_θ represents an angular error rather than a percentage since errors in angular measurements were assumed to depend only on the measuring technique. The accuracy of the measured variables depends on the instrumentation available to the pilot and thus it is this second group of random variables that is affected by the introduction of navigation aids. Table B.1 lists the values of the standard deviations (one sigma (σ) values for the error percentages identified above) corresponding to the various specified navigation aids.

The selection of appropriate random error values for a given set of mean values and standard deviations is accomplished prior to each simulation run by use of a subroutine (Subroutine GAUSS).

3. FAILURE CONDITIONS

The performance analysis required that some means be provided for discriminating against flights which approach marginal inflight or terminal conditions. While in actual flight, the astronaut would not allow such conditions to develop (e.g., touching down with large horizontal velocities) the math model did not include additional guidance laws which would prevent such situations from occurring. The frequency with which these conditions occur provides a means for assessing the sensitivity of a flight to errors, since they will be encountered more often in the presence of large errors

than with small errors. Thus, the failure statistics should not be interpreted as a reliability measure for mission completion but rather as a measure of error sensitivity.

The following conditions determine if a flight is rated a failure.

(1) If the vehicle reaches the burn-out weight before it touches down. $m_f < m_{dry}$

(2) If the flight trajectory is so low that the line-of-sight angle is less than 1°.

$$\sigma \leq 1^\circ$$

(3) If the vehicle does not touch down within 200 feet of the specified target.

$$\text{Final value } \left| X_{fin} \right| > 200$$

(4) If the vehicle velocities at touchdown are excessive.

$$\left| \dot{X}_{fin} \right| + \left| \dot{h}_{fin} \right| > 10 \text{ ft/sec}$$

TABLE B-1

RANDOM VARIABLES NOT AFFECTED BY NAVIGATION EQUIPMENT

		Mean Value	Standard Deviation	Major Source of Uncertainty
Initial Range	X_o – ft	$\bar{X}_o$	600.	Exact target not defined at lift-off
Initial Mass	M_o - slugs	$\bar{M}_o$	0.6	Payload and Fuel not precisely known
Maximum Thrust Level	T_{max} – lb	300.	6.	Temperature and Pressure Variation in the propellant system
Minimum Thrust Level	T_{min} - lb	50.	1.	
Specific Impulse	I_{sp} - sec	280.	1.	
Boost Duration	t_B – sec	$\bar{t}_B$	1.	Variations in pilot reaction time

APPENDIX C

CONTROL SYSTEM ANALYSIS AND SIMULATION

1. SIMULATION

The four vehicle configurations simulated for the study are summarized in Table C-1.

Two simulations of varying complexity were used in the study. A simplified six degree of freedom simulation with fixed vehicle mass properties (no variations in inertia, mass, or center of gravity (cg)), and no products of inertia or trim requirements was used for the parametric manual TVC studies described in Section 6.4 of the text. For the final evaluation of the selected manual configuration and the stability augmentation studies (Sections 6.5 and 6.6) an expanded simulation was used which included the variation in vehicle mass properties due to propellant burnoff, varying cg offset trim requirements, and the significant vehicle product of inertia terms.

TABLE C-1

VEHICLE CONFIGURATIONS SIMULATED

Configuration	Pitch	Yaw	Roll
I	Pivots high	Diff. gimballing	Diff. throttling
II	Pivots low	Diff. gimballing	Diff. throttling
III	Translating thrusters	Diff. gimballing	Diff. throttling
IV	Pivots high	Diff. gimballing	Pivots (high)

1.1 Simulation Facility (Figures C-1 and C-2)

The six degree of freedom simulation facility utilized in the study consisted of a 3 axis pilots station driven in pitch, roll, and yaw, used in conjunction with a TV camera-model-track (CMT) which projected a lunar scene on a 9 x 12 ft screen in front of the operator's station from which the pilot derived the three translational cues (down range, lateral, and altitude). Since the pilot is an integral part of the control system, the sensing of body motion is an important cue which enables controllability of these simple vehicles. The 3-axis pilots station provided ±60 degrees rotation in pitch, ±180 degrees in yaw and ±15 degrees in roll.

The vehicle equations of motion and control system characteristics were programmed on an analog computer. The pilot control inputs (pitch, yaw, roll and throttle)

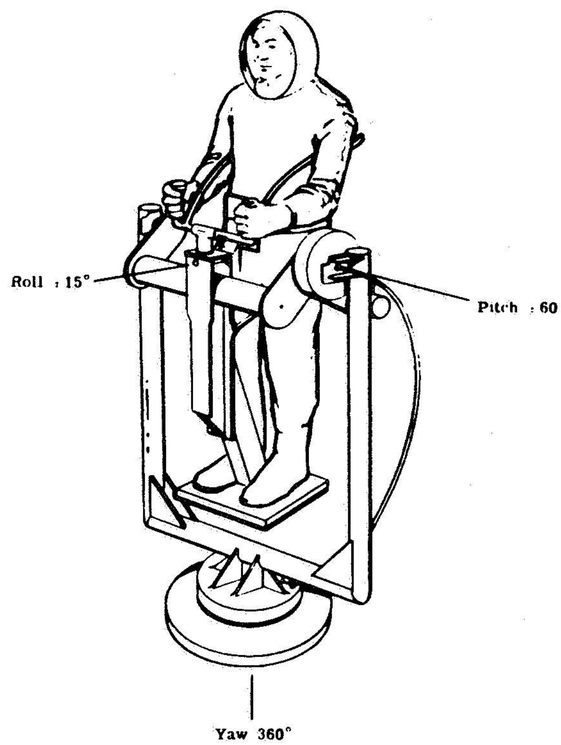


Figure C-1. 3-Axis Simulator

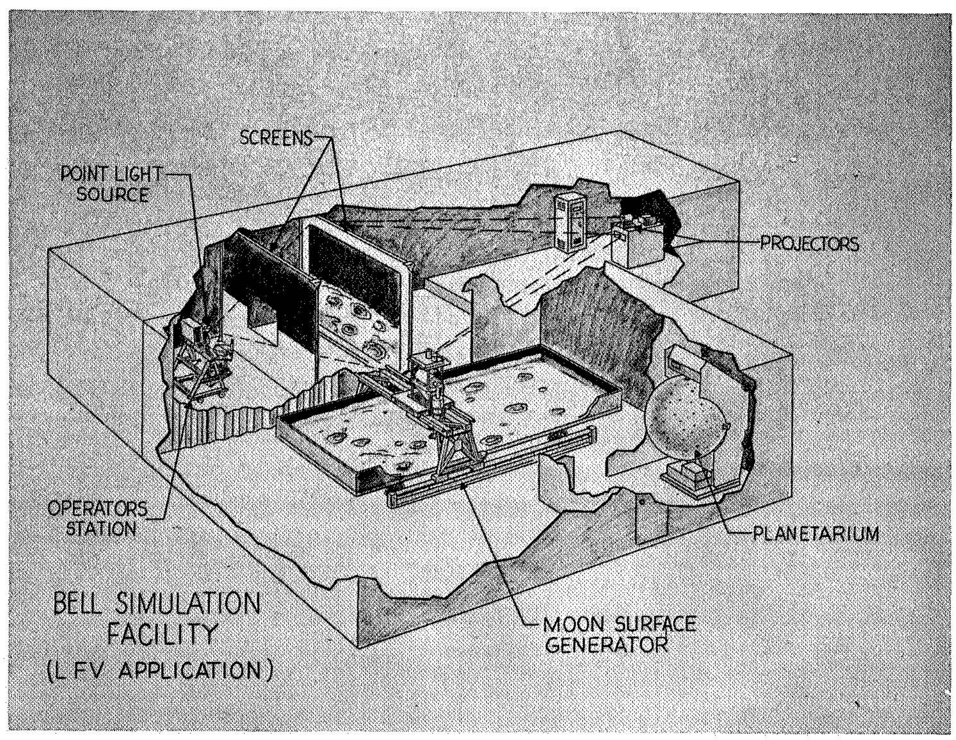


Figure C-2. Visual Simulation Facility

were processed on the analog and six position commands were generated for driving the three position servos on the CMT and the three position servos on the 3 axis gimballed pilots station. A simple simulation flow block diagram is shown in Figure C-3.

Two pilot hand controller arrangements were used in the study: (1) a handle bar arrangement similiar to those presently used on the earth rocket belt, pogo vehicle, and jet belt, and (2) an Apollo (block one) hand controller.

1.2 Analog Simulation

Figure C-4 summarizes in block diagram form the analog simulation of the vehicle equations of motion.

1.2.1 Vehicle Mathematical Model

A general mathematical model of the vehicle is shown in Figure C-5 which defines the vehicle parameters for configuration I through IV when the appropriate conditions listed are applied.

1.2.2 Assumptions

Small angle assumption ($\sin \alpha = \alpha$, $\cos \alpha = 1.0$) were assumed for the gimballed thruster angles (δ_θ, δ_ψ, δ_ϕ) and for the vehicle euler roll (ϕ). The roll cockpit gimbal is limited to ±15 degrees.

1.2.3 Equations of Motion

The vehicle equations of motion for configuration I through IV of Table C-1 are summarized in Table C-2 and are cross referenced by equation numbers (1 through 6) to the associated block in the analog simulation block diagram (Figure C-4). Configuration I which was used in both the selected configuration studies (Section 6.5) and in the stability augmentation studies (Section 6.6) was the only vehicle used in the expanded simulation which included variable vehicle mass properties (due to propellant burnoff), trim requirements and product of inertia terms.

The remaining equations in the analog simulation which are common to all vehicles are as follows:

$$\left.\begin{aligned} p &= \int \dot{P}\,dt \\ q &= \int \dot{q}\,dt \\ r &= \int \dot{r}\,dt \end{aligned}\right\} \quad \text{Equation 7}$$

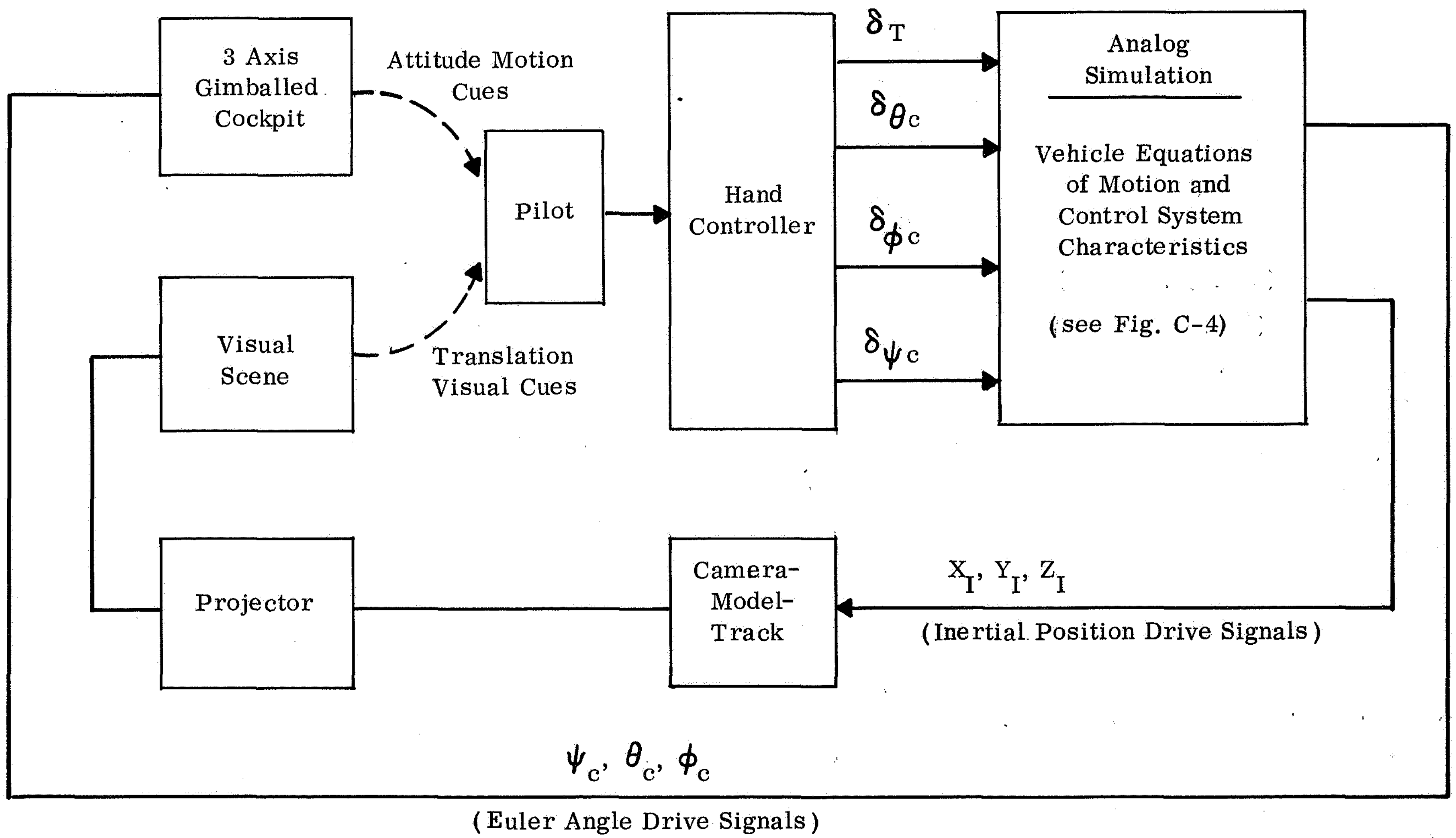


Figure C-3. Simulator Flow Diagram

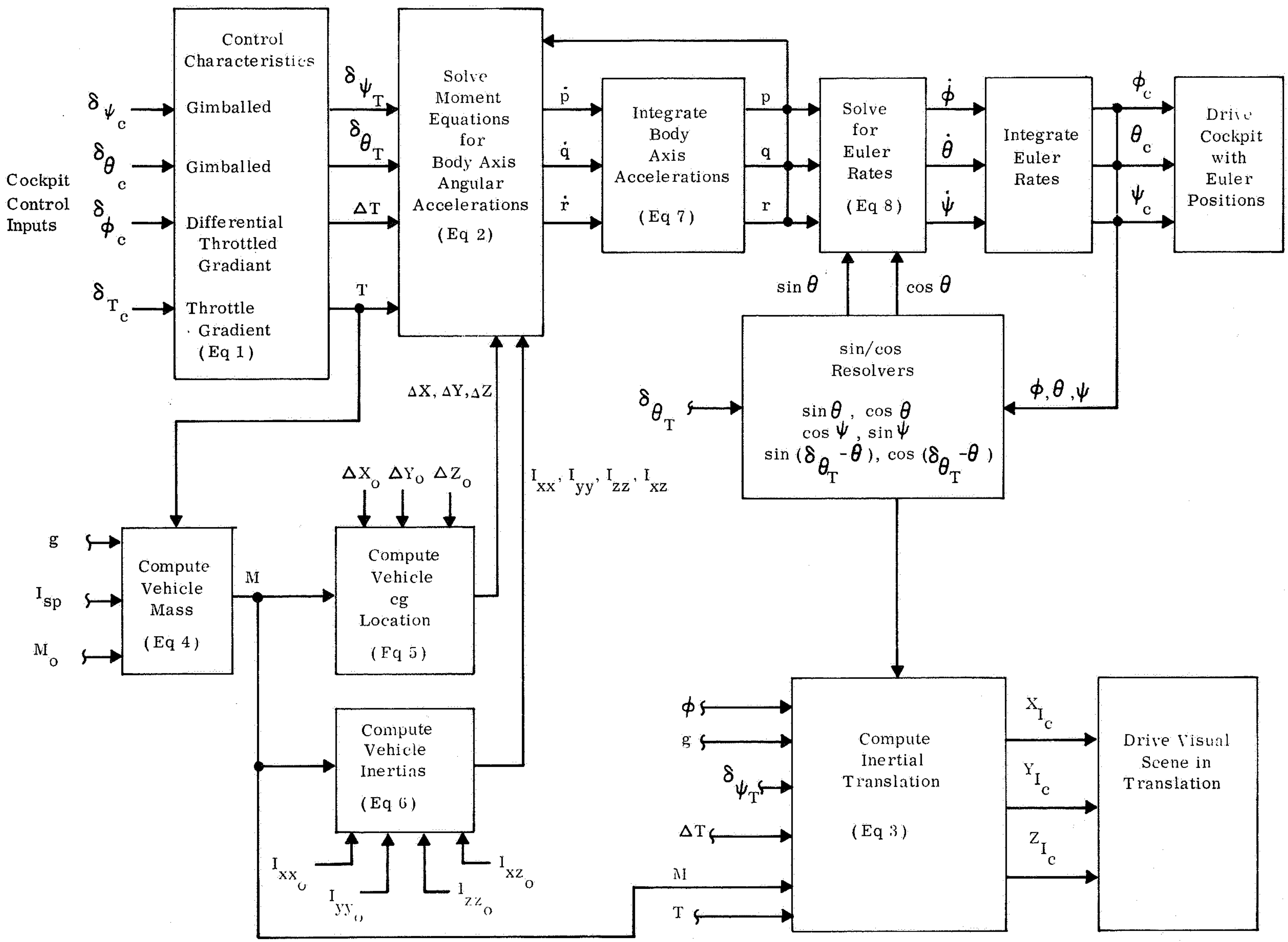
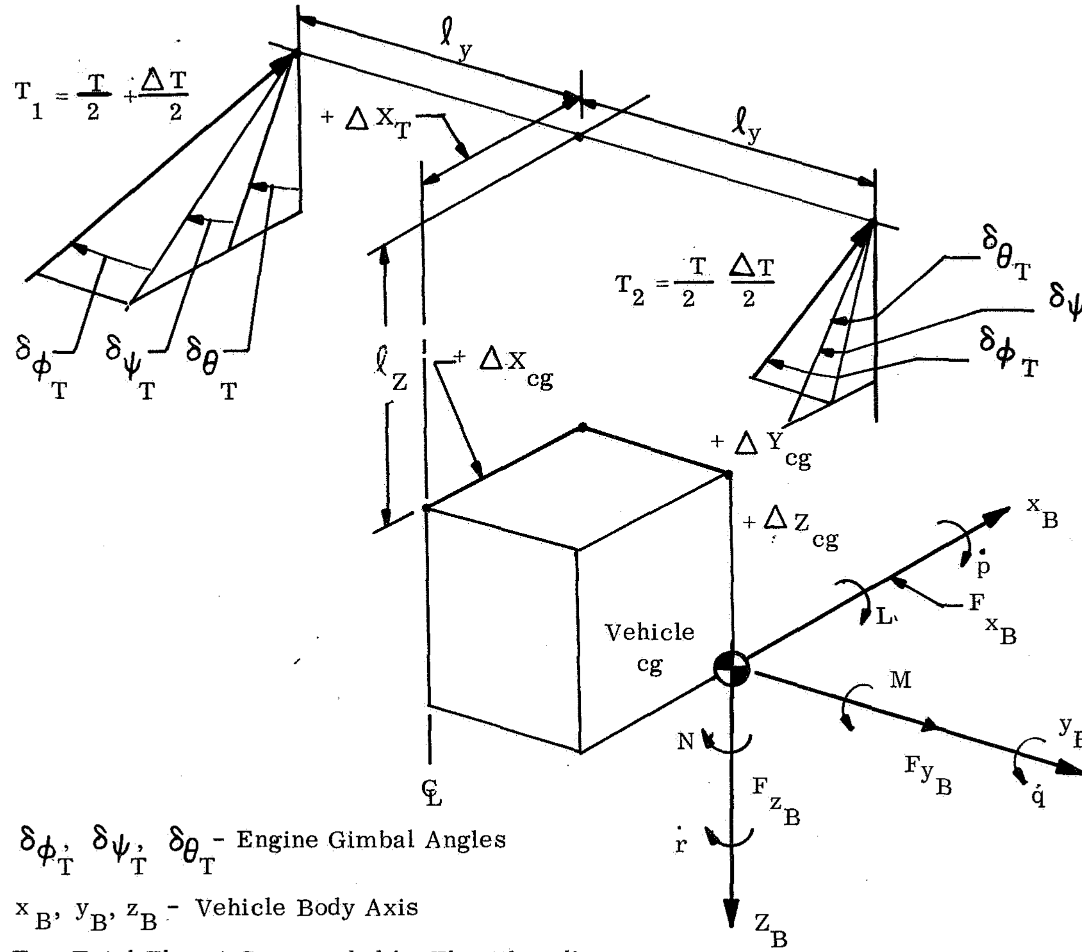


Figure C-4. Analog Simulation Block Diagram of the Vehicle Equations of Motion



$\delta\phi_T$, $\delta\psi_T$, $\delta\theta_T$ - Engine Gimbal Angles

x_B, y_B, z_B - Vehicle Body Axis

T - Total Thrust Commanded by Throttle - lb

ΔT - Total Differential Thrust Commanded by Roll Controller for Differential Throttling - lb

ΔX_T - Fore and Aft Travel of Translating Thrusters in Pitch - ft

	Pitch	Yaw	Roll	Condition
Config I	Pivots High	Gimballing	Dif Throttling	$\delta\phi_T = 0 \quad \Delta X_T = 0$
Config II	Pivots Low	Gimballing	Dif Throttling	$\delta\phi_T = 0 \quad \Delta X_T = 0 \quad \ell_z = -\ell_z$
Config III	Translating Thrusters	Gimballing	Dif Throttling	$\delta\phi_T = 0 \quad \delta_\theta = 0$
Config IV	Pivots High	Gimballing	Gimballed	$\Delta X_T = 0 \quad \Delta T = 0$

Figure C-5. Mathematical Model

TABLE C-2

SUMMARY OF EQUATION OF MOTION FOR CONFIGURATION I THROUGH IV

		Configuration No. 1	Configuration No. 2	Configuration No. 3	Configuration No. 4
Control Characteristics (Eq. 1)	Yaw	δ_{ψ_T} (rad) $= \frac{K_\psi}{57.3}\,\delta_{\psi_c}$ (deg)	Same as Configuration No. 1	Same as Configuration No. 1	Same as Configuration No. 1
	Pitch	δ_{θ_T} (rad) $= \frac{K_\theta}{57.3}\,\delta_{\theta_c}$ (deg)		ΔX_T (ft) $= K_\theta\,\delta_{\theta_c}$ (deg)	Same as Configuration No. 1
	Roll	ΔT (lb) $= K_\psi\,\delta_{\phi_c}$ (deg)		Same as Configuration No. 1	δ_{ϕ_T} (rad) $= \frac{K_\phi}{57.3}\,\delta_{\phi_c}$ (deg)
		K_ψ, K_θ - controller to thruster linkage (N.D.) K_θ - total differential thrust per deg (lb/deg) ΔT - total differential thrust between engine (lb)		K_θ - $\frac{\text{Thruster Translation}}{\text{Deg. of Controller}}$	K_ϕ = Roll controller to thruster linkage (ND)
Body Axis Angular (Eq. 2) Accelerations		$\dot p$ (rad/sec^2) $= \frac{\Delta T \ell_y}{I_{xx}} + \left[\frac{T\,\Delta Y_{cg}}{I_{xx}} + \frac{I_{xz}}{I_{xx}}\dot r - \frac{(I_{zz}-I_{yy})}{I_{xx}}qr\right]^*$ $\dot q$ (rad/sec^2) $= -\frac{(T\delta_{\theta_T} + T\delta_{\theta_T})\ell_z}{I_{yy}} - \left[\frac{(T\delta_{\theta_T} + \Delta T\delta_{\psi_T})\Delta z}{I_{zz}} - \frac{T\Delta X_{cg}}{I_{yy}} - \frac{I_{xx}-I_{zz}}{I_{yy}}pr\right]^*$ $\dot r$ (rad/sec^2) $= \frac{(\Delta T\delta_{\theta_T} + T\delta_{\psi_T})\ell_y}{I_{zz}} + \frac{(\Delta T\delta_{\psi_T} + T\delta_{\phi_T})\Delta Y_{cg}}{I_{zz}}$ $+ \left[\frac{I_{xz}}{I_{zz}}\dot p - \frac{I_{yy}-I_{xx}}{I_{zz}}pq\right]^*$	$\dot p = \frac{\Delta T \ell_y}{I_{xx}}$ $\dot q = -\frac{(-T\delta_{\theta_T} + \Delta T\delta_{\psi_T})(-\ell_z)}{I_{yy}}$ $\dot r = \frac{(-\Delta T\delta_{\theta_T} + T\delta_{\psi_T})\ell_y}{I_{zz}}$	$\dot p = \frac{\Delta T \ell_y}{I_{xx}}$ $\dot q = \frac{-\Delta T\,\delta_{\psi_T}\ell_z}{I_{yy}} + \frac{T\Delta X_T}{I_{yy}}$ $\dot r = \frac{T\,\delta_{\psi_T}\ell_y}{I_{zz}}$	$\dot p = \frac{T\ell_z}{I_{xx}}\delta_\theta$ $\dot q = -\frac{T\,\delta_{\theta_T}\ell_z}{I_{yy}}$ $\dot r = -\frac{T\,\delta_{\psi_T}\ell_y}{I_{zz}}$
Inertia Linear (Eq. 3) Accelerations		$\ddot X_I\left(\frac{ft}{sec^2}\right) = \frac{T}{M}\cos\psi\sin(\delta_{\theta_T}-\theta) - \frac{T\Phi}{M}\sin\psi + \frac{\Delta T\,\delta_{\psi_T}}{M}\cos\psi\cos(\delta_{\theta_T}-\theta)$ $\ddot Y_I\left(\frac{ft}{sec^2}\right) = \frac{T}{M}\sin\psi\sin(\delta_{\theta_T}-\theta) + \frac{T\Phi}{M}\cos\psi + \frac{\Delta T\,\delta_{\psi_T}}{M}\sin\psi\cos(\delta_{\theta_T}-\theta)$ $\ddot Z\left(\frac{ft}{sec^2}\right) = -\frac{T}{M}\cos(\delta_{\theta_T}-\theta) + \frac{\Delta T}{M}\delta_\psi\sin(\delta_{\theta_T}-\theta) + g$	$\ddot X_I = \frac{T}{M}\cos\psi\sin(-\delta_{\theta_T}-\theta) - \frac{T\Phi}{M}\sin\psi + \frac{\Delta T\delta_{\psi_T}}{M}\cos\psi\cos(-\delta_{\theta_T}-\theta)$ $\ddot Y_I = \frac{T}{M}\sin\psi\sin(-\delta_{\theta_T}-\theta) + \frac{T(\Phi)}{M}\cos\psi + \frac{\Delta T\delta_{\psi_T}}{M}\sin\psi\cos(-\delta_{\theta_T}-\theta)$ $\ddot Z_T = -\frac{T}{M}\cos(-\delta_{\theta_T}-\theta) + \frac{\Delta T}{M}\delta_\psi\sin(-\delta_{\theta_T}-\theta) + g$	$\ddot X_I = \frac{T}{M}\cos\psi\sin(-\theta) - \frac{T\Phi}{M}\sin\psi + \frac{\Delta T\,\delta_{\psi_T}}{M}\cos\psi\cos(-\theta)$ $\ddot Y_I = \frac{T}{M}\sin\psi\sin(-\theta) + \frac{T\Phi}{M}\cos\psi + \frac{\Delta T\,\delta_{\psi_T}}{M}\sin\psi\cos(-\theta)$ $\ddot Z_i = -\frac{T}{M}\cos(-\theta) + \frac{\Delta T}{M}\delta_\Phi\sin(-\theta) + g$	$\ddot X_I = \frac{T}{M}\cos\psi\sin(\delta_{\theta_T}-\theta) - \frac{T}{M}(\Phi + \delta_\phi)\sin\psi$ $\ddot Y_I = \frac{T}{M}\sin\psi\sin(\delta_{\theta_T}-\theta) + \frac{T}{M}(\Phi + \delta_\phi)\cos\psi$ $\ddot Z_I = -\frac{T}{M}\cos(\delta_{\theta_T}-\theta) + g$
Computing (Eq. 4) Vehicle Mass		m (slugs) $= m_o - \left[\int \dot m\,dt\right]^*$ $\quad \dot m = \frac{T}{g_e I_{sp}}$ $\quad I_{sp}$ = 280 sec (assumed constant)	$m = m_o$	Same as Configuration No. 2	Same as Configuration No. 2
Computing Changing (Eq. 5) Vehicle cg Offset		ΔX_{cg} (ft) $= [\Delta X_o + f_1(m)]^*$ ΔY_{cg} (ft) $= [\Delta Y_o + f_2(m)]$ ΔZ_{cg} (ft) $= [\Delta Z_o + f_3(m)]$ $f_n(m)$ (n = 1,2,3) = cg variation assumed a linear function of mass (m); established from full to empty propellant condition for a given payload.	$\Delta X_{cg} = \Delta Y_{cg} = \Delta Z_{cg} = 0$		
Computing Variable (Eq. 6) Inertia		I_{xx} (slug ft^2) $= I_{xx_o} + [f_4(m)]^*$ $I_{yy} = I_{yy_o} + f_5(m)$ $I_{zz} = I_{zz_o} + f_6(m)$ $I_{xz} = I_{xz_o} + f_7(m)$ $f_n(m)$ (n = 4,5,6,7) = Moment of inertia variation assumed linear function of mass (m) established from full to empty propellant condition for a given payload.	$I_{xx} = I_{xx_o}$ $I_{yy} = I_{yy_o}$ $I_{zz} = I_{zz_o}$ $I_{xz} = 0$		

(*Note: Terms set equal to zero for simplified 6 DOF Sim.)

$$\left.\begin{aligned} \phi &= p + \psi \sin\theta \\ \theta &= q - r\phi \\ \psi &= \frac{1}{\cos\theta}(r - q\phi) \end{aligned}\right\} \quad \text{Equation 8}$$

1.3 Frequency Response of Gimballed Cockpit

The frequency response characteristics of the three axes of the gimballed cockpit are shown in Figures C-6 through C-8 for sinusoidal input amplitude of ±3 degrees and in Figures C-9 through C-11 for an amplitude of ±1 degree. Both the uncompensated and compensated response is shown in each figure.

The compensation was attained by providing lead in the input to each gimbal servo loop. For example, the input θ_i to the pitch servo consisted of the desired pitch attitude θ_c of the rocket plus a constant, τ, times the rate of change θ_c of the desired pitch attitude, or

$$\theta'_c = \theta_c + \tau \dot{\theta}_c$$

Based on transient response tests using ramp inputs, it was originally estimated that τ should be approximately 0.2 seconds for pitch and yaw, and 0.125 seconds for roll. These values were used for all runs subsequent to the parametric studies of Section 6.4. Frequency response measurements made at the conclusion of the program (shown in Figure C-6 through C-11) illustrate that this compensation was ideal for pitch and yaw, where it resulted in essentially flat response in the important range below 3.0 rad/sec. Apparently, however, a value of $\tau = 0.2$ should also have been used in roll. The 0.125 sec compensation actually used in roll did, however, reduce the phase shift from 30° in the uncompensated case to 14° in the compensated case at 3 rad/sec.

2. MECHANICAL CONTROL AUGMENTATION (EQUATIONS OF MOTION)

The Lagrange equations were used to obtain the equations of motion for the mechanical system shown in Figure 6.39, Section 6.0.

$$\frac{d}{dt}\left(\frac{\partial T}{\partial q_i}\right) - \frac{\partial T}{\partial q_i} + \frac{\partial U}{\partial \delta_e} + \frac{\partial D}{\partial \delta_e} = Q_i \qquad \text{(Eq. 1)}$$

$$\left.\begin{aligned} &\text{Kinetic energy:} && T = \frac{1}{2} I_e \dot{\delta}_e^{\,2} \\ &\text{Potential energy:} && U = \frac{1}{2} K_T (\ell_c \delta_c - \ell'_e \delta_e)^2 + \frac{1}{2} K_e (\ell_e \delta_e)^2 \\ &\text{Dissipation function:} && D = \frac{1}{2} B (\ell'_e \dot{\delta}_e - \ell_c \dot{\delta}_c)^2 \end{aligned}\right\} \qquad \text{(Eq. 2)}$$

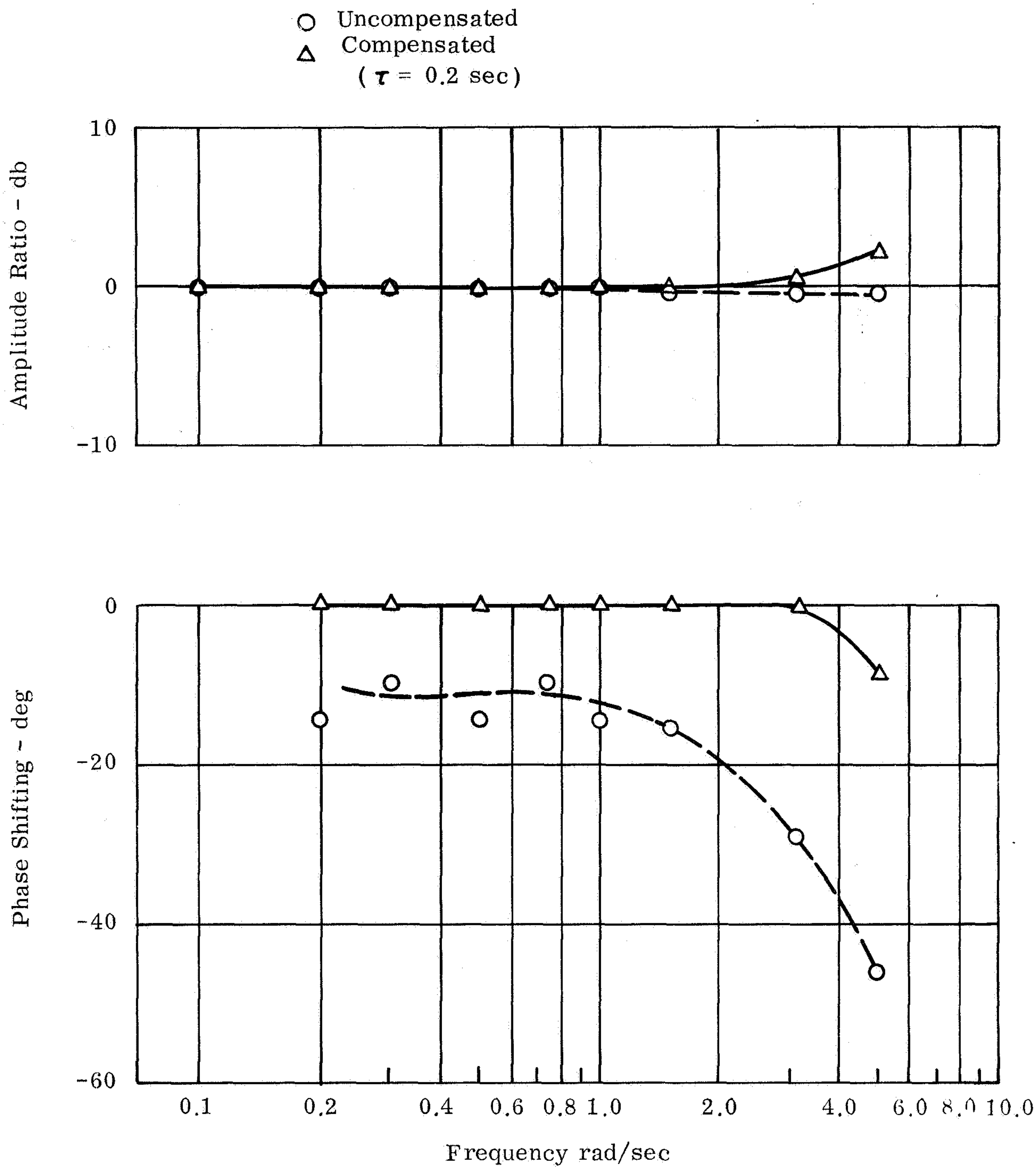


Figure C-6. Cockpit Frequency Response - Pitch Axis (amplitude ±1.0 degrees)

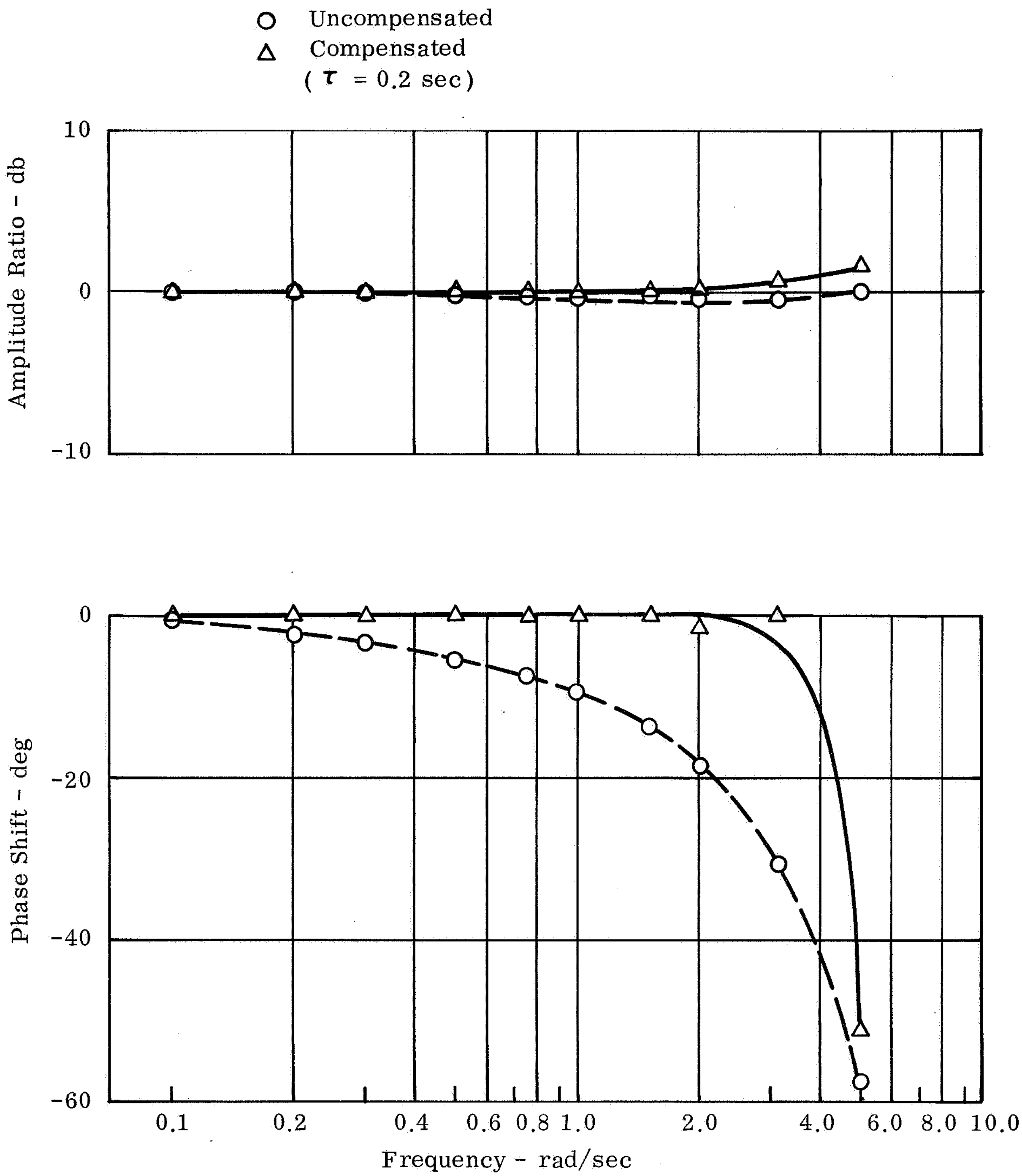


Figure C-7. Cockpit Frequency Response - Pitch Axis
(amplitude = ±3.0 degrees)

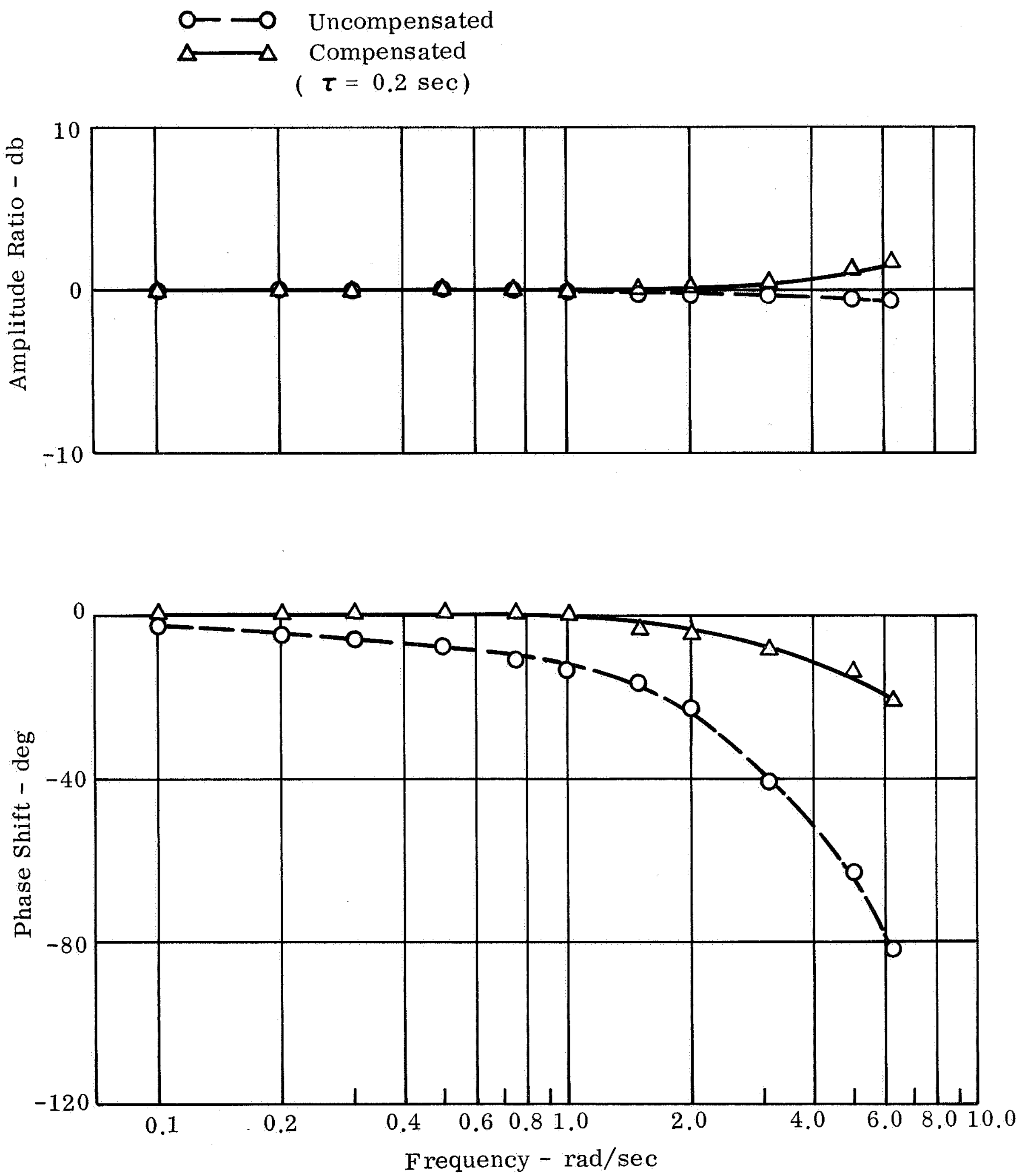


Figure C-8. Cockpit Frequency Response - Yaw Axis
(amplitude ±1.0 degrees)

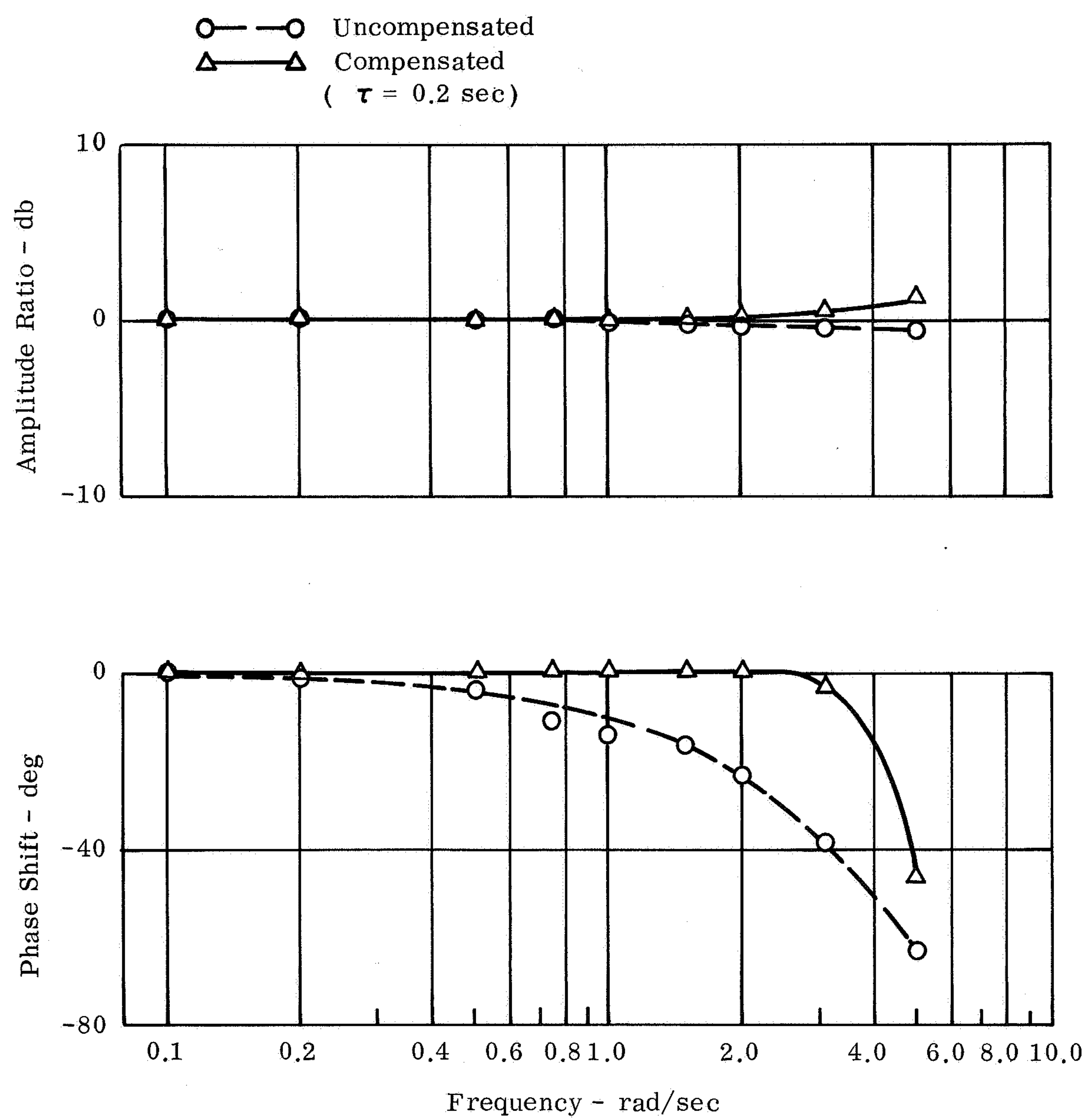


Figure C-9. Cockpit Frequency Response - Yaw Axis (amplitude ± 3.0 degrees)

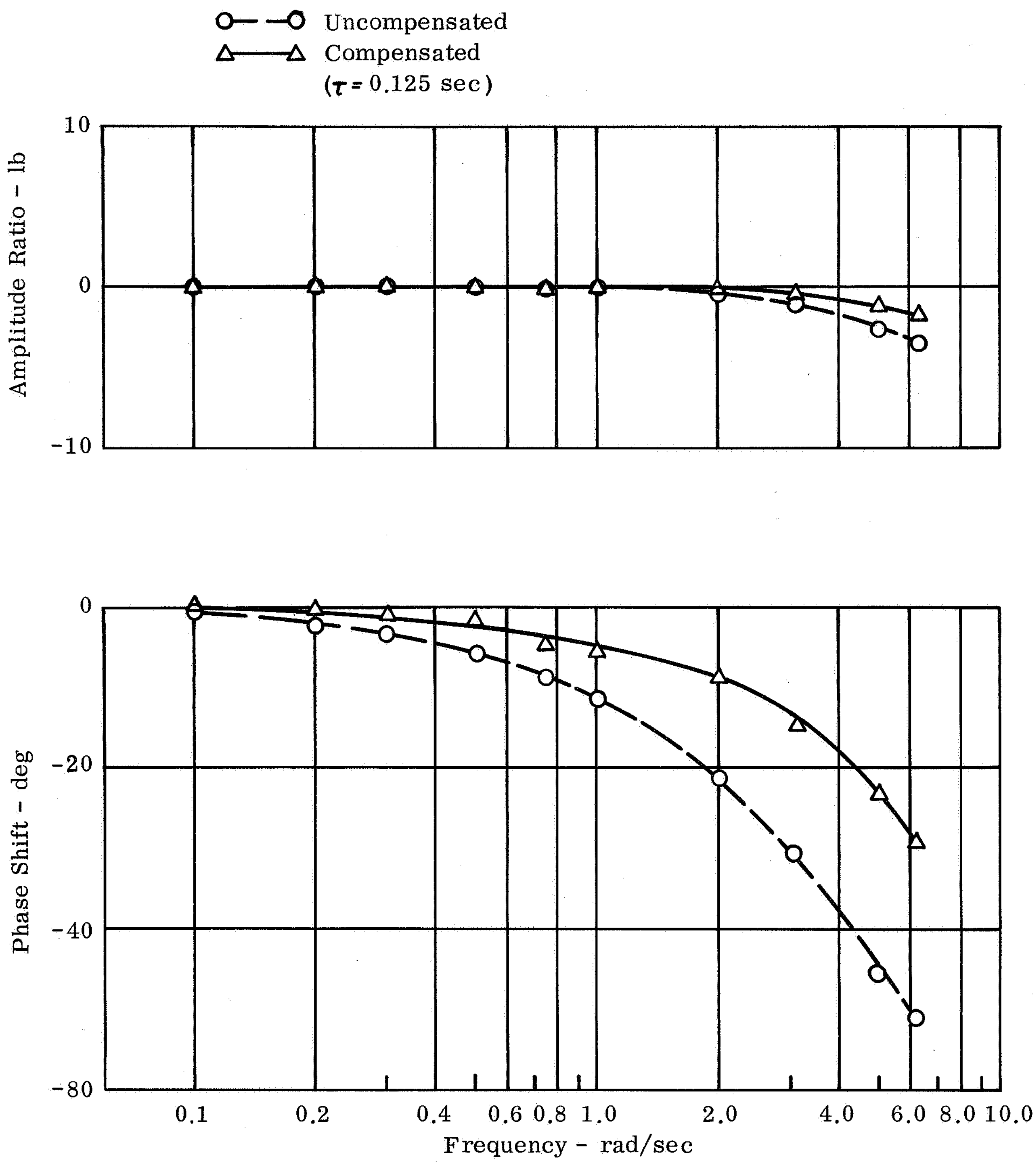


Figure C-10. Cockpit Frequency Response - Roll Axis
(amplitude ±1.0 degrees)

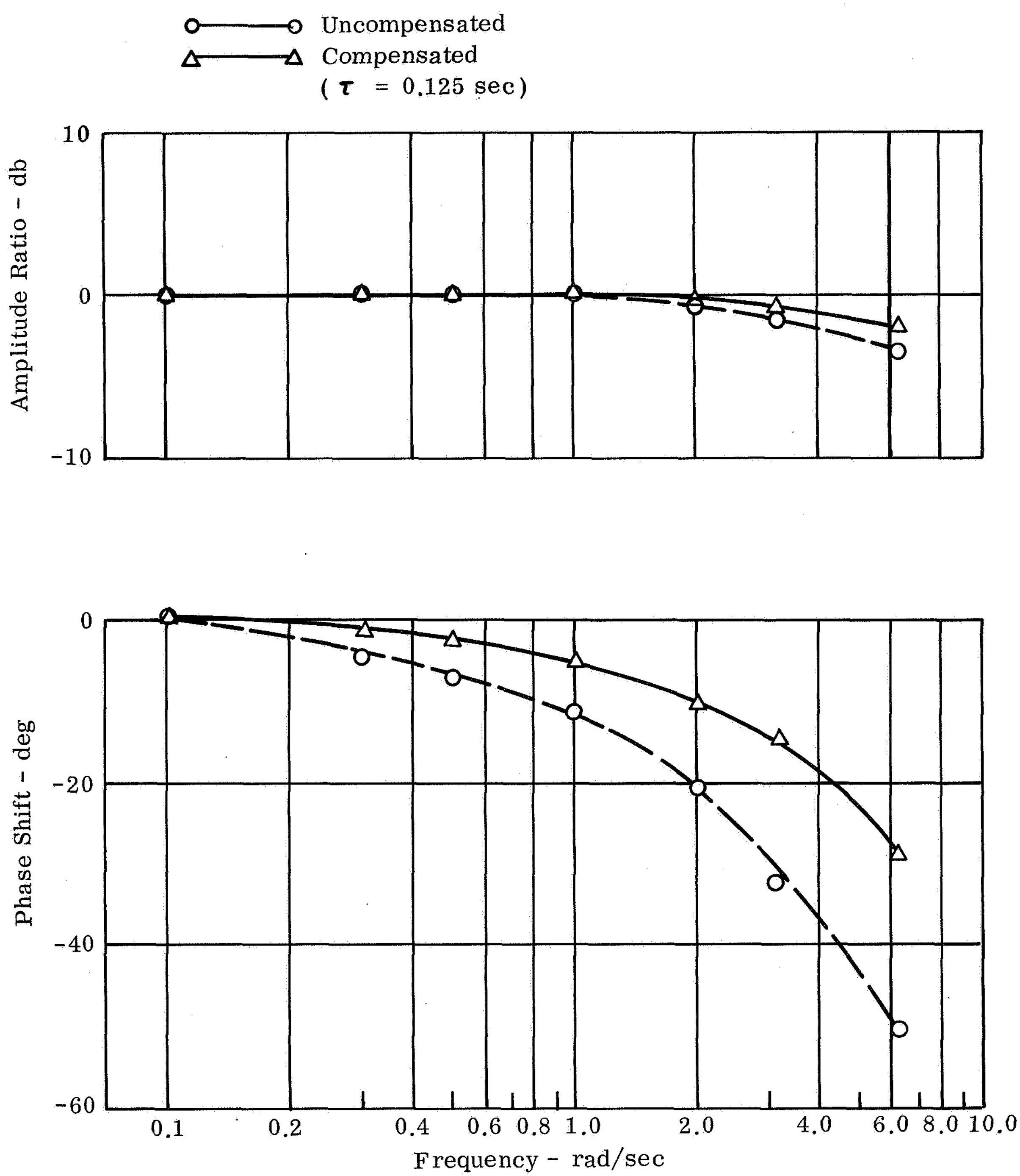


Figure C-11. Cockpit Frequency Response - Roll Axis
(amplitude ±3.0 degrees)

Substituting the expressions of equation 2 into equation 1 and performing the indicated operations results in the two differential equations which describe the system dynamics:

$$\left.\begin{aligned} & I_e \ddot{\delta}_e + B \ell_e'^{2} \dot{\delta}_e + (K_T \ell_e'^{2} + K_e \ell_e^{2}) \delta_e = \ell_e' \ell_c (B \dot{\delta}_c + K_T \delta_c) \\ & \ell_e' \ell_c (B \dot{\delta}_e - K \delta_e) + \ell_c^{2} (K_T \delta_c - B \dot{\delta}_c) = F \ell_H \end{aligned}\right\} \quad \text{(Eq. 3)}$$

The former equation describes the relationship between the controller and engine motions. The latter equation relates the hand force to engine and controller motion.

For the preliminary study, only the dynamic relationship between controller and engine motion were of interest. The Laplace transform of the equation becomes:

$$\frac{\delta_e}{\delta_c} = \frac{\ell_e' \ell_c (B S + K_T)}{I_e S^2 + B \ell_e'^{2} S + (K_T \ell_e'^{2} + K_e \ell_e^{2})} \qquad \text{(Eq. 4)}$$

Rearranging equation 4:

$$\frac{\delta_e}{\delta_c} = \frac{\ell_c / \ell_e'}{\alpha} \; \frac{\alpha \tau S + 1}{\left(\frac{S}{\omega}\right)^2 + \tau S + 1}$$

where $\alpha = \dfrac{K_T \ell_e'^{2} + K_e \ell_e^{2}}{K_T \ell_e'^{2}}$

$$\tau = \frac{B \ell_e'^{2}}{K_T \ell_e'^{2} + K_e \ell_e^{2}} = \frac{2\zeta}{\omega}$$

$$\omega_n = \sqrt{\frac{K_T \ell_e'^{2} + K_e \ell_e^{2}}{I_e}}$$ - engine natural frequency.

The expression relating the commanded vehicle angular acceleration to controller input becomes

$$\frac{\ddot{\theta}}{\delta_c} = \frac{T\ell}{I_v}\left[\frac{\ell_c / \ell_e{}'}{\alpha} \; \frac{\alpha \tau S + 1}{\left(\frac{S}{\omega}\right)^2 + \tau S + 1}\right]$$

where $\frac{T\ell}{I_v} = \frac{\ddot{\theta}}{\delta_e}$ The vehicle control sensitivity based on engine deflection.

For the parametric studies, it was assumed that the engine frequency could be made sufficiently large compared to the input frequency ($\approx$0.5 cps) and sufficiently well damped that the engine inertia effects can be neglected. The second order denominator then can be approximated by the single order lag (τ S + 1).

The final transform reduces to:

$$\frac{\ddot{\theta}}{\delta_c} = K \, \frac{\alpha \tau S + 1}{\tau S + 1}$$

$$\text{where } K = \frac{T\ell}{I_v} \, \frac{\ell_c / \ell_e{}'}{\alpha} \text{ or } \frac{T\ell}{I_v} \, \frac{K_2}{\alpha}$$

Similar expressions were used for roll and yaw.

3. DESCRIPTION OF EVALUATION MEASURES

Quantitative measures used in the study were the standardized pilot (Cooper) rating scale described in Table C-3, and objective workload and performance measures.

3.1 Pilot Workload Measure

The pilot was given a secondary task to perform while flying the vehicle. The pilot's performance on the secondary task provided a measure of the workload associated with the primary task of flying the vehicle.

The secondary task used was a bipolar nulling task. A meter was randomly displaced to the left or right by a signal from an analog random sampling circuit. The pilot operated two push buttons, one on each hand grip (yaw and throttle) of the handle bar controller. The pilots task was to null the meter when the primary task of flying permitted the time. The left button nulled the meter only when the needle was deflected to the left and the right button nulled the meter only when it was deflected to the right. Once the pilot released the button, the meter was immediately repositioned.

TABLE C-3

REVISED PILOT RATING SCALE WITH ADDITIONAL CHANGES INCORPORATED

Controllable Capable of being controlled or managed in context of the task or flight phase, with available pilot attention.	Acceptable to Pilot *Pilot compensation, if required to achieve acceptable performance in task is feasible. *May have deficiencies for which pilot desires improvement, but adequate for task or flight phase.	Satisfactory to Pilot *Meets all demands and expectations. *Clearly adequate for the task or flight phase. *Good enough without improvement.	*Excellent, highly desirable	1
			*Good, pleasant, well behaved	2
			*Fair. Some mildly unpleasant characteristics. *Good enough for task or flight phase without improvement.	3
		Unsatisfactory to Pilot *Reluctantly acceptable. *Deficiencies which warrant improvement. *Performance adequate for task or flight phase with feasible pilot compensation.	*Some minor but annoying deficiencies. *Effect on performance is easily compensated for by pilot. *Improvement is requested.	4
			*Moderately objectionable deficiencies. *Reasonable performance requires considerable pilot compensation. *Improvement is needed.	5
			*Very objectionable deficiencies. *Requires best available pilot compensation to achieve acceptable performance. *Major improvements are needed.	6
	Unacceptable to Pilot *Deficiencies which require mandatory improvement. *Inadequate performance for task or flight phase even with maximum feasible pilot compensation.		*Major deficiencies but controllable. *Performance inadequate or pilot compensation required for minimum acceptable performance in task or flight phase is too high. *Requires mandatory improvement for acceptance.	7
			*Controllable with difficulty in task and finger phase. *Requires substantial pilot skill to retain control and continue mission.	8
			*Marginally controllable in task or flight phase. *Requires maximum available pilot skill to retain control.	9
Uncontrollable *Control will be lost during some portion of task and flight phase.			*Uncontrollable in task or flight phase.	10

A calibration or baseline reference was established by determining the number of correct responses per second that the pilot could perform in a two minute period when he devoted his entire attention to that task. By comparing this baseline response rate with the response rate during a flight, a quantitative measure of the workload involved in the flight task was obtained.

In the study, the secondary task results were presented in either of the two following forms.

$$\text{(A)} \quad \frac{\text{Response rate for the flight (Based on correct responses only)}}{\text{Calibration (Baseline) response rate}}$$

$$\text{(B)} \quad \text{W.L.} = 1 - \frac{\text{Response rate for the flight (Based on correct responses only)}}{\text{Calibration (Baseline) response rate}}$$

For the former ratio, the larger the ratio the easier the primary task of flying (pilot could devote more time to operating secondary task). The latter form of presenting the results is a direct indicator of workload. The lower the value, the lower the workload associated with the primary task.

3.2 Performance Measures

3.2.1 Hover Accuracy

The average position error and average hover velocities for a 60 second hover task were computed on the analog computer using the following equations:

$$\bar{r} = \frac{\int^{\Delta t} \sqrt{x_e^{\,2} + y_e^{\,2}}\, dt}{\Delta t} \qquad \bar{h} = \frac{\int^{\Delta t} |z_e|\, dt}{\Delta t}$$

$$\bar{\dot{r}} = \frac{\int^{\Delta t} \sqrt{x'^{2} + y'^{2}}\, dt}{\Delta t} \qquad \bar{\dot{h}} = \frac{\int^{\Delta t} |\dot{z}|\, dt}{\Delta t}$$

$\bar{r}$ = Average position error from the desired hover point in a horizontal plane parallel to the ground (ft)

$\bar{\dot{r}}$ = Average hover velocity in a horizontal plane parallel to the ground (ft/sec)

$\bar{h}$ = Vertical position error from the desired hover point (ft)

$\bar{\dot{h}}$ = Average vertical hover velocity (ft/sec)

x_e, y_e, z_e = Downrange, lateral, and vertical position error from the desired hover point, respectively (ft)

$\dot{x}, \dot{y}, \dot{z}$ = Down range, lateral, and vertical hovering velocities (ft/sec)

Δt = Integration interval for which the averages were determined (60 seconds)

3.2.2 ΔV - Characteristic Velocity (ft/sec)

The following defines the equations used on the analog computer for determining ΔV requirements

$$\Delta V = \nu - \nu_o = \int \frac{T}{M} dt$$

$$M = M_o - \frac{T}{g_e I_{sp}} \quad (\text{ note } \dot{M} = \frac{T}{g_e I_{sp}})$$

T = Total vehicle thrust which was a function of throttle position (lb)

M = Instantaneous vehicle mass

g_e = Earth gravity (32.2 ft/sec^2)

I_{sp} = $\dfrac{\text{lb of thrust}}{\text{lb/sec of propellant flow}}$ - Simulation studies assume a value of 280

4. SIMULATION PILOTS

The following summarizes the background of the pilots used during the study.

PILOT A - Experienced simulator pilot

PILOT B - Experienced, rocket belt, pogo, and jet belt operator (all manual thrust vector controlled vehicles); and fighter pilot. Has flown kinesthetic controlled vehicles free flight.

PILOT C - Test pilot. Experienced helicopter and VTOL pilot. Has flown kinesthetic and thrust vector controlled vehicles.

PILOT D - Engineer and simulator pilot used in previous Manned Flying System (MFS) and Lunar Flying Vehicle (LFV) Studies.

4.1 Rotation/Translation Characteristics of Various Control Configurations

Table C-4 outlines the derivation of the basic single axis translational characteristics associated with the vehicle control configurations discussed in Section 6.4 of text.

Small angles were assumed for the vehicle attitude angle (θ) and the gimballed thruster deflection (δ_T). $T \approx W$ was also assumed.

TABLE C-4

BASIC TRANSLATIONAL RESPONSE CHARACTERISTICS
FOR SEVERAL VEHICLE CONTROL CONFIGURATIONS

Mathematical Steps		Pivoted Thruster (above cg)	Pivoted Thruster (below cg)	Translating Thruster	Differential Throttling
Equations of Motion (Laplace)	$X(S) =$	$\frac{g}{S^2}\left[\theta(S) + K_\theta \delta_c(S)\right]$	$\frac{g}{S^2}\left[\theta(S) - K_\theta \delta_c(S)\right]$	$\frac{g}{S^2}\left[\theta(S)\right]$	$\frac{g}{S^2}\left[\theta(S)\right]$
	$(S) =$	$\frac{T\ell K_\theta}{IS^2}\, \delta_c(S)$	$\frac{T\ell K_\theta}{IS^2}\, \delta_c(S)$	$\frac{TK_\ell(57.3)}{IS^2}\delta_c(S)$	$\frac{K_T\ell(57.3)}{IS^2}$
Substituting step input $\delta_c(S) = \frac{\delta_c}{S}$ and $\theta(S)$ expression into $X(S)$	$\frac{X(S)}{\delta_c} =$	$\frac{gK_\theta\,(S^2 + \frac{T\ell}{I})}{S^5}$	$\frac{gK_\theta\,(\frac{T\ell}{I} - S^2)}{S^5}$	$\frac{g\ TK_\ell(57.3)}{IS^5}$	$\frac{gK_T\ell(57.3)}{IS^5}$
Time history solution $\frac{X(t)}{\delta_c}$ $\left(\frac{ft}{deg\ step\ \delta_c}\right)$	$\frac{X(t)}{\delta_c} =$	$\frac{g}{57.3}\left[\frac{\theta_{sens}t^4}{24} + \frac{K_\theta t^2}{2}\right]$ $\theta_{sens} = \frac{T\ell K_\theta}{I}$	$\frac{g}{57.3}\left[\frac{\theta_{sens}t^4}{24} - \frac{K_\theta t^2}{2}\right]$ $\theta_{sens} = \frac{T\ell K_\theta}{I}$	$\frac{g}{57.3}\left[\frac{\theta_{sens}t^4}{24}\right]$ $\theta_{sens} = \frac{TK_\ell 57.3}{I}$	$\frac{g}{57.3}\left[\frac{\theta_{sens}t^4}{24}\right]$ $\theta_{sens} = \frac{K_T\ell 57.3}{I}$

T = total thrust (lb)
ℓ = thruster moment arm (ft)
I = vehicle inertia (lb sec^2/ft)
g = gravity (ft/sec^2)

δ_c = controller input (deg)

θ_{sens} = control sensitivity (deg/sec^2/deg)

K_θ = thruster to controller linkage gain (deg/deg)

K_ℓ = thruster translation per deg of δ_c (ft/deg)

K_T = total diff thrust per deg of δ_c (lb/deg)

APPENDIX D

QUANTITY OF HARDWARE INCLUDED IN COST ESTIMATE

DEVELOPMENT PROGRAM

Human Factors (soft) Mockup (1)
OMLFV Mockup Interface/Change Control (1)
OMLFV Subsystems, Parts and Components (see Test Matrix)
LSE Systems (3 Sets)
OMLFV Vehicle (3 - 2 from DVT Components)

PFRT PROGRAM

OMLFV Subsystems, Parts and Components (see Test Matrix)
LSE Systems (2)

QUALIFICATION TEST PROGRAM

OMLFV Subsystems, Parts and Components (see Test Matrix)
OMLFV Vehicles (2)
LSE Systems (2)
GSE Systems (2)
Training Vehicles (2 - manrating)

TRAINING PROGRAM

Flight Trainers (6)

LUNAR OPERATIONS PROGRAM

OMLFV Vehicles (6)
LSE Systems (6)

SUPPORT PROGRAM

GSE Systems (3)

SPARES

20% of one flight system cost
20% of one training vehicle cost

APPENDIX E

OMLFV PRELIMINARY PARTS BREAKDOWN

1. Primary and Secondary Structure

Quantity	Nomenclature
	Main Structure
1	Floor Assembly
1	Radiation Shield
2	Mounting Fixtures (Propellant Tanks)
1	Mounting Fixture (Helium Tank)
2	Tank Insulation Boots (Propellant Tanks)
1	Tank Insulation Boot (Helium Tank)
2	Handholds and Grips
1	Mounting Fixture (Controls)
1	Payload Pallet
	Landing Gear
4	Struts
4	Foot Pads
4	Corner Fittings
4	Pins
4	Strut Insulation Sleeves

2. Propulsion System

Quantity	Nomenclature
	Engines
2	Bipropellant Shutoff Valves
2	Throttle Valves
2	Engine Mounts
2	Injectors
2	Chambers
2	Nozzles
	Pressure System
1	Pressure Tanks
2	Vent Valves
1	Gas Isolation Valve
1	Gas Filters
1	Regulator
2	Quad Check Valves
2	Relief Valves
	Plumbing
1	Gas Pressure Transducer

Quantity	Nomenclature
	Fuel System
1	Tank with Baffles
1	Propellant Filters
1	Fill and Drain Valve
1	Fuel Isolation Valve
	Plumbing (including flex line)
1	Quantity Sensing Probe
	Oxidizer System
1	Tank with Baffles
1	Oxidizer Filters
1	Fill and Drain Valve
1	Oxidizer Isolation Valve
	Plumbing (including flex line)
1	Quantity Sensing Probe
1	Quantity Sensing System Controller

3. Flight Control System

Quantity	Nomenclature
	Engine Differential Throttle Control
1	Throttle Controller
1	Controller Attachment
1	Controller Tube
1	Control Rod
1	Throttle Link
1	Differential Roll Link
	Engine Gimbal Control
1	Yaw Controller
1	Controller Attachment
1	Control Rod
1	Yaw Link
1	Thrust Vector Position Controller
1	Yaw Control Cable

4. Electrical Power System and Displays

Quantity	Nomenclature
1	Pressure Vessel with Mount
1	Pressure Vessel Cap
1	Seal
1	Battery Mount Assy
1	Battery Mount Slide Lock
1	Battery

Quantity	Nomenclature
1	External Harness
1	Power Switches
1	Power Switch Mount
1	Instrument Panel
1	Dual Reading Pressure Gage
1	Dual Reading Quantity Gage
1	Helium Pressure Gage

APPENDIX F

OMLFV SPECIFICATION NO. CP 7335-947001

1. SCOPE

This part of this specification establishes the requirements for performance and design of one type-model-series of equipment identified as a one man lunar flying vehicle (OMLFV) CEI 733501A. This CEI is used to provide manned mobility on the lunar surface to accomplish reconnaissance and exploration sorties. This CEI requires propellants supplied from the LM and the lunar support equipment for mission deployment.

2. APPLICABLE DOCUMENTS

The following documents form a part of this specification to the extent specified herein. In the event of conflict between documents referenced here and other detailed content of Sections 3, 4, 5, and 10, the detail requirements of Sections 3, 4, 5 and 10 shall be considered a superseding requirement.

3. PROJECT DOCUMENTS

NPC 500-1	Apollo Program Configuration Management Manual
SPECIFICATIONS	
Military	
MIL-I-8500	Interchangeability
MIL-P-27402	Propellant, Hydrazine, Unsymmetrical Dimethylhydrazine
NASA	
MSC-PPD-2	Propellant, Nitrogen Tetroxide, Inhibited
Bell Aerosystems	
EC 7335-947001	Engine Assembly Detail Specification
EC 7335-947002	Control System Detail Specification
EC 7335-947003	Propellant Supply Subsystem Detail Specification
EC 7335-947004	Pressurization Subsystem Detail Specification

STANDARDS

Military

MIL-STD-130	Identification and Marking
MIL-STD-841	Critical High Temperature Alloys
MIL-STD-1247	Fluid Line Identification Working
MS-33586	Definition of Dissimilar Metals

Bureau of Mines

Standard	Helium, Grade A

DRAWINGS

Bell Aerosystems

7335-099001	General Arrangement, OMLFV
7335-099012	Storage and Deployment, OMLFV
7335-099013	Inboard Profile, OMLFV

BULLETINS

Airforce Navy Aeronautical

Bulletin No. 438	Age Control Synthetic Rubber Parts

OTHER PUBLICATIONS

Manuals

Bell Aerosystems

Pre-Flight Checkout Procedures
Flight Operation Procedures
Post Flight Checkout Procedures

3.1 PERFORMANCE

The OMLFV shall have the operational capability shown in Figure F.1.

3.1.1 Functional Characteristics

3.1.1.1 Primary Performance Characteristics

The OMLFV shall have the following mission/use performance characteristics:

(a) Payload - 0 to 370 lb

(b) Radius of Operation - 22,500 ft

(c) Duty Cycle Average Specific Impulse - 270 seconds

(d) Capability to takeoff and land on lunar soil

(e) Capability to accomplish 30 sorties

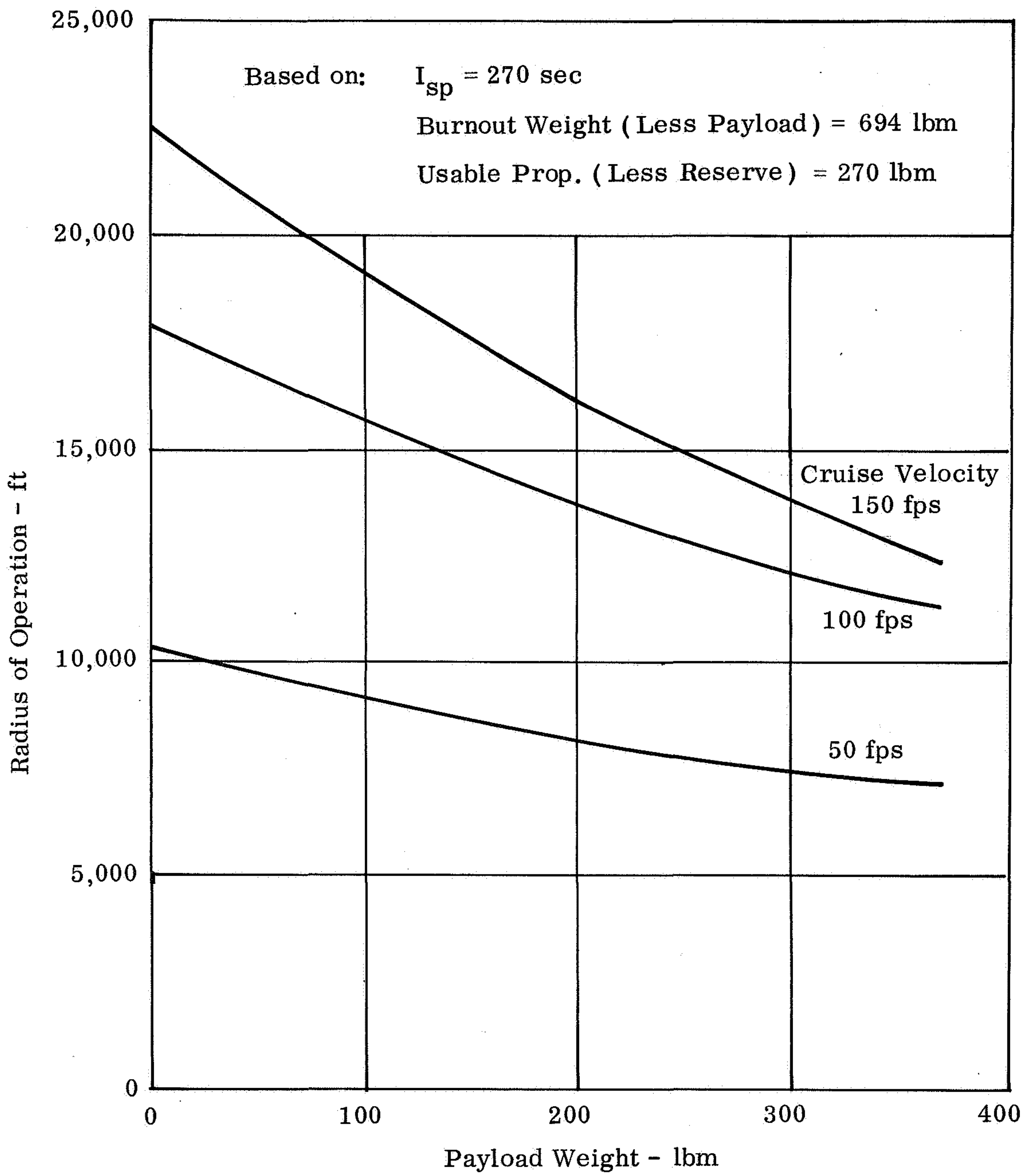


Figure F.1. Maximum Radius of Operation as a Function of Payload

(f) Use propellants supplied from LM (N_2O_4 and Anhydrous Hydrazine/UDMH)

(g) Controllability

(h) Flying Qualities

(i) Deployable from LM by astronaut with the centerline of LM 15° from the local vertical in any direction.

3.1.1.2 Secondary Performance Characteristics

The OMLFV shall have the following design process performance characteristics:

(a) Ground servicable

(b) Emergency shutdown (malfunction)

3.1.2 Operability

3.1.2.1 Reliability

The OMLFV shall provide an inherent reliability to ensure mission success equal to or in excess of . The meantime between failures shall be hours, assuming exponential distribution of times to failure.

3.1.2.2 Maintainability

The OMLFV shall have a meantime to repair of . The maintenance man-hours per operating hour shall not exceed

3.1.2.2.1 Maintenance Requirements

3.1.2.2.2 Maintenance Repair Cycle

3.1.2.2.3 Service and Access

Access doors shall provide ease of servicing equipment and replacement of expended components of the OMLFV. Propellant servicing shall be facilitated by accessibility to quick disconnects located on the propellant supply panel.

3.1.2.3 Useful Life

The OMLFV shall have a shelf life of years with minimum maintenance requirements. The OMLFV shall have an operating life (mission) of 30 sorties.

3.1.2.4 Natural Environment

The OMLFV shall be designed to withstand the elements of a lunar environment.

3.1.2.5 Transportability

The maximum dimensions and weight of the OMLFV shall be such as to provide ease of transportation. The OMLFV shall be packaged and packed for shipment using commercially accepted standards suitable for the equipment type.

3.1.2.6 Human Performance

Human performance/lunar engineering requirements incorporated into the design of the OMLFV equipment shall satisfy the program/system requirements.

3.1.2.7 Safety

The OMLFV shall be designed to operate in a manner which will preclude or limit hazard to personnel and/or equipment. Procedures implemented in the manufacture, test, transport, storage, and operation or maintenance of the OMLFV shall comply with the NASA practices and regulations applicable to the program/system requirements. An astronaut in-flight restraint system functionally similar to that shown in Figure F.2 shall be provided.

3.1.2.8 Induced Environment

The OMLFV shall be designed to withstand the induced environment criteria incident to transportation, storage, and deployment.

3.2 OMLFV DEFINITION

3.2.1 Interface Requirements - The functional and physical interface requirements for the OMLFV shall be as shown on Drawing 7335-099012.

3.2.1.1 Schematic Arrangement

The mechanical interface of the OMLFV with the LM is shown on Drawing 7335-099012.

3.2.1.2 Detailed Interface Definition

3.2.1.2.1 Mechanical

(a) Latching fittings mounted on the platform and the deck of the vehicle shall engage mating fittings located on the lower support beam and upper beam assemblies attached to the lunar module.

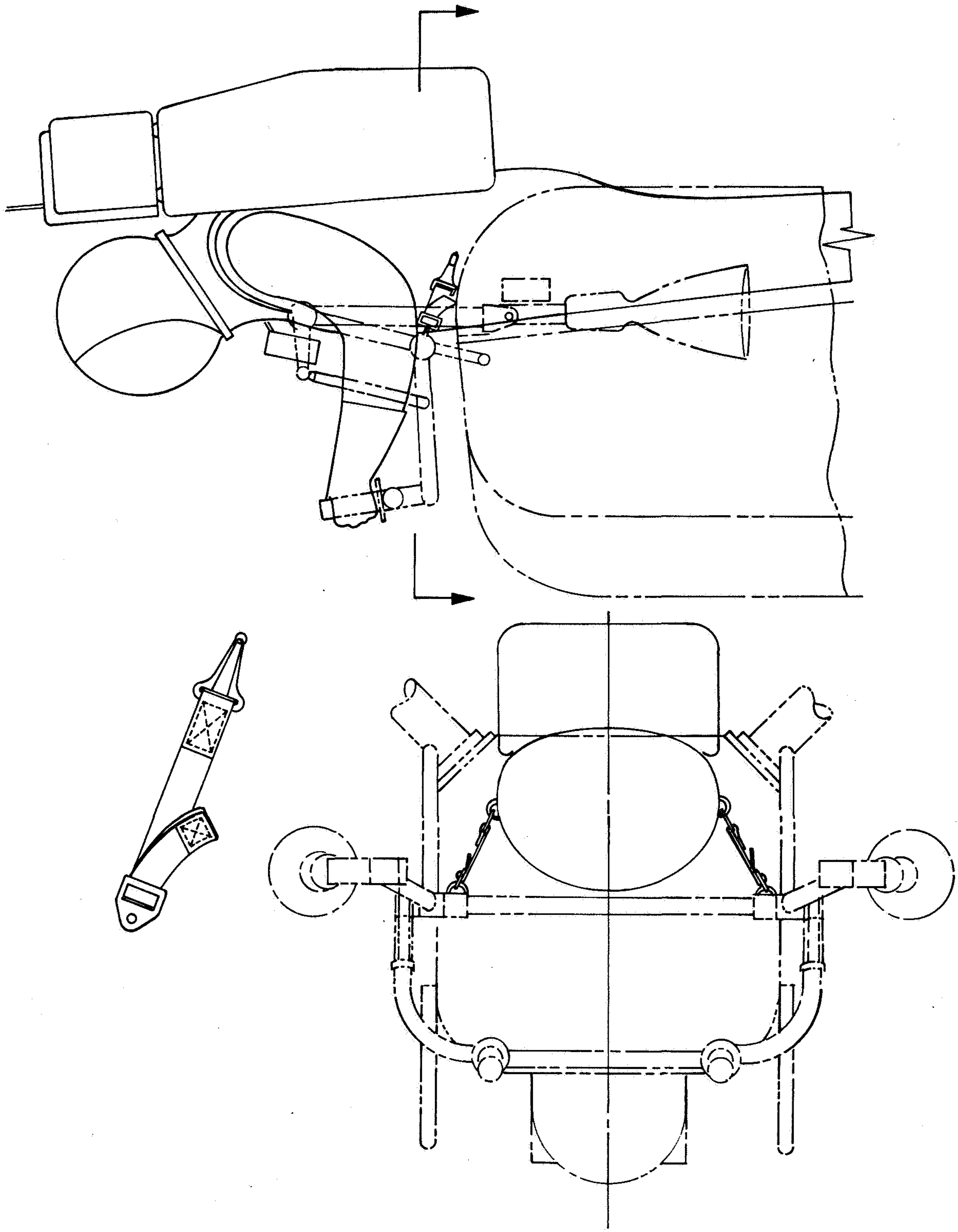


Figure F-2. Astronaut Restraint System

(b) The quick disconnect fittings for propellants and pressurant supply shall be compatible with the corresponding elements of the lunar support equipment.

3.2.1.2.2 Functional

(a) Filters contained in the propellant and pressurant supply elements of the lunar support equipment shall prevent the entry of particles which may preclude the proper operation of the propulsion system.

(b) The upper support fitting shall accept the terminal of deployment boom lanyard.

3.2.1.2.3 Procedural

3.2.2 Component Identification

3.2.2.1 Government Furnished Property List

(a) Nitrogen tetroxide (N_2O_4) Inhibited - conforming to Specification MSC-PPD-2. Propellant - Oxidizer

(b) A 50/50 blend of hydrazine (N_2H_4) and unsymmetrical dimethlyhydrazine (UDMH) - conforming to Specification MIL-P-27402. Propellant - Fuel

(c) Helium - Bureau of Mines Grade A.pressurizing gas.

3.2.2.2 Engineering Critical Components List

(a) Engine Assembly	EC7335-947001
(b) Control System	EC7335-947002
(c) Propellant Supply Subsystem	EC7335-947003
(d) Pressurization Subsystem	EC7335-947004

3.2.2.3 Logistics Critical Components List

The procuring agency will establish requirements for the selection of logistics critical components.

3.2.3 Technical Manuals

The following technical manuals shall be required for operation of the OMLFV and will be supplied as authorized by the approved documents requirements list.

Preflight Checkout Procedures
Flight Operation Procedures
Post-Flight Checkout Procedures

3.3 DESIGN AND CONSTRUCTION

The OMLFV shall be designed and constructed in accordance with the applicable requirements of NASA approved specifications and standards.

3.3.1 General Design Features

The three-view plan and inboard profile of the OMLFV, showing configuration, shape, and nominal dimensions appears on Drawings 7335-099001 and 7335-099013.

3.3.1.1 Size

The overall dimensions of the OMLFV shall be as shown on the general arrangement drawing (No. 7335-099001).

3.3.1.2 Weight

The total dry weight of the OMLFV shall not exceed 235.1 pounds. The total wet weight (propellants loaded) of the OMLFV shall not exceed 541.7 pounds. The estimated weight breakdown shall be as follows:

Detail Weight Summary (No Payload)

Structure	71.5	
Induced Environmental Protection	15.6	
Launch and Recovery (Landing Gear)	37.2	
Main Propulsion	71.6	
Orientation Controls	21.9	
Prime Power source	3.0	
Instrumentation	9.6	
Personnel Provisions	4.7	
Dry Weight		235.1
Residual Propellant and Service Items	6.6	
Subtotal		241.7
Full Thrust Propellant	300.0	
Dry Weight		541.7

3.3.2 Selection of Specifications and Standards

All specifications or standards, other than those established and approved for use by NASA shall be approved by the Procuring Agency prior to incorporation in this specification.

3.3.3 Materials, Parts, and Processes

The materials, parts, and processes selected, which are not in accordance with a NASA recognized or other approved specification shall be substantiated so that the selection meets the performance characteristics satisfactory for the intended purposes.

3.3.4 Standard and Commercial Parts

Standard and commercial parts shall be used wherever they are suitable for the purpose and shall be identified by their part numbers. The use of nonstandard parts will be acceptable only when standard parts have been determined to be unsuitable.

3.3.5 Moisture and Fungus Resistance

Materials which are not nutrients for fungus shall be used whenever possible. Materials which are nutrients for fungus may be used in hermetically sealed assemblies and other accepted and qualified applications.

3.3.6 Corrosion of Metal Parts

The use of dissimilar metals in direct contact shall be prohibited unless suitably protected against electrolytic corrosion. Dissimilar metal shall be as defined in MS-33586.

3.3.7 Interchangeability and Replaceability

The components, assemblies, and parts of the OMLFV shall be completely interchangeable with respect to installation and performance in general accordance with the requirements of MIL-I-8500. Access doors and panels shall be readily removed and replaced by quick disconnects.

3.3.8 Workmanship

The OMLFV, including all parts and assemblies shall be constructed and finished in accordance with NASA-STD-   (MIL-STD-   ). Thoroughness of soldering, wiring, impregnation of coils, marking of parts and assemblies, plating, painting, riveting, machine screw assemblage, welding, brazing, and freedom of parts from burrs shall comply with NASA-STD   (MIL-STD-   ).

3.3.9 Electromagnetic Interference

The electrical and electronic interference generated by operation of the OMLFV shall not exceed tolerable limits. The OMLFV shall not be susceptible to interference generated by other electrical and electronic sources within the operating environment of the OMLFV.

3.3.10 Identification and Marking

Equipment, assemblies, components, and parts shall be marked for identification in accordance with MIL-STD-130. All interface connections shown on the installation drawings shall be permanently worked. All fluid lines shall be marked in accordance with MIL-STD-1247. Age control of synthetic rubber parts shall be in accordance with ANA Bulletin No. 438. Parts and assemblies fabricated from critical high temperature alloys shall be marked in accordance with MIL-STD-841.

3.3.11 Storage

The OMLFV shall have a maximum storage duration capability of with minimum maintenance required. Storage environment shall be within the following limits:

(a) temperature:

(b) pressure:

(c) humidity:

4. QUALITY ASSURANCE PROVISIONS

4.1 PHASE I, TEST/VERIFICATION

The testing to be accomplished shall consist of design verification tests, preliminary flight rating tests and qualification tests.

4.1.1 Engineering Test and Evaluation (Design Verification Tests)

The design verification tests shall be performed to acquire data to support the design and development process. These tests shall be accomplished at the component, subsystem and installation, and vehicle levels.

a. Component Tests - The following propulsion system components shall be subjected to design verification testing.

(1) Thrusters

(2) Propellant Tanks

(3) Bipropellant Throttle Valve

(4) Propellant Tank Vent Valve

(5) Gas Regulator

(6) Gas Pressurant Tank

(7) Gas Isolation Valve

(8) Propellant Filters

(9) Gas Filter

(10) Propellant Fill/Drain/Valves

(11) Propellant Pressure Relief Valves

(12) Propellant/Gas Quick Disconnects

(13) E.B. Weld Tube Joints

(14) Passive Insulation Samples

The following structure components shall be subjected to design verification testing:

(1) Primary Structure

(2) Landing Gear

(3) Plume Shield

The following controls components shall be subjected to design verification testing:

(1) Pitch/Yaw/Roll Linkages and Bearings

(2) Controller - Throttle

(3) Controller - Yaw

(b) Subsystems and installations tests - The following subsystems and installations shall be subjected to design verification testing:

(1) Flight Controls Subsystem

(2) Engine Assembly (Thruster/Valves)

(3) Pressurant Tank Installation

(4) Fuel Tanks Installation

(5) Oxidizer Installation

(c) Vehicle Tests - Three vehicles shall be subjected to the design verification tests. One vehicle shall be subjected to static/thermal tests. One vehicle shall be subjected to dynamics tests. One vehicle shall be subjected to propulsion tests.

4.1.2 Preliminary Qualification Tests (Preliminary Flight Rating Test)

Preliminary flight rating tests shall be performed to achieve interim acceptance of performance and design characteristics. These tests shall be accomplished at the component and subsystem levels.

(a) Component Tests - The oxidizer tank and the fuel tank shall be subjected to dynamics tests.

(b) Subsystems and Installations - The engine assembly (thruster/valves) shall be subjected to PFRT.

4.1.3 FORMAL QUALIFICATION TEST

Qualification tests shall be performed to demonstrate that the performance and design requirements have been satisfied. These tests shall be accomplished at the component, subsystem and installations and vehicle levels.

(a) Component Tests - The following propulsion system components shall be subjected to qualification testing:

(1) Thrusters

(2) Propellant Tanks

(3) Bipropellant Throttle Valve

(4) Gas Regulator

(5) Quick Disconnects

(6) Gas Pressurant Tank

(b) Subsystems and Installations Tests - The following subsystems and installations shall be subjected to qualification testing:

(1) Flight Controls Subsystem

(2) Engine Assembly (Thruster/Valves)

(3) Instrumentation Installation

(c) Vehicle Tests - Two vehicles shall be subjected to the qualification tests. One shall be subjected to static/thermal tests. One shall be subjected to dynamics tests.

4.1.4 Reliability Test and Analyses

The data acquired and recorded during performance of the tests of paragraphs 4.1.1, 4.1.2, and 4.1.3 shall be used for reliability analysis.

4.1.5 Engineering Critical Component Qualification

The qualification test requirements of the items listed in paragraph 3.2.2.2 shall be defined in the specifications listed therein. The testing may be accomplished at the subsystem level or at lower (installation or component) level as specified in paragraph 4.1.3.

4.2 PHASE II INTEGRATED TEST REQUIREMENTS

Tests shall be performed to demonstrate the capability to successfully integrate the vehicle and lunar and ground support equipment as elements of the lunar flying system.

5. PREPARATION FOR DELIVERY

Not Applicable. Requirements for preparation of produced hardware for delivery and shipment shall be contained in Part II of this specification.

6. NOTES

6.1 SUPPLEMENTAL INFORMATION

6.1.1 Technical Data

Technical data shall be supplied in accordance with the requirements of the authorized Document Requirements List.

6.1.2 NPC500-1

Exhibit II of NPC500-1 was used as a guide in the preparation of this specification.

6.2 ALTERNATE SOURCE QUALIFICATION

These requirements shall be established when applicable.

10. APPENDIX

APPENDIX G

ADDITIONAL SUPPORTING STUDIES OF A ONE-MAN LUNAR FLYING VEHICLE

I. INTRODUCTION

The lunar flyer study conducted by Bell Aerosystems during the first half of 1969, under Contract NAS 9-9044, has recommended a concept for a small lunar flying vehicle. The vehicle is useful for exploration and rescue on the lunar surface.

The recently completed study has identified several areas in which early supporting research and technology could provide design and mission planning data or provide verification of items which will increase confidence in follow-on program cost and schedule estimates.

Four of these tasks, given top priority, are discussed in Section II and are proposed herein, because they will help to verify the safety with which the vehicle can be flown, the suitability of the vehicle handling qualities, and the feasibility of training astronauts on earth for safe flight on the moon.

These tasks are proposed as a follow-on to the current Study of a One Man Lunar Flying Vehicle, Contract NAS 9-9044.

Section V describes additional supporting tasks recommended for early but less urgent funding and execution.

II. TECHNICAL DISCUSSION

A. SAFETY, ABORT, AND HAZARD ANALYSIS

The usefulness and feasibility of flying mobility for lunar exploration and rescue has been well established, by design studies and earth flight test experience. However, the manner in which the lunar vehicle is employed is dependent on an assessment of the risk in comparison to the scientific return. A qualitative analysis will be useful in exposing the higher risk hardware items, where larger safety margins or design changes should be considered, or the high risk tasks, where operational procedures should be modified to reduce the probability or consequence of human error.

A future quantitative analysis will permit comparison with other vehicles, other parts of the Apollo system, or alternate versions of the vehicle, provided the same failure rates and ground rules are used for each vehicle or system being compared. The quantitative analysis is not proposed at this time.

Assessment of risk depends on an evaluation of the probability and the effect of hardware failures, and of human errors. Methods for assessing risk due to hardware failures have been widely used. A modified form of the Apollo Procedure for Failure Mode, Effects, and Criticality Analysis, Document RA-006-013-1A is proposed herein.

Methods of assessing risk due to human error in accomplishing complex tasks requiring judgment are less well developed. However, it is possible to compare the factors which affect the task of flying on the moon, with flying on earth, and thus extrapolate earth experience, for which data are available,to lunar flight. It is proposed that this be accomplished by comparison of the detailed operations required, and verified by comparison of the overall flight task. In this way, a probability of safe lunar flight can be related to actual earth flight data.

It is necessary that a "broad brush" type of analysis be performed early in the program, in order to isolate critical functional subsystems. With this requirement in mind, the analysis should be divided into three major task categories:

(1) Gross Hazard Analysis

(2) Failure Mode Effect Analysis

(3) Hazard/Risk Assessment (Human Reliability)

1. Gross Hazard Analysis

The Gross Hazard Analysis will identify all of the system critical functional subsystems. The categories criteria for hazard evaluation shall be as follows:

Category 1 (safe) -	No system effect
Category 2 -	The failure permits safe return of the vehicle to the LM site but mission objectives are lost.
Category 3 -	The failure permits safe landing of the vehicle, remote from LM, but the vehicle is incapable of flying back to LM.
Category 4 (catastrophic) -	The failure results in injury or loss of life for the pilot.

A benefit from the development of a Gross Hazard Analysis, is the fact that the Failure Mode Effects Analysis is reduced in size because subsystem failures that are determined to be in the safe categories do not require additional detail analysis.

2. Failure Mode and Effects Analysis

The second task to be performed will be the generation of the system failure mode and effects analysis (FMEA). This analysis will be performed for subsystem components that fall into categories 2, 3, and 4. In this manner the more critical components failure modes will be identified, to provide analytical support for component detail design recommendations to either eliminate the hazard or provide monitoring of critical functions to preclude development of an undesirable event.

To insure the most comprehensive system safety analysis, the FMEA will be divided into three sections to represent the three major Lunar Flyer functional parts which are: Propulsion, Flight Controls and Structure/Landing Gear. In addition, each one of these functional analyses will be divided into two operational phases, servicing and flight. The functional analysis for flight operation will also be divided into

three phases of flight; take off, in flight, and landing with varied quantities of propellant and pressurant gas loads. Figure II-1 is a block design of a typical analysis breakdown.

The generation of the FMEA will be an outgrowth of subsystem reliability logic block diagrams, that will be developed to identify subsystem component functional interdependency.

Documentation of the FMEA will be presented in the format illustrated in Figure 2-2 of NASA document RA-006-013-1A entitled, "Failure Mode Effect and Criticality Analysis".

3. Hazard/Risk Assessment (Human Reliability)

The typical reliability prediction usually assumes that there is no equipment degradation or failure due to the human element in the system. In other words, it is assumed that man's reliability is equal to 1.0. Typically in performing Failure Modes and Effects Analyses, man is often considered; particularly with regard to initiating corrective actions (for example, assuming manual control of a space vehicle in the event that the automatic stabilization and control system fails) but again it is usually assumed that his reliability is equal to 1.0. These assumptions have been made as a matter of expediency due to the deficiencies in the state-of-the-art in human reliability assessment. Up until recently human reliability assessment has been difficult if not impossible, due primarily to the lack of data regarding human error (or failure) rates. This problem still exists to a certain degree, but it is now possible to describe behavior as a function of the probability of success or failure in terms specific enough to permit logical engineering judgments about risks and probable performance. This approach provides a mathematical assessment of risk in situations where the only other alternative is an educated guess; or assuming that man's reliability is equal to 1.0 (particularly when actual experience indicates, for example, that the overall reliability of the X-15 personnel subsystem = 0.84, see Reference 1).

The objective of the approach outlined is two fold: (1) to provide an estimate of man's reliability in flying the One Man Lunar Flyer over the lunar surface and (2) to attempt to validate this estimate by extrapolation from earth derived experience.

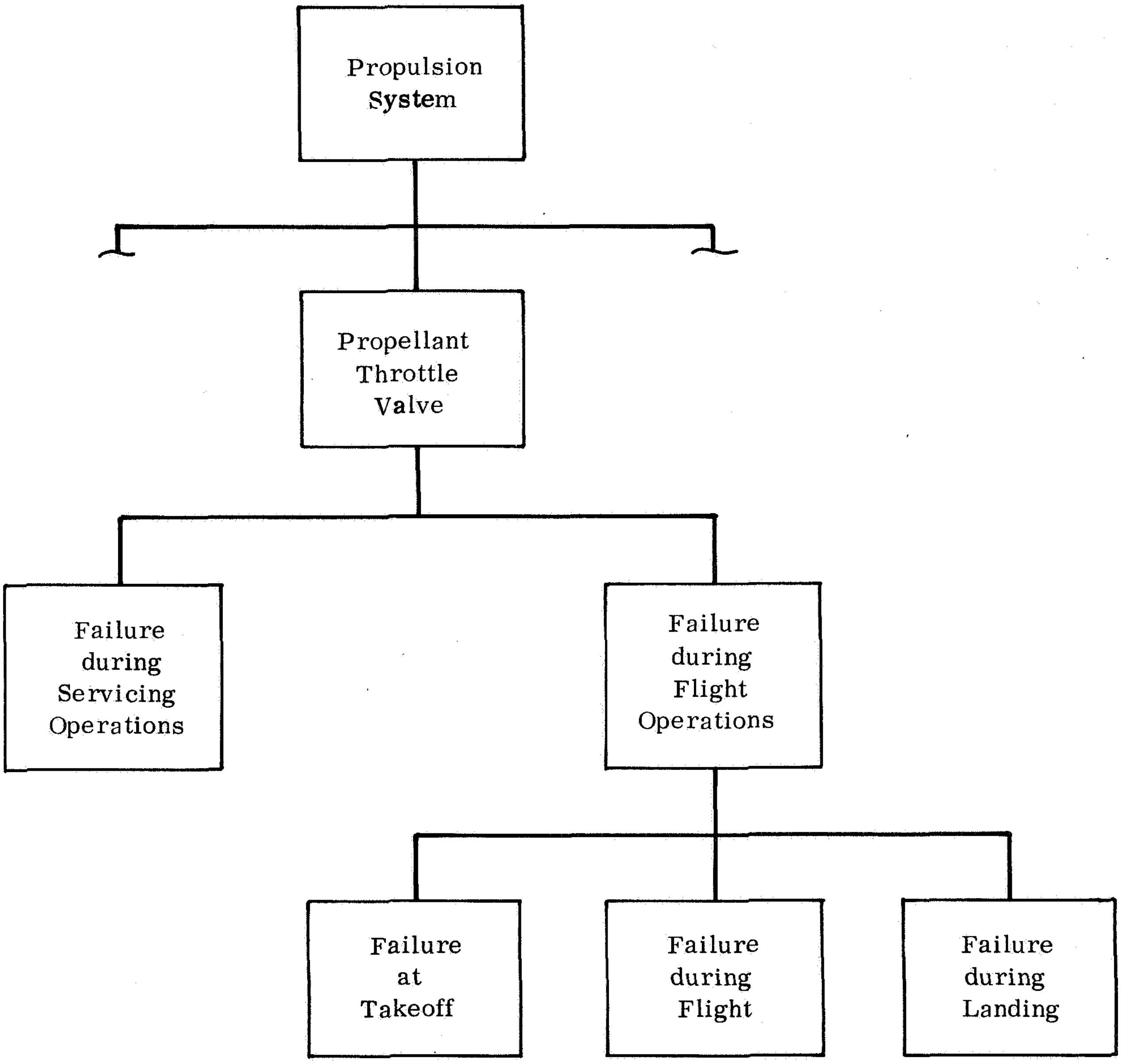


Figure II-1. Safety and Abort Analysis Breakdown

The first step is to determine what tasks the OMLFV operator is required to perform in flying the lunar flyer over the lunar surface, for a typical mission profile.

Operator task time estimates will be developed by listing sequentially, all of the operations required to be performed to accomplish a given function and the synthesis of a task time for each operation.

Operator task time is the sum of information processing time, visual transition time, reach time, and manipulation time; while total event time is the sum of operator task time plus system wait time.

Information processing time is derived by determining the amount of information (H) contained in the task as follows:

$$H = \log_2 N \text{ (bits)}$$

where N = number of possible alternatives or states

(For example: If the system operator must correctly select one position on a six-position rotary switch, the information content of this task is: $H = \log_2 6 = 2.585$)

H is then divided by man's average information processing rate and adding a constant equivalent to man's simple reaction time to yield information processing time. Thus:

$$\text{Infor. Proc. Time} = 0.20 \text{ sec.} + \frac{H \text{ (bits)}}{3 \text{ (bits/sec.)}}$$

Visual transition, reach and manipulation time are obtained from method-time-measurements data (for example: Kreager, D. W. and Boyha, F. H. Engineered Work Measurement. New York: Industrial Press, 1957).

This analysis, thus provides the baseline for the OMLFV human reliability analysis and also allows the estimation of operator workload as described below.

Workload estimates will be developed by comparing operator event time (as derived from the Operator Task Time Estimates) with the time available to perform the task;

i.e.: $\text{workload} = \dfrac{\text{operator event time}}{\text{time available}} \times 100\%.$

A work overload condition exists when the time required to perform a given task exceeds the time available to perform the task.

This technique has been applied to the Apollo Lunar Landing Program analysis with outstanding results (see Reference 2).

The next step in the development of the human reliability assessment is the development of a task oriented reliability logic block diagram using techniques similar to those described in Section 2.2 of Reference 3.

DATA STORE (Reference 4) is an index of equipment operability developed by the American Institute for Research. It contains quantitative information on the reliability of performing various tasks. The individual tasks are broken into small behavior segments, which lend themselves to use in any situation. In those cases where DATA STORE contains information on similar, but not identical tasks a certain amount of extrapolation will be required. In addition, the data will require modification, via "K" factors to adjust for different environmental and operating stresses.

Using the block diagrams previously developed the individual task reliability estimates will be combined into an overall OMLFV mission human reliability estimate. This estimate can then be compared with actual experience data on human reliability by operating earth bound vehicles, such as helicopters to determine the relative crew safety of flying the OMLFV over the lunar surface.

4. Validation of OMLFV Human Reliability Estimate

Since pilot error rates (at the gross level) for flying fixed and rotary wing aircraft are well documented, it should be possible to extrapolate this data to the OMLFV mission via application of appropriate "K" factors, such as:

K_1 - Gust load differences
K_2 - Illumination differences
K_3 - Pressure suit differences

K_4 - Terrain differences

K_5 - Gravity differences

K_6 - Handling qualities differences

K_7 - Work load differences

K_n - Other factors

The value of human reliability obtained in this manner will then be compared with the OMLFV human reliability estimate to validate that estimate.

During later phases of the OMLFV program, the data base developed above provides a quantitative estimation of operator interaction with the OMLFV in a mission oriented context. As a development tool, this crew performance data base can be used for:

Engineering simulation and test

Mission analysis and planning

Development of training plan, manuals and course content requirements

Full mission, part task and systems trainer design and development

As an operation tool, this crew performance data base can be used for:

Developing operational handbooks, manuals and checklists

Training guides for training in operational vehicles

Development of mission rules and OMLFV operational plans

If desired, the human reliability assessments can be integrated into the Failure Mode, Effects and Criticality Analysis. The generic human error rate (E_G) can be determined since (E_G) = 1 - Reliability (where the human reliability values has been extrapolated from DATA STORE). In particular, it appears that a criticality assessment, utilizing procedures such as those of Section 3 of Reference 3, would be highly desireable inputs into the development of the OMLFV training program.

The approach outlined above provides a logical and mathematically valid technique to estimate the crew safety aspects of operation of the OMLFV. It will permit comparison of this estimate with earth-bound vehicle experience such as rocket-belt,

and fixed and rotary wing aircraft. An approach has also been outlined which will permit earth-bound experience to be extrapolated to the OMLFV to validate the human reliability estimates obtained.

5. References

1. Wilson, R. B. and Gaffney, J. L., "Man's Reliability in the X-15 Aerospace System." Paper presented at the Symposium on the Quantification of Human Performance, Univ. of N. Mex., Aug. 1969.

2. Gross, R. L., Lindquist, O. H. et al - "The Application, Validation and Automation of a Method for Delineating and Quantifying Aerospace Flight Crew Performance." Paper presented at the Symposium on the Quantification of Human Performance, Univ. of N. Mex., Aug. 1964.

3. Procedure for Failure Mode. Effects and Criticality Analysis. Report RA-006-013-1A. Washington, D.C.: NASA.

4. Munger, Sara J., Smith, R. W. and Payne, O. An Index of Electronic Equipment Operability: Data Store. Report AIR C43-1/62-RP(1). Pittsburgh, Pa.: Amer. Inst. for Research, Jan. 1962.

B. FLIGHT RESEARCH/TRAINING PLAN AND SIMULATOR PRELIMINARY DESIGN

A high degree of confidence in the ability of astronauts to safely fly a simple one man lunar flying vehicle can be attained by conducting a well planned and coordinated flight research and training program. The OMLFV study established a preliminary flight research and training plan. It is important at this stage of lunar flying vehicle development to prepare a more detailed plan, establish simulator requirements, and prepare preliminary designs of simulation vehicles and facilities so that an early start can be made on the flight research program, and to establish the vehicle cost and schedule.

1. Flight Research and Training Activities

Vehicles and facilities for conducting research and training for flight in the vicinity of the moon are already in use or under development. These include Icarus, FLEEP, and the LLRF at Langley Research Center; the LLTV and 1/6 g sloping floor at Manned Spacecraft Center; the Rail/Tether Suspension System and Rocket Pogo Vehicle at Bell; and many visual simulation facilities at NASA Centers and in industry. These vehicles and facilities must be investigated to determine their applicability to one man lunar flying vehicle flight research and training. In addition, modifications to these facilities and equipment such as are suggested in this section should be investigated.

Modified equipment includes a Pogo configuration of the Bell turbojet powered earth gravity free flight vehicle, a larger rocket powered earth gravity free flight vehicle to carry a pressure suited operator for flight duration up to one minute and a turbojet powered 1/6 g tethered vehicle, for longer duration and distance than the present Bell and LRC facilities.

The control sensitivities which provide the best handling qualities for the OMLFV were established during the OMLFV study and can be well simulated in a tether facility. Earth gravity free flight vehicles, however, have different dynamic characteristics than lunar vehicles and research is required to determine what control characteristics the free flight vehicles should have when considering the differences in dynamics and the desire to simulate lunar vehicle characteristics.

It is proposed that this research be conducted using the visual simulation facility currently available at Bell. By varying g level, linkage gain and pivot to cg distance, a wide range of vehicle characteristics can be simulated and investigated.

A training program for both flying and servicing the LFV has been established during the current program. Improvement in this plan can be made by conferring with cognizant NASA training personnel and crews. Advantage should be taken of the unique experience being gained by those who have already been trained and worked in the lunar environment. Additionally, the training program as currently defined assumes continuity of training. However, this must be integrated with other training requirements associated with the lunar mission. Furthermore, NASA guidance and/or policy regarding criteria and proficiency standards - "When is the trainee ready for the next step?" will be helpful. In these ways, then, consultation with NASA personnel will be beneficial in modifying and detailing the existing training plan.

In initial lunar operations the flying vehicle will be operated conservatively and cautiously (at short ranges, low speeds, and low altitudes). As lunar experience is gained longer range and higher speed sorties will undoubtedly be flown. Earth rocket belt and jet belt experience has demonstrated safe flight at altitudes up to 75 ft and speeds up to100 ft/sec for VFR flight (flight without instruments). The question arises: "At what speeds, altitudes, and ranges, does it become necessary to add instrumentation and what kind of instrumentation is required for safe and efficient flight in the lunar environment?" It is therefore proposed that an early research program be implemented using a helicopter to acquire some preliminary information towards answering this question and to assist in defining a more comprehensive flight research program.

The early program involves the use of a pilot flying with and without instruments (with a safety co-pilot) at altitudes ranging from 25 to 500 ft and at speeds ranging from 20 to 90 knots. Data will be recorded by photographing the instrument panel. The pilot's task will be to maintain the speed and altitude set up prior to the test run. From the data the following statistics will be derived and analyzed: (1) speed variance, (2) altitude variance, (3) mean speed, (4) mean altitude. Additional short range flights (1 to 5 miles) from point to point on the surface, with and without instruments will be made to note differences in flight profile and times of flight. These flights will be made over unfamiliar terrain in remote areas to simulate as much as possible flight over unfamiliar lunar features.

An early short pilot training program using existing vehicles and facilities and NASA personnel (astronauts or test pilots) as trainees would contributed towards a better definition of the flight research and training plan. This would serve the dual purpose of providing information for training plan refinement and introducing NASA personnel to manually controlled rocket powered flight. A ten week program involving the use of the visual simulation facility, the 1/6 g tether facility and vehicle, and the 1 g vehicle (with safety tether and free flights) is proposed to be conducted with two NASA trainees.

2. Simulator Vehicles Preliminary Design

It is mandatory that the astronaut be asked to conduct no flights on the moon which he has not first simulated on earth, including free flights over a similar flight path. The flight dynamics of a lunar vehicle differ from the flight dynamics of the same vehicle flown on earth, because the 1/6 lunar gravity calls for 1/6 of the rocket thrust required on earth. However, the mass of the vehicle is the same on the moon as on the earth, thus the 1/6 thrust moving the same mass results in a six times slower response. In order for the lunar vehicle to accomplish the same translation response as it would on earth, it must tilt six times as far as it would on earth. Earth VTOL and helicopter pilots report that both attitude and translation cues are important, so it is desirable to reproduce on earth the correct lunar attitude/translation relationship.

The lunar landing training vehicle does this exactly by supporting 5/6 of the vehicle earth weight with a gyro stabilized jet engine. However, this results in a vehicle which is functionally and operationally complex. For the simple lunar flyer, it is highly desirable to keep the earth trainer or earth research vehicle as simple as the lunar vehicle. Fortunately a means of providing lunar simulation with simple vehicles is available by part task simulation.

The tasks involved in flying a VTOL vehicle can be divided into two categories: those required for short period vehicle attitude and velocity stabilization and control, and those required for long period flight path control. Instead of combining lunar simulation of both the short period and long period tasks in the same vehicle, two separate, but simple vehicles can provide the same result. The short period stabilization and control handling qualities can be simulated exactly by using a simple lunar

type vehicle mounted on a gimbal and supported by a 5/6g tether on an overhead rail, similar to the LRC/LLRF, or the Bell lunar tethered vehicle. Such a vehicle can reproduce the lunar tilt angle/translation acceleration relationship and thus provide correct lunar handling qualities. For the long period flight path simulation, a simple 1g prototype lunar vehicle using jet engine or rocket propulsion can be provided to allow free flight on earth to reproduce exact lunar flight paths. Vehicles and facilities of this type, to be investigated during the proposed study, are described in the following paragraphs.

a. Extended Range Tethered Vehicle

The present Bell tethered vehicle is shown in Figure II-2. It is suspended with a constant tension support on an overhead rail to provide a flight envelope in one plane with a maximum altitude of 10 feet and a range of 150 feet. It provides a flight duration of 40 seconds with a pressure suited or shirt sleeved operator. This vehicle has been extremely useful for research of flyer handling qualities and as part of a pilot training program in preparation for earth gravity free flight. However, its performance limitations and the large variation in weight during flight detract from the full utilization of this tethered flight concept. The flight envelope limits the vehicle to flights of low speed and short distance. The short flight duration increases the concentration level, since during a large portion of each flight the pilot is thinking about the landing. The variation in vehicle weight with propellant consumption (approximately 8%) results in a large change in the simulated gravity during flight since the overhead suspension system supporting force is constant. The present propulsion system throttle ratio of 4:1 limits the thrust at takeoff and does not permit operation at a low enough thrust level at landing without inadvertant thrust cutoff. These deficiencies can be reduced or eliminated by using a turbojet powered vehicle and a longer overhead rail.

The Williams Research Corporation WR-24 engine is ideally suited to this application and is in production. The engine, 11 inches in diameter and 15 inches in length, can easily be accommodated by a configuration much the same as that of Figure II-2. Nearly double the maximum thrust of the present system will be available with a throttling ratio of about 10:1, and the jet version would provide five minutes of

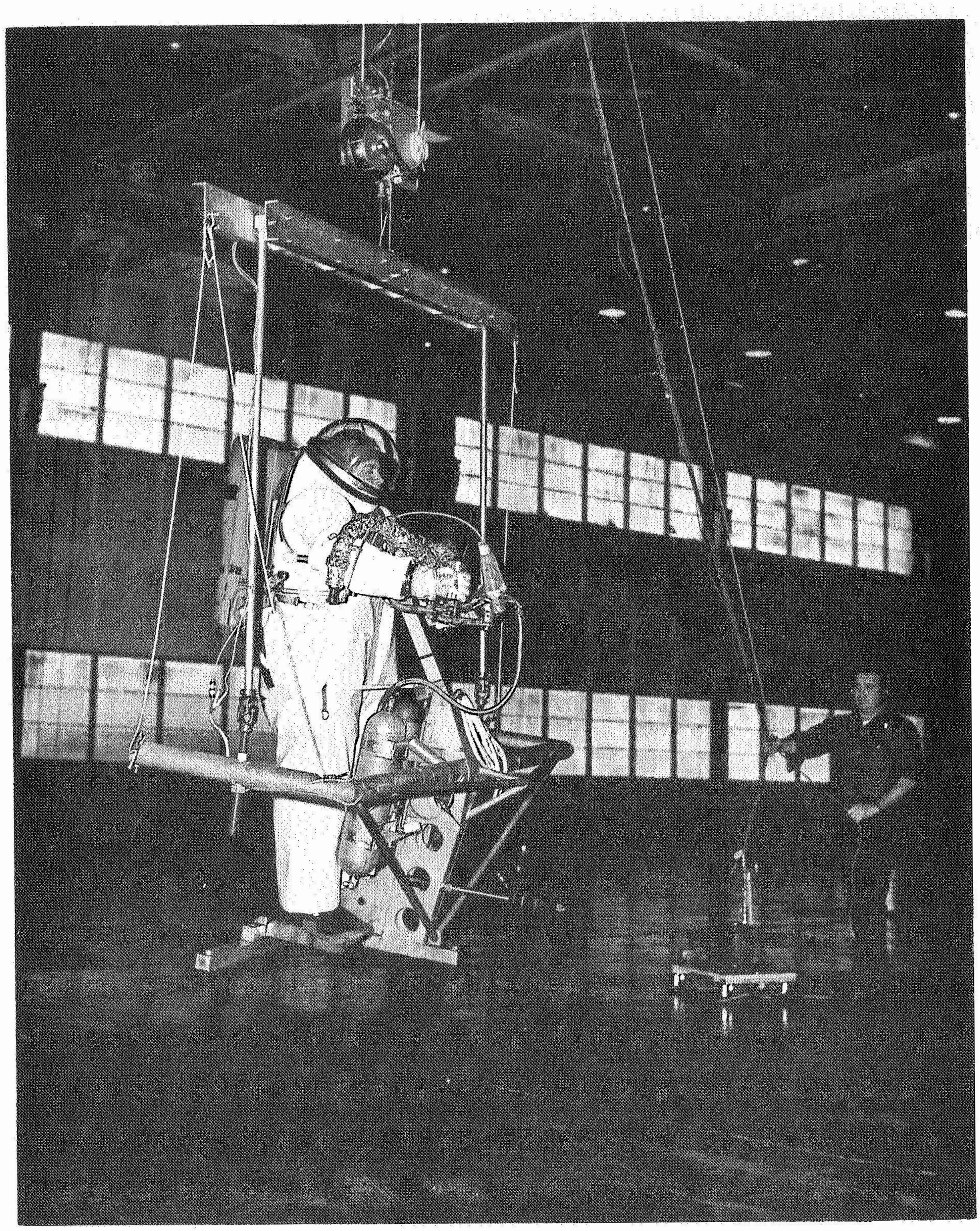


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Figure II-2. Hover Tests at Bell, Operator in RX2A Apollo Hard Suit

flight time versus the presently available 40 seconds. The following table presents an estimated comparison of the characteristics of the present rocket powered 1/6 g vehicle and the proposed jet version:

	Present 1/6 g Rocket Pogo	1/6 g Jet Pogo
Structural weight - lb	195	215
Propulsion system wt - lb	60	95
Fuel weight - lb	49	15
Ballast weight - lb*	171	150
Pilot weight - lb	180	180
Instrumentation weight - lb	15	15
Gross takeoff weight - lb	670	670
Maximum thrust - lb	133	250
Minimum thrust - lb	33	25
Endurance - sec	40	300

* Ballast can be off-loaded to accommodate a pressure suit and PLSS.

The fuel consumption of 15 pounds will reduce the variation in desired g simulation during flight to one-third the present variation.

In order to utilize the capabilities of the jet pogo, a longer overhead rail is desirable. A 1000-foot rail would permit maximum velocities up to about 100 ft/sec for example. Such a rail is available at an installed price of approximately $25.00 per foot. The rail can be straight or it can be circular if sufficient length cannot be obtained for a straight run.

It is proposed that a study be conducted for a conceptual design of a turbojet powered lunar tethered vehicle. The study will include analysis to determine the performance capabilities, research potential, and the effects of characteristics such as engine gyroscopic effects, aerodynamic moments and torque. Concurrently, a study will be made of an extended facility and means by which the 5/6 g suspension system can be improved to increase the fidelity of lunar simulation. A resources plan and schedule will be developed for the vehicle and facility.

b. Rocket Powered Free Flight Trainer

The requirement that an astronaut simulate lunar flight on earth before flying on the moon includes a requirement for earth free flight in a pressure suit. The presently available earth flight vehicles are limited to shirtsleeve flight and a flight duration of twenty seconds with an accompanying flight envelope of 500 feet distance, 80 feet altitude and a velocity of 80 feet per second. The shorter flight time and more rapid reactions of this vehicle to control inputs requires greater concentration and is more difficult to fly than the tethered lunar simulation vehicle described elsewhere.

In order to obtain a larger free flight envelope and one that is practical for pressure suited flight, the flight envelope must be extended through the use of a larger vehicle. Ideally a turbojet powered vehicle would be used for this envelope extension. However, there are no existing turbojet engines which can meet the thrust requirements for a vehicle carrying a pressure suited operator. Therefore, it is proposed that a preliminary design study be conducted on a rocket powered free flight earth prototype lunar vehicle which will carry a pressure suited operator and extend the flight time and flight envelope.

The present one man Pogo vehicle is shown in Figure II-3. This configuration can be enlarged as suggested by Figure II-4 without making radical departures from the present vehicle arrangements yet providing a design similar to the current recommended lunar configuration. The hydrogen peroxide propellant capacity can be increased from 47 pounds to about 300 pounds which will provide one minute of flight and a range up to one mile with a pressure suited operator. An estimated mass breakdown is:

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Figure II-3. Rocket Pogo Stick in Flight

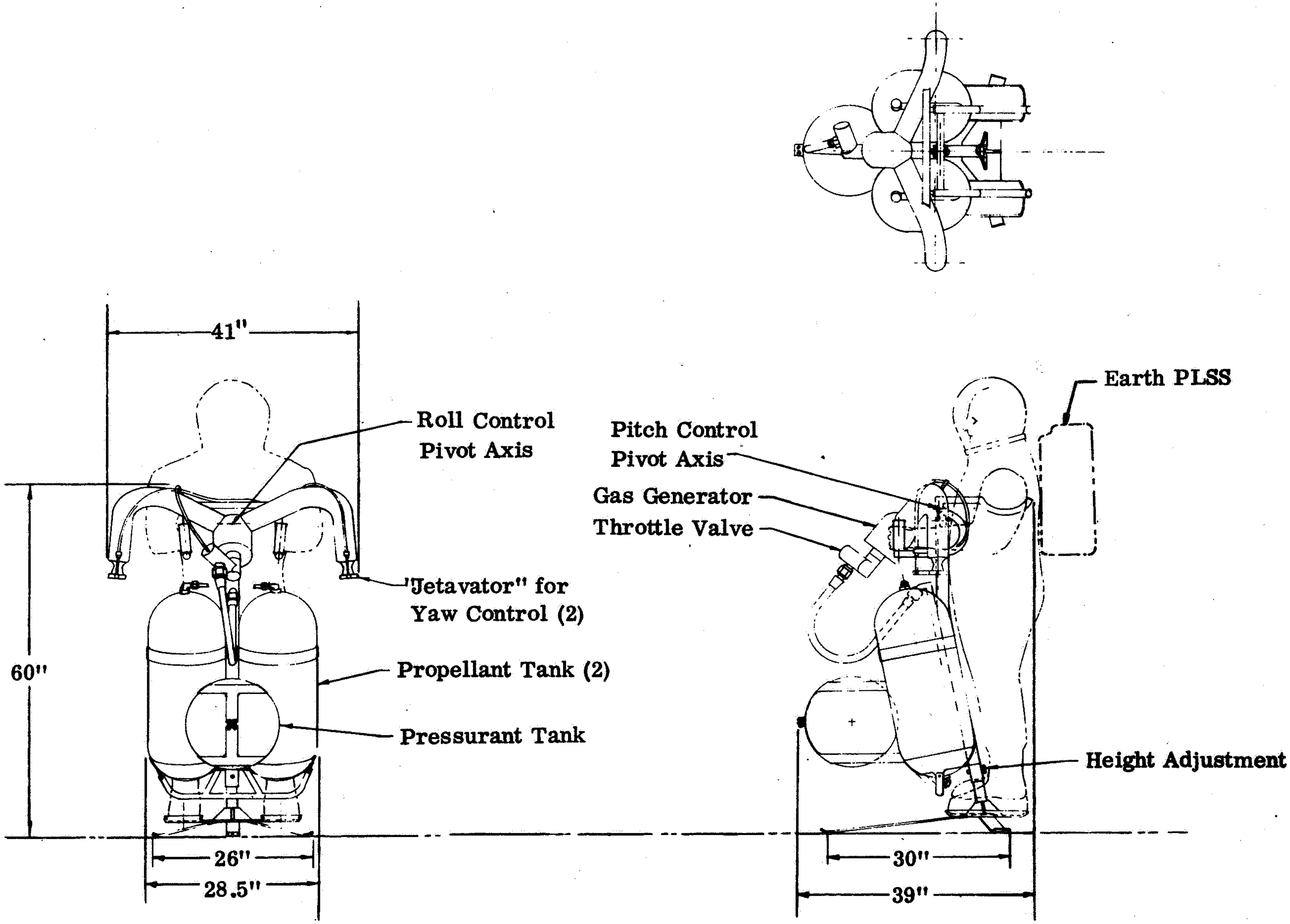

Figure II-4. Personnel Flying Device, Standing

	Weight - lb
Landing Gear	8.4
Structure	16.7
Engine and Control Assy.	39.0
H_2O_2 Tankage	46.0
Helium Tanks	32.8
Valves and Plumbing	15.8
Empty Weight	158.7
Operator	180.0
Pressure Suit and PLSS	100.0
Radio and Battery	10.0
H_2O_2	311.0
Helium	4.0
Gross Weight	763.7

The hydrogen peroxide tanks are G-1 stainless steel oxygen bottles, increased in length and modified to increase the exit port diameter and add vent tubes and anti-vortex baffles. The gas generator and thrusters are similar to those used in the rocket belt family of vehicles and the LLRV lift rockets. The gas generator is increased in diameter to provide 1000 pounds of thrust and the thrusters are sized to provide 500 pounds of thrust each. The throttle valve can be a sleeve valve similar to those used on the rocket belt family of vehicles or a ball valve as used on the NASA - Langley LLRF LM and Icarus vehicles.

The use of a hydrogen peroxide system will result in a highly reliable vehicle as demonstrated by past experience with reaction control systems on the X-1, X-15, Mercury, Centaur, Dyna-Soar, the lift rockets on the LLRV and the rocket belt vehicles. A hydrogen peroxide rocket vehicle can be made available in a relatively short time. The overall design is to include the requirements for a safety tether for training flights that will accommodate the weight of such a vehicle. Vehicle preliminary specifications will be established and a preliminary selection of components will be made. A vehicle resources plan and schedule will be developed.

c. Jet Powered Free Flight Vehicle

A turbofan powered vehicle based on the existing Bell jet belt can serve as a free flight prototype lunar vehicle for duplicating complete lunar trajectories on earth. The jet belt is shown in flight in Figure II-5. The general arrangement of the vehicle is very similar to the Bell rocket belt. It is powered by a Williams Research Corporation WR-19 twin spool turbofan engine and utilizes thrust vectoring for attitude control. The jet engine is positioned vertically at the back of the operator with the exhaust upward. A bifurcated duct divides the flow and directs it to each side of the operator and downward. Jetavators are located at the exit of each duct for three axis attitude control and are actuated by hand controllers whose functions and motions are the same as those of the rocket belts.

The prototype lunar version would place the engine in front of the operator mounted to a "pogo" type structure with a landing gear. The vehicle would carry an operator in shirtsleeves. The vehicle performance is sensitive to system dry weight since the margin of excess thrust at takeoff is limited by the WR-19 engine thrust. Although some engine uprating is possible, it is anticipated that the pilot recovery system will be heavier than the one presently used. The final trajectory capability will therefore depend on the system weight and the degree of thrust uprating attained at the time of vehicle fabrication.

It is expected that the capabilities will be sufficient to fly all trajectories that the first generation lunar vehicle can accomplish. The short period control characteristics will differ from those of a lunar vehicle, as mentioned previously, but lunar navigation and flight path control tasks can be investigated, the need for instruments can be evaluated, and the vehicle used as part of the astronaut training program.

It is proposed that a preliminary design study be conducted for an earth free flight jet powered prototype vehicle based on the present jet belt vehicle propulsion and control concepts and that a resources plan and schedule for such a vehicle be developed.

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Figure II-5. Jet Belt in Flight

C. COMPARISON OF BELL AND NR VEHICLE HANDLING QUALITIES ON BELL SIMULATOR

NASA has provided a comparison of the lunar vehicle concepts recommended by Bell and North American, which indicates a difference in the flight control system approach. Bell has recommended a simple, mechanically linked, unaugmented, flight control system. North American has recommended a more complex, electrically linked, augmented, flight control system.

Of considerable importance in selection of the best approach is an evaluation of the influence of the simulators used by Bell and North American in arriving at their respective recommendations. To aid in this evaluation, it is proposed to set up both the Bell and the North American vehicle and control system characteristics on the Bell visual simulator, and conduct flights using NASA subjects as pilots.

The Bell visual simulator employing a vehicle driven in pitch, roll, and yaw, and a TV projected lunar scene driven in three translational axes, provides a means of rapid and safe comparison of many control systems and system characteristics.

The switch back and forth between the Bell and NR vehicles can be accomplished in a matter of hours since vehicle and control system characteristics are provided by an easily reprogrammed analog computer and the three axis vehicle used on the simulator is already equipped with interchangeable Bell recommended handle bar type controller, and North American recommended LM type side arm controller. Furthermore, both types of controller can be evaluated, with and without augmentation.

D. 1G FREE FLIGHT TESTS WITH A PRESSURE SUITED SUBJECT

The precision and safety with which a pilot in shirtsleeves can control a small rocket supported flying vehicle, using a simple, all-mechanical thrust vector control system has been demonstrated by 10 pilots in over 3000 flights in earth gravity, in a variety of vehicle configurations, and by three pilots in over 60 flights in simulated lunar gravity with the Bell Pogo vehicle on the NASA-LLRF, and on a lunar tether and overhead trolley system at Bell. Pressure suited flight has been demonstrated by two pilots in 27 flights in simulated lunar gravity with the Bell Pogo vehicle on the NASA-LLRF. Free flight with a pressure suited pilot has not yet been attempted, because the

thrust available in the Bell one man vehicle is insufficient to lift the added weight of a pressure suit and PLSS. Because of the low velocity, acceleration, and range limits of the lunar tether systems and other unwanted simulator side effects on the pilot, it is desirable that safety and precision be demonstrated by free flight with a pressure suited operator. Vehicle and control input motion data can be obtained during flight for the validation of ground based simulators.

It is proposed that the Bell two-man pogo shown in flight in Figure II-6 be modified to carry one man in a pressure suit and PLSS, as shown in Figure II-7, and instrumented.

The control concept is identical to that used on Bell one-man vehicles. Controller locations and motions are nearly identical to those in the Pogo vehicle used with a pressure suited operator in simulated lunar gravity on the NASA-LLRF.

The modification to the vehicle will consist of adding a frame work for the PLSS unit as shown in Figure II-7 and adding the sensors and telemetry system to obtain the following data:

(a) Vehicle attitude - pitch, roll and yaw (gyros)

(b) Roll, pitch, yaw, and throttle controller positions (potentiometers)

The control sensitivities will be fixed at a value which has resulted in pilot ratings of 3 to 3-1/2 when flying in shirtsleeves.

The flight program will be conducted by a pilot with previous Pogo experience. Training flights will be made with a safety tether. When sufficient proficiency has been attained, free flights will be made.

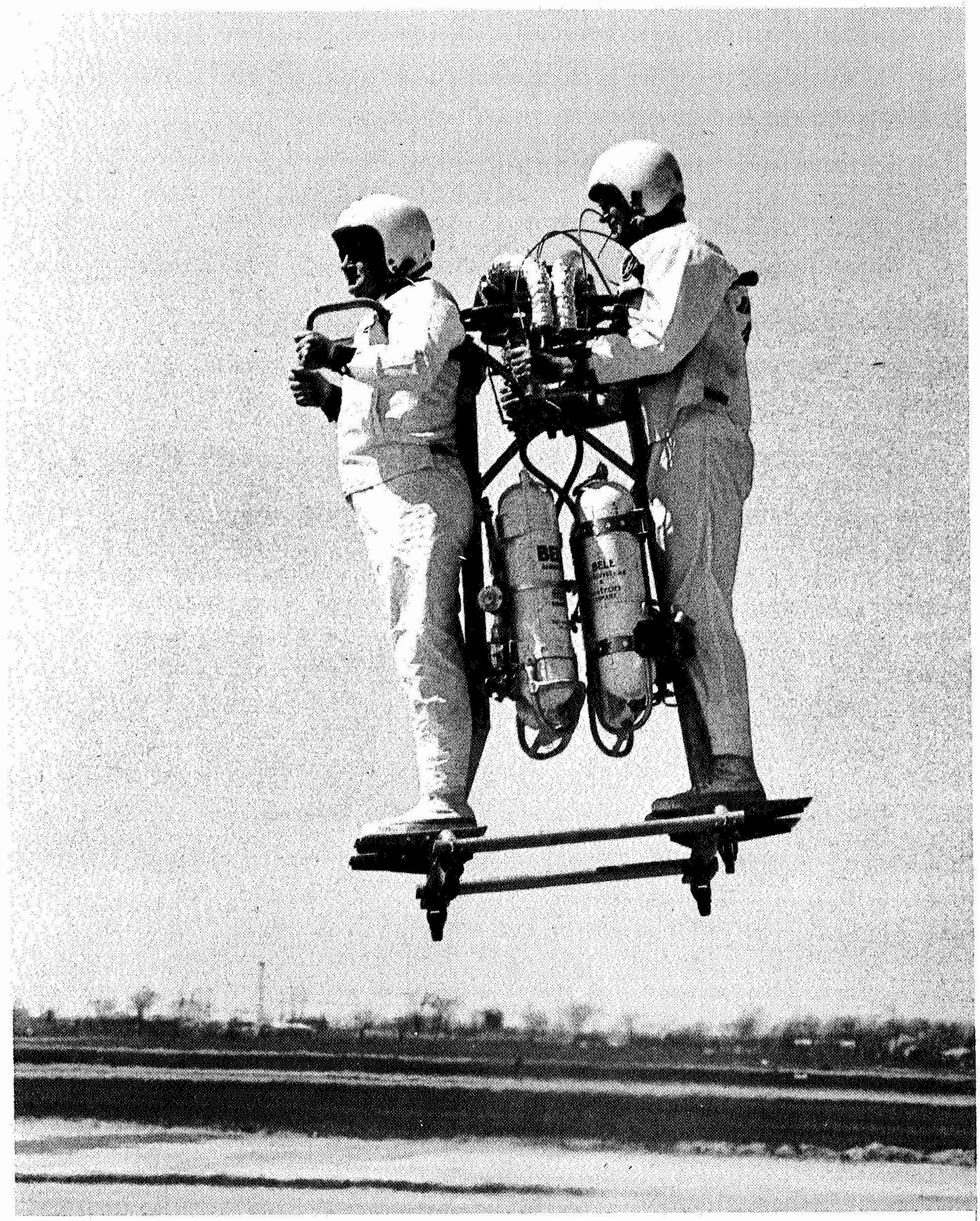

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Figure II-6. Two-Man Rocket Pogo

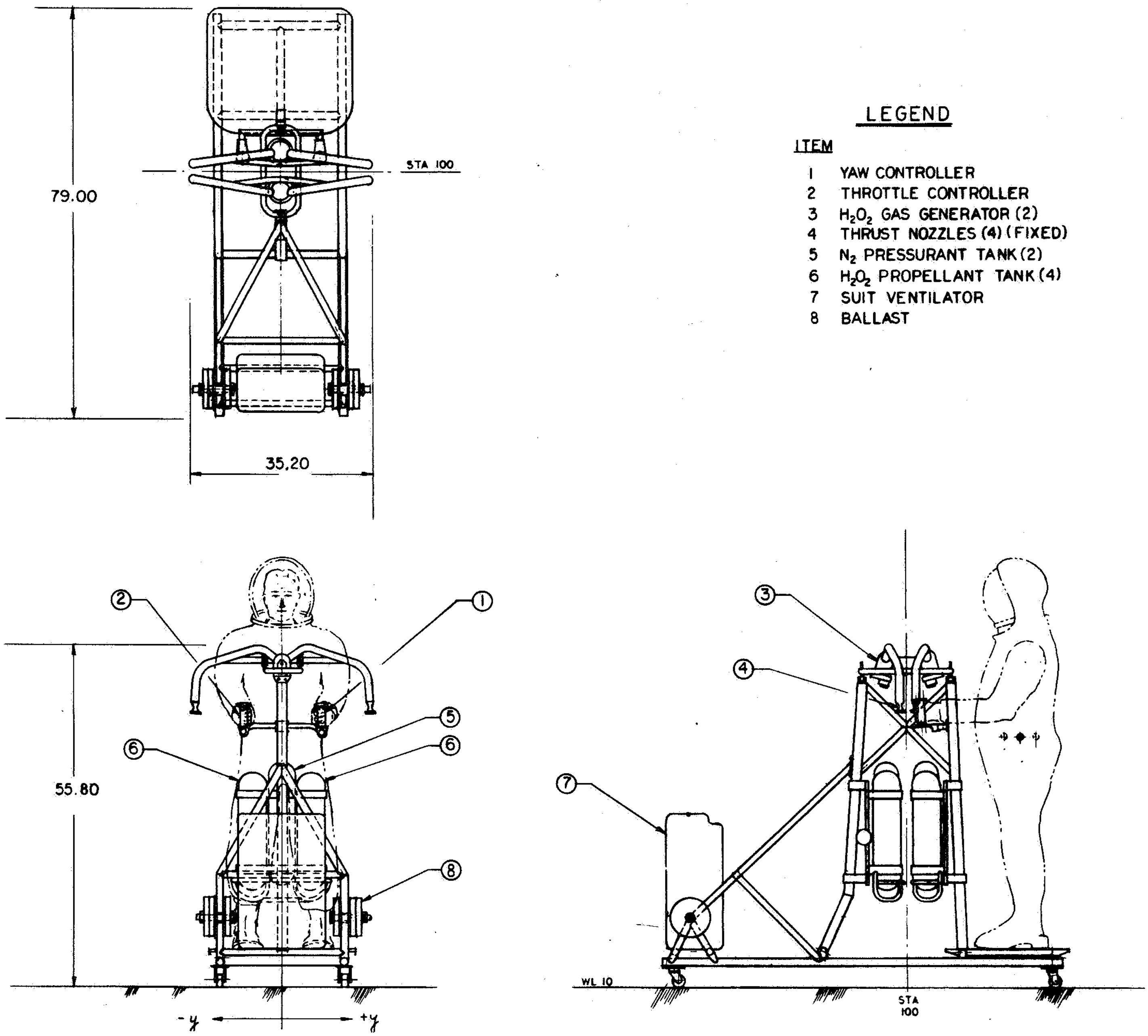


Figure II-7. Fixed Thruster Flying Vehicle

III. WORK STATEMENT

A. SAFETY ABORT AND HAZARD ANALYSIS

1. Conduct a System Gross Hazard Analysis, to identify failure modes which will result in loss of mission objectives, loss of capability to return to LM, or loss of life.
2. Conduct a System Failure Mode and Effects Analysis, and provide reliability logic diagrams.
3. Summarize the potential failure modes into the three categories listed in Task (1).
4. Develop a mission profile and perform a task analysis of OMLFV operation on the lunar surface.
5. Develop task oriented reliability logic block diagrams.
6. Develop elemental task activity reliability estimates using Data Store.
7. Combine the human reliability estimates from (6) in accordance with the logic developed in (5) to obtain an overall hazard and risk assessment.
8. Validate the assessment obtained in (7) by extrapolation from earth-bound vehicle experience.

B. FLIGHT RESEARCH/TRAINING PLAN AND SIMULATOR PRELIMINARY DESIGN

1. Flight Research and Training Planning Tasks

   (a) Conduct visual simulation studies to establish the desired rotational and translational sensitivities and proper combinations for research and training vehicles.

   (b) Conduct helicopter flights to establish procedures and preliminary information on helicopter flight research program (preliminary flight profile information).

   (c) Establish research flight parameters and procedures and data handling and analysis procedures.

   (d) Determine desired characteristics and requirements of visual simulation for flight research and training. Investigate existing visual facilities and possible modifications to make them suitable for flight research and training.

   (e) Confer with NASA training personnel and crews to assure compatibility of the training plan developed during the OMLFV study with NASA training

philosophy and other astronaut training activities. Modify the existing plan as necessary to reflect the information gained from these conferences and other tasks of this study.

(f) Investigate facilities and vehicles (existing, under development, and proposed) for lunar flight simulation to determine applicability to one man flying vehicle flight research or training. Incorporate the use of those facilities which are appropriate into the overall plan.

(g) Prepare an integrated flight research and flight training plan. This will include a definition of:

(1) Parameters to be investigated and program required
(2) Facilities and equipment required
(3) Sequence of activities and schedule
(4) Man hours required

2. Preliminary Training Program

Conduct preliminary training program using NASA subjects and existing visual simulators, 1/6 g tether vehicles (rocket powered) and facilities, and 1 g vehicles.

3. Preliminary Design of Simulators

a. Conduct a conceptual design study for an extended range, turbojet powered lunar tethered vehicle and overhead rail. The study will include analyses to determine (a) the vehicle performance capabilities and research potential and the effects of turbojet engine operation on lunar simulation fidelity and (b) the extended range vehicle facility requirements including an investigation of methods of improving the 5/6 g suspension system characteristics.

b. Conduct a preliminary design study of a hydrogen peroxide earth gravity free flight vehicle to provide a flight duration of one minute for a pressure suited operator with a PLSS. Establish vehicle preliminary specifications and make a preliminary selection of propulsion system components. Determine the safety tether requirements for initial training flights.

c. Conduct a preliminary design study of a turbofan powered earth gravity free flight pogo vehicle to provide up to 10 minutes flight time and the capability to duplicate lunar flyer trajectories on earth with a shirtsleeved operator. Investigate pilot recovery systems.

4. Resources Planning Tasks

Provide a resources plan for the vehicles and facilities defined in Task 3 above. This will include schedule, manpower required and cost data.

C. COMPARISON OF BELL AND NR VEHICLE HANDLING QUALITIES ON THE BELL SIMULATOR

1. Conduct a two week simulation test program using two or three NASA subjects as pilots, to compare the Bell and North American vehicle handling qualities, both with and without augmentation, as follows:

1st, 2nd, 3rd days:	Pilot training flights on Bell configuration
3rd day, 2nd shift:	Switch simulator to NR configuration
4th, 5th, 6th days:	Pilot training flights on NR configuration
7th day:	Pilot evaluation flights on NR configuration
7th day, 2nd shift:	Switch simulator to Bell configuration
8th day:	Pilot evaluation flights on Bell configuration
9th day, 2nd shift:	Simulator changes as required for additional tests to be determined as result of first 8 days experience
9th, 10th days:	Additional flights as required

2. Collect, process, and analyze data on accuracy of pilot performance and propellant consumed in accomplishing specific tasks. Correlate with pilot opinion rating data.

3. NASA Support Required

(a) Continued loan of the Apollo Block I side arm 3 axis attitude controller.

(b) Data on the NR vehicle mass characteristics and control system characteristics.

(c) Two or three test subjects, preferably with previous flight and simulator experience, to be at Bell Aerosystems for a two week period.

D. 1 G FREE FLIGHT TESTS WITH A PRESSURE SUITED SUBJECT

1. Modify the present Bell two-man Pogo vehicle to carry a pressure suited operator and PLSS.
2. Conduct approximately 60 flights with the operator in a pressurized space suit. These flights will be up to 20 seconds duration. Initial flights will be made with a safety tether until sufficient proficiency has been attained to fly free.
3. Provide color motion picture coverage of representative free flight.
4. Prepare a final report describing the equipment and test results and summarizing the operator and observer comments.
5. Provide a telemetry system and instrument the vehicle to obtain vehicle attitude and control position histories during flight.
6. NASA Support Required - Loan of an Apollo pressure suit and support equipment for a six week period.

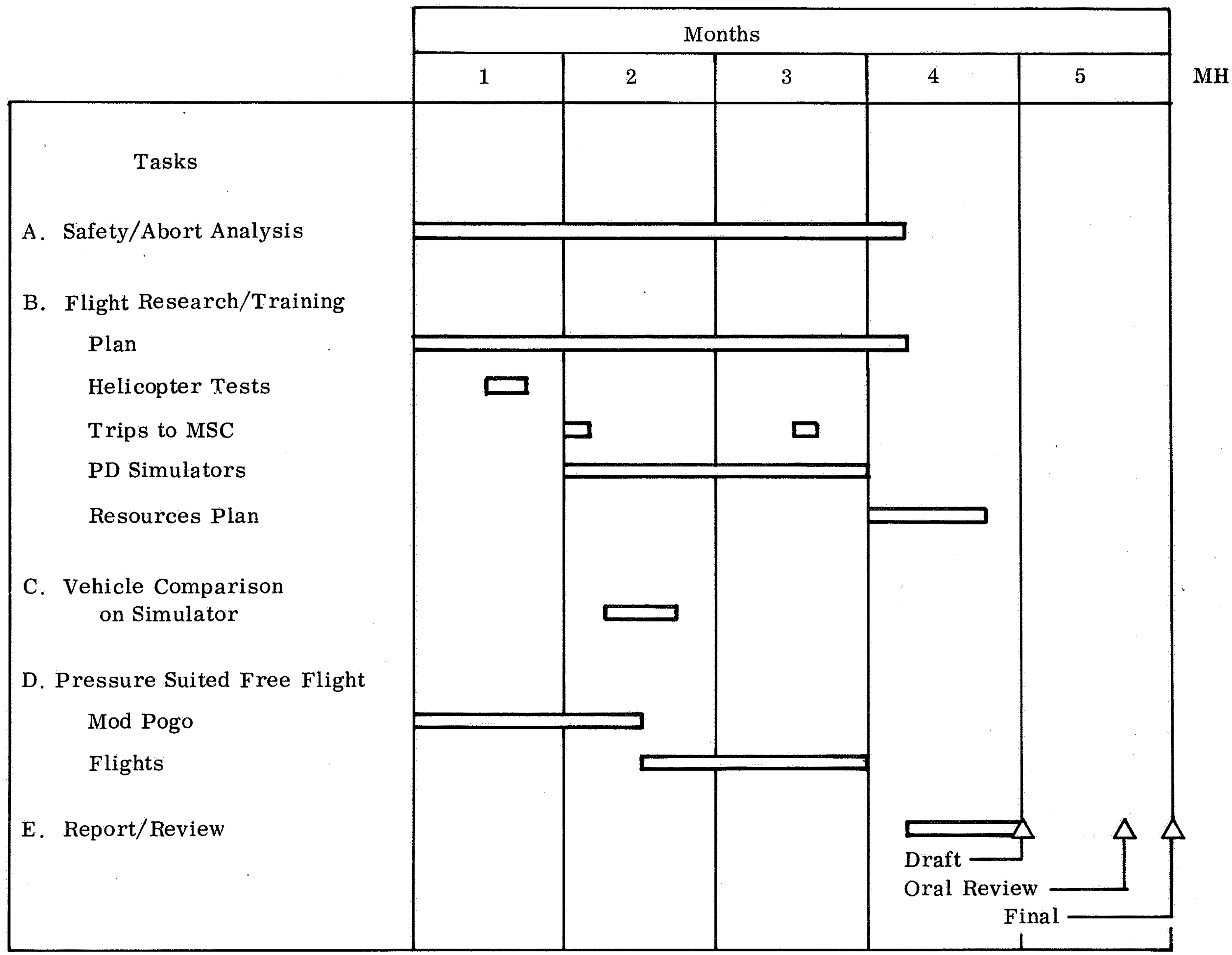
E. FINAL REPORT AND REVIEW

1. Provide 101 copies and one reproducible copy of a final report. The final report will include a section on each of the task areas studied during the follow-on program. Printing will be in accordance with requirements specified in Contract NAS 9-9044.

2. Conduct an oral review at Houston at the end of the follow-on study period, covering the tasks studied during the follow-on program.

3. Provide two copies of each slide or Vuegraph used in the oral review. Provide 101 copies of a brochure reproducing the slides or Vuegraphs used in the oral review.

IV. SCHEDULE



Months
1
2
3
4
5
MH
Tasks
A. Safety/Abort Analysis
B. Flight Research/Training
Plan
Helicopter Tests
Trips to MSC
PD Simulators
Resources Plan
C. Vehicle Comparison on Simulator
D. Pressure Suited Free Flight
Mod Pogo
Flights
E. Report/Review
Draft
Oral Review
Final

V. ADDITIONAL RECOMMENDED TASK DESCRIPTIONS

Attached are outlines of additional supporting research and technology tasks, recommended because they will provide design data required for follow-on lunar flyer development, mission planning data, or provide verification of items which will increase confidence in follow-on program cost and schedule estimates.

Task 1 will provide a design analysis for a hydrogen peroxide propulsion system for the lunar flyer, which can be available at an earlier date and lower cost than the bi-propellant system.

Tasks 2, 3 and 4 will provide data on operational use of the flyer.

Tasks 5 and 6 will provide design data for the flight control and landing gear systems.

Task 7 will demonstrate engine operating performance.

1. EARLY FLYER HYDROGEN PEROXIDE PROPULSION SYSTEM

A. BACKGROUND

Recent flyer studies indicate that the Lunar Flyer will require rocket thrust of about 300 pounds. The present Bell Rocket Belt Hydrogen Peroxide Propulsion System provides 300 pounds thrust and has demonstrated extremely high reliability in over 3000 earth flight tests. Preliminary analysis indicates that the incorporation of the rocket belt hydrogen peroxide propulsion system on the One Man Lunar Flying Vehicle could provide an earlier operational Lunar Flyer, with minimum development risk. The vehicle would provide sufficient lunar performance capability to conduct useful scientific exploration, and to verify lunar flight handling qualities prior to incorporation of the later LM propellant system.

B. TASKS

(1) Design the installation of the rocket belt hydrogen peroxide system on the pogo One Man Lunar Flying Vehicle. This would be the same basic vehicle which would later use a LM propellant propulsion system.

(2) Investigate each component of the hydrogen peroxide system for application to lunar flight.

(3) Design the modification of the throttle valve range from 180-300 lb to 50-300 lb. (A thrust range from 33 to 300 lb has already been demonstrated, using the same 300 lb gas generator.)

(4) Increase the nozzle expansion ratio from 3:1 to 30:1 to take advantage of vacuum I_{sp}.

(5) Investigate flight weight tankage to carry approximately 300 lb of hydrogen peroxide for a storage period of 2 to 3 months.

(6) Provide a hydrogen peroxide system preliminary design, performance estimates, and cost and schedule estimates.

2. FLYER ORIENTED APPLICATION ANALYSIS

A. BACKGROUND

Most previous lunar scientific mission studies have been conducted by personnel who were unfamiliar with flyers and assumed that exploration would be done on foot or with wheeled vehicles. Thus, scientific experiments were designed to be compatible with surface mobility. Bell funded studies have indicated that scientific time at remote sites can be extended, scientific return increased by in-flight remote sensing, and new experiments devised not possible with surface mobility, if experiments are designed with flying mobility as a consideration. In addition, several modes of surface rescue are available with a flyer, which could enhance total mission safety and probability of success.

B. TASKS

(1) Analyze specific Apollo lunar landing sites, such as Marius Hills region, for application of flying mobility to increase remote site exploration time.

(2) Investigate specific payloads and vehicle propellants required to accomplish missions analyzed in Task (1).

(3) Investigate increase of scientific return by accomplishing high resolution geologic reconnaissance by remote sensing during flights from LM to remote sites and return.

(4) Investigate methods of improving scientific return for specific experiments described in the Santa Cruz Report, such as accurate and rapid photographic survey of exploration sites, deployment of surface experiment packages, placing of seismic charges, etc.

(5) Establish methods of using a flyer for fast emergency return to LM of a walking or riding astronaut, or emergency aid or resupply to an EVA astronaut. Determine the maximum rescue or aid range from LM.

(6) Investigate the application of a stripped down minimum weight flying seat as the seat on a lunar roving vehicle, as an emergency return device. Determine the maximum practical one-way return range, carrying the astronaut only, and the flying seat weight.

3. LUNAR EXPLORATION SIMULATION USING FLYING MOBILITY

A. BACKGROUND

The U.S. Geological Survey Branch of Astrogeology at Flagstaff, Arizona, has conducted simulations of lunar geological explorations using walking and surface vehicle mobility. No such simulation has been done using flying mobility. It is recommended that typical flying sorties be simulated, with a pressure suited subject, using a helicopter to provide the mobility which a lunar flyer would provide on the moon. This could be accomplished by having the subject ride as passenger in a helicopter, providing verbal commands to the helicopter pilot. He could thus simulate his ability to navigate and explore from the air, recognize items of interest, and evaluate the suitability of this type of exploration.

B. TASKS

(1) Provide helicopter and pilot, suitable for carrying a pressure suited subject as passenger, modified as required to provide downward visibility typical of the lunar vehicle.

(2) Conduct sorties, to be planned by U.S.G.S.

(3) Provide final report (U.S.G.S.).

4. FULL SCALE MOCKUP/PRESSURE SUIT TESTS

A. BACKGROUND

A portion of the astronaut's EVA time will be consumed in unloading the lunar flying vehicle from the LM descent stage, and servicing the flyer with propellants and pressurizing gas. Time estimates to conduct these tests have been provided in the current flyer study by breaking the operation into specific subtasks required and estimating the time for each operation. However, estimates of time required for basic operations vary widely. A full scale simulation using a lunar weight vehicle and a pressure suited subject in a lunar gravity suspension would provide time line data with a higher degree of confidence, for use in mission planning.

B. TASKS

(1) Fabricate a full scale lunar weight mockup of the lunar vehicle. Fabricate a full scale mockup of one quadrant of the LM descent stage, and unloading and servicing equipment, including disconnect fittings.

(2) Assemble a 1/6 g pilot suspension from existing components.

(3) Conduct unloading and service operations to obtain time line data and recommendations for modifications to decrease time for unloading and servicing.

5. HANDLING QUALITIES/CONTROL AUGMENTATION

A. BACKGROUND

The Bell study of the One Man Lunar Flying Vehicle has recommended a simple mechanically linked flight control system. This system has demonstrated safe, reliable, and precise flight control on thousands of free flights of rocket belts, pogo vehicles, and flying chairs, and has proven acceptable in simulated lunar gravity flight tests on the Langley Research Center LLRF with both shirt sleeve and pressure suited operators.

The all mechanical system consists of a handle bar type controller which is mechanically linked directly to pivoted engines, for control of vehicle pitch, roll and yaw attitude. This type of controller provides proportional command of vehicle angular acceleration, and does not include what is commonly called "stability augmentation". Stability augmentation results in command of vehicle angular rate rather than angular acceleration, and is commonly provided in space vehicles by using rate gyro feedback and electronic control systems. Because stability augmentation has proved beneficial in other vehicles, Bell has conducted tests in the visual simulation facility to determine the benefit of stability augmentation for the lunar flyer. Tests using a handbar type controller indicated that although handling qualities were acceptable without stability augmentation, the addition of stability augmentation decreased pilot workload and improved the pilot opinion rating of the vehicle.

As an additional check, the handle bar controller was replaced with a LM type three-axis hand controller. This was flown on the visual simulator both with and without stability augmentation. It was found, with stability augmentation that handling qualities were about the same with the LM type hand controller as with the handle bar controller. However, without stability augmentation, the LM type hand controller was unacceptable.

In view of these findings, Bell recommends the employment of the simple mechanical system, in order to avoid the design and operational complexity of the electronic control system. However, Bell recommends further investigation of methods of incorporating stability augmentation in such a manner as to not degrade the inherent high reliability of the mechanical system.

It has been found, by analog simulation, that stability augmentation, or rate command, can be provided by introducing a parallel spring/damper between the handle bar controller and the gimballed engines. This results in handling qualities equal to the electronic system. However, it does not allow for vehicle attitude trim control as CG shifts, as does the unaugmented system. Trim adjustment can be obtained by the addition of a second spring, with some degradation in the stability handling qualities. Additional investigation is required to determine the optimum parameters for flight control and trim control or the need for other methods of providing trim control. Investigations should be conducted on the mass and inertia effects of the engines on the spring damper system, and the entire system should be optimized for a pressure suited operator, both with and without augmentation.

B. TASKS

(1) Conduct additional analog simulation testing of mechanical control augmentation, to establish the benefits to be gained for individual and combined control axes, and to optimize parameters for control and for trim. Conduct similar tests using a pressure suited subject to optimize control parameters and control handle location, and direction and magnitude of control motion, for pressure suited operation, with and without control augmentation.

(2) Using the results of Task (1), fabricate a prototype mechanical augmentation system and install it on the fixed base vehicle used in the visual simulation facility. Incorporate the equipment required to provide lunar vehicle engine mass effects. Conduct tests to verify results found in Task (1).

(3) Install the prototype augmentation system on the Bell lunar tethered vehicle and conduct flight tests.

(4) Install the prototype mechanical augmentation system on the Bell free flight Pogo vehicle to evaluate its value for application to a training vehicle.

(5) Accomplish the preliminary design of the mechanical augmentation system on the One Man Lunar Flying Vehicle.

(6) Install the system on the Langley Research Center backpack test vehicle (ICARUS), and, when available, on the FLEEP vehicle. Conduct tests with shirt sleeve and pressure suited subjects.

6. LANDING GEAR ANALYSIS

A. BACKGROUND

Bell has compared various vehicle and landing gear configurations using a digital computer program. This program analyzes landing dynamics in two dimensions, that is, in the pitch plane, and provides time histories of loads, deflections, and vehicle stability as a function of vehicle, landing gear, flight path, lunar surface, and soil parameters. This program is the only one known which can analyze the strut/pad or helicopter type, as well as the LM type gear. It has been verified by comparison with previous analyses on other computer programs verified by model drop tests. In order to establish final landing gear design requirements, it will be necessary to expand the Bell program to three dimensional analysis to include vehicle yaw.

B. TASKS

(1) Expand the present Bell two-dimensional program to three dimensions.

(2) Analyze a typical lunar flying vehicle to establish the worst combination of landing conditions, to be used for landing gear design.

(3) Establish landing gear design/performance requirements.

7. PROPULSION SYSTEM TESTING

A. BACKGROUND

Present One Man Lunar Flying Vehicle studies have established a requirement for rocket engines in the 100 to 150 lb thrust class, using LM propellants. Several vendors have demonstrated the capability of providing a throttleable engine of this size. However, in some cases the demonstrations of throttling were conducted by changing propellant feed pressure, a method not suitable to lunar vehicle use, and in other cases maximum engine operating temperature was higher than is desirable for maximum engine life. Since the engine/valve assembly is the single most costly component, and a long lead item on the lunar flyer, confidence in the program cost and schedule would be increased by early demonstration tests of an engine which meets the thrust, throttling, stability and temperature margin requirements of the lunar flyer.

B. TASKS

(1) Design and fabricate two test prototypes of a rocket engine of 150 lb thrust, 6:1 throttle ratio, and with a temperature margin compatible with 3-1/2 hours engine life.

(2) Conduct tests to demonstrate stable operation, maximum thrust, throttle ratio, specific impulse as a function of percent throttle, temperature as a function of percent throttle, and engine life for repeated typical mission duty cycles.