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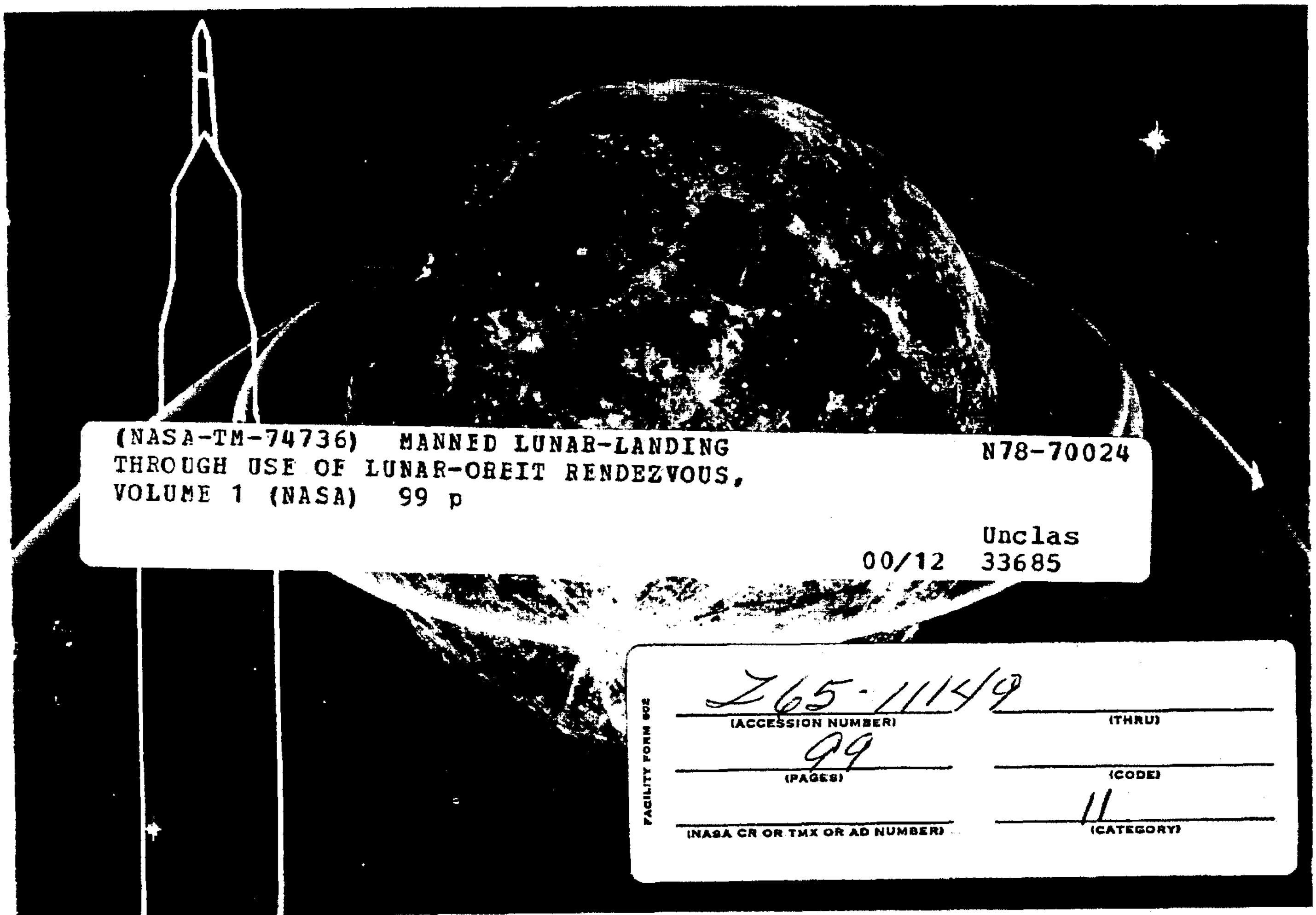

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MANNED LUNAR-LANDING through use of LUNAR-ORBIT RENDEZVOUS

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F O R E W O R D

In the course of conducting research on the problem of space rendezvous and on various aspects of manned space missions, Langley Research Center has evolved what is believed to be a particularly appealing scheme for performing the manned lunar landing mission. The key to the mission is the use of lunar rendezvous, which greatly reduces the size of the booster needed at the earth.

More definitely the mission may be described essentially as follows: A manned exploration vehicle is considered on its way to the moon. On approach, this vehicle is decelerated into a low-altitude circular orbit about the moon. From this orbit a lunar lander descends to the moon surface, leaving the return vehicle in orbit. After exploration the lunar lander ascends for rendezvous with the return vehicle. The return vehicle is then boosted into a return trajectory to the earth, leaving the lander behind.

The significant advantage brought out by this procedure is the marked reduction in escape weight required; the reduction is, of course, a direct reflection of the reduced energy requirements brought about by leaving a sizable mass in lunar orbit, in this case, the return capsule and return propulsion system.

This report has been prepared by members of the Langley Research Center to indicate the research that has been conducted, and what a complete manned lunar landing mission using this system would entail. For further reference, main contacts are John D. Bird, Arthur W. Vogeley, or John C. Houbolt.

J.C.H.
October 31, 1961

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SUMMARY AND CONCLUSIONS

Studies made at Langley Research Center of various schemes for performing the manned lunar landing mission indicate that the lunar rendezvous method is the simplest, most reliable, and quickest means for accomplishing the task. This technique permits a lunar exploration to be made with a single C-3 booster. A first landing is indicated in March 1966, with a possibility of an attempt as early as November 1965. These dates do not require changes in previously established Apollo, C-1, and C-3 development schedules. Further, the lunar rendezvous approach contains a number of features which tend to raise the schedule confidence level; the most important of these are:

(a) The Apollo vehicle, the lander, and the rendezvous experiment can all proceed on an independent parallel basis, thus avoiding schedule conflicts; further, the overall development is simplified because each vehicle has only a single function to perform.

(b) The lunar rendezvous approach permits complete system development to be done with C-1, which will be available and well developed, and makes the entire C-3 picture exceptionally clean and simple, thus resulting in a minimum cost program.

In amplification of these general remarks, the following specific conclusions are drawn from the technical studies which are summarized in the body of this report:

A. Mission Approach and Scheduling:

1. The lunar rendezvous method requires only a single C-3 or C-4 launch vehicle. Earth orbital weights required for various system arrangements are summarized in figure 1. (See also tables VI and VII later in the text.)

2. The lunar rendezvous method schedules the first landing in March 1966.

3. The lunar rendezvous method does not require that the Apollo vehicle be compromised because of landing considerations.

4. The lunar rendezvous method allows the landing vehicle configuration to be optimized for landing.

5. The lunar rendezvous method requires only C-1 boosters for complete system development.

6. The lunar rendezvous method provides for complete lander checkout and crew training in the lunar landing, lunar launch, and rendezvous docking operations on the actual vehicle.

B. Funding:

The lunar rendezvous method results in a program cost which will be less than the cost of other methods for the following reasons:

1. Requires fewer (20 to 40 percent) large boosters than other programs.

2. Requires no Nova vehicles.

3. Requires less C-3 or C-4 vehicles than other programs.

4. Programs most flights on best-developed booster (C-1).

5. Requires a minimum of booster ground facilities, because large boosters are avoided and because of a low launch rate.

The lunar rendezvous method can be readily paralleled with some other program at least total program cost.

C. Lunar Rendezvous:

The lunar rendezvous under direct, visual, pilot control is a simple reliable operation which provides a level of safety and reliability higher than other methods as outlined below.

D. Safety and Reliability:

1. The lander configuration is optimized.

2. The single-lander system permits safe return of the primary vehicle in event of a landing accident.

3. The two-lander system provides a rescue capability.

4. Crews can be trained in lunar landing, lunar launch, and rendezvous docking operations in the actual vehicle.

5. Requires fewest number of large booster flights.

6. Provides for most flights on best-developed booster (C-1).

E. Abort Capability:

1. An abort capability meeting the basic Mercury-Apollo requirements can be provided.

2. This abort capability can be provided with no additional fuel or weight penalties.

F. Lunar Lander Development:

1. Lunar lander design is optimized for landing.

2. Being essentially separate from Apollo, development can proceed with a minimum of schedule conflict.

3. Research, development, and checkout can be performed on ground facilities now under procurement and which will be available in time to meet the program schedule.

G. Development Facilities:

1. The lunar rendezvous method requires no additional booster ground facilities (see item B-5).

2. The ground facilities required for rendezvous-operations development are now being procured and will be ready.

3. The ground facilities for lander development and checkout are now being procured and will be ready.

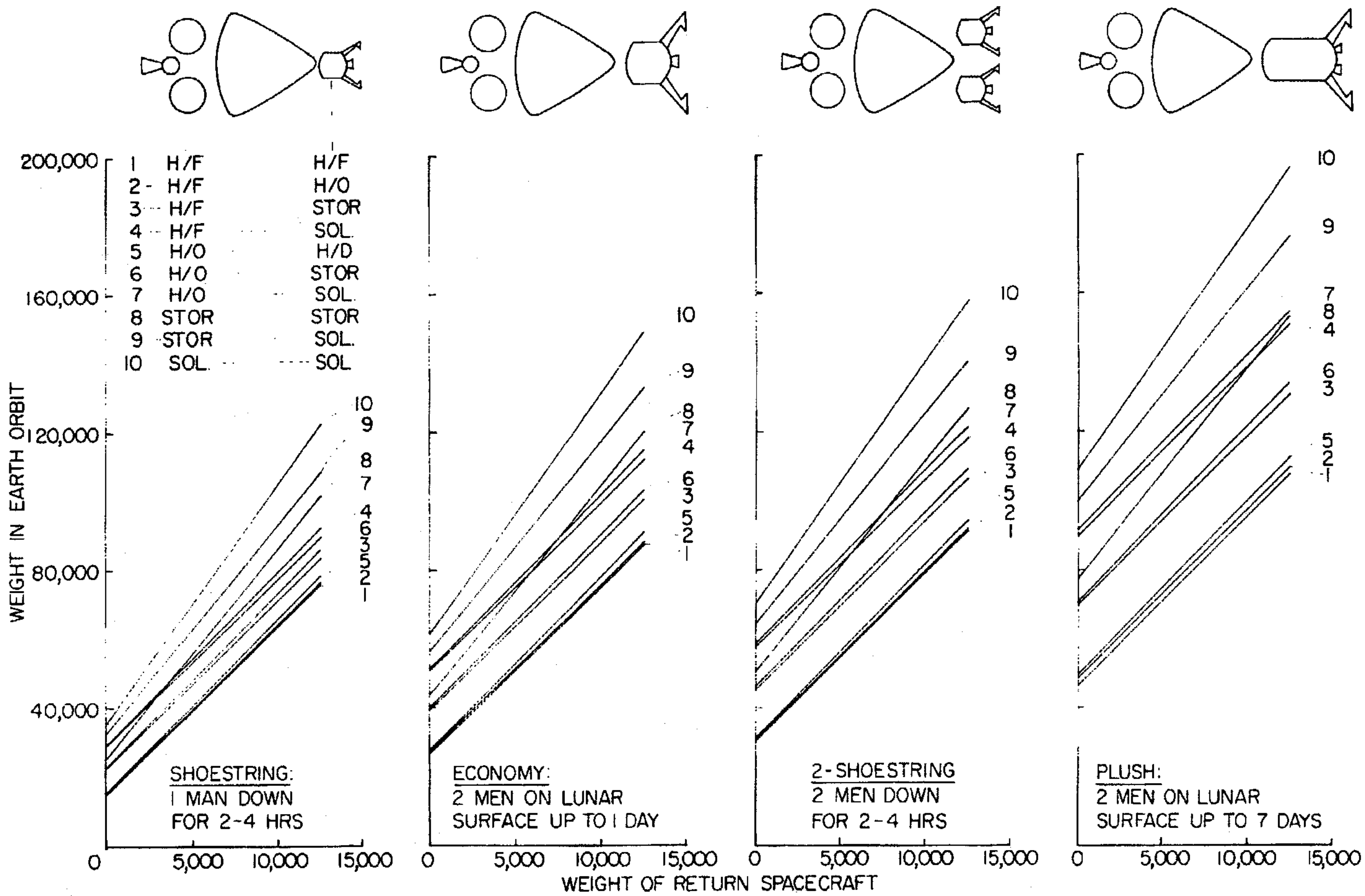


Figure 1.- Summary of earth orbital weights.

INTRODUCTION

For several years Langley Research Center has been actively studying various aspects of the general problem of rendezvous in space. For the past year and a half attention has been focused on using rendezvous to accomplish the manned lunar landing mission, with specific attention being given to the use of lunar rendezvous, because of its attendant benefits. During this time presentations of the basic concepts of the lunar rendezvous approach and specific research results have been given before various groups, including:

1. Space Task Group - Langley briefing December 10, 1960
2. Administrator's briefing, December 14, 1960
3. Space Exploration Council, January 5, 1961
4. NASA Intercenter Rendezvous meeting, February 27-28, 1961
5. Lundin Committee, May 1961
6. Heaton Committee, June 1961
7. Apollo Conference Review, July 1961
8. Golovin Committee, August 1961
9. Space Task Group briefing, August 1961

The purpose of this report is to put under one cover the various pieces of information, facts and figures that have been disseminated, and more particularly to outline a detailed lunar landing program using the lunar rendezvous approach, as was requested by the Large Launch Vehicle Planning Group in a telegram, dated August 24, 1961. The study presented gives consideration to a number of system configurations involving various return capsule weights, fuel combinations, and associated mass fractions, including those requested by the planning group.

The report is divided basically into three parts: Part I encompasses the mission approach and weights involved, the scheduling, funding, safety and reliability aspects, and the required development program. Part II contains the technical discussion of the various phases of the mission. Part III, which is an appendix, indicates the additional facilities and studies which are underway and planned at Langley Research Center in support of a manned lunar landing.

PART I

PROGRAM FOR MANNED LUNAR LANDING THROUGH USE OF LUNAR RENDEZVOUS

MISSION DESCRIPTION

Mission profile and vehicle concept.- The major trajectory phases for accomplishment of manned lunar landing and return to earth through the use of the lunar orbit rendezvous technique are illustrated in figure 2. It is apparent that a number of these phases will be common to any lunar landing mission and that items 5 through 8 represent particular considerations which apply to the lunar orbit rendezvous mission, in place of other particular considerations for other missions.

The vehicle components used in the various phases of the mission are shown in figure 3. As noted on the figure, and as discussed in subsequent sections, several components, including the reentry vehicle and the propulsion and fuel components for injection into the moon-earth return trajectory, are left in a lunar orbit when the lander goes down to the surface of the moon. On return from the lunar surface, the lander accomplishes a rendezvous with the components left in the lunar orbit, after which injection is made into the moon-earth return trajectory.

The characteristic velocity increments involved in the various phases are shown in table 1. For the purposes of the present study, the mission commences from a 300-mile earth orbit or coasting orbit. The velocity increments are listed in two columns. The first column contains the velocity increments corresponding to impulsive thrust application. The second column contains the realistic velocity increments corresponding to finite thrust levels and thrusting times, with additional allowances for inclusion of pilot control in the propulsion system. The values in the second column have been used in this study in arriving at the necessary mass ratios and corresponding weights for accomplishment of the mission.

It should be noted that a conservative approach has been taken in defining the velocity increments used in this study. In general, nonoptimum conditions have been assumed for each phase. While this approach tends to penalize the results somewhat, it is considered to be the logical approach to a parametric study and contributes to the confidence in the results obtained. Optimization studies are being carried out in a number of areas, and the results of these studies will be used in detailed planning for specific missions.

Detailed discussions of research programs and research results directed to accomplishment of various phases of the lunar orbit rendezvous mission are included in subsequent sections of this report. Some trajectory and operational considerations for the various phases of the mission are discussed below.

1. Launch from Cape Canaveral.- This is common to all missions since Cape Canaveral has been designated as the launch site for lunar exploration.

2. Establishment of earth orbit or coasting orbit.- The lunar orbit rendezvous mission can be accomplished either with or without earth orbit rendezvous. In the former case, the benefits of earth orbit rendezvous can be utilized. In any event, a coasting orbit or coast phase, as opposed to direct injection, is required to give complete freedom for injection into the earth-moon transfer trajectory. As discussed in detail in reference 1, use of a coast phase allows freedom in choice of the time of the month (i.e., lunar declination) for the mission, allows launch azimuths within the range safety requirements, allows some freedom in the design of the trajectory to avoid part of the Van Allen radiation belts, and permits the design of the trajectory plane to be nearly co-planar with the moon-earth orbital plane.

3. Injection into the earth-moon transfer trajectory.- The choice of velocity, or energy, and the corresponding time of flight for the transfer trajectory is a compromise between very short flight times and the velocity increment required to establish the vehicle in a close lunar orbit. A $2\frac{1}{2}$- to 3-day flight time to the moon is satisfactory. The lunar rendezvous mission has the safety advantage resulting from the fact that the transfer trajectory can be designed as a free return circumlunar trajectory, which would be utilized if a decision or malfunction prohibited going into the lunar orbit. Such a circumlunar trajectory will give a safe return into the earth's atmosphere without compromising the normal trajectory and with an injection velocity increment less than that indicated in table 1 (see, for instance, reference 2). Injection considerations involving inclination of the trajectory plane to the moon's orbital plane, the declination of the moon, injection angle and azimuth angle, etc., are discussed in some detail in references 1 and 3.

4. Midcourse correction.- Common to all missions. The velocity increments included in table 1 are typical values resulting from midcourse guidance studies.

5. Establishment of lunar orbit.- A nearly equatorial lunar orbit is desirable from rendezvous considerations, especially if the exploration time on the lunar surface is to be of the order of a week. Such an orbit gives the advantages of landing capability at any selected location

along the lunar equator, the landing can be accomplished during each orbital period ($\approx$ 2 hours), and return to the orbiting vehicle can be accomplished during any orbital period. For a 1- or 2-day surface exploration time, the orbit can be more inclined to the lunar equatorial plane. In any event, preliminary studies (for instance, reference 4) indicate that establishment of equatorial orbits involves only small additional velocity increments. Studies of optimum conditions for establishing equatorial orbits are currently in progress.

6. and 7. Descent from orbit and return to orbit.- These items are discussed in detail in subsequent sections.

8. Injection into moon-earth return trajectory.- A requirement on the return trajectory is that the reentry vehicle have the capability for making a landing in the continental United States, which is a function of the inclination of the return orbit to the earth equatorial plane and the range capability of the vehicle, as discussed in reference 2. This requirement is common to all lunar landing techniques. A number of studies are in progress to determine optimum situations for establishing desired return trajectories, with preliminary results indicating that little or no additional velocity increment is required for injection from lunar orbit when the moon has negative declination and is descending. Other studies are in progress to define optimum injection velocity and injection longitude for minimization of trajectory dispersion at the earth. The velocity increments used in table 1 are conservative in that no reductions due to optimization studies have been included.

9. Midcourse correction.- Common to all missions.

10. and 11. Reentry, touchdown, and recovery.- These procedures will be common to all missions. The lunar orbit rendezvous mission imposes no additional requirements on the reentry techniques or on the atmospheric range requirements than other lunar landing missions.

TABLE I.- CHARACTERISTIC VELOCITY INCREMENTS FOR

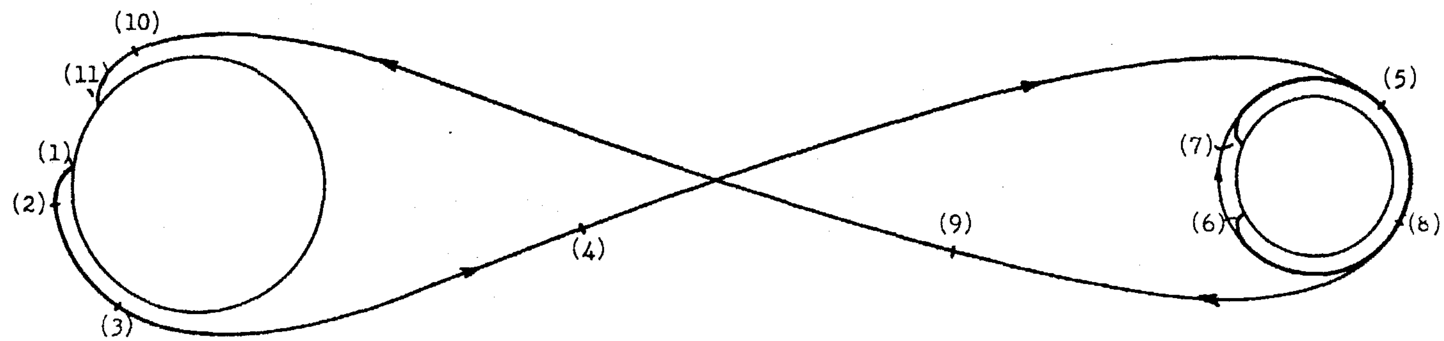
LUNAR ORBIT RENDEZVOUS MISSION

Phase	VΔ, impulsive	ΔV, finite time and piloting allowance
1. Launch from Cape Canaveral	------	------
2. Establishment of earth orbit, or coasting orbit (300-statute-mile altitude)	------	------
3. Injection into earth-moon transfer trajectory[1]	10,183	11,100
4. Midcourse correction		200
5. Establishment of lunar orbit[2]	3,534	3,640
6. Descent from orbit	5,630	6,798
7. Return to orbit	5,630	7,468
8. Injection into moon-earth trajectory[3]	3,175	3,461
9. Midcourse correction		200
10. Reentry	------	------
11. Touchdown and recovery	------	------

[1]Designed for $2\frac{1}{2}$ - day earth to moon flight time.

[2]Includes allowance for 10° change of plane to achieve equatorial lunar orbit.

[3]Includes allowance for 10° change of plane to achieve desirable inclination of return trajectory to earth equatorial plane.



(1) Launch from Cape Canaveral

(2) Establishment of Earth Orbit, or Coasting Orbit

(3) Injection into Earth-Moon Trajectory

(4) Midcourse Correction

(5) Establishment of Lunar Orbit

(6) Descent from Orbit

(7) Return to Orbit

(8) Injection into Moon-Earth Trajectory

(9) Midcourse Correction

(10) Reentry

(11) Touchdown and Recovery

Figure 2.- Mission profile for lunar orbit rendezvous mission.

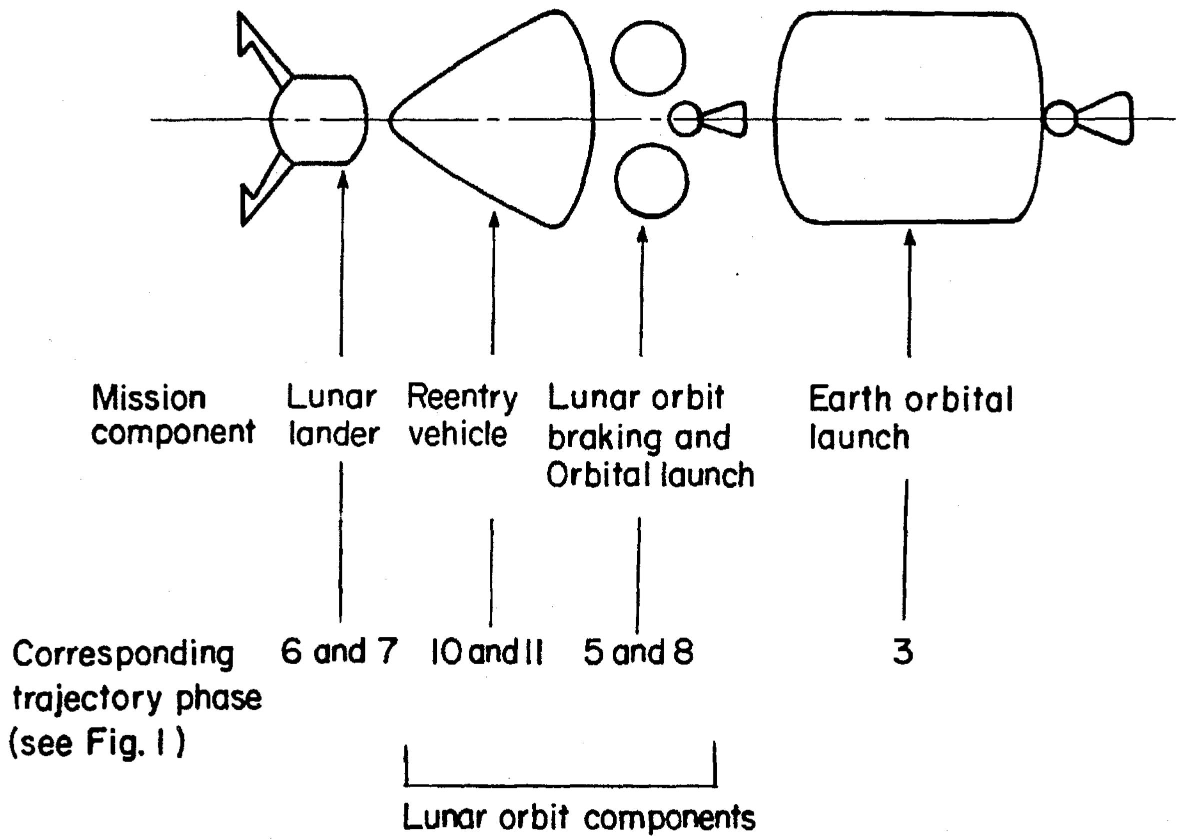


Figure 3.- Schematic of lunar orbit rendezvous mission vehicle.

VEHICLE WEIGHTS AND BOOSTER REQUIREMENTS

As noted in the section "Mission Profiles," earth and lunar orbiting are assumed for all the estimates made herein. The condition called the direct mission is one in which the spacecraft which returns to earth is also the vehicle which lands on the moon. The condition called "lunar lander vehicle" is one in which the lander vehicle separates from the basic vehicle and lands while the basic vehicle remains in moon orbit.

Calculations have been made for the mission profile and velocity increments previously discussed. Four fuels were used in various combinations for these calculations, these fuels are listed in table II along with the basic tank, engine, and associated structural weights. The weight ratios listed are practical values based on experience and projection and were established as ground rules for these studies. For all cases considered, staging after launch from earth orbit (or coasting phase) and on the moon are assumed. Staging here is used to mean the separation from empty tanks, and engines and structure no longer needed. For the lunar lander case this assumption implies that the system used to brake into lunar orbit and to launch from orbit back to earth are the same. The fuels used for the various conditions considered are shown on table III.

Consideration of boiloff and insulation for cryogenic fuels was made by making general calculations of mass ratios and fuel requirements to establish approximate fuel quantities required for the missions. The insulation weight and boiloff has been calculated for typical tanks under the environmental condition typical for the moon landing mission. These are shown on figures 4 and 5. These results were applied to the approximate fuel quantities required and conservative values of insulation and boiloff weight ratios were established. These are shown in table IV. The values listed in table IV as "used" were used in a general recalculation of the mission weight ratios. The values for the insulation were applied to the tank, engines, and associated structural weight ratios. The values for boiloff were used effectively to establish new characteristic velocity increments for the cases where applicable.

Three landing vehicles were considered for these calculations. These are given in table V. The basic weight breakdown of these vehicles without fuel and tankage is given in table V; the basic weights are 1,270, 2,234, and 3,957 pounds. Weights prior to descent to the lunar surface are dependent on the fuel used and are given in table VI; the lander weights range from 4,100 pounds for a one-man "shoestring" machine using H/F to 24,600 pounds for a two-man "plush" machine using solid propellants. These machines were considered with each of the three basic earth return vehicle weights established by the ground rules of this study to determine total weights in earth orbit prior to launch to the moon and in addition the corresponding weights launched to the moon. These weights are listed in tables VII and VIII for the various fuel combinations listed in table III.

In addition, calculations of earth orbital and orbital launched weights were made for four direct lunar missions using H/F, H/O, solid, and storable propellants throughout (with exception of the earth orbital launch phase) as indicated by footnotes in table III. These results are given in tables IX and X along with the results for the corresponding lunar orbit rendezvous missions. The substantial advantage of the lunar orbit rendezvous mission over the direct lunar mission in terms of reduced earth orbit weight required is readily apparent in table IX.

It is readily apparent on examination of tables VII and VIII that many of the missions considered may be accomplished by a direct boost with a C-3 vehicle so that no earth rendezvous is necessary. Many others involving heavier landers and lower specific impulse fuels may be accomplished by a direct C-4 launch. The use of orbital refueling operations and assembly will enable the accomplishment of many of the missions with only C-1 boost capability. The specific numbers of earth launch boosters required to accomplish the various missions is not considered here because of classification.

A particularly interesting combination involves the use of two small lander vehicles. This combination has a rescue capability not possessed by direct or other forms of lunar landing missions. If the first lander vehicle is damaged on landing, the second vehicle can effect a rescue and return to the orbiting "mother" vehicle in a matter of hours. The earth orbit weights for this mission with H/F propulsion are within C-3 boost capability. Such a mission would not necessitate earth orbit rendezvous.

TABLE II.- FUELS CONSIDERED

Fuel	Specific impulse	Tanks, engines, and associated structural weight ratio (a)
Hydrogen/fluorine	440	[b]0.10
Hydrogen/oxygen	425	[b].10
N_2O_4/UDMH	315	.08
Solid propellant	290	.12

[a]Ratio of weight of tanks, engine, and associated structure to fuel weight.

[b]Insulation for cryogenics not included in these values.

TABLE III.- FUEL ARRANGEMENTS CONSIDERED

Case	Earth orbital launch	Lunar braking	Lunar landing	Lunar take-off	Lunar orbital launch
[a]1	H/O	H/F	H/F	H/F	H/F
2	H/O	H/F	H/O	H/O	H/F
3	H/O	H/F	Storable	Storable	H/F
4	H/O	H/F	Solid	Solid	H/F
[a]5	H/O	H/O	H/O	H/O	H/O
6	H/O	H/O	Storable	Storable	H/O
7	H/O	H/O	Solid	Solid	H/O
[a]8	H/O	Storable	Storable	Storable	Storable
9	H/O	Storable	Solid	Solid	Storable
[a]10	H/O	Solid	Solid	Solid	Solid

[a]Comparative calculations were made for the direct mission in these cases.

TABLE IV.- INSULATION AND BOILOFF WEIGHTS

(Percent of fuel weight)

Fuel and tank	Insulation		Boiloff	
	Range	Used	Range	Used
Lander case				
Earth orbital launch	0.82 to 0.98	1.0	Negligible	
Lunar orbiting and orbital launch	1.9 to 2.0	2.0	0.4 to 0.46	0.5
Lunar deorbit and landing	1.1 to 2.1	2.0	0.1 to 0.17	0.2
Lunar take-off and rendezvous	1.6 to 2.0	2.0	0.3 to 0.5	0.5
Direct case				
Earth orbital launch	0.7 to 0.85	1.0	Negligible	
Lunar orbiting, deorbit, and landing	1.0 to 1.3	1.5	0.07 to 0.10	0.1
Lunar take-off, rendezvous, and orbital launch	1.6 to 2.0	2.0	0.3 to 0.5	0.5

TABLE V.- WEIGHT BREAKDOWN OF LANDERS WITHOUT FUEL TANKS

Item	Shoestring (a)	Economy (b)	Plush (c)
Navigation and guidance	130	239	239
Communications	70	70	70
Power	225	280	370
Life support	100	200	668
Men and space suits	200	400	400
Reaction controls and fuel	100	200	250
Basic thrust control	50	100	125
Attaching gear	15	25	35
Structure or enclosure	150	250	900
Landing gear	50	70	100
Scientific payload	50	100	150
Specimens and lunar samples	50	100	150
Contingency	80	200	500
Totals, pounds	1,270	2,234	3,957

[a]Shoestring - 1 man, down and up.

[b]Economy - 2 men, 24-hour mission.

[c]Plush - 2 men, 7-day mission.

TABLE VI.- LANDER WEIGHTS

Configuration	Basic weight	Weight in lunar orbit			
		H/F	H/O	Storable	Solid
Shoestring	1,270	4,100	4,200	6,200	7,900
Economy	2,234	7,250	7,500	10,800	13,900
Plush	3,957	12,750	13,300	19,100	24,600

TABLE VII.- WEIGHT IN EARTH ORBIT

Case	Lunar braking and lunar orbital launch fuel	Lunar landing and lunar take-off fuel	For 8,500-lb return vehicle				For 11,000-lb return vehicle				For 12,500-lb return vehicle			
			Shoestring lander	Economy lander	Two shoestring landers	Plush lander	Shoestring lander	Economy lander	Two shoestring landers	Plush lander	Shoestring lander	Economy lander	Two shoestring landers	Plush lander
1	H/F	H/F	56,400	67,900	71,500	88,100	68,600	80,100	83,700	100,400	76,000	87,500	91,000	107,600
2	H/F	H/O	56,900	68,900	72,300	90,000	69,200	81,100	84,500	102,200	76,400	88,500	91,700	109,600
3	H/F	Storable	64,200	80,900	86,800	111,300	76,400	93,100	99,100	123,500	83,700	100,500	106,300	130,700
4	H/F	Solid	70,500	92,300	99,100	131,300	82,700	104,500	111,400	143,500	89,900	111,800	118,700	150,800
5	H/O	H/O	58,500	70,700	74,100	92,300	71,000	83,200	86,600	104,800	78,600	90,800	94,200	112,400
6	H/O	Storable	65,800	82,900	88,900	113,600	78,500	95,400	101,400	126,400	86,100	103,100	109,000	133,900
7	H/O	Solid	72,300	94,400	101,400	134,200	84,800	107,100	114,100	146,600	92,300	114,600	121,600	154,300
8	Storable	Storable	76,700	95,400	101,900	129,200	92,000	110,800	117,300	144,300	101,200	119,900	126,400	153,500
9	Storable	Solid	83,700	108,200	115,700	151,600	99,100	123,200	131,100	166,700	108,100	132,400	140,100	175,800
10	Solid	Solid	94,400	120,900	129,500	168,200	111,800	138,600	146,900	185,900	122,400	149,000	157,300	196,300

TABLE VIII.- WEIGHT LAUNCHED TO MOON

Case	Lunar braking and lunar orbital launch fuel	Lunar landing and lunar take-off fuel	For 8,500-lb return vehicle				For 11,000-lb return vehicle				For 12,500-lb return vehicle			
			Shoestring lander	Economy lander	Two shoestring landers	Plush lander	Shoestring lander	Economy lander	Two shoestring landers	Plush lander	Shoestring lander	Economy lander	Two shoestring landers	Plush lander
1	H/F	H/F	21,700	26,100	27,500	33,900	26,400	30,800	32,200	38,600	29,200	33,700	35,000	41,400
2	H/F	H/O	21,900	26,500	27,800	34,600	26,600	31,200	32,500	39,300	29,400	34,000	35,300	42,100
3	H/F	Storable	24,700	31,100	33,400	42,800	29,400	35,800	38,100	47,500	32,200	38,600	40,900	50,300
4	H/F	Solid	27,100	35,500	38,100	50,500	31,800	40,200	42,800	55,200	34,600	43,000	45,700	58,000
5	H/O	H/O	22,500	27,200	28,500	35,500	27,300	32,000	33,300	40,300	30,200	35,000	36,200	43,200
6	H/O	Storable	25,300	31,900	34,200	43,700	30,200	36,700	39,000	48,600	33,100	39,600	41,900	51,500
7	H/O	Solid	27,800	36,300	39,000	51,600	32,600	41,200	43,900	56,400	35,500	44,100	46,800	59,300
8	Storable	Storable	29,500	36,700	39,200	49,700	35,400	42,600	45,100	55,500	38,900	46,100	48,600	59,000
9	Storable	Solid	32,200	41,600	44,500	58,300	38,100	47,400	50,400	64,100	41,600	50,900	53,900	67,600
10	Solid	Solid	36,300	46,500	49,800	64,700	43,000	53,300	56,500	71,500	47,100	57,300	60,500	75,500

TABLE IX.- COMPARISON OF WEIGHTS IN EARTH ORBIT FOR SEVERAL LUNAR ORBIT RENDEZVOUS AND DIRECT LUNAR MISSIONS (11,000-POUND RETURN VEHICLE, ECONOMY LANDER)

Case	Earth orbital launch	Braking	Landing	Take-off	Lunar orbital launch	Weight in earth orbit, lb	
						Lunar orbit rendezvous	Direct
1	H/O	H/F	H/F	H/F	H/F	79,300	180,624
5	H/O	H/O	H/O	H/O	H/O	82,200	193,322
8	H/O	Storable	Storable	Storable	Storable	108,000	338,461
10	H/O	Solid	Solid	Solid	Solid	134,000	533,981

TABLE X.- COMPARISON OF WEIGHTS INJECTED TO MOON FOR SEVERAL LUNAR ORBIT RENDEZVOUS AND DIRECT LUNAR MISSIONS (11,000-POUND RETURN VEHICLE, ECONOMY LANDER)

Case	Earth orbital launch	Braking	Landing	Take-off	Lunar orbital launch	Weight injected, lb	
						Lunar orbit rendezvous	Direct
1	H/O	H/F	H/F	H/F	H/F	30,500	69,233
5	H/O	H/O	H/O	H/O	H/O	31,600	74,100
8	H/O	Storable	Storable	Storable	Storable	41,700	129,732
10	H/O	Solid	Solid	Solid	Solid	51,500	204,675

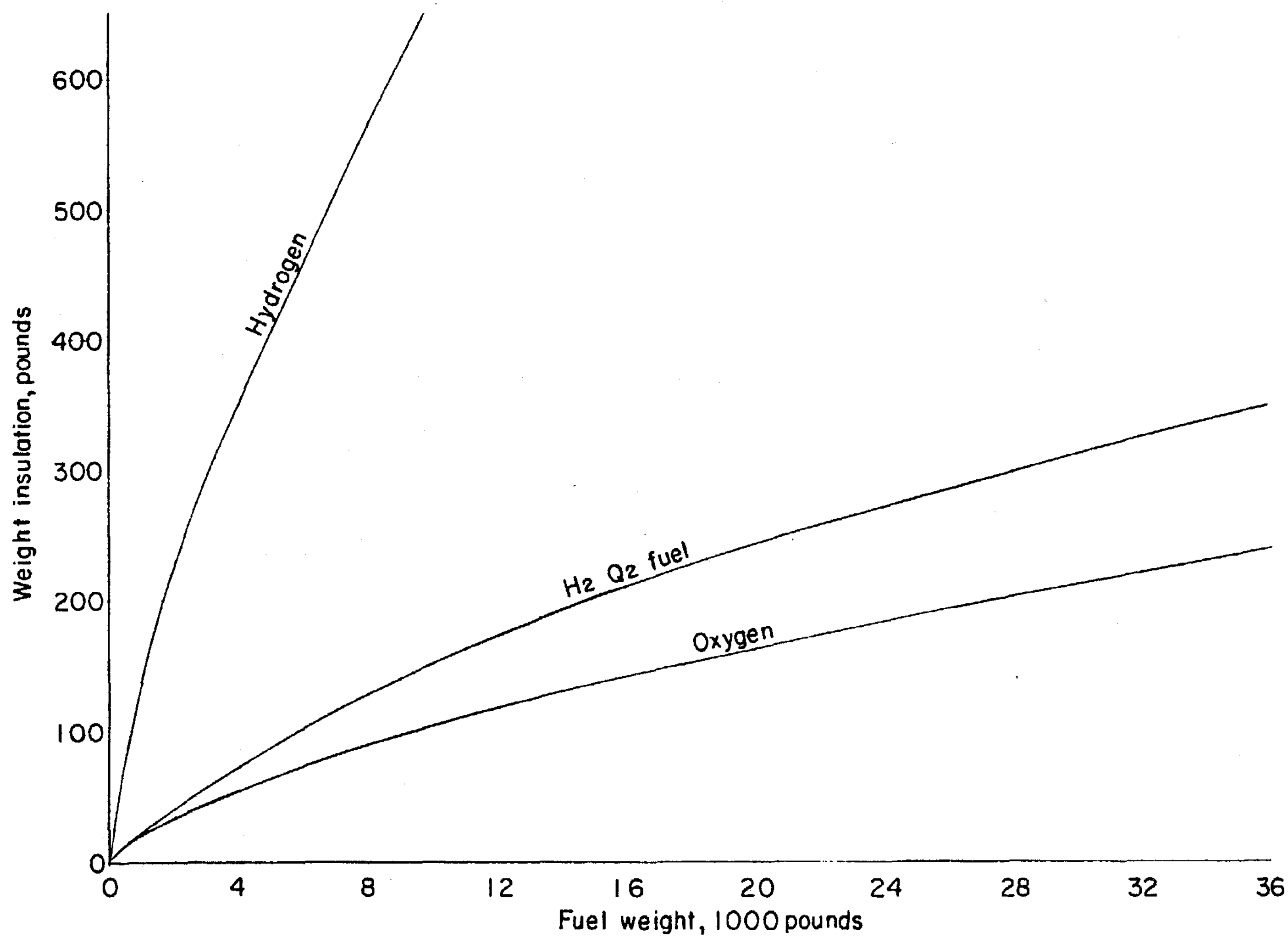


Figure 4.- Weight of 2 inches of reflective insulation as a function of fuel weight.

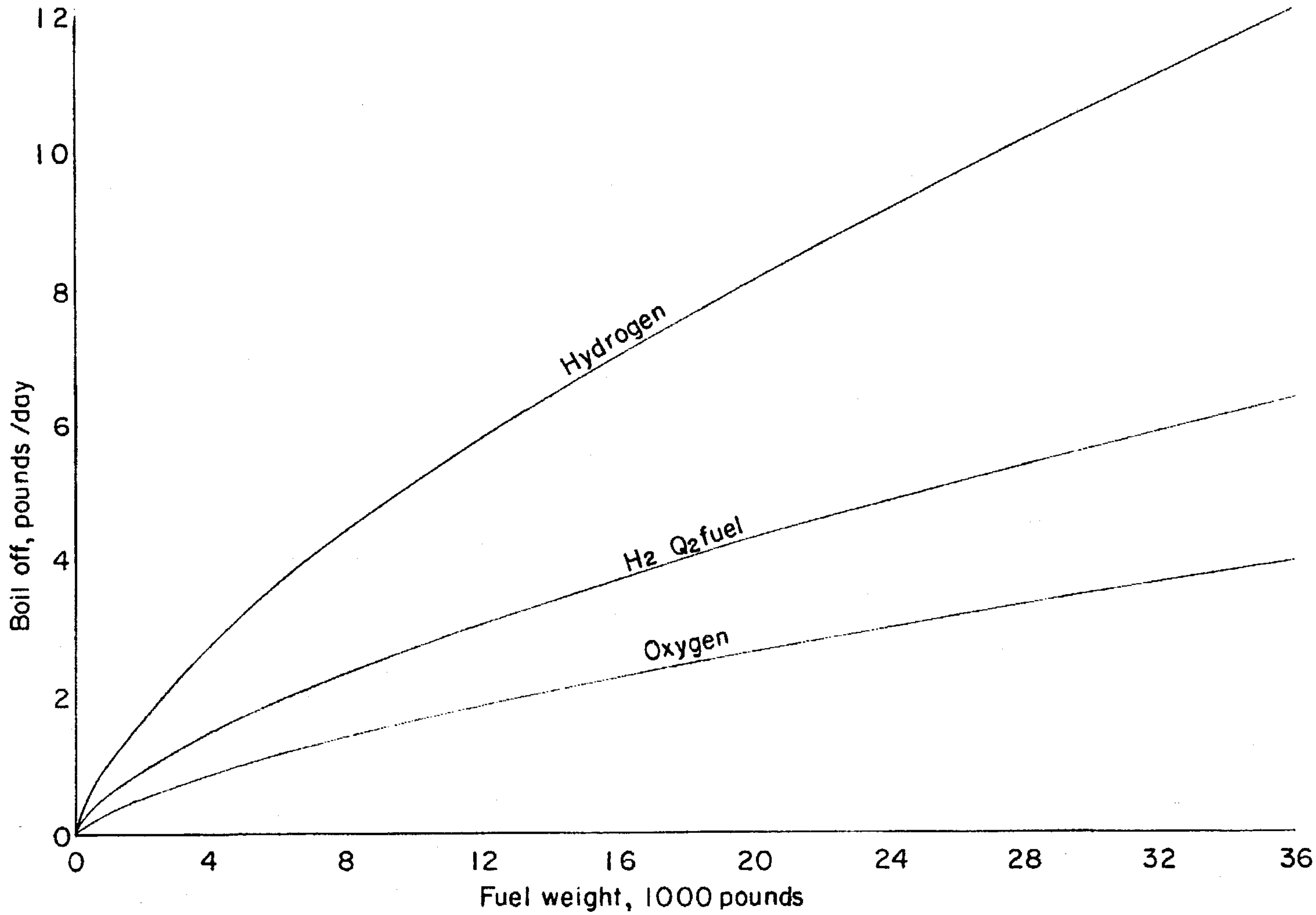


Figure 5.- Fuel boiloff in near earth orbit for tanks with 2 inches of reflective insulation.

PROGRAM PLAN AND SCHEDULING

In planning its mission approach, the Lunar Rendezvous group gave careful consideration to the desirability of accomplishing a manned lunar landing at the earliest practicable date. This study group has always felt, for reasons which will be discussed later in this section and also elsewhere in the report, that the lunar-rendezvous approach is the most logical means of reaching this objective.

Pacing items common to all schemes for lunar landing are development of large boosters (C-1, C-3 or C-4, Nova) and construction of launch facilities. A particularly critical factor is the C-3 development program. The Heaton group emphasizes this item by noting that both the Fleming and Heaton dates for mission accomplishment are liable to slip 5 months unless the C-3 program can be accelerated a like amount. Because the large booster schedules already seem to be optimistic, this group decided it would be unrealistic to make use of the vehicle-availability ground rule (perfect vehicles in perfect supply) in this study. Therefore, this group has used throughout the same basic development schedules used by both the Fleming and Heaton studies. Because of this, it was considered unnecessary to conduct a PERT analysis of this mission. The group is confident that such a study would not disclose any major conflicts.

The resulting mission flight plan is presented in table XI. It will be noted that a starting date of July 1961 has been assumed in order to permit direct comparison to be made with the Fleming and Heaton plans. On this basis, the lunar-rendezvous method calls for a first lunar landing in March 1966 with the possibility of an attempt as early as November 1965. This date is the same as that of Heaton Mission C, 4 months ahead of Heaton Missions A and B, and 17 months ahead of the Fleming Mission. This study emphasizes again the time to be gained through use of rendezvous, whether at the earth (Heaton studies) or at the moon.

At first glance, little difference in schedule appears evident between the earth-rendezvous and the lunar-rendezvous methods. However, hidden differences advantageous to lunar rendezvous exist. These differences are of the kind which tend to provide a margin of confidence in the ability to meet the lunar-rendezvous schedule.

Before discussing these factors, however, it would be appropriate to dispose of that part of the lunar-rendezvous plan (through C-1 development) which represents only minor changes from the Fleming and Heaton plans. To aid in this discussion, a comparison of flight requirements of the lunar-rendezvous method with the Fleming plan and the three Heaton plans has been made and is given in table XII.

In table XII it will be noted that differences exist between the plans in the number of Argo D-8, Thor, Agena Geophysical Observatory, Centaur Surveyor, and Centaur Recoverable Biomedical flight requirements. The reduced number of flights noted for these activities do not imply any deemphasis of the importance of these programs. Rather the missing flights are those which, according to the Fleming and Heaton schedules which this group has adopted, were scheduled to occur after the last of four lunar-rendezvous landings will have been made. The same situation exists with regard to the C-3 Prospector program.

In these support programs particular emphasis should be given to those projects which are designed to determine the lunar-surface conditions at the landing site. In this area this study group feels that present plans are not completely adequate and should be critically reviewed. (This problem is not yet, however, considered to be critical schedulewise.)

This group, in agreement with the Heaton study, recommends a reduction in Atlas 18-orbit missions because of the added Mark II Rendezvous flights.

This group has reduced (to 5 from the Heaton group's 8) the number of Agena Mark II Rendezvous flights in view of the 8 added C-1 Rendezvous flights.

Table XI indicates that four landing attempts have been programmed, reflecting the same degree of conservatism as the Heaton studies. The program presented also permits the capability of attempting a landing on any of the four elliptic or circumlunar flights scheduled to start in November 1965. In effect then, the four lunar landing vehicles shown represent 100-percent spares.

By use of the same basic schedules, the present mission schedule would tend to contain the same amount of optimism or pessimism as the other studies. However, the group feels that the lunar-rendezvous approach contains significant features that tend to generate a higher level of confidence in the schedule presented than do the other studies. As an aid in this discussion, figure 6 has been prepared which shows the interrelationship of major events in the program. The attractive features of lunar rendezvous are as follows:

1. Use of rendezvous.- Use of the rendezvous concept (whether at earth or at moon) permits lunar landing to be accomplished with boosters of the C-3 or C-4 class. This factor avoids the uncertainties of the Nova development program.

2. Rendezvous at moon.- By rendezvousing at the moon the earth take-off weight is reduced to the point where only a single C-3 or C-4 is required. This factor avoids the considerable pressure applied to the C-3 or C-4 facilities development program by eliminating the high firing rate associated with the Heaton earth-rendezvous approach.

The lunar-rendezvous approach separates the reentry function from the lunar landing function. This feature provides the following additional advantages:

3. Simplifies Apollo development.- The Apollo vehicle need not be designed with lunar landing considerations in mind. This factor should ease somewhat the Apollo development.

4. Simplifies lunar landing development.- The lunar landing vehicle can be optimized since it need not be designed under the more restrictive requirements of a vehicle which must also perform reentry.

5. Parallel development possible.- Separation of function into two vehicles permits development of both vehicles with a minimum of schedule conflict.

The lunar-rendezvous approach results in a lunar landing vehicle which is smaller and lighter than other vehicles (where the entire system is landed). The advantages of this factor are as follows:

6. Entire system development accomplished with C-1.- The small size of the lunar lander permits complete rendezvous development and final check-out with Apollo to be done with C-1 rather than requiring C-3 or C-4. Since Apollo development also is accomplished on C-1, this means that no development work will be contingent upon C-3 or C-4 scheduling (about which the Heaton study has expressed concern). It is indicated that the lunar-rendezvous approach could tolerate 4 or 5 months slippage in C-3 and still meet the Heaton Mission A and B first-landing date. (This group considers that there is little likelihood of C-4 meeting the present C-3 schedule so that Heaton Mission C is perhaps unrealistic.)

7. Lunar landing and lunar launch operations development improved.- The Langley Research Center is actively pursuing the design, development, and construction of Lunar Descent and Landing Research Facilities. With "go-ahead" recently received, these facilities will be available in time to do the necessary research and development on these problems and will also be usable for lunar lander check-out and crew training on the actual vehicle for the lunar landing, lunar launch, and the rendezvous operation.

In summary, the lunar-rendezvous approach to a manned lunar landing suggests a first landing attempt in March 1966, with a possibility of making an attempt as early as November 1965. This program is based on NO changes in previously assumed development schedules. Finally, the lunar-rendezvous approach contains a number of features which tend to raise the schedule confidence level. The most important of these are:

A. The lunar-rendezvous approach, because of separation of functions permits essentially separate and parallel development programs of Apollo, lunar lander, and rendezvous operations with a minimum of schedule conflict.

B. The lunar-rendezvous approach permits complete system development to be done with C-1 which will be available and well developed, thus avoiding any uncertainties associated with C-3 or C-4.

TABLE XI.- MASTER FLIGHT PLAN - LUNAR RENDEZVOUS

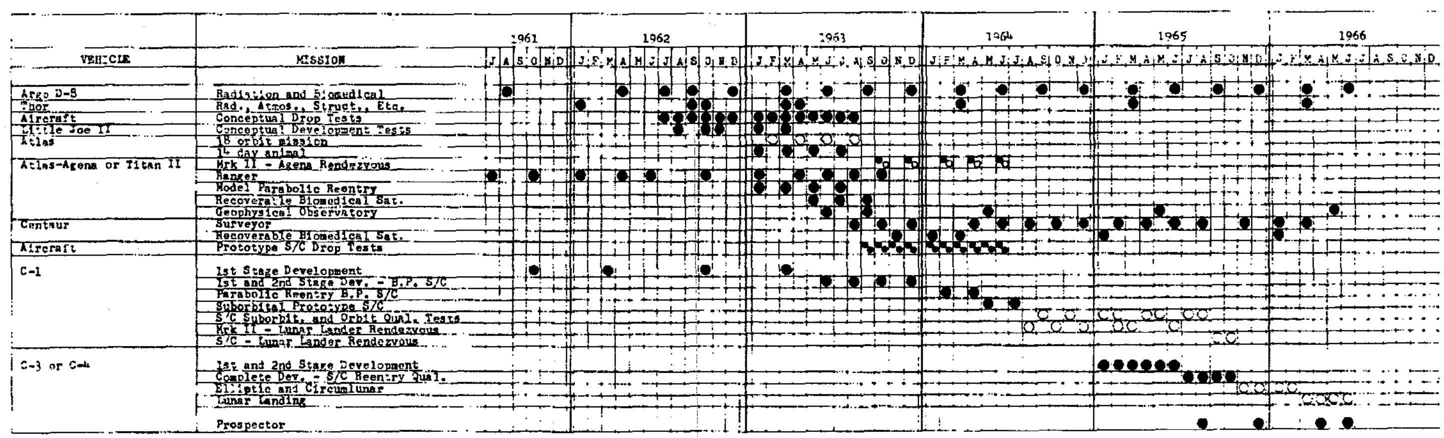


TABLE XII.- COMPARISON OF FLIGHT REQUIREMENTS FOR ACHIEVING MANNED LUNAR LANDING

Vehicle	Mission	Fleming group	Heaton Mission A	Heaton Mission B	Heaton Mission C	Lunar rendezvous
Argo D-8	Radiation and biomedical	25	22	22	22	19
Thor	Radiation, atmosphere, structure, etc.	9	9	9	9	8
Aircraft	Conceptual drop tests	14	14	14	14	14
Little Joe II	Conceptual development tests	5	5	5	5	5
Atlas	18-orbit	8	4	4	4	4
	14-day animal	4	4	4	4	4
Agena	Mark II rendezvous		16	16	16	10
	Ranger	11	11	11	11	11
	Model parabolic reentry	4	4	4	4	4
	Recoverable biomedical	3	3	3	3	3
	Geophysical observatory	6	5	5	5	5
Centaur	Surveyor	18	18	18	18	15
	Recoverable biomedical	6	6	6	6	5
Aircraft	Prototype S/C drop tests	20	20	20	20	20
C-1	First-stage development	3	4	4	4	4
	First- and second-stage dev. (B.P. S/C)	4	4	4	4	4
	Reentry + Landing and T.O. dev. (Fleming)	11	2	2	2	2
	Suborbital prototype S/C	2	2	2	2	2
	S/C suborbit and orbit qualification	8	8	8	8	8
	Mark II - lunar lander rendezvous					6
	S/C - lunar lander rendezvous					2
C-3 or C-4	First- and second-stage development	6	6	6	6	6
	S/C reentry qualification	4	4	4	4	4
	Elliptic and circumlunar	9	11	11	4	4
	Orbital operations (R-3 or R-4)		32	24	13	
	Prospector	8	7	7	7	4
	Lunar landing		8	8	10	4
Nova	First-stage development	3				
	First- and second-stage development	3				
	Complete development	7				
	Lunar landing	4				
Number of landing attempts programmed		2	4	4	5	4
Percentage spares provided		100	60	100	100	100

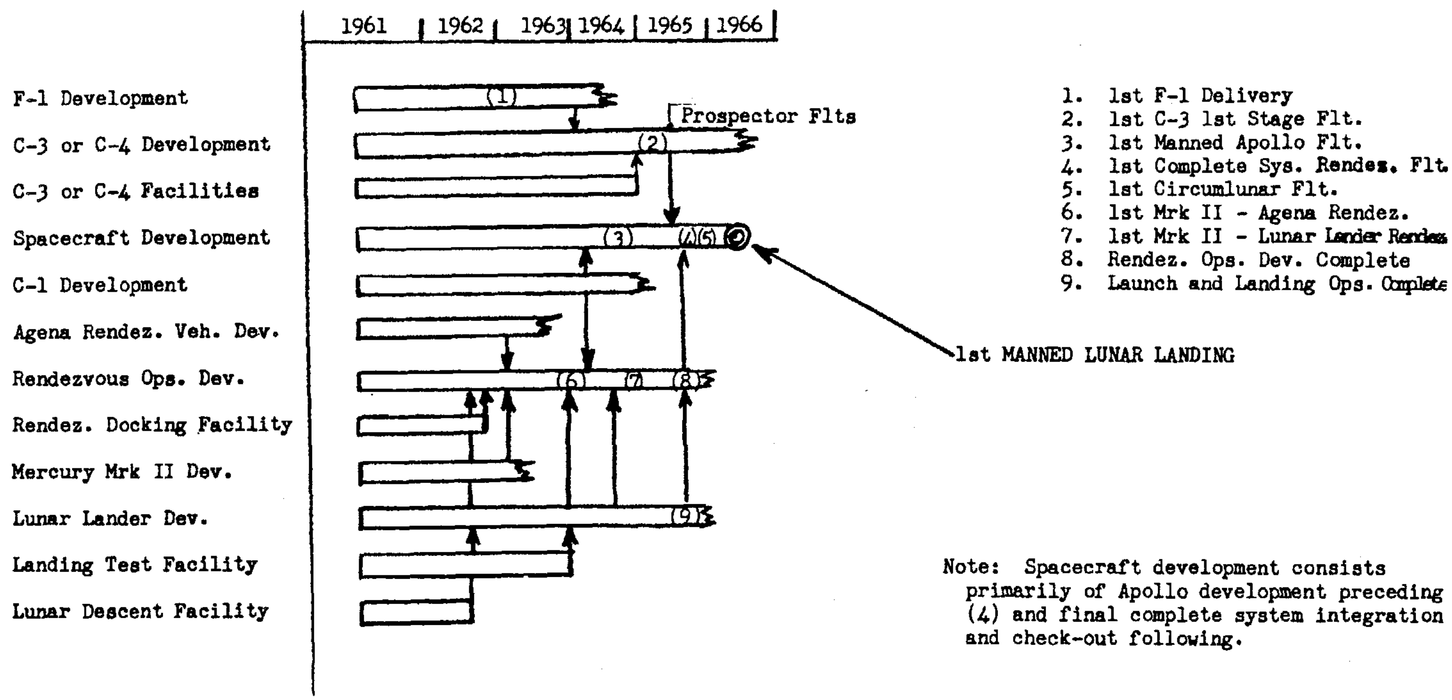


Figure 6.- Interrelationship of major events.

FUNDING

Because of the short time available for this study, it has not been possible to study costs in any detail. Total program funding can, however, be discussed in general terms with the aid of table XIII.

Table XIII summarizes the vehicle requirements of the Fleming, Heaton A, Heaton B, Heaton C, and the Lunar Rendezvous programs. Since development costs would generally be about the same for all methods, it is considered that a good indication of relative total program cost can be obtained by comparing the numbers of vehicles required.

All missions assume 34 aircraft drops.

All missions require roughly the same number of small booster flights. The differences shown on table XIII are due to dropping off the flights indicated in the Fleming and Heaton reports that occur after the lunar landing flights of this report, or to added rendezvous development flights.

Although this mission approach required as many or more C-1 vehicles than the other approaches, it should be noted that it requires less than half as many C-3 vehicles as required by the Heaton missions and even less than required by the Fleming mission. No Nova vehicles are required.

Aside from the fewer total numbers of vehicles of smaller size required, it is extremely significant from a cost standpoint that the most flights are made with the best-developed vehicle. This is contrary to all of the other approaches.

Some cost savings should be realizable because of separation of reentry and lunar landing functions which should ease development problems.

It appears evident that the total program cost of lunar rendezvous should be less than the Heaton costs.

Finally, parallel Fleming-Lunar Rendezvous programs should cost less than parallel Fleming-Heaton programs.

TABLE XIII.- SUMMARY OF VEHICLE REQUIREMENTS

Vehicle	Fleming group	Heaton Mission A	Heaton Mission B	Heaton Mission C	Lunar Rendezvous
Aircraft drops	34	34	34	34	34
Small boosters	99	107	107	107	93
Large booster:					
C-1	28	20	20	20	28
C-3 or C-4	27	68	60	44	22
Nova	17				
Total	72	88	80	64	50

SAFETY AND RELIABILITY

In a manned operation the most important consideration concerns crew safety. A second item of importance is reliability related to the probability of completing a mission. Finally, if an important factor in the program is accomplishment of the mission at the earliest practicable date, then safety and reliability are also very important during the development phase. In all of these areas the lunar rendezvous approach as outlined herein is considered to have significant advantages over other methods so far proposed.

Crew safety.- Crew safety is, of course, of prime importance. In this area this group has followed the basic approach of previous studies of providing the crew with an abort capability. Therefore, the lunar rendezvous method should provide equal safety with other methods from earth launch to lunar orbit. All methods are also equal from lunar orbit to reentry.

Rendezvous at the moon is the single operation which differentiates this approach from other approaches. It must be admitted that this factor tends to reduce safety and reliability (when considered by itself) to some slight degree. The use of lunar rendezvous, however, leads to improvement in the safety and reliability of the landing operation far greater than that lost because of the rendezvous operation, as is discussed later in this section. Further this group believes, based on its studies which are discussed elsewhere, that the lunar rendezvous will itself be a simple, reliable operation. This group further believes that rendezvous at the moon should be easier and more reliable than rendezvous at the earth because of the less stringent circumstances which exist at the moon. Rendezvous at the moon requires that consideration be given to the necessity for providing an abort capability during the landing and return-to-lunar orbit operation. This group has studied this problem by analytical and piloted-simulation means and concludes that a satisfactory abort capability can be provided. (A brief discussion of this problem is given in another section of this report.)

Abort system reliability is a large factor in all methods. Nevertheless, a 100-percent reliable system cannot be guaranteed. Therefore, any system which subjects the crew to situations which tend to increase the probability of abort must be considered to have decreased safety and reliability. From this viewpoint, important factors in comparing the various proposed methods are the number of flights required and the development status of the various boosters used.

Based on the above considerations, the following factors are listed as being advantageous to the lunar rendezvous method:

1. This method requires fewer flights than other methods.

2. This method programs the greatest number of flights on the most highly developed booster (C-1).

3. A single C-3 (or C-4) launch will be safer than a single Nova launch.

4. A single C-3 (or C-4) launch will be safer than multiple C-3 (or C-4) launches in an earth-rendezvous method.

This group believes that whatever approach is taken, the most hazardous operation of the entire mission is likely to be the landing itself. Here we should compare whatever hazards are introduced by rendezvous at the moon with the gains in safety and reliability which this method provides over other methods.

Studies of the lunar-landing problem made by this group and described elsewhere, backed by experience with VTOL aircraft, helicopter and X-15 leads us to the firm conclusion that the safest, most reliable landing system is the one which makes maximum use of pilot capabilities (backed by suitable instrumentation). Such a system is most readily achieved only if the system can be designed and developed without restrictions imposed by other requirements (such as reentry). The lunar rendezvous method is the only method so far advanced which permits the landing vehicle to be essentially a single-purpose machine, optimized to perform a lunar landing.

Also, the lunar rendezvous concept results in a small landing vehicle. This factor makes possible a pilot-training program which is considerably more satisfying than those previously proposed. With the Lunar Landing Research Facility (now under procurement at Langley) the crews can be trained in the landing, lunar launch, and rendezvous operations <u>in the actual vehicle</u> under realistic conditions. It would be exceedingly valuable to provide a training program for all approaches which could match the program suggested by this study. This, however, becomes more and more difficult as the size of the landing vehicle is increased.

Despite the facts that the lunar-rendezvous method provides an optimum lander configuration and a realistic training program, it must be recognized that the first lunar-landing attempt will be a unique and trying experience. In the event that an accident occurs during landing and no return to orbit is possible, the basic vehicle can return to earth with first-hand information on the conditions encountered and the cause of the accident. The probability of such an accident with a lander vehicle is certainly no greater than that for a direct-landing Apollo for which no return to earth is possible at all.

Further, it is evident from this discussion on lander and booster weights that a number of configurations allow two landers to be carried. Such an arrangement would have distinct safety and reliability advantages.

With two vehicles the following procedures will lead to a greater probability of success than is otherwise possible. One lander vehicle is landed with one man. After the landing (successful or otherwise) the second vehicle lands with one man, the third remaining in orbit as before. The two men then return in one vehicle back to orbit. Although the first landing attempt might result in vehicle damage so that a return to orbit would be hazardous, one would not expect the accident to be damaging to the pilot due to the low lunar gravity. This first man down, having experienced a landing and being in radio contact with both the orbiting vehicle and the second lander, could greatly improve the chances of the second lander by providing GCA (ground-controlled approach) assistance.

One final comment is in order. Although this study indicates that the mission can be accomplished with C-3, this group feels that reliability and safety could be considerably improved if C-4 were used. The increased capability of this vehicle would permit a more conservative approach to be taken in design, would permit larger fuel reserves, etc. Finally, this vehicle would upgrade the mission because fewer airstarts are required than with C-3.

Mission accomplishment and development reliability.- Much of the arguments advanced above apply equally well here and will not be repeated. Many of the factors regarding schedule confidence discussed under Mission Approach also apply here. Of particular significance in this method is that all development is done on C-1, our most highly developed vehicle.

DEVELOPMENT PROGRAM

Lunar Lander Development

Development of the Lander should start as soon as possible. Since the Lander can be designed without giving consideration to earth-reentry problems, it is the opinion of this group that the vehicle should be much easier to develop than Apollo so that its development schedule should be easily compatible with Apollo.

A significant feature of the lunar rendezvous approach is the relatively small size of the Lander. This makes it possible to develop the vehicle, check-out hardware, and provide realistic pilot training in both the lunar landing and lunar launch operations through use of the Landing Research Facility proposed. The Lander rendezvous functions are developed and training is performed as stated under Rendezvous Operations Development.

Rendezvous Operation Development

The rendezvous operation is a vital element in this approach to a manned lunar landing. A thorough systematic development program is there required to perfect this technique. Such a program has been provided as indicated in figure 6.

The Rendezvous Docking Research Facility now under procurement at the Langley Research Center will be available about 9 months before the first Mark II Agena flight and about 19 ahead of the first rendezvous flight with the Lunar Lander.

The Mark II Agena flights are designed to confirm in flight the ground-based simulation studies which indicate the ease of manually controlling rendezvous.

The Mark II Lunar Lander flights with C-1 are designed to check out the Lander rendezvous systems and provide additional pilot training.

The Spacecraft (Apollo) Lunar Lander rendezvous flight on C-1 are primarily final integrated system check outs.

It should be noted that C-1 is capable of carrying either Mark II or Apollo plus the Lunar Lander with sufficient fuel for accomplishing several separations and rendezvous operations per flight.

All of the rendezvous development flights with Mark II or Apollo require the pilot to remotely control the rendezvous vehicle to him.

This type of rendezvous is not as simple (because of the telemetry link required) as the type of rendezvous wherein the pilot is in the rendezvous vehicle (which is the case for the proposed Lunar-Rendezvous operation). It is necessary to do this different operation because of safety considerations. However, in being slightly more complex, these experiments would be conservative demonstrations.

Pilot training in rendezvous docking in the actual Lunar Lander vehicle would be accomplished in the Landing Research Facility.

All rendezvous development is carried out on either Atlas or C-1 and is not affected by C-3 scheduling.

The basic concept of rendezvous has been examined in a number of analytical studies. Furthermore, the feasibility of performing a rendezvous under pilot-control has been demonstrated by means of a number of simulation studies conducted at Langley Research Center. These studies have range from all-instrument methods to purely visual techniques and have all indicated that rendezvous under pilot-control should be a simple, reliable operation.

Three specific simulation studies that have been made are described in Part II of this report, and are:

1. Investigation of an all-instrumented pilot-controlled rendezvous

2. Visual control of rendezvous

3. Visual technique for determining rendezvous parameters

Instrumentation Development

In this study only those pieces of equipment which are or will be within the state of the art have been recommended. Some of these items are currently under development. However, developmental effort will be required on such items as the computer and radar altimeter in order to optimize the equipment for the specific requirements.

Development Facilities for Lunar Rendezvous Mission

The lunar mission employing lunar-rendezvous requires generally the same kinds of subsystems check out, environmental test, etc., facilities as are required by the other methods and will not be discussed further.

The booster facilities required are less than required by other methods due to the fewer total number of flights and to avoidance of multiple launches.

The rendezvous operation employed by this concept requires unique facilities. Development and training in the launch, midcourse, and terminal phases of the rendezvous operation can be carried out on relatively simple ground-based equipment of the types shown in figure 7 that has been used previously to demonstrate the feasibility of rendezvous. The docking phase requires special facilities. An analog docking facility is required for preliminary development. Such a facility is now under procurement at Langley. Facilities are also required for research, development, vehicle checkout, and pilot training in rendezvous and docking, lunar-landing, and lunar launch. In the method proposed all these facilities and purposes are included in the Lunar Landing Research Facilities now under procurement at Langley.

Details of the Analog Docking Facility and the Lunar Landing Facilities being procured at Langley and which will fully meet the requirements of the lunar-rendezvous mission are as follows:

Analog docking facility.- A Rendezvous Docking Facility is presently in procurement at Langley. This facility will consist of an overhead support system which provides three translational degrees of freedom for a three-axis gimbal system. The three-axis gimbal system, already available, will provide three degrees of rotational freedom. The entire apparatus will be designed so that it can operate in conjunction with a ground-based mock-up of another space vehicle. The rendezvous maneuver will be controlled through existing analog facilities from either the cockpit in the gimbal system, the ground based mock-up, or a third position simulating assembly of two vehicles visually from a remote capsule.

The facility is designed so that it can be used to obtain answers in the major problem areas of the docking operation such as:

1. The guidance, control, and propulsion requirements of both automatic and piloted systems.

2. The required instruments and visual aids necessary for a pilot to effect the maneuvers.

3. The operational and design requirements for coupling and transfer (personnel and materials) systems.

4. The impact loads developed during coupling and how the loads affect the structural design of the coupling systems and the vehicles.

The versatility of the facility makes it capable of providing limited information in other phases of interest such as:

1. The final touchdown phase of either a manually or automatically controlled lunar soft landing.

2. A manually or automatically controlled launch from the lunar surface to rendezvous with an orbiting satellite.

The Docking Facility has been approved and at the present time the contract specifications are being completed. The over-all facility is scheduled to be completed and in operation by February 1963. The attached photo, figure 8, is an illustrative model of the facility.

Lunar descent and take-off research facility.- A six-degree-of-freedom, fixed-base analog simulator is currently being constructed to determine the ability and efficiency of pilots to control the deorbit and descent phase of the lunar landing, and the return to lunar orbit. This facility consists of a 10-foot radius spherical screen, a cockpit and controls for a pilot, wide-angle optical system, and two models of the moon scaled so that an altitude range from 1,000 miles to 1 mile can be simulated. See figure 9. The optical system scans the models of the moon, with a wide-angle viewing lens, and projects the viewed portion on the 10-foot spherical screen through a wide-angle projection lens. The models move to follow the tracks of the vehicle over the moon. The pilot will use the controls in response to the information obtained from the display of the moon and other instruments in the cockpit to control the flight of the vehicle. The angular motions of the vehicle will be simulated by the appropriate motions of the projection head. Analog and digital computing equipment concurrently available at the Langley Research Center will be used to simulate the vehicle motions and control the visual display.

Lunar landing research facility.- A full-scale research facility to study the problem of a human pilot controlling a lunar landing is presently in a contract negotiation stage. The present schedule calls for a completion date of February 1963. This facility will be capable of handling a vehicle up to 20,000 pounds and will allow six degrees of freedom. Linear motions allowed will be 400 feet lengthwise, 50 feet crosswise, and 200 feet vertically. Initial velocities of 50 ft/sec horizontal and 30 ft/sec vertical will be obtainable. A gimbal system will allow freedom in pitch, roll, and yaw and will attach to an overhead cable supporting 5/6 of the vehicle weight. An overhead support system will be servo-driven to keep the support cable in a true vertical.

An artist's conception of the facility is shown in figure 10. This facility will be used in research in handling problems and piloting techniques for all types of lunar landing vehicles. It will also permit check-out of the final vehicle and provide pilot training in lunar landing, lunar launch, and rendezvous.

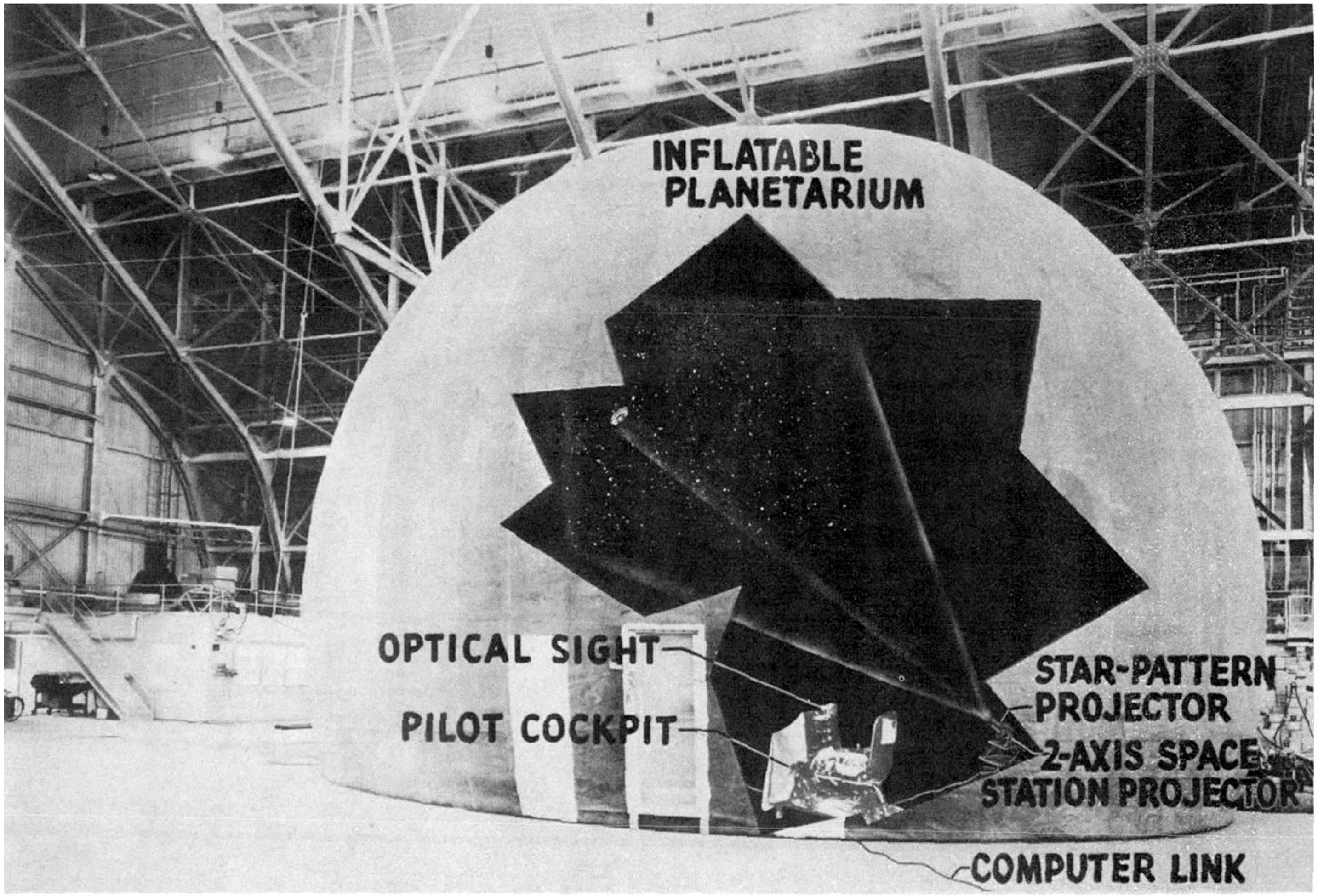


Figure 7.- Pilot-controlled lunar take-off simulation. L-1292

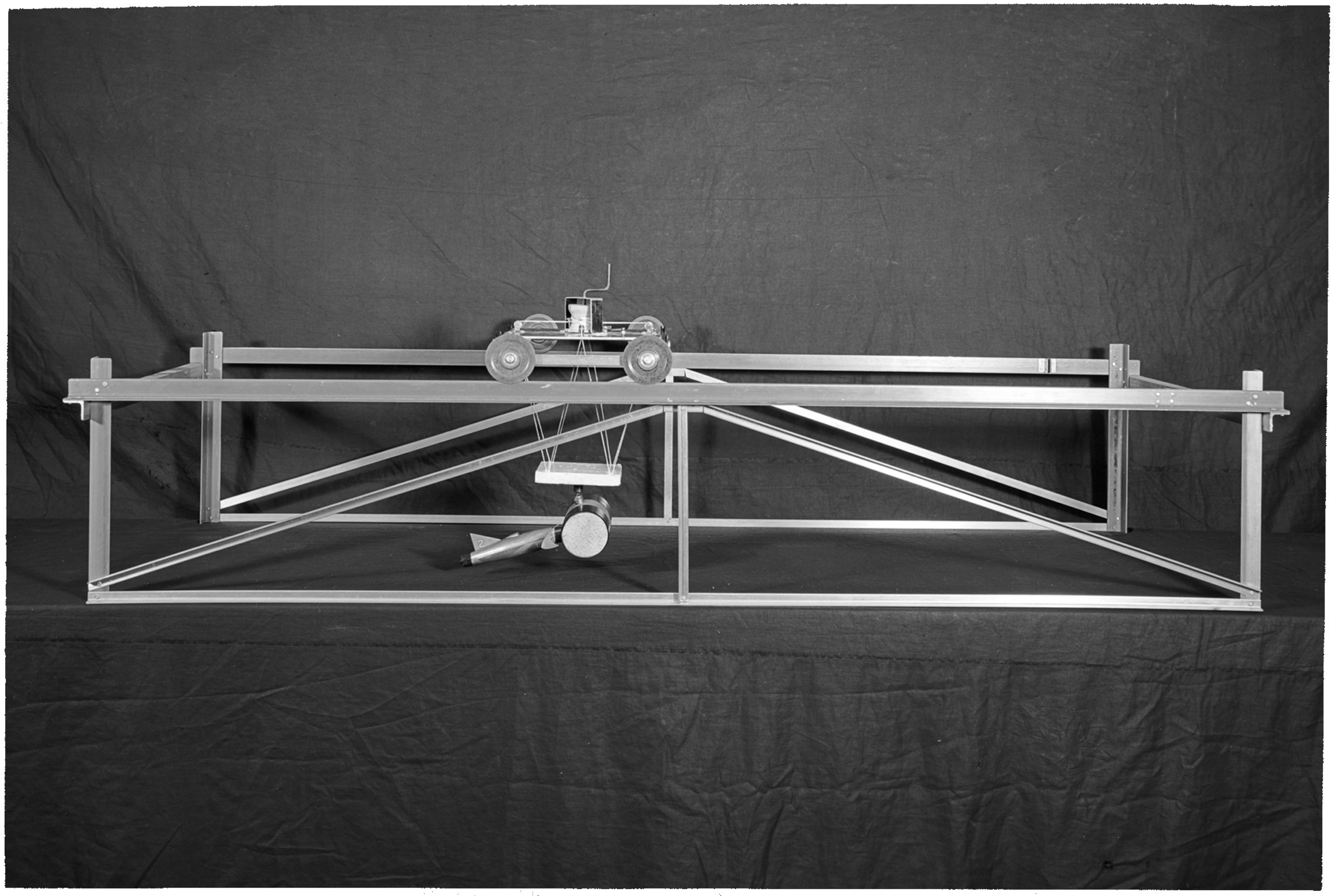


Figure 8.- Docking facility conceptual model. L-60-7882

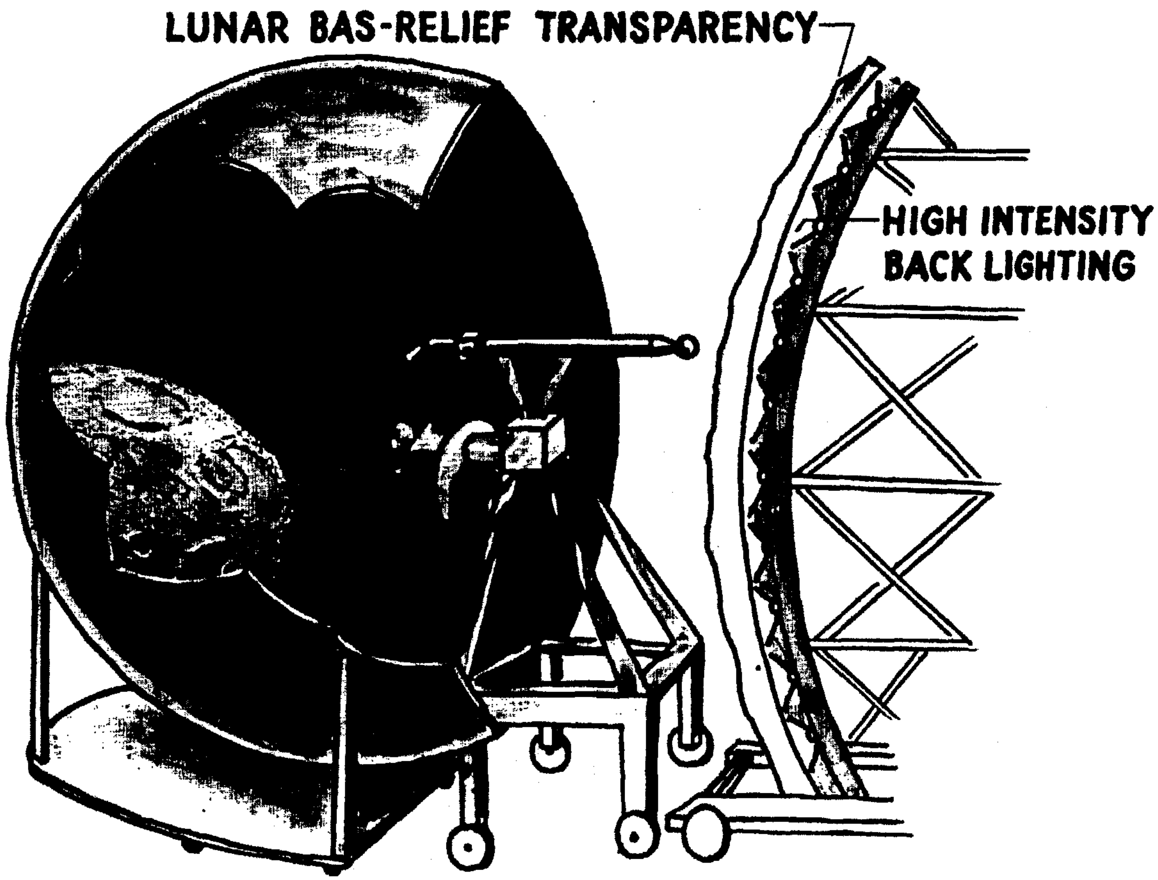


Figure 9.- Lunar-letdown visual simulator. L-1296

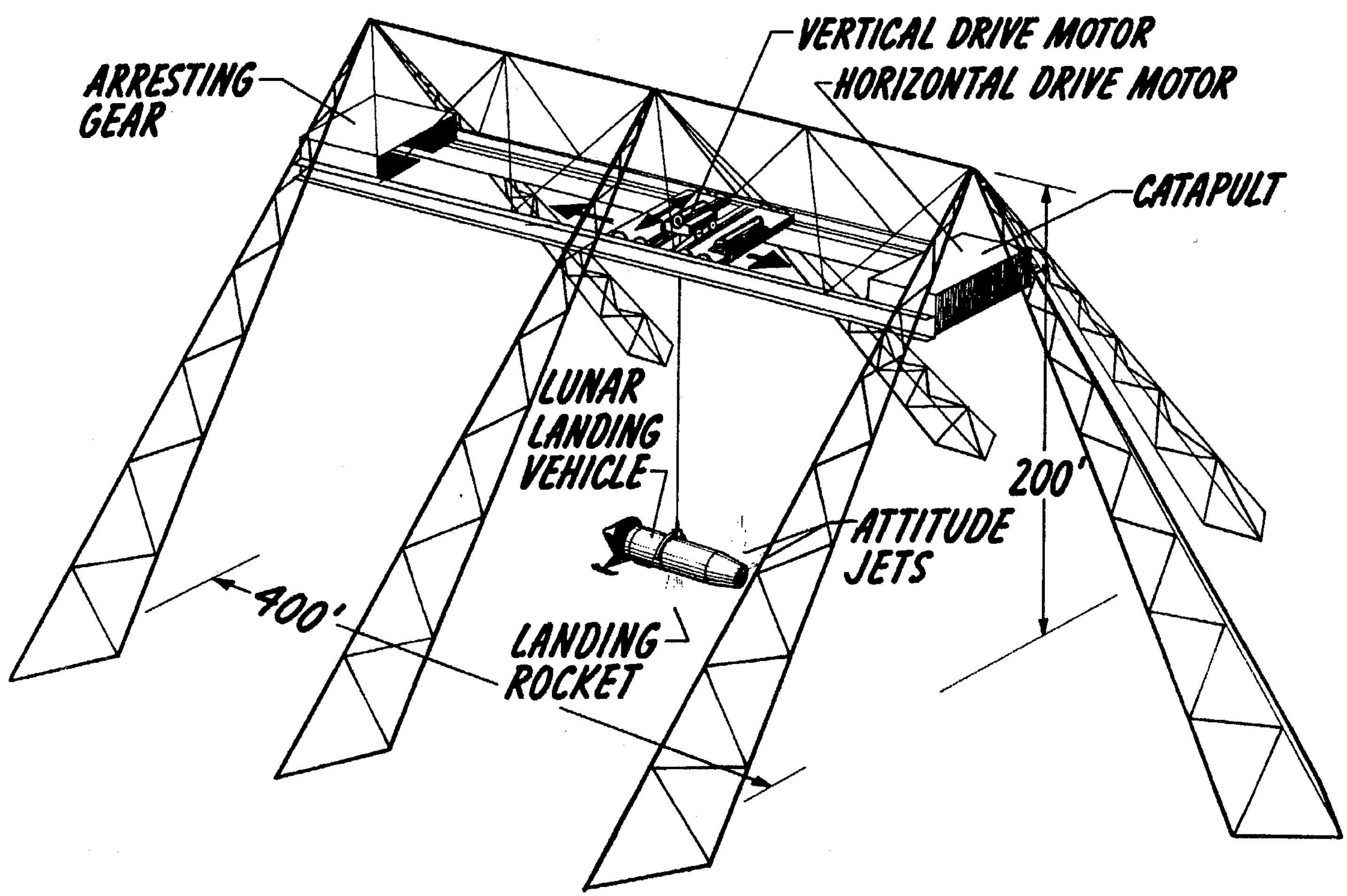


Figure 10.- Lunar-landing research facility.

PART II

TECHNICAL ASPECTS OF THE MISSION

GUIDANCE INTO A LUNAR ORBIT

After appropriate midcourse corrections the space vehicle will be on a hyperbolic trajectory approaching the moon. Basically the problem of establishing a close circular orbit consists of application of retro-thrust in such a manner as to make the point of closest approach occur at a selected altitude above the lunar surface, and simultaneously modifying the velocity to the magnitude and direction required for a circular orbit. Presuming that the vehicle is in the proper orbit plane, the parameters which determine the orbit characteristics are altitude, radial velocity, and circumferential velocity. The desired circular orbit conditions are zero radial velocity, some selected altitude, and a circumferential velocity equal to circular orbital velocity at that altitude. These three parameters therefore must be measured quite carefully and displayed to the pilot, so that he can apply any required corrections through his control of thrust and vehicle orientation. Various schemes for measuring these quantities were considered, and the following scheme is proposed as being sufficiently accurate and reliable for the mission. See instrumentation for details. During the approach to the moon the local vertical to the moon is determined by use of a horizon seeker (IR or conical-scan radar). An inertial table is then erected with one axis along the local vertical, a second axis in the plane of the trajectory, and the third axis normal to the trajectory plane. This could be accomplished using appropriate stellar reference. The inertial table is then slaved to the local vertical. The radial velocity component can then be obtained by integration of the radial acceleration. The altitude above the lunar surface can be obtained initially by optical measurement of the lunar disk, and then by integration of the radial velocity component. The circumferential velocity is obtained from the torqueing required to slave the inertial table to the local vertical and from the radial distance to the center of the moon. Close to the moon, it is proposed to obtain the altitude and the velocity components from doppler radar rather than through the inertial table. The table could still be used as a backup system.

In order for a pilot to control the orbit characteristics, it is necessary that he be presented a display indicating the vehicle orientation and altitude rates in addition to the three orbit parameters already mentioned. It is then up to the pilot to apply correct thrust to establish an orbit. If one assumes that the pilot is attempting to establish a 50-mile-altitude circular orbit, and that he terminates thrust at an altitude of about 50 statute miles, table XIV indicates the sensitivity of the orbit to various errors at thrust termination. The system proposed for measuring the various quantities required for orbit determination (see Instrumentation) has the accuracies shown in table XV for the indicated range of the quantities involved.

Comparison of the figures of tables XIV and XV shows that if thrust termination occurs at about 50-mile altitude with the probable errors in radial and circumferential velocity, the error in pericynthion altitude will be about 12 miles. It appears then, that a 50-mile-altitude orbit is a safe nominal orbit to establish with the proposed sensor equipment.

The ability of a pilot to acquire a predetermined orbit depends on three primary systems; the sensors to determine orbit parameters and vehicle orientation, panel display, and the control system. A fixed-base analog study to determine the capability of pilots to establish a circular orbit about the moon was reported in reference 5. The scope and some of the results are summarized in the following paragraphs.

It was assumed that a manned space vehicle was approaching the lunar surface on a hyperbolic trajectory which would have a point of closest approach at an altitude of about 56 miles and a velocity at that point of 8,500 feet per second. The pilot's task was to establish a circular orbit at a 50-mile altitude. The pilot was given control of the thrust (along the vehicle longitudinal axis) and torques about all three body axes. A sketch of the approach ballistic trajectory and the controlled path of the vehicle are presented as figure 11.

The information display given to the pilot was a hodograph of the vehicle rate of descent and circumferential velocity, an altimeter, and vehicle attitude and rate meters (fig. 12). The general procedure used in the investigation was to permit the pilots to become familiar with the instrumentation, controls, and indicated vehicle dynamics by flying a simple "nominal" trajectory for which the operating mode was specified. This trajectory had a miss distance of 294,000 feet and a velocity of 8,466 feet per second at this point. The trajectory could be modified to result in a circular lunar orbit by applying a constant thrust in the plane of the velocity vector and normal to the local vertical. The required thrust level resulted in an initial deceleration of 0.26 earth "g" and had to be applied when the altitude above the lunar surface was 383,700 feet, and terminated at an altitude of 264,000 feet.

The "nominal" trajectory which the pilots were asked to fly is shown on figure 13. The ballistic portion is the uncontrolled, hyperbolic approach trajectory. The controlled portion is the portion in which retro-thrust is applied to establish the circular orbit. The pilots had no difficulty flying this nominal trajectory, as is indicated in figure 14 and used only about 2 percent of the initial fuel weight more than the computed fuel weight to accomplish the task perfectly. This result was to be expected since the piloting procedure was specified for the nominal trajectory.

On an actual lunar mission, it would be expected that the space ship would be injected toward the moon on a particular nominal trajectory such as that of the preceding paragraph. However, because of injection errors, and small residual errors after midcourse corrections, the trajectory approaching the moon would be somewhat different from the desired nominal trajectory. At the time that the investigation reported in reference 5 was being conducted, the probable corridor at the moon due to the various errors was estimated to be about ±20 miles. On this basis, the initial conditions for the problem included altitude variations of ±100,000 feet and velocity variations (in radial and circumferential velocity) of ±200 feet per second. The range of initial conditions and combinations of miss distance and velocity for the various trajectories is shown in table XVI.

The results of the investigation showed that the pilots soon became adept at flying the simulator, and could manage the "off-nominal" trajectories with little or no difficulty (see fig. 15, for example). The indicated fuel consumption generally was about 1 to 3 percent of the initial vehicle mass more than that required by use of a two-impulse Hohman maneuver. Since publication of these results, a number of sources have indicated that the accuracy of reaching a specific point in space relative to the moon after suitable midcourse correction is within ±2 miles in position and 2 feet per second in velocity. The results of reference 5, therefore appear to be conservative.

TABLE XIV.- SENSITIVITY OF ORBIT TO VARIOUS ERRORS AT THRUST TERMINATION

Type of error	Effect
1 fps in final speed	1 mile in pericynthion
1 mile in altitude	3 miles in pericynthion
$1/10^{\circ}$ in flight-path angle	3 miles in pericynthion
1/2 in retro-rocket angle	6 miles in pericynthion
$1/10^{\circ}$ in heading angle	$-1/6^{\circ}$ orbital inclination
$1/2^{\circ}$ in retro-rocket heading	$-1/3^{\circ}$ orbit inclination

TABLE XV.- ACCURACIES OF PROPOSED ORBIT DETERMINATION SYSTEM

Quantity	Maximum value	Accuracy
Altitude	60 miles	±1 mile
Altitude rate	2,000 fps	±7 fps
Circumferential velocity	8,500 fps	±10 fps
Pitch altitude		$\pm .05^{\circ}$
Bank angle		$\pm .05^{\circ}$
Azimuth angle		$\pm .10^{\circ}$

TABLE XVI.- RANGE OF INITIAL CONDITIONS AND COMBINATIONS OF MISS DISTANCE AND VELOCITY FOR THE VARIOUS TRAJECTORIES

Initial conditions			Pericynthion	
Radial velocity, fps	Circumferential velocity, fps	Altitude, ft	Altitude, ft	Circumferential velocity, fps
[a]-2,867	7,575	1,000,000	294,800	8,466
-2,867	7,775	1,000,000	342,600	8,617
-2,867	7,375	1,000,000	238,600	8,320
-3,067	7,575	1,000,000	213,700	8,582
-3,067	7,775	1,000,000	267,800	8,729
-3,067	7,375	1,000,000	153,100	8,442
-2,667	7,575	1,000,000	374,100	8,355
-2,667	7,775	1,000,000	419,600	8,512
-2,667	7,375	1,000,000	322,800	8,203
-2,867	7,575	900,000	198,900	8,475
-2,867	7,575	1,100,000	390,500	8,457

[a]Nominal trajectory.

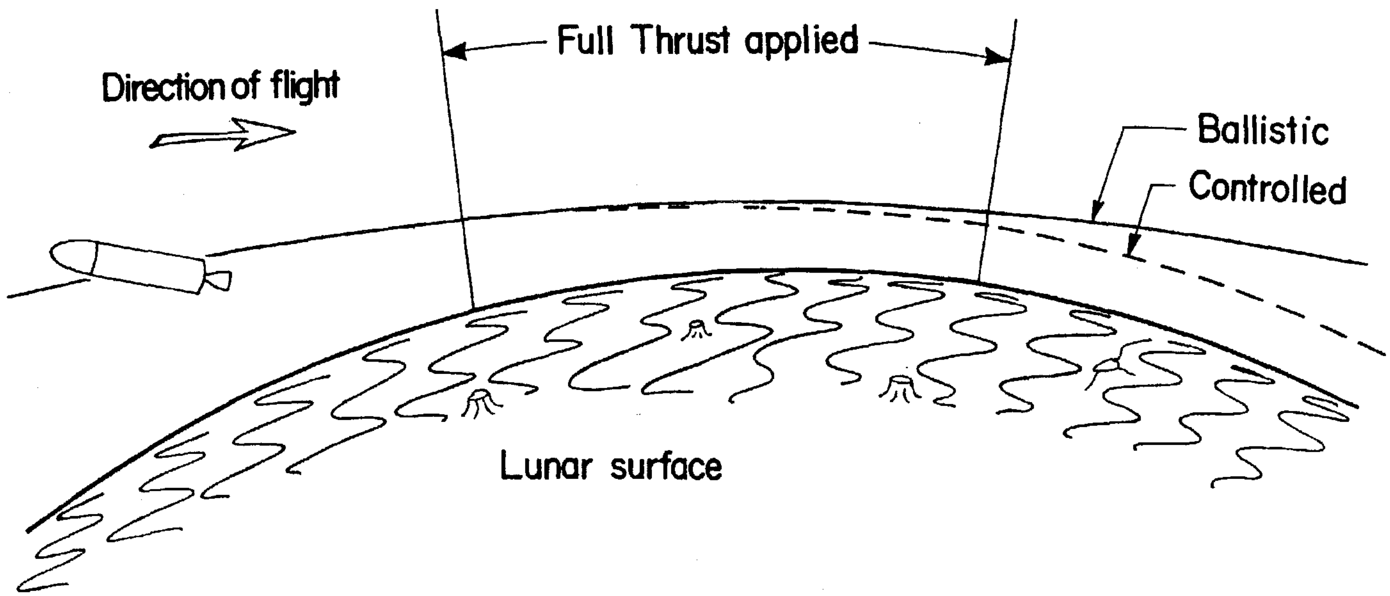


Figure 11.- Ballistic and controlled flight paths.

Figure 12.- Cockpit used for simulation study. L-60-7360

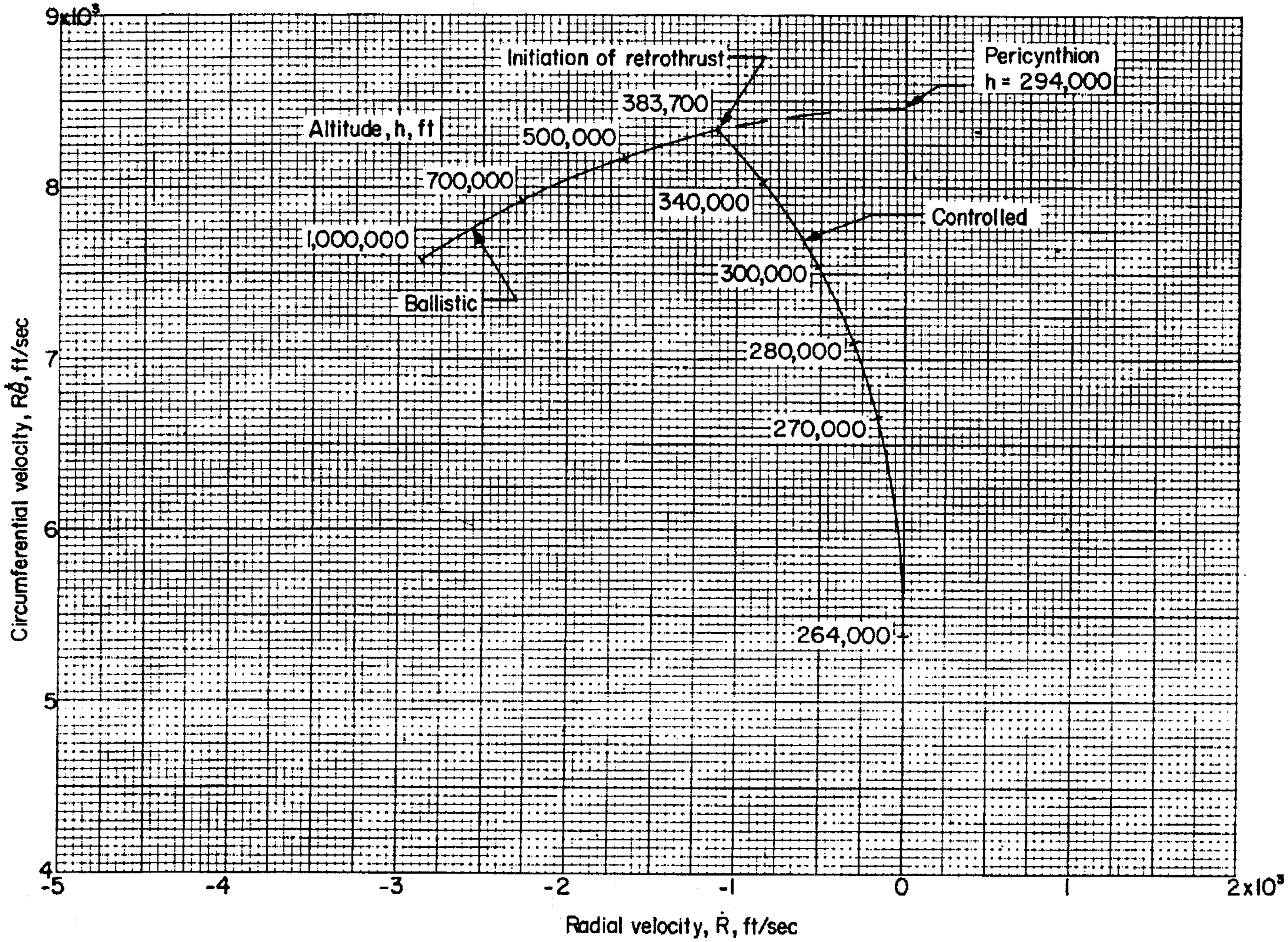


Figure 13.- Nominal trajectory in the hodograph plane.

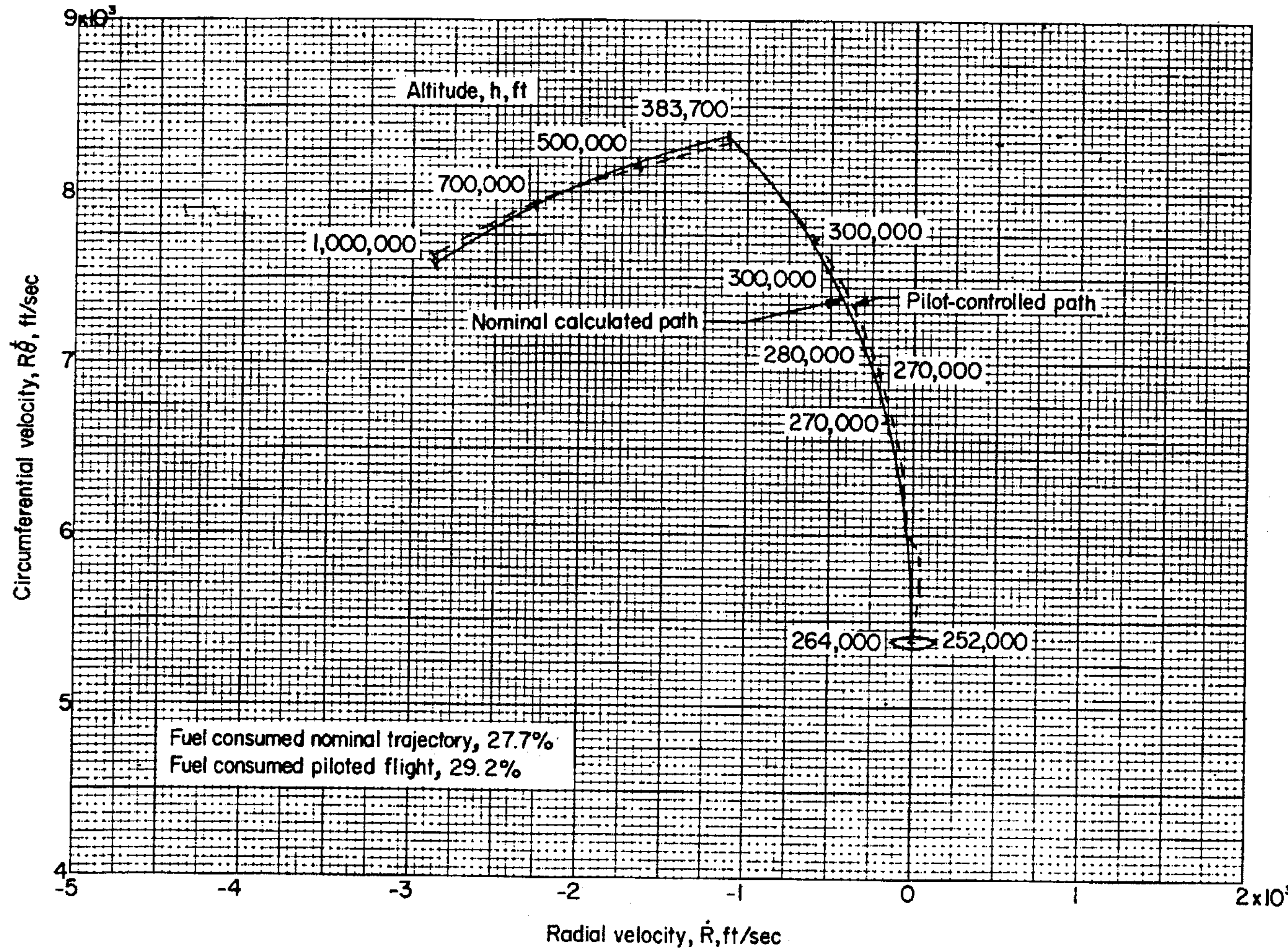


Figure 14.- Typical piloted flight of the nominal trajectory (fuel consumption given in percent of initial vehicle mass).

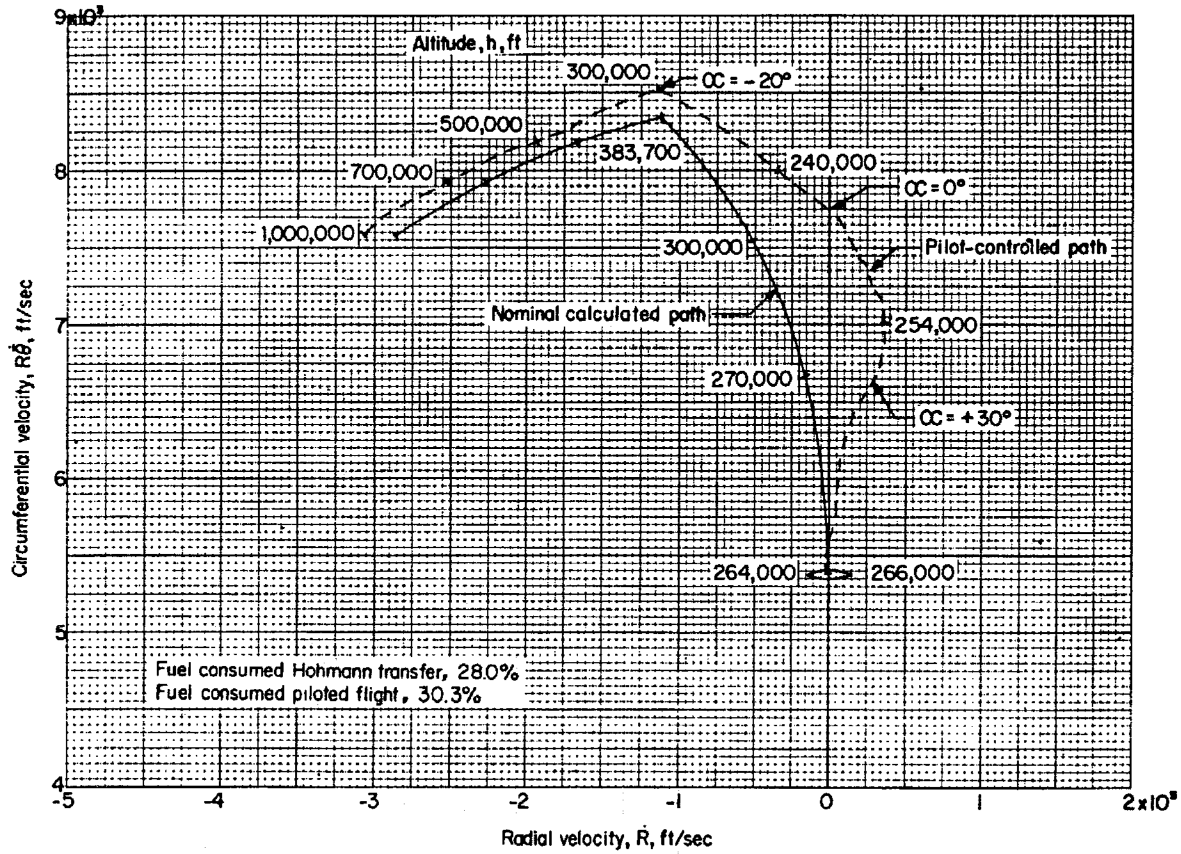


Figure 15.- Piloted flight of an off-nominal trajectory during which full thrust was applied continuously from thrust initiation until vehicle achieved desired orbital conditions. Thrust initiation altitude, 300,000 feet (fuel consumption given in percent of initial vehicle mass).

DESCENT TO LUNAR SURFACE

Lunar letdown.- After establishing an initial lunar orbit, the pilot can modify the orbit characteristics by proper application of thrust. This might be desirable in order to pass the orbit over a specific point of the lunar surface or to establish a more precise circular orbit. The orbit characteristics can be determined quite accurately at this stage by use of very simple devices as indicated in table XVII. The next phase is the departure of the Lunar Lander from the mother ship, and a subsequent deorbit, letdown, and landing. The departure of the lander will be accomplished by applying a small thrust to separate the vehicles into such a position that the rocket plume from further thrust application will not envelope the mother ship. At this point the two vehicles are in nearly coincident orbits. There are a variety of thrusting procedures which could be used during the deorbit and letdown phases. Analytical studies have shown that the deorbit and letdown phases can be accomplished economically by applying a small impulse to initiate deorbit, followed by a coast period, and finally a second, longer, retro-period to bring the vehicle to rest on the lunar surface. Typical computed values of the characteristic velocity for the deorbit and landing maneuver are given in table XVIII for various initial thrust-to-initial-weight ratios (assuming constant thrust). The characteristic velocity associated with a Hohmann transfer (impulsive thrust, 180° surface travel) is 5,630 feet per second. These results show that only a 5-percent ΔV penalty in characteristic velocity is incurred by the use of a thrust-to-weight ratio as low as 0.430. Low accelerations reduce the piloting problem since motions occur within the pilot reaction times.

A six-degree-of-freedom, fixed-base analog simulator study is currently under way to determine the ability and efficiency of pilots to control the deorbit and letdown phase of the lunar landing, and the control and display requirements of the pilots. Photographs of the control console used in initial studies are shown as figure 16. The pilot is given control of thrust along the vehicle longitudinal axis, and moment control about all three body axes. The display as used in initial tests showed altitude, altitude rate, circumferential velocity, vehicle angular rates, and vehicle attitude.

In the actual lunar landing vehicle the altitude, radial velocity, and tangential velocity components are to be measured using onboard pulse doppler radar (Appendix) vehicle attitude and angular rates can be measured by use of an inertial table or, if practical, by visual observation of the lunar surface and horizon.

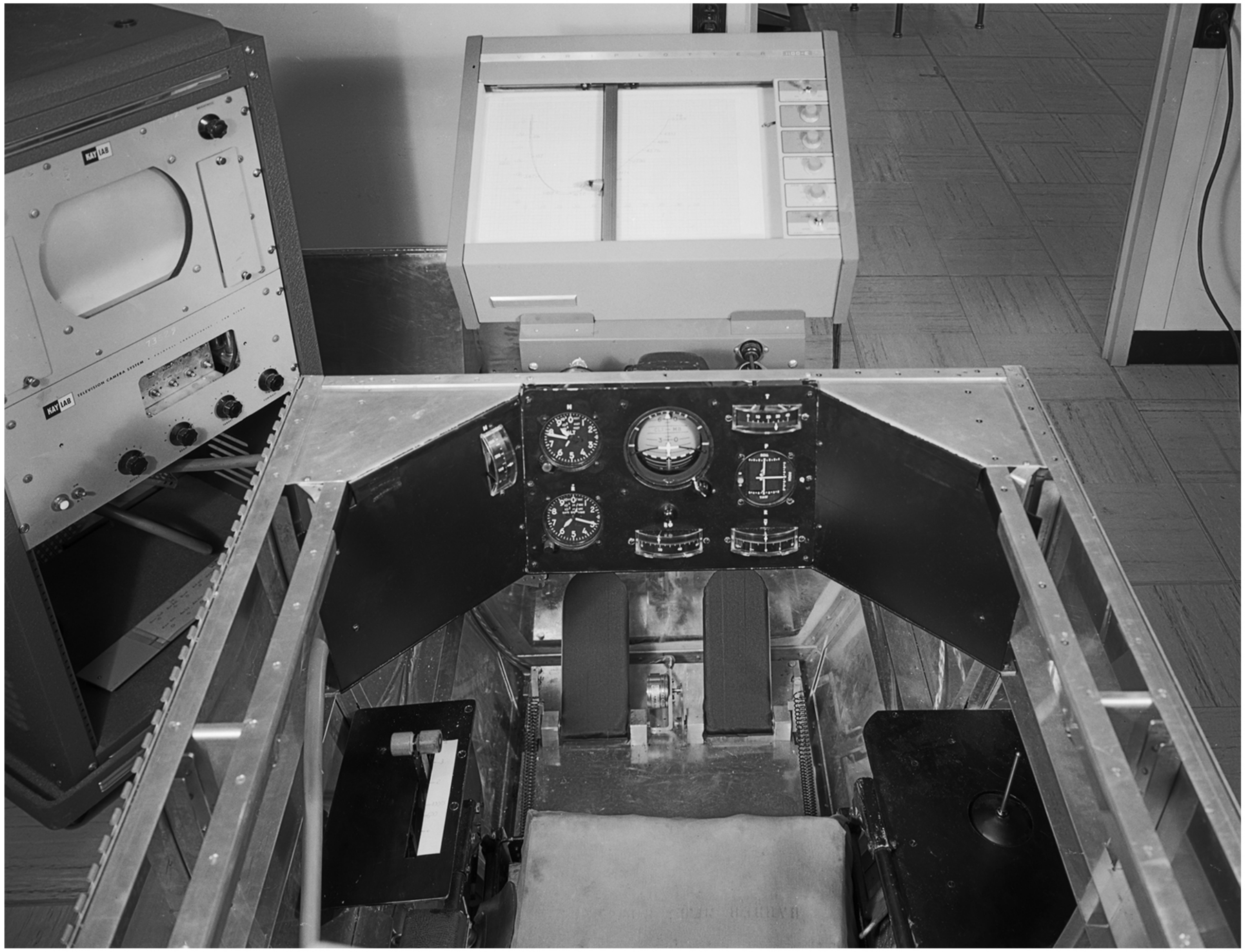
TABLE XVII.- ACCURACY OF DETERMINATION OF ORBIT CHARACTERISTICS

Orbit Characteristic	Precision
Altitude (from optical ranging), mile	±1
Velocity (from altitude and orbit period), fps	±6
Flight-path angle (from period and altitude), deg	±0.05
Orbital inclination (from observation of lunar surface), deg	±0.07

TABLE XVIII.- TYPICAL COMPUTED VALUES OF CHARACTERISTIC VELOCITY

FOR THE DEORBIT AND LANDING MANEUVERS

T/ω_o	0.250	0.286	0.430	1.000	2.000
ΔV	6,350	6,230	5,920	5,690	5,650



(a) Closeup of display and controls. L-61-6517

Figure 16.- Control console used in initial simulator study of lunar letdown.

(b) General arrangement. L-61-6516

Figure 16.- Concluded.

Lunar landing technique.- Because of several compelling reasons, it has generally been agreed that all lunar landings, whether by direct (entire system) descent or by means of lunar rendezvous, will be made from a lunar orbit. This orbit will have been established at an altitude of about 50 miles by methods described in a separate section of this study. (Although more detailed mission analysis may revise this choice of orbital attitude either upward or downward, the actual attitude employed will have little effect on the basic landing operation as herein proposed.)

The landing technique to be described is the result of a number of lunar landing studies conducted and in progress at the Langley Research Center. These studies are described in detail in Part III of this report. Both analytical and pilot-controlled simulation investigations have been made. Completely instrumented and completely visual landing maneuvers have been examined. From these studies two important conclusions seem to be emerging. These may be stated as follows:

1. A completely visual landing operation appears feasible. Although we do not recommend an uninstrumented landing vehicle (instrument aids for the pilot are of considerable help and can be provided with no large weight penalty) it is of considerable comfort and a large plus value in safety and reliability to know that such an operation is possible.

2. Pilots prefer braking rocket(s) that have a thrust level of from 1/4- to 1/2-earth "g." Thrust levels less than 1/4-earth "g" do not seem to provide enough positive response and thrust levels much over 1/2-earth "g" tend to become overly sensitive.

Basically, the lunar landing operation is a problem in orbital mechanics, which can be discussed readily with the aid of figure 17. This figure shows the characteristic velocity requirements for a variety of landing maneuvers. All of these maneuvers are initiated by firing a braking thrust which reduces the vehicle velocity by a small amount. This step in the operation puts the vehicle on an elliptical orbit which, depending on the size of the initial thrust period, carries the vehicle varying amounts around the lunar surface before the touchdown point is reached.

The lower curve on figure 17 shows the characteristic velocity required if the vehicle had an infinite-thrust braking rocket. In this hypothetical situation, the vehicle approaches the lunar surface at high velocity. Just as the surface is reached, an impulsive braking thrust is applied which reduces the velocity to zero. The particular point on this curve at 180 degrees of travel is called a "Hohmann Transfer" and is theoretically the most efficient landing maneuver possible.

In the practical case, however, finite-thrust braking rockets are used, requiring a relatively long braking period during which gravity acts to cause a loss in efficiency. The two upper curves on figure 17 illustrate what happens with braking rockets of about the size which pilots find to be most satisfactory (1/4- to 1/2-earth "g"). It is seen that the efficiency loss (by reference to the shaded area which shows a 5-percent loss) is about 10 percent. These particular curves represent landing maneuvers in which the final approach is made by a "gravity turn." This simply means that the braking thrust is always applied to oppose the flight velocity, which results in the flight path becoming more and more steeply inclined, reaching a vertical descent just at touchdown.

The two separate points on figure 17 show a landing maneuver which is only about 5 percent less efficient than a Hohmann Transfer. The maneuver is in fact, a variation of a Hohmann Transfer. The only difference is that, when 180 degrees of travel has occurred and the vehicle is flying horizontally (tangent to the surface), the velocity is reduced while maintaining constant altitude (instead of instantaneously with impulsive thrust). The vehicle ends up in a hovering condition a short distance above the surface, from which a vertical descent to touchdown is made. This basic maneuver is the one which is presently recommended. Not only is it relatively efficient but, from a pilot's viewpoint, it most nearly duplicates the landing approach of conventional airplanes.

It should be noted that figure 17 tends to indicate that considerable leeway is available in the landing trajectory so long as at least 45 degrees or more is traversed. While this may be so if a "direct" (entire system) landing is being considered, factors related to the abort situation for the lunar-rendezvous method favor choice of a landing maneuver covering more nearly 180 degrees. (These factors are discussed in detail in the section of the report covering abort.)

A typical landing operation might proceed through the following steps:

1. During the initial lunar orbits, a final selection of the landing area will be made. This area will probably be in earth-shine, thus avoiding the bright glare and black shadows on the sunlit side.

2. The pilot will then enter the lander, perform final checks, separate his vehicle from the parent vehicle, and place it a short distance to one side so that when he fires his retro-thrust it will not affect the parent vehicle. (Perhaps this position may be held by means of a simple extensible-rod device.)

3. When the vehicles have travelled about half-way around the moon from the landing area, the retro-thrust will be applied to place the

lander on a suitable landing trajectory. For a 50-mile orbit this retro-thrust must decrease the lander velocity by about 60 feet per second. To avoid large errors in this step, the large braking rocket(s) will not be used. Probably the small attitude-control rockets will be adaptable for this operation.

4. The lander will coast for about one hour on its landing trajectory. During this time the pilot in the lander and the crew in orbit will continuously monitor progress, making whatever small corrections are necessary. It should be noted that the lander and orbiting vehicles will be in direct visual and radio contact during the entire operation. Should it be desirable to abort during this phase, only a small amount of thrust is required (primarily to change the flight path several degrees upward to place the lander on a course for rendezvous).

5. When the lander passes over a preselected point about 100 miles from the touchdown point, at an altitude of several thousand feet, braking thrust will be applied. This thrust will be applied to reduce flight velocity, with a downward component sufficient to maintain a reasonably constant altitude. The pilot will maintain his altitude by either visual observation or by means of a radar altimeter in an operation not unlike an airplane on final approach. As his velocity decreases, the pilot will gradually lower his altitude and will finally reach a hovering condition close to the surface. During this phase the pilot will use throttling and/or gimballing to control his altitude and to reach his touchdown area. In this connection the use of two braking-rocket engines capable of being gimballed sideways seems attractive. Such an arrangement should greatly alleviate the jet-blast effects (visibility obscurement and vehicle damage) while also tending to prevent damage to the touchdown area.

6. While hovering (for up to a minute or more) the pilot will select his touchdown point and will then lower his vehicle to the surface. Even during this phase, as discussed elsewhere, abort will be possible.

Finally, it should be observed that the procedure outlined never results in a high vertical velocity. Consequently, positive control of altitude with the braking rocket(s) provided will be possible. With the gimballed two-engine configurations proposed, moreover, it will be possible to control altitude and abort even after failure of one engine.

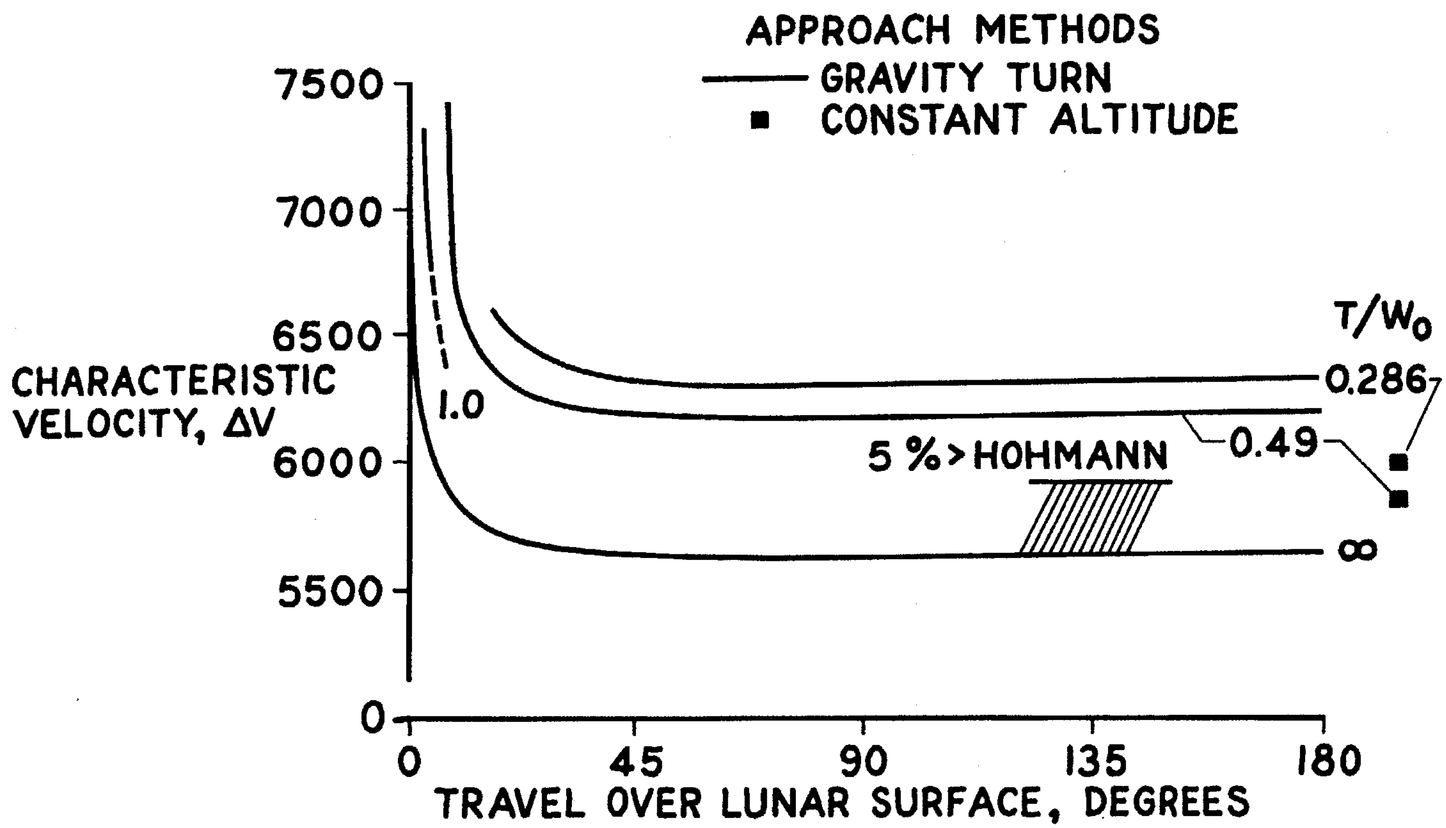


Figure 17.- Lunar-landing performance. L-1308

LUNAR LAUNCH AND RENDEZVOUS

General considerations.- The launch to rendezvous involves an injection phase, a coast phase, and a terminal rendezvous phase. See figure 18. The launch trajectory is chosen so that the lunar satellite is always in view of the lunar launch vehicle. This stipulation avoids the necessity for a "blind" launch in which the lunar launch vehicle is required to launch itself prior to appearance of the orbiting vehicle. It is anticipated that control of the injection phase as well as the other phases of lunar launch will be manual; however, inertial and radar sensing as well as optical line-of-sight sensing will be used to insure success of the entire rendezvous operation.

Much work has been completed at Langley in the area of manned rendezvous control, particularly in the area of terminal rendezvous control. Some of this work is described at the end of this section. More recently work has been in progress relative to manned control of the entire lunar rendezvous operation. Indications are that simple line-of-sight optical sensing, perhaps backed up by range measurement and an inertial attitude reference will prove adequate for the lunar rendezvous. Such control may prove to be suitable as a basic rendezvous plan and certainly would serve in the event of loss of inertial and radar sensing.

Because of the absence of an atmosphere and the attendant interference with vision the lunar rendezvous should be much easier to perform than the earth rendezvous if appropriate lighting conditions are chosen. In this respect a side of the moon away from the sun may be most appropriate.

Launch and injection phase of rendezvous.- Launch of the lunar lander is initiated when the elevation of the line of sight between the lander and orbiting vehicle reaches an appropriate range of values. Corrections during launch for offset of the launch point from the orbital plane of the orbiting vehicle are minimized by use of the near equatorial lunar orbit mentioned previously and by selection of the landing point with consideration of the stay time and the rotational rate of the moon. The inclination of the lunar equator to the lunar orbital plane is never more than about 6°. A stay on the moon of 7 days could thus result in an offset from the lunar orbital plane of 6°. This effect results from the 90° rotation of the moon which carries the landing site out of the lunar orbital plane. If a 3° plane change were made on landing and another 3° plane change made on take-off 7 days later, a total expense in mass ratio of 2.6 percent would be incurred. The plane change required for a 24-hour lunar mission is insignificant of course.

Figure 19 shows the elevation angle of the line of sight between the lander and orbiting vehicle at launch as a function of the coast angle and in launch to rendezvous. For a 20° coast around the lunar surface an elevation of the line of sight of 4.8° at launch is required. For longer coast angles, higher elevation angles will be required. The flight time to rendezvous is shown in figure 20 as a function of coast time. Times from 10 to 33 minutes are required for the direct rendezvous assumed here. In the event that it is necessary to maintain the orbiting vehicle in view for a longer period of time, then a lunar orbital altitude higher than the 50 miles assumed for this study may be used, but no real necessity for a higher orbital altitude seems to exist.

In the powered injection phase two guidance plans are to be available if the pilot elects to follow inertial sensing rather than line of sight. In each case control of the vehicle is manual. In one plan a complete inertial system senses accelerations and by use of a digital computer logic arrives at instantaneous pitch, roll and yaw rates which will achieve the desired burnout condition and insure intercept. In the other plan an open-loop tilt program with respect to an inertial reference is employed. In each case the required information is presented to the pilot for appropriate control action. The control plan in each case is appropriate to the satellite elevation angle from the lunar lander at launch and is followed until the required velocity for coast to the terminal phase is acheived. This procedure will carry the lunar lander to about 10 miles altitude if executed at an initial acceleration of three lunar G.

An inertial sensing system of modest quality as given in table XIX will be employed during the injection phase. The positional accuracies of this system will be satisfactory for the beginning of the terminal phase. See table XX. Further refinement of the positional and velocity uncertainties at burnout will occur during the radar and optically monitored coast phase which follows.

It is expected that the open-loop plan of injection guidance will be somewhat less precise than the rate command system, but not too much so in that both plans depend on the same inertial components. The primary difference is that one plan has computer derived command rates while the other has programmed attitude commands.

Coast phase of rendezvous.- Radar and optical monitoring of the coast phase will be employed to insure success of the rendezvous mission. In this phase gross injection errors will be corrected and vernier adjustments made to insure a close approach to the conditions required for the terminal rendezvous phase. References 6 and 7 discuss the use of radar ranging and appropriate orbital mechanics for the estimation of the propulsive corrections required for rendezvous. For the 1σ radar measurement uncertainties given in table XXI with the last course correction made 5 miles from the beginning of the terminal phase of rendezvous, reference 6 indicates a radius of uncertainty at terminal phase initiation of as little as 250 feet. These results are for a rendezvous trajectory in the vicinity of the earth and for radar capabilities somewhat more attractive than would be employed for the lunar rendezvous because of weight limitation. Nevertheless these results are representative of the power of optimized homing techniques in achieving conditions appropriate to the beginning of the terminal phase of rendezvous.

Gross errors in the performance of the inertial guidance system may be erased in the early phase of coast at little expense in mass ratio. In this respect the radar system complements the inertial system in that although the inertial system is capable of sufficient accuracy when performing correctly, the radar and optically guided coast can compensate for substantial deficiencies at moderate expense in mass ratio.

In the unlikely event that the rendezvous is not completed as planned because of launch delay, the chasing technique discussed in references 8 and 9 may be employed. Ample instrumentation is provided to monitor such an operation, and a rendezvous could be effected a few orbits later than originally planned.

Simulation studies.- Three simulation studies which have been made at Langey Research Center and which indicate the ability of the pilot to perform rendezvous are as follows:

1. Investigation of an all-instrument pilot-controlled rendezvous: A simulator investigation was made of a pilot-controlled terminal rendezvous maneuver. The pilot was presented all of the required parameters on cockpit instruments. See figure 21. The investigation was conducted primarily to determine the control requirements, instruments, thrust levels, and fuel and time relations for a wide range of intial conditions, thrust misalinements, and damping. Results, reported in reference 10 indicate that a human pilot can control the rendezvous maneuver successfully in the presence of relatively severe conditions if adequate vehicle control and flight-data presentation are provided, and do so with only slightly more than minimum fuel. Misalinement thrusts equal to 90 percent of attitude control power can be overcome, noncoplanar condition can be controlled, and a single thrust rocket is sufficient for all translational control. Additional work with instrumented space control problems is planned to include studies of such effects as radar noise and other interference to instrument accuracies.

2. Visual control of rendezvous: A simulation study has been made of a pilot's ability to control the terminal phase of rendezvous using visual cues to determine relative angular motion. Figure 22 illustrates the procedure. The right side of figure 22 shows the required attitude and thrust controls and range and range-rate instruments needed for a pilot to control his vehicle to a target which has an identifying light. Visual sightings through the hatch indicate motion of the target relative

to the star (inertially fixed) background. For the condition shown, the pilot establishes a proper intercept by rolling to aline a lateral rocket in the plane of the motion arrow, thus orienting a thrust vector to "chase" the target vehicle. A collision course is indicated when the target is stopped relative to the stars. Then the second phase of the maneuver, that of braking the closure rate as the vehicles draw together, begins. This phase of the operation requires a knowledge of range R and range rate $\dot{R}$. In this study, these two parameters were displayed to the pilot on suitable instruments, but could also be obtained by the visual techniques described in item 3 below, thereby eliminating the need for the two display instruments.

In some instances, such as the resupply of an orbiting station, it would be desirable to launch the unmanned supply vehicle to within rendezvous range and then have the pilot remotely control the supply vehicle to his station. In this case, the attitude of the remote vehicle would have to be presented to the pilot in the control station to enable him to orient the remote rockets properly. This is the case studied in this simulation study and is identical with the system that would be used in the early phases of this development program. This case is represented by the left portion of figure 22. Results indicate that a pilot can detect the angular motion of the target through 1 milliradian in 10 seconds, establishing rates as low as 0.1 milliradian per second, and can perform the rendezvous maneuver precisely. Also, the transition from the terminal maneuver to the final docking maneuver is easier visual-to-visual than for the instrument-to-visual case.

3. Visual techniques for determining rendezvous parameters: Analytical and simulation studies have been made of a pilot's ability to control the terminal phase of rendezvous using only visual measurements. The analytical phase derived techniques for transforming visual measurements of relative angular motion and thrusting times into range and range rate between the vehicles. The simulation phase was conducted to prove the feasibility of the analytical techniques as well as the ability of the pilot to utilize these techniques. Results show that a pilot can successfully control rendezvous visually using a star background to measure angular motion, timing a known thrust level used to arrest the angular motion, and from these quantities computing range and range rate. At 50 miles, range can be determined within 2 miles or 4 percent, and range rate can be determined within 5 percent at 1,000 feet per second closure rate. These values are well within safe and efficient limits.

A technical note presenting the detailed results of the study is being prepared.

TABLE XIX.- LUNAR LAUNCH INERTIAL GUIDANCE ERROR PARAMETERS 3σ

Gyros:	
Time dependent drift rate	1°/hour
Acceleration dependent drift rate	1°/hour/g
Anisoelasticity	0.25°/hour/g^2
Accelerometers:	
Zero uncertainty	3×10^{-4}g
Scale factor uncertainty	5×10^{-4}g/g
Platform alinement:	
Angular uncertainty -	
Pitch, min	2
Roll, min	2
Yaw, min	10
Lunar acceleration:	
Uncertainty	1.6×10^{-4}g

TABLE XX.- VELOCITY AND POSITIONAL UNCERTAINTIES

Velocity uncertainties 3σ at burnout of lunar launch:	
Longitudinal, ft/sec	6
Vertical, ft/sec	6
Lateral, ft/sec	18
Positional uncertainties 3σ at end of 20° coast period:	
Vertical, ft	3,590
Lateral, ft	7,980

TABLE XXI.- RADAR MEASURING ERRORS 1σ

Range, ft	30
Range rate, ft/sec	3
Angle, radians	5×10^{-4}
Angular rate, radians/sec	1×10^{-5}

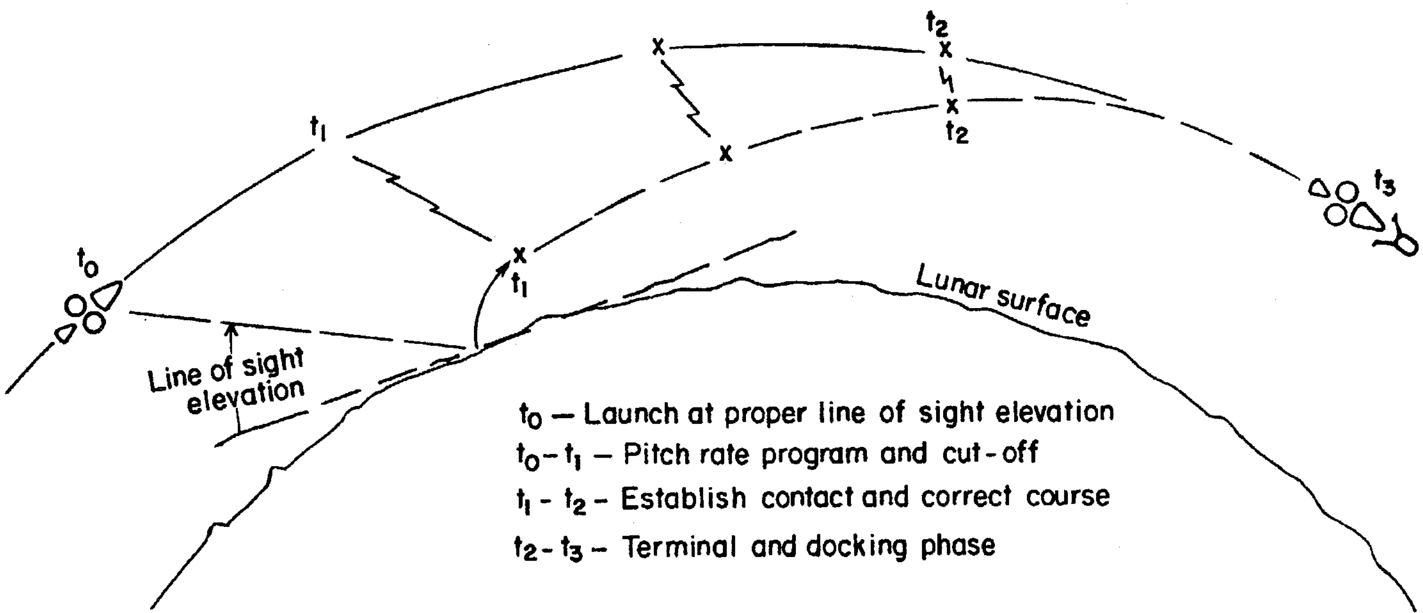


Figure 18.- Sequence for lunar rendezvous.

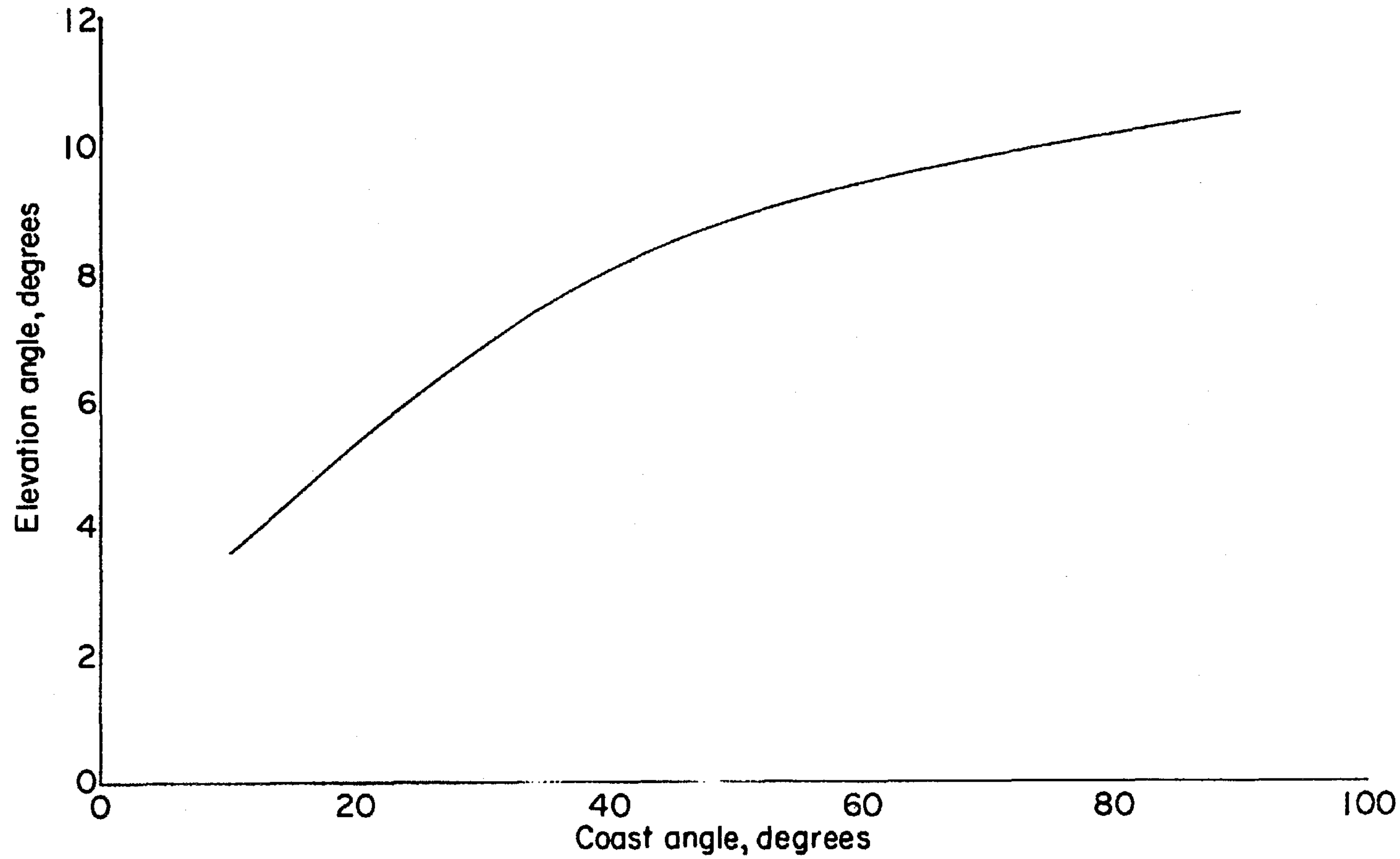


Figure 19.- Elevation of orbiter above horizon at take-off of lander as a function of coast angle.

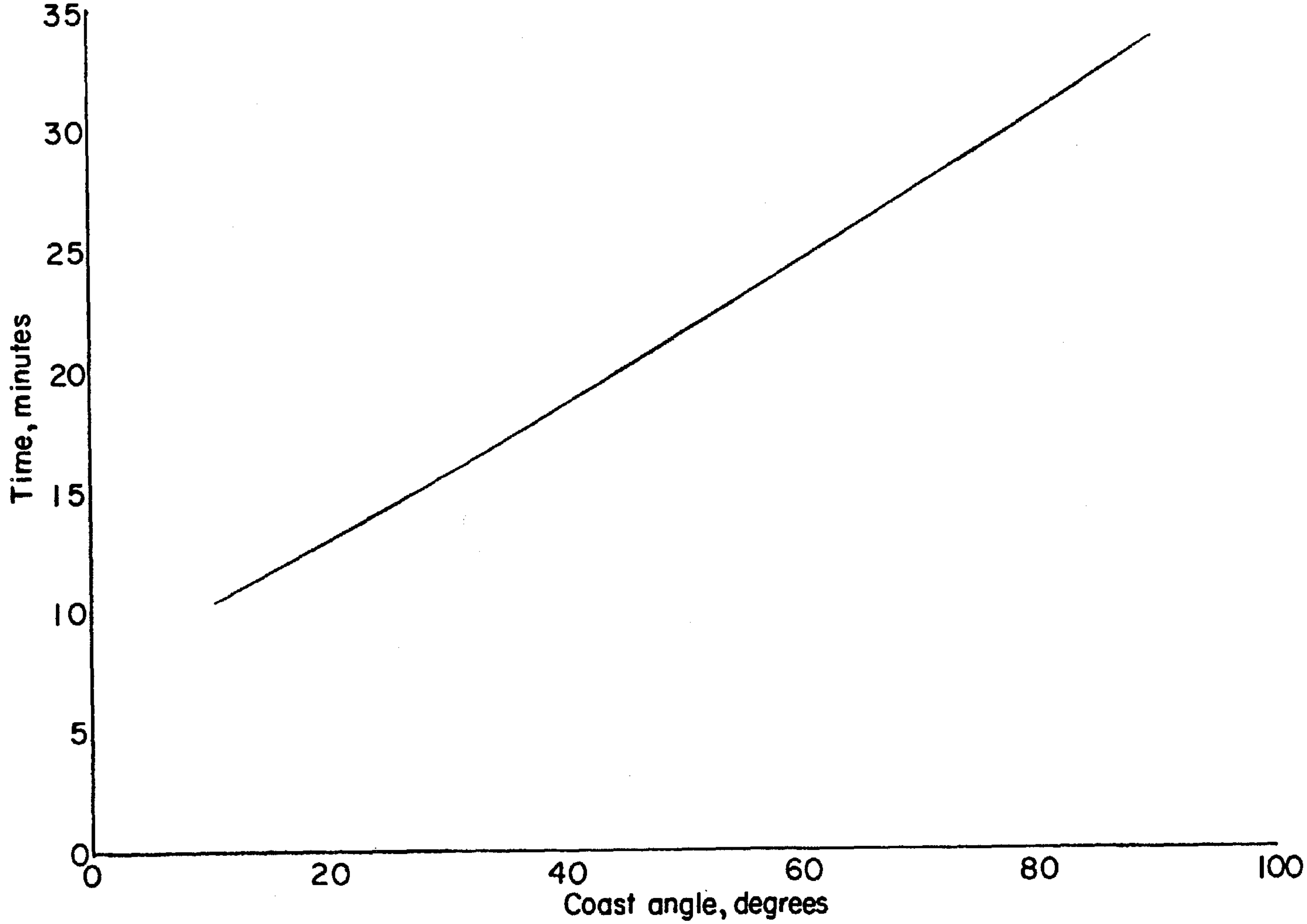


Figure 20.- Time to lunar rendezvous as a function of coast angle from boost cutoff.

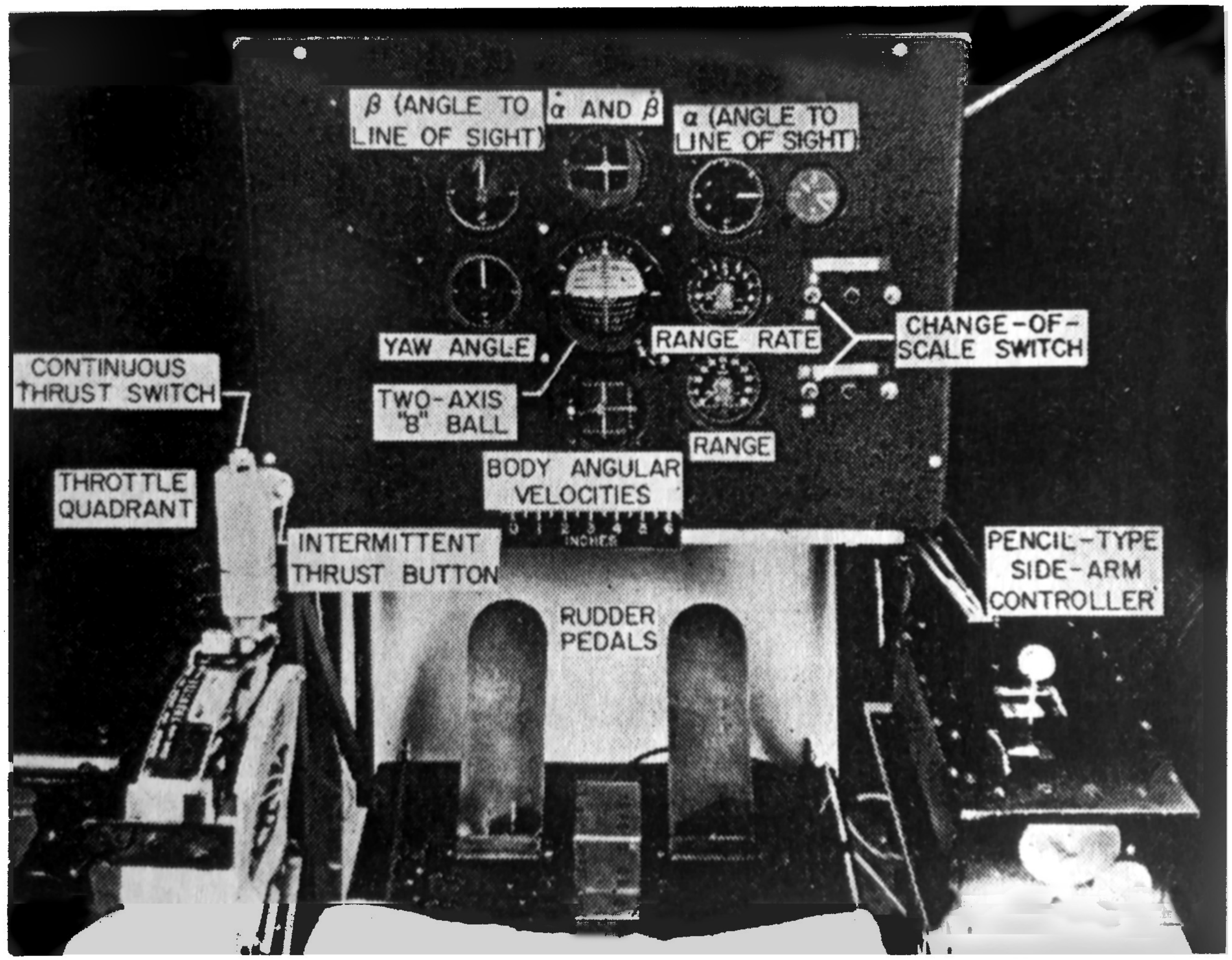


Figure 21.- Photograph of cockpit display. L-60-4266

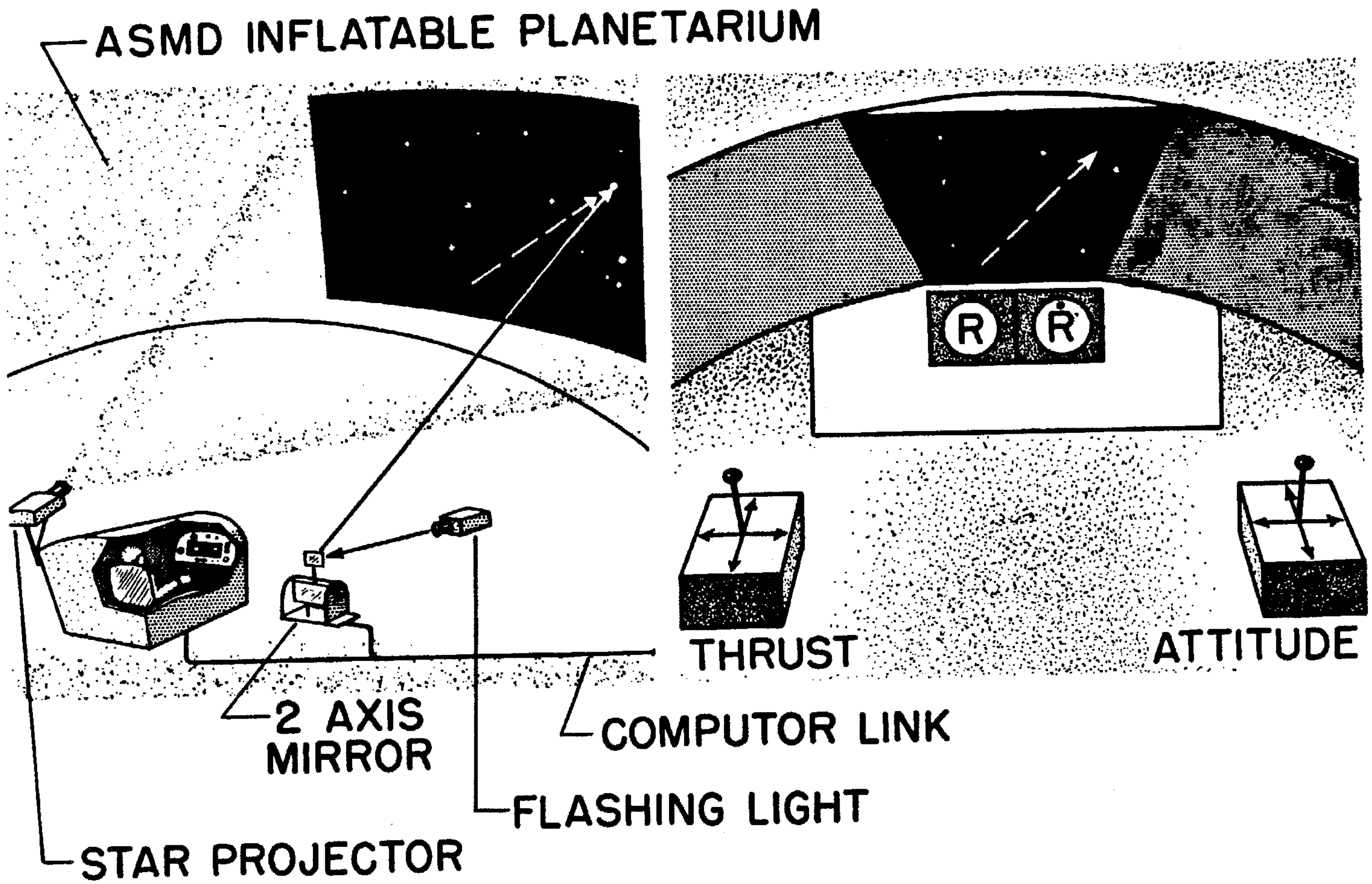


Figure 22.- Visual terminal phase simulation. L-1663-10

ABORT IN ALL PHASES

Abort techniques for all methods of achieving a manned lunar landing are the same for the various phases of the operation from earth launch to lunar orbit. The problems are also the same from lunar orbit to reentry.

In the lunar-rendezvous method, the lunar-landing phase incorporates a unique problem because here the landing module must separate from the orbiting command station, and in case of abort must return and rendezvous with the orbiting link. Therefore, this section deals only with the abort problem associated with lunar landing.

For abort considerations, the most critical phase of the landing operation occurs just prior to touchdown of the lander on the lunar surface. Should, for some reason, an abort be required at this time, the landing vehicle should have the capability of returning to the station in approximately one orbital period of the station. This time limit for the return is based upon an assumed maximum allowable difference between the time of detection of a solar flare and the time that a radiation hazard would exist at the moon of about one and one-half hours. Also, the velocity increment required for the return should not be excessive. Consequently, landing maneuvers must be tailored such that the landing vehicle is always in a position favorable for rendezvous with the station. Essentially, this dictates the use of a Hohmann type transfer orbit, if abort is elected at the end of hovering.

An analytical study of the problems associated with abort in landing and return to a lunar orbiting space station has been made. Results of the study indicate that the landing maneuver should be chosen such that, at the point of touchdown, the space station should not have proceeded down range so far that a Hohmann type transfer return could not be made by the landing vehicle. This technique for landing would insure an economical return should an abort situation arise. If the space station exceeds this limit, the landing vehicle would have to resort to a chasing technique for which the fuel requirements could become excessive.

Figure 23 shows the impulsive velocity requirements for landing and return and the positions of the two vehicles after 1 minute of hovering by the landing vehicle above the lunar surface for landings from 50- and 100-mile orbits. Also shown on figure 23 are the displacement angles for a Hohmann transfer return to the orbiting vehicle. Hence figure 23 shows that the landing vehicle would be in a favorable position for return if the orbital angle to touchdown is at least 165° from the position at which the landing maneuver is initiated for the 50-mile-orbit case, and at least 63° for the 100-mile-orbit case. It is evident from figure 23 that landing from a 100-mile orbit is less sensitive to abort requirements than landing from a 50-mile orbit. Also, the velocity increment required for landing from the 100-mile orbit is only about 100 ft/sec greater than that required for the 50-mile orbit. Consequently, by proper choice of the orbital altitude and transfer maneuver for landing, the velocity requirement for abort should not exceed the requirement for a normal take-off and rendezvous maneuver.

A simulation study has been conducted of a pilot's ability to perform the abort from a lunar landing. In the simulation study, the pilot was furnished with range, closure rate, and the attitude information of the vehicle. The pilot detected angular motion by visual observation of a simulated orbiting vehicle against a simulated star background.

Results of the simulation show that a pilot can control the abort maneuver visually, and do so using less than 20-percent greater fuel than that required using impulsive thrust. A 20-percent greater fuel consumption would correspond to a velocity increment of about 6,900. Therefore, the 7,500-foot-per-second characteristic velocity increment adopted in the vehicle analysis section is more than adequate to allow abort under the worst conditions.

It should be noted that, if abort is required during initial letdown or during final approach before the lander vehicle velocity is appreciably reduced, the fuel requirements are very modest. Here the main requirement is primarily to change the flight-path direction only a few degrees upward to reach a proper rendezvous course.

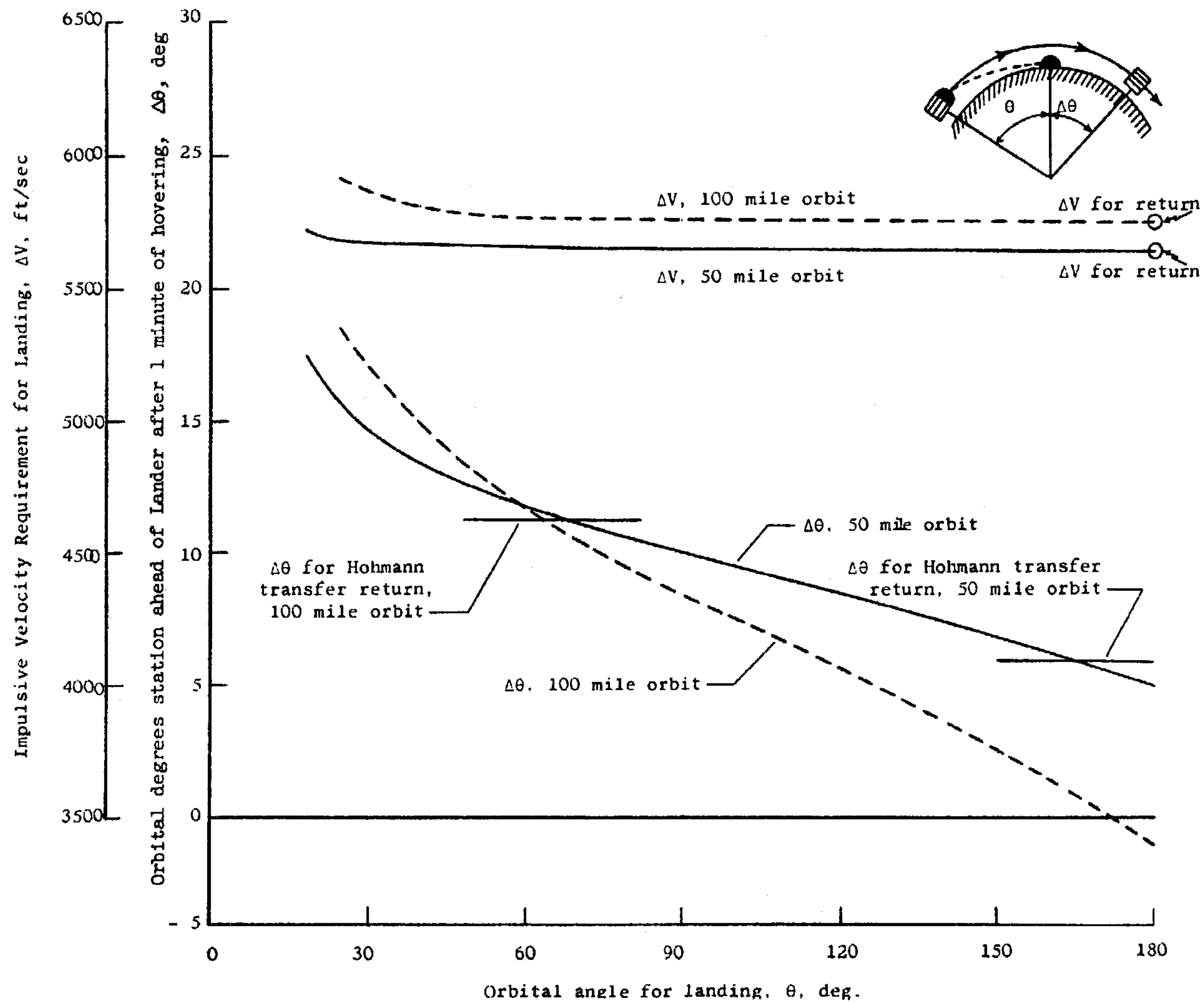


Figure 23.- Impulsive velocity requirements for landing.

INSTRUMENTATION, COMMUNICATIONS, LIFE SUPPORT, AND AUXILIARY POWER

The lunar landing concept under study involves the establishment of a 50-nautical-mile lunar orbit using an Apollo type vehicle (mother ship) and a subsequent lunar landing using a smaller landing vehicle. This lander (or BUG) will contain only the equipment deemed necessary to accomplish the landing and subsequent rendezvous with the mother ship. The operational technique considered imposes more stringent instrumentation requirements than might be imposed if greater dependence could be placed on visual manned control. Greater dependence on manned control will require less equipment and consequently the instrumentation can be deduced from this more comprehensive study. The various phases of the overall mission may be identified on figure 24. The measurements which are required for each phase along with the estimated ranges and accuracies are given in table XXII.

Navigation and Guidance

The landing and rendezvous mission has been divided into several phases as illustrated in figure 24 and listed in table XXII. These phases are as follows:

A. Lunar Orbit - This phase involves the insertion of the mother ship into a lunar orbit at an altitude of about 50 nautical miles. The establishment of the orbit and the determination of the orbital parameters will be accomplished using equipments carried onboard the mother ship. These parameters will be used as initial conditions in programming the navigational schemes for the Bug. Table XXII lists the range and accuracy of the measurements required during this phase, as well as the subsequent phases.

B. Deorbit - This phase commences with the separation of the Bug from the mother ship and terminates at an altitude of 500 to 1,000 feet. The parameters to be measured pertinent to the descent maneuver are altitude h, rate of change of altitude $\dot{h}'$, tangential velocities (lateral V_{TT}), and along course V_{TC}, time T, and the vehicle attitude angles (pitch θ, roll ϕ, and yaw ψ).

In addition, continuous tracking of the mother ship will be accomplished to permit the immediate initiation of an abort maneuver should it become necessary. These measurements are range r to the mother ship, range rate $\dot{r}$, in-plane line-of-sight angle α, out-of-plane line-of-sight angle β, and the rates of change of these angles ($\dot{\alpha}$ and $\dot{\beta}$).

The optimum abort maneuver will be computed continuously during this phase as discussed in the section dealing with the onboard computer.

C. Landing - Beginning at an altitude between 500 and 1,000 feet, it may be possible to perform the landing maneuver using visual references and no additional instrument aids. However, since the possibility exists that a dust cloud may seriously limit the field of view, the following measurements will be made and presented to the pilot: altitude, rate of change of altitude, the tangential velocity components, and the vehicle attitude angles.

D. Launch - After completing the mission on the lunar surface the mother ship will be acquired and its position determined using the onboard radar. This information will be fed into the computer and the time to launch will then be computed based on the mother ship's orbital parameters which were stored in the computer prior to the initiation of the deorbit phase. Having performed this computation the launch phase will commence after the mother ship completes the next orbit and comes into view again. The mother ship is acquired on radar with the line-of-sight sensors, and propulsion is initiated at the precomputed time. The primary control elements during the launch phase will be the inertial platform and radar altimeter. The other parameters, r, $\dot{r}'$, α, β, $\dot{\alpha}'$, and β', are also being obtained during this phase. The launch phase will be completed with engine cut-off at the attainment of a predetermined position and velocity. This altitude will be about 9 miles at a range angle of about 24°.

E. Rendezvous - This phase commences at engine cut-off and terminates at a range of a few hundred feet from the mother ship. Primary control parameters during this phase will be r, $\dot{r}'$, α, β, $\dot{\alpha}'$, and $\dot{\beta}'$. Corrective maneuvers will be performed as necessary to ensure rendezvous within allowable tolerances.

F. Docking - Docking has been assumed to be accomplished through visual aids by the pilot.

To satisfy the guidance requirements for the mission phases shown in table XXII, a preliminary evaluation indicates that a digital computer will be needed on the Bug.

The chart shown in figure 25 separates the mission into computing programs. It has been assumed that monitoring for an abort will be made during deorbit and landing. In the event that abort is necessary, transfer to the rendezvous program will be made.

Table XXIII lists typical subroutines used for programming the tasks shown in figure 25. Table XXIV provides a brief description of these subroutines.

Figure 26 gives flow diagrams using these subroutines to program the required tasks.

It is felt that a computer can be developed to satisfy the proposed lunar-landing project with the following characteristics:

Power - 80W, weight - 45 lb, volume - 0.4 cu ft

The estimates of computer power, weight, and volume were based on the following:

A. Computing the subroutines listed in table XXIII.

B. The use of solid-state components.

C. The use of welded circuit modules.

D. The use of a bit organized memory of 22 bits per word with parity check.

E. The use of present computer technology.

Equipments which will be required to accomplish the navigation and guidance functions are listed in table XXV along with estimates of the power and weight requirements. It is assumed that this equipment will not be required to operate all the time while it is on the lunar surface. A total of 6 hours operation time has been assumed. The total power requirement, then, is about 3,300 watt-hours.

Communications

Since the vehicle will contain a minimum of equipment, communications will be furnished for the purpose of voice communication only. It is necessary to provide the capability for continuous communication to the earth and for intermittent communication to the mother ship while it is in the line of sight. It is planned to provide this capability not only in the Bug, but also in the backpack the man will carry during his sojourn on the lunar surface. For increased reliability each system will provide this capability independent of the other. It is planned to utilize S-band frequencies to be in consonance with other Apollo equipment as presently planned.

The weight and power requirements for this equipment are included in the summation in table XXVI.

Life Support System

The assumptions made for the design of this system were as follows:

(1) Two vehicles: 1 man each, one with scientific payload for lunar exploration, and one with 1 man payload for rescue.

(2) Length of use: $4\frac{1}{2}$ hours - 7 days

$4\frac{1}{2}$ hours - 12 hours

(3) Lunar environment: Time of landing and exploration will be during lunar night, illuminated by earth light.

(4) Power and water will be available from an onboard fuel cell with additional water storage.

(5) Shirtsleeve environment onboard lunar landing vehicle except for emergencies, and during times of lunar exploration.

(6) Vehicle will lose heat, at least at rate of equipment input to maintain a comfortable temperature, during time on lunar surface.

(7) Additional heat losses due to equipment use during landing and launch, and due to the occupants' metabolic heat, will be accomplished by water evaporation to the lunar environment.

(8) Atmosphere aboard landing vehicle, and in space suit will be 200-260 mm Hg. of oxygen, and less than 8 mm Hg. of carbon dioxide, with a comfortable humidity level.

The recommended equipment for each vehicle, based on the preceding assumptions is as follows:

The environmental control system for each of these vehicles will consist of two nearly identical packages, which will serve as back up for each other. See figure 27. One of these packages will be for the provision and maintenance of the vehicle environment and the other will be attachable to the spacesuit for purposes of lunar exploration. Both systems will be resupplied periodically from onboard stores, and will have a period of use of about 16 and 8 hours, respectively. Weight, without supplies, about 35 pounds each.

Since these two systems will not be operating at the same time, the power requirements should not overlap, except for very short periods. This power is used to operate fans for air circulation through the system and for recharging suit fan battery. For cooling, about 0.8 pound of water per hour will be required (using a heat exchanger efficiency of 50 percent - fuel cell of 1-kw capacity will produce about 1.1 pounds of water per hour). This cooling will be controllable.

The supplies required for each vehicle are:

Oxygen - Cryogenic or supercritical 4.4 pounds + 50 percent (bottle) = 6.6 pounds per day, in quantities of 1.46 pounds of oxygen per bottle.

Lithium hydroxide (LiOH) stored onboard vehicle. Used either in suit or vehicle system. 5.6 pounds per day + 1 pound per day (canister). Canisters loaded weight 2.2 pounds each.

Drinking water - Stored onboard vehicle at about 5 pounds per day. It is possible, except for emergency, to use water from fuel cell. With some additional weight, about $2\frac{1}{2}$ pounds per day can be recovered from atmosphere.

Food - Rations similar to C-rations can be used at about 2 pounds per day.

The requirements for additional supplies and equipment are:

After about 1 day it becomes necessary to provide some additional facilities and supplies for the occupant of the lunar landing vehicle, such as, sanitary facilities, food preparation areas, etc., at a fixed cost of about 30 pounds.

Cabin repressurization supplies:

Because an air lock will not be practical from a weight standpoint for this lunar landing vehicle, there will be required for each exploration of the surrounding terrain, a supply of oxygen to repressurize the vehicle. This will be dependent upon the volume of the vehicle, but will probably be approximately 5 pounds per day which will make up leakage also.

The instrumentation requirements are:

There is the necessity of providing for the occupant a minimum presentation of his environment and remaining supplies with appropriate alarms as follows:

Oxygen partial pressure
Carbon-dioxide partial pressure
Total pressure
Oxygen supply remaining
Battery or power supply condition

These items should be presented for both the vehicle and suit packs.

For periods of lunar landing beyond about 1 day it may be necessary to monitor the occupant of this vehicle especially during sleeping periods. There is a good possibility that this may be accomplished at a small cost by modulating the existing voice link during those periods of time when no communication is being conducted. Two possible measurements for this might be respiration rate and pattern and electrocardiograph.

Cabin lighting:

Some lighting of the vehicle cabin and auxiliary lamps on the space suit will probably be needed with an average power requirement of about 25 watts.

Equipment utilization:

During descent and launch both the space suit and the vehicle environmental control systems will be used to provide emergency backup. At other times, the cabin will provide a shirtsleeve environment. If a malfunction of either system should occur the space suit will be used with the remaining system. Although there is provision for supply usage by either system for the duration of the mission, the mission should be aborted because of lack of backup.

Rescue:

The rescue vehicle shall be used in the event that the exploration vehicle is not capable of safely achieving rendezvous. This vehicle will have the capabilities of transporting from the lunar surface back to the mother ship the man, complete with suit and suit pack. Thus, during the rescue maneuver there will be two men, with suits and packs, plus a vehicle pack for backup.

Alternate systems:

As indicated on the weight-time chart (fig. 28), after a period of about 3 days an alternative system can be used for the vehicle environmental control. The weight of this equipment is somewhat greater than the lithium hydroxide package, but because it is a regenerative type system, it uses fewer supplies.

This system would utilize radiators to recover the moisture from the atmosphere, and to remove by freeze-out the carbon dioxide which, in the preceding system, is removed by lithium hydroxide.

Other considerations:

Some attention should be paid to the possibilities of reducing the weight of this vehicle by the following means:

Utilization of oxygen from fuel cell supplies saving weight by more efficient storage (or more fuel stores, if oxygen is used as an oxidizer).

Utilization of water from fuel cell for drinking and achieving cooling by radiators or other means.

Utilization of urine and other liquid waste for cooling.

Auxiliary Power

Table XXVI presents a summation of the estimated weight and power requirements for three mission durations, viz., 24 hours, 48 hours, more than 48 hours. These particular durations were chosen to reflect differences in the weights of the type of subsystems selected. For example, the minimum life support subsystem selected can operate for a maximum of 24 hours. For longer periods, it is more economical weightwise to use another type subsystem which has a higher initial weight, but requires fewer stores in pounds per hour (see table XXVI). Similarly, it appears to be more economical to use batteries for missions up to 48 hours, and for longer periods the use of a fuel cell is contemplated. The weights indicated for the secondary power source are computed on the basis of a battery efficiency of 24 watt-hours per pound for durations of 24 and 48 hours. The tare weight for fuel cells shown for durations exceeding 48 hours, includes 60 pounds of batteries for backup purposes. The comparison between batteries and the fuel cell is presented as a function of mission duration in figure 29.

The power requirements for the navigation and guidance equipment is assumed to be independent of mission duration, as previously indicated.

This study has been based on the necessity for all the measurements shown in table XXII. In the event that subsequent research indicates that greater dependence can be placed on the man, then the instrumentation required may consist only of items 2, 5, 7, 8, and 9 in table XXV. A simplified version of the computer (item 6) would also be required. It is estimated that this equipment would weigh 110 pounds and require 180 watts of power.

The weight and power estimates presented in this preliminary feasibility study are based on the best information presently available. Some development effort will be required on items 1 through 6 in table XXV.

TABLE XXII.- INSTRUMENT REQUIREMENTS FOR LUNAR LANDING VEHICLE

Measurement	Lunar orbit		Deorbit		Landing		Launch		Rendezvous	
	Range	Accuracy	Range	Accuracy	Range	Accuracy	Range	Accuracy	Range	Accuracy
h	50 naut. miles	±1 naut. mile	50 naut. miles to 500 ft	±500 ft ±20 ft	500 ft to TD	±2 ft	50 naut. miles 10 naut. miles	±1 naut. mile ±100 ft		
$\dot{h}$	200 ft/sec	±1 ft/sec	1,000 ft/sec		200 ft/sec	±4 ft/sec	2,000 ft/sec	±6 ft/sec		
T	7,200 sec	±1/2 sec	1,500 sec	±1 sec						
V_{TC}	5,500 ft/sec	±6 ft/sec	5,500 ft/sec	±6 ft/sec	50 ft/sec	±6 ft/sec	6,000 ft/sec	±6 ft/sec		
V_{TT}	0	0	700 ft/sec	±6 ft/sec	50 ft/sec	±6 ft/sec	500 ft/sec	±6 ft/sec		
θ		±0.05°		±0.05° ±.10°		±2°		±0.05°		
φ		±0.05°		±0.05° ±.10°		±2°		±0.05°		
ψ		±1.0°	Direct abort	±1.0° ±2.0°		±2°		±0.2°		
r			500 naut. miles 10 naut. miles	±1.0 naut. mile ±1/4 naut. mile			500 naut. miles 20 naut. miles	±1,000 ft ±100 ft	20 to 10 naut. miles 10 to 1/4 naut. miles	±100 ft ±50 ft
$\dot{r}$			2,000 ft/sec	±6 ft/sec			5,000 ft/sec 500 ft/sec	±20 ft/sec 8.5 ft/sec	1,000 to 300 ft/sec 300 to 0 ft/sec	±10 ft/sec ±8.5 ft/sec
α			0° to 120°	±1/2°			120°	±1/2°	90°	±1/2°
β			±90°	±1/2°			±90°	±1/2°	90°	±1/2°
$\dot{\alpha}$			60°/sec	0.05°/sec			60°/sec	0.05°/sec	60°/sec	0.05°/sec
$\dot{\beta}$			60°/sec	0.05°/sec			60°/sec	0.05°/sec	60°/sec	0.05°/sec

TABLE XXIII.- SUBROUTINES CONSIDERED FOR LUNAR LANDING SCHEME

SR-A. Orbital elements for specified position and velocity

SR-B. Kepler orbit to specified point

SR-C. Pericenter determination

SR-D. Kepler orbit time of flight

SR-E. Velocity at specified point on given Kepler orbit

SR-F. Rendezvous thrust initiation

SR-G. Rendezvous thrust termination

SR-H. Free-fall velocity

TABLE XXIV.- EXPLANATION OF SUBROUTINES CONSIDERED

SR-A. Computes for a given position and velocity the in-plane orbital elements for the Kepler orbit in the form $ax + bz + c = (x^2 + z^2)^{1/2}$.

SR-B. Computes the in-plane flight-path angle required for given present position and speed (magnitude of velocity) to yield a Kepler orbit passing through a specified aim point. In the case where V is less than the minimum energy speed, this is indicated and the program gives the real part of the answer.

SR-C. Computes the coordinates of pericenter for given orbital elements.

SR-D. Computes the time of flight along a Kepler orbit, given by its elements, between two specified points.

SR-E. Calculates the in-plane velocity components at a given point on a Kepler orbit specified by its elements.

SR-F. Assumes the rendezvous vehicle is moving essentially colinearly with a known target. The rendezvous vehicle thrust initiation time is computed on the basis of the relative range and range rate so that, for nominal performance, relative range and range rate will be nulled simultaneously.

SR-G. Accounts for deviations from nominal performance in determining the rendezvous vehicle thrust termination time so that an intercept at low relative velocity is achieved in a desired time. The relative velocity is also computed for final vernier correction.

SR-H. Free-fall velocity $V = V_o + (2ah)^{1/2}$.

TABLE XXV.- WEIGHT AND POWER SUMMARY FOR NAVIGATION

AND GUIDANCE INSTRUMENTATION

Parameter	Equipment	Weight, lb	Power, watts
(1) $h - \dot{h} (h \geqq 2{,}000 \text{ ft})$	Radar altimeter	25	75
(2) $h - \dot{h} (h \leqq 2{,}000 \text{ ft})$	Radar altimeter	15	---
(3) $V_{TC} - V_{TT}$	Doppler radar	16	40
(4) $r - \dot{r}, \alpha, \beta, \dot{\alpha}, \dot{\beta}$	Radar	54	200
(5) $\theta, \phi, h, \dot{h}, V_{TT}, V_{TC}$	Inertial platform	60	100
(6)	Computer	45	80
(7) p, q, r	Rate gyros	8	25
(8)	Pilot display	11	30
(9) $\alpha, \beta, \dot{\alpha}, \dot{\beta}$	Visual line of sight	5	2
Totals		239	552

TABLE XXVI.- SUMMATION OF WEIGHT AND POWER REQUIREMENTS

Area	24-hr mission		48-hr mission		Missions exceeding 48 hr	
	Weight	Power, watt-hr	Weight	Power, watt-hr	Weight	Power/day
Communications	20	480	40	960	20	480 watt-hr
Navigation and guidance	239	3,300	239	3,300	239	3,300 (total)
Life support[a]	100	1,200	158	2,400	[b]110 lb + 32 lb/day	1,200 watt-hr
Secondary power supply	225		300		270 lb + 4 lb/day	
	584		737		639 lb + 44 lb/day	

[a]Does not include man and suit estimated at 200 pounds per man.

[b]The 32-pound estimate includes 16 pounds per day of water to be used for cooling.

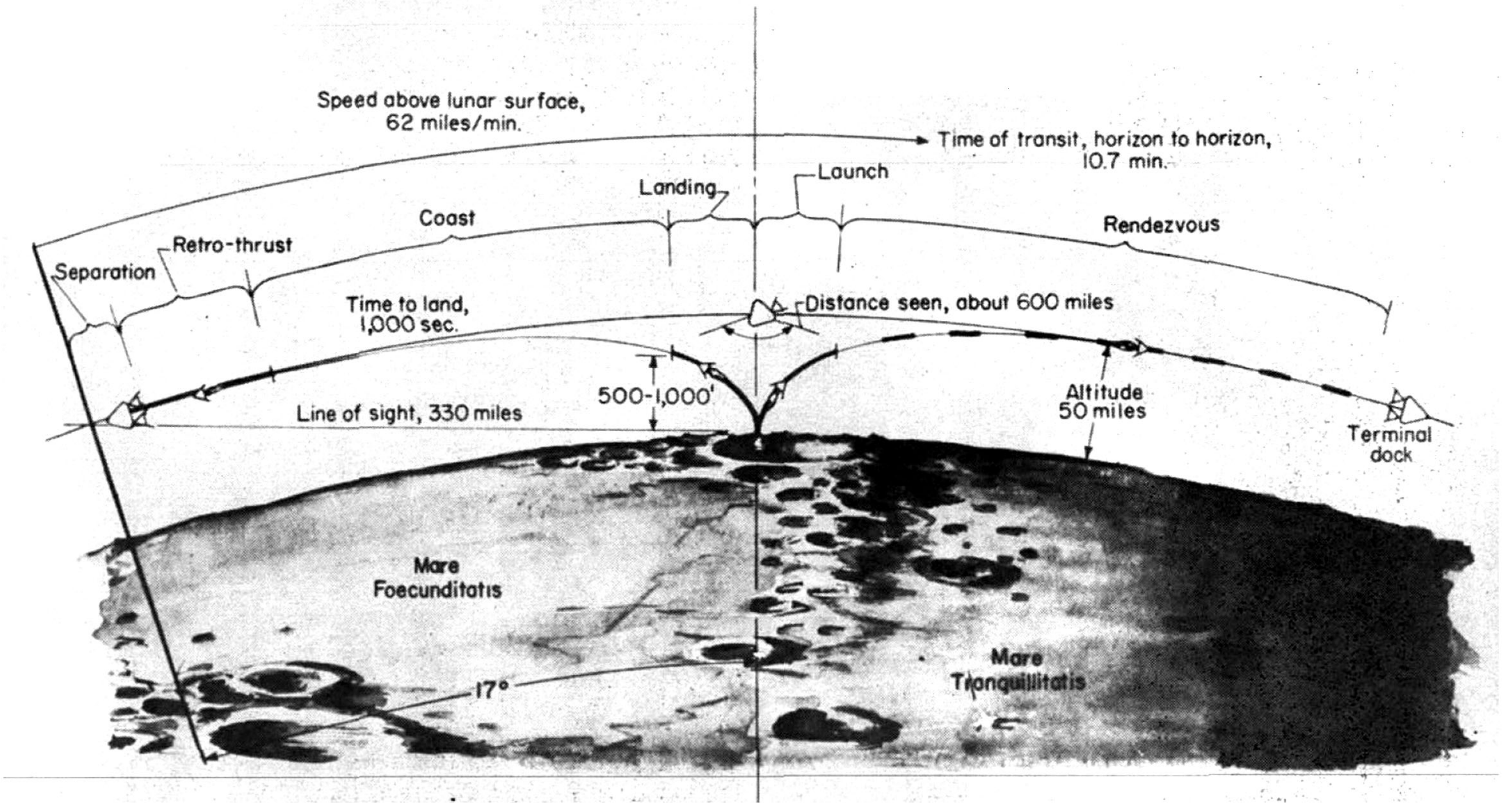


Figure 24.- Lunar-landing mission phases.

I. Mission Computer Task Flow Chart

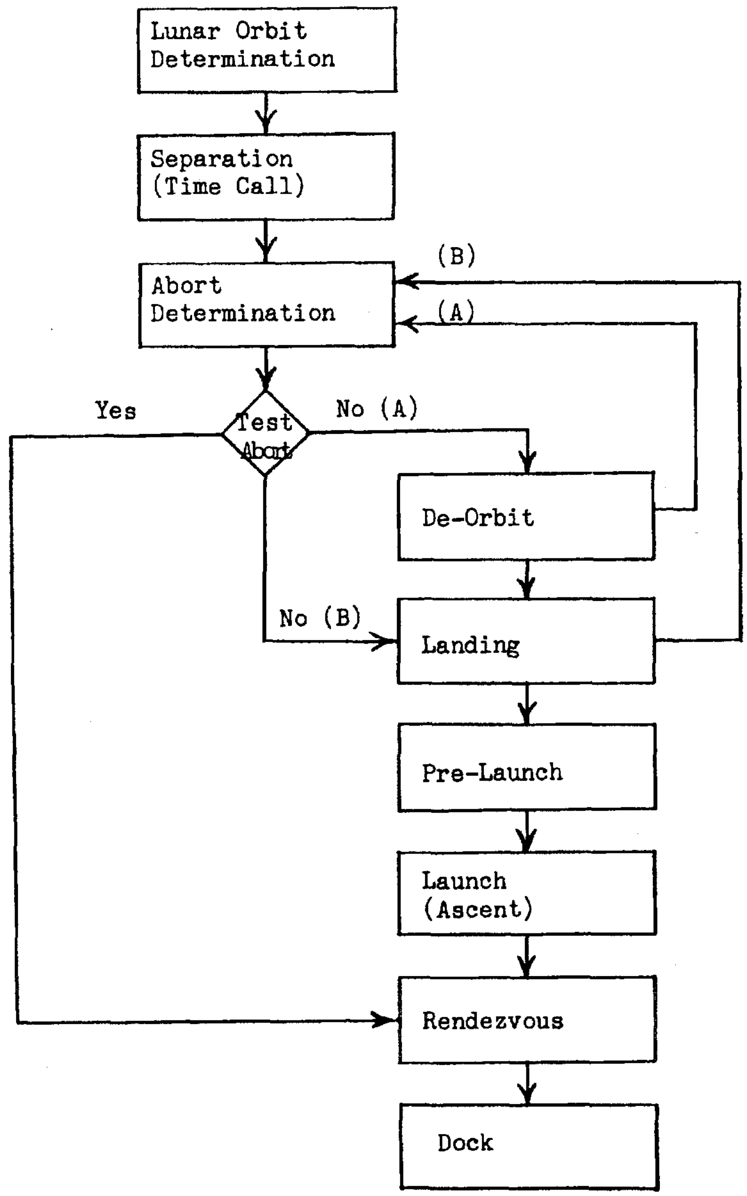


Figure 25.- Computer considerations for lunar landing using rendezvous at moon.

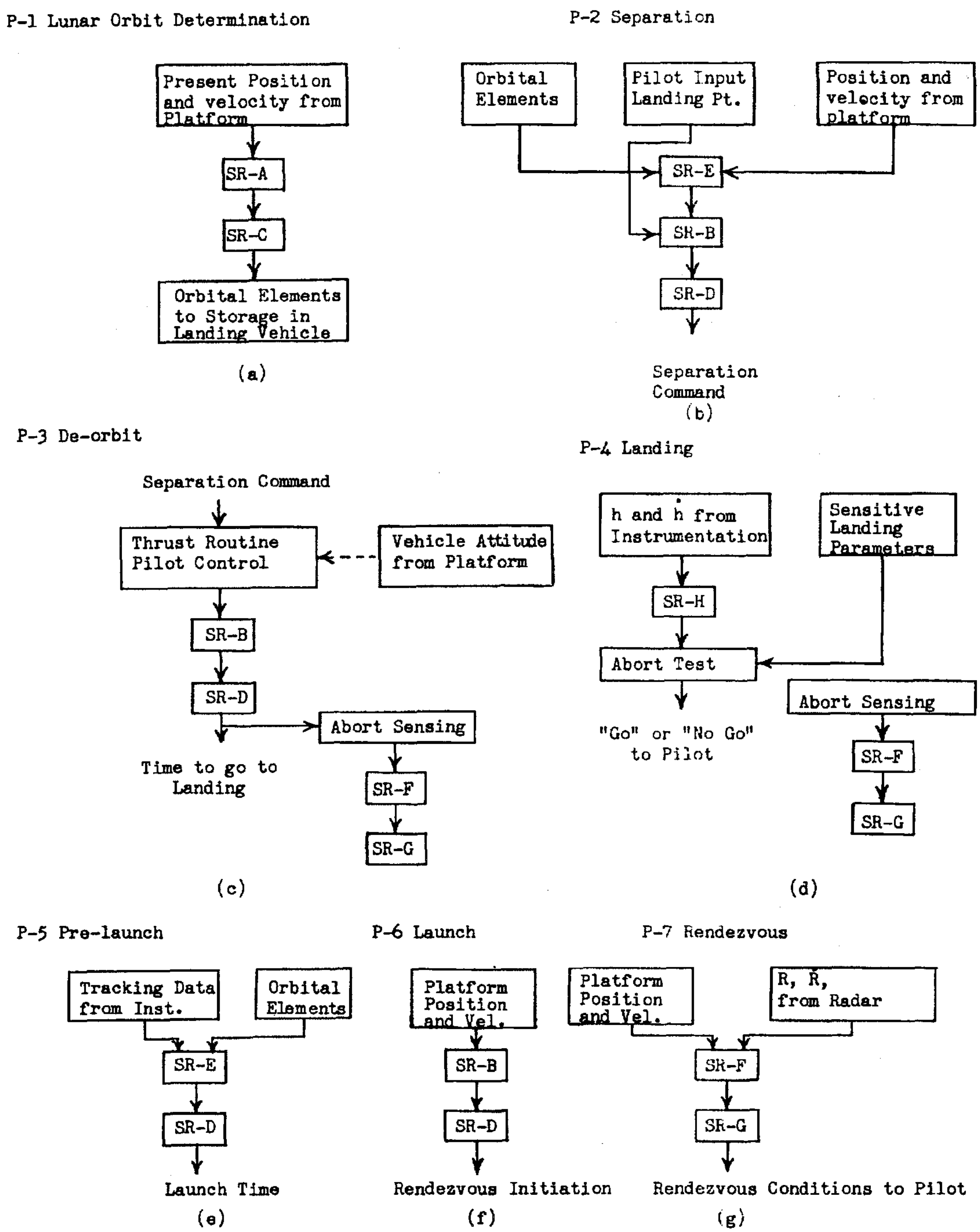


Figure 26.- Lunar-landing computer programs.

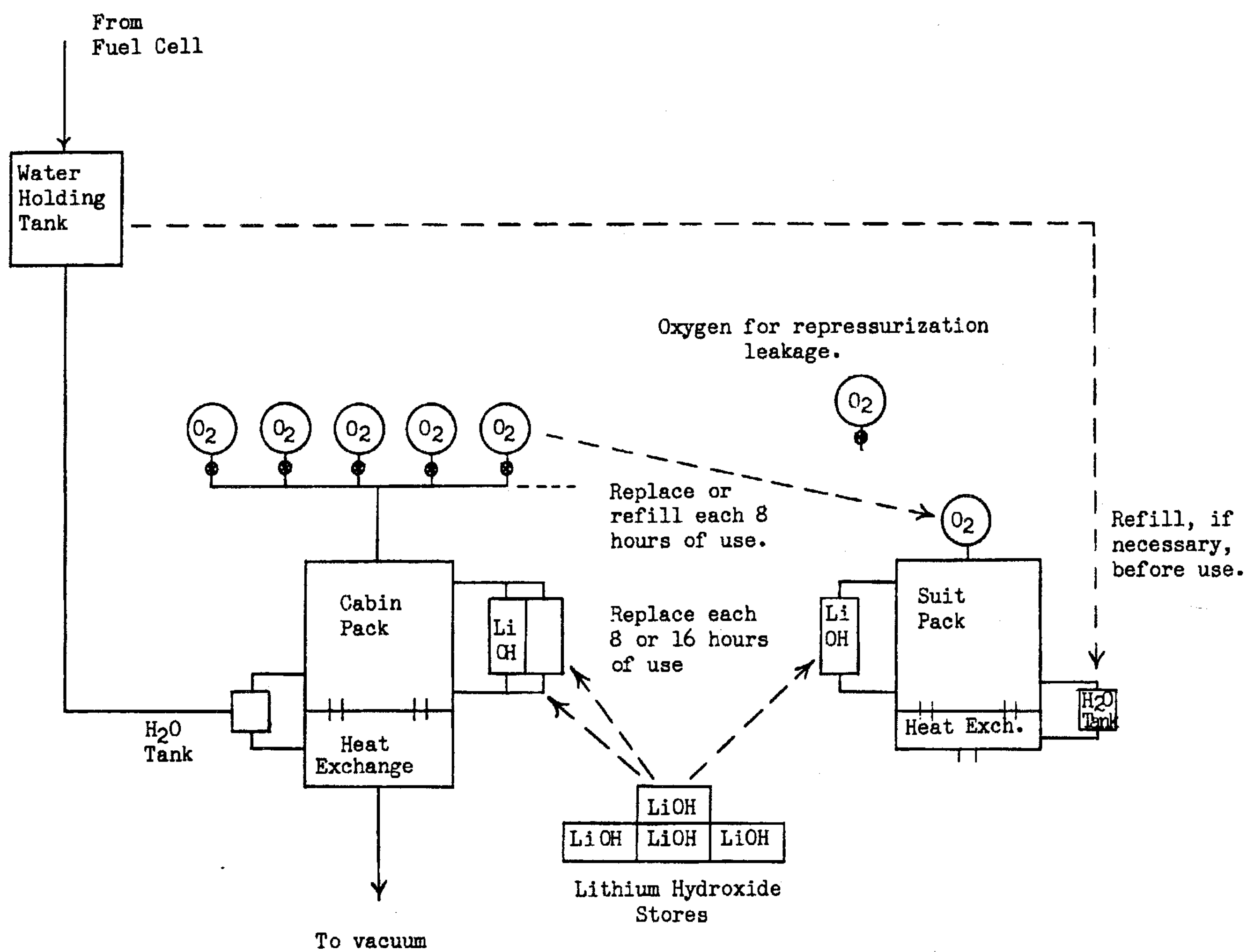


Figure 27.- Environmental control system.

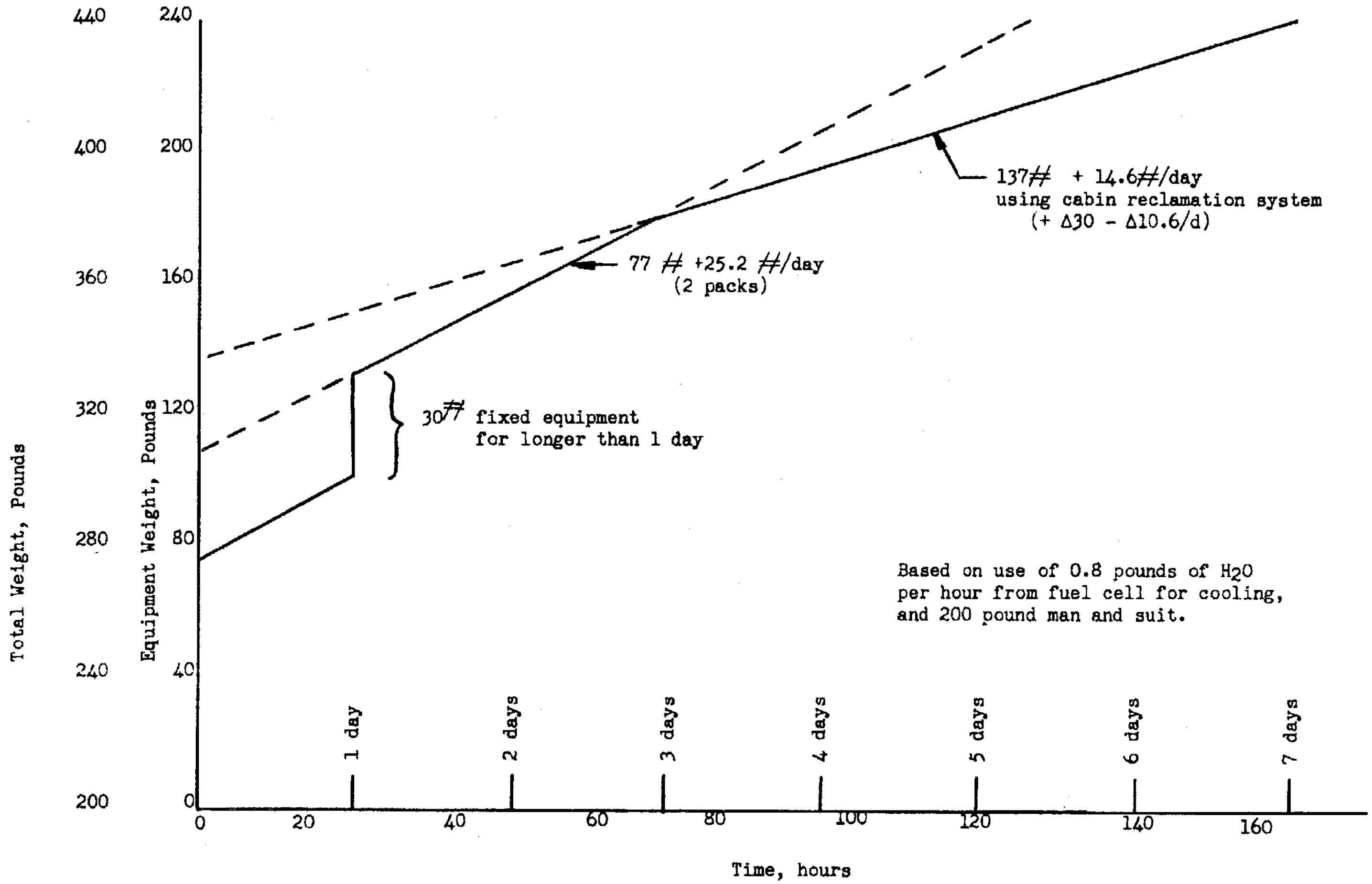


Figure 28.- Estimated weight of ECS vs. mission duration.

Batteries - 24 watt hrs/lb.

Power for descent and rendezvous maneuvers - 3780 watt hrs.

Power for period of lunar stay - 70 watt/hrs/hr.

$P_T = 3780 + (T_L - 4)\ 70$

$W = P_T/24$

Fuel Cell - 700 watt capacity - cell + controls W_T = 200 lbs.

backup batteries W_T = 70 lbs.

fuel + tankage = 1.2×10^{-3} lbs/watt hr.

$P_T = 3780 + (T_L - 4)\ 70$

$W = 270 + 1.2 \times 10^{-3}\ P_T$

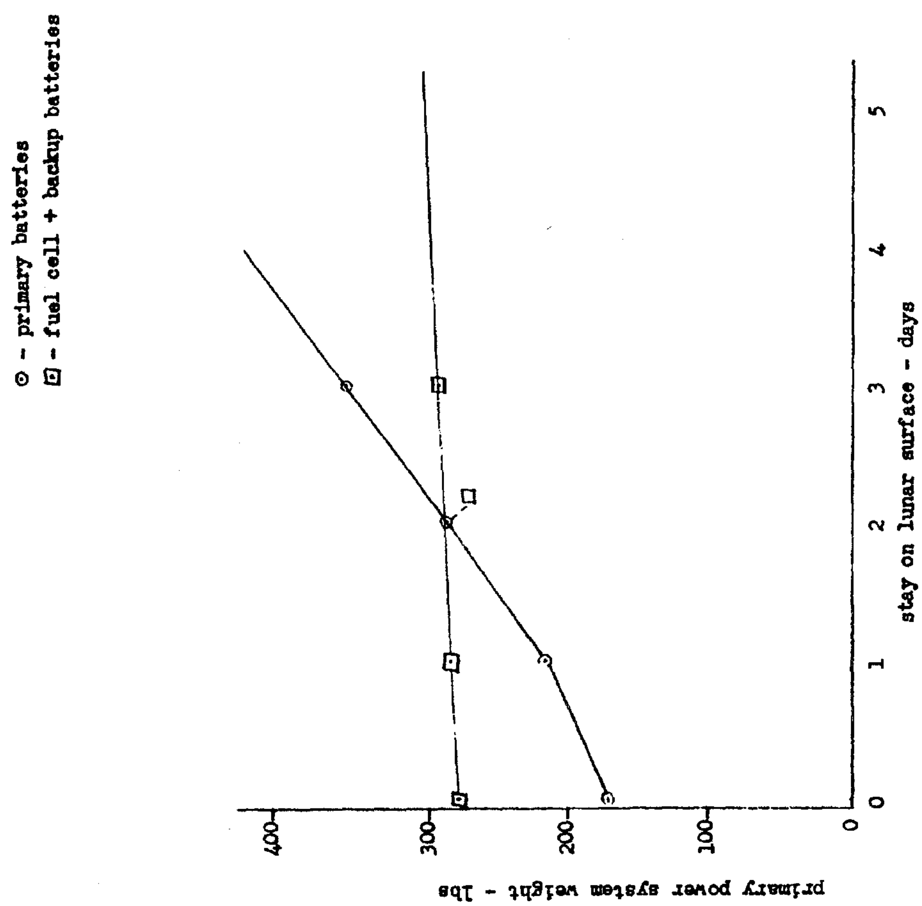


Figure 29.- Weight and power summary for navigation and guidance instrumentation.

THE RADIATION HAZARD

The Apollo spacecraft is assumed to be the basic vehicle for either the direct landing case or that with a separate landing vehicle. Thus the only difference in the cases regarding the problem of radiation occurs during the period from lunar deorbiting and letdown to rendezvous after lunar take-off. This period of initial landing as specified for Apollo is 26 hours. According to Dr. Trutz Foelsche (ref. 11) the general galactic cosmic radiation is 0.5 rem per week. Thus for the period involved the basic radiation would be 0.07 rem which is negligible relative to the nominally accepted dose of 5 rem for the total mission (Apollo specifications). Presuming no damaging radiation from the moon itself the landing phase entails no danger unless a solar event occurs.

Reference 11 indicates that with nominal shielding (2 g/m^2) the May 1959 low-energy high-flux flare had about 2,000 rep total flux and the February 1956 high-energy flare had about 350 rep total flux. The initial rates also from reference 11 were 200 rep per hour and 100 rep per hour, respectively. If the crew remained exposed on the moon surface for 26 hours, they would accrue about the entire dose of 2,000 rep or 350 rep, respectively, far in excess of the emergency dose allowed in the Apollo specifications of 100 rem. Clearly then some action and/or shielding is required when on the moon surface. First, if the landing is on the dark side of the moon (possibly in earth light) the moon itself provides a sensible shield for the basic radiation onslaught. Otherwise, the moon explorers have two courses, first to seek shelter on the moon surface in caves or digging in (2 feet of lunar material should adequately shield), and second by returning to the the orbiting spacecraft to the sanctity of its shielding. The former course of action could be initiated on landing as a precaution but may seem senseless preparation for a 1-day stay.

For the situation of returning to orbit, 2 hours (one orbit) should be a maximum time required before launch plus up to 30 minutes to rendezvous. At the initial rates previously noted a dose of 500 rep and 250 rep for the May 1959 and February 1956 flare, respectively, could be encountered. This assumes that no action is taken until the radiation reached the moon. These doses are, of course, far in excess of the 100 rep emergency dose specified in the Apollo specifications. Shielding equivalent to 25 grams of water per square centimeter, as noted in reference 11, would adequately protect from this hazard. Such shielding would be an excessive burden for the lander and, for that matter, for the basic Apollo vehicle. Prior prediction of solar activity must therefore be an inherent part of the lunar landing scheme. As noted in reference 11, predictions of the absence of major solar events appear possible for times of 2 to 4 days. A prediction for the 1-day-landing period of initial moon landings seems possible and thus the landing could be postponed or aborted in the case of predicted activity. Predictions for longer periods, as for the entire initial flight (7 days), are not reliable unless the flight is made during solar minimum or quiet periods.

It would appear then that dependence on predictions for the 1-day-landing period is required. The landing being made or aborted based on the best criteria evolved by that time. For longer periods on the moon than the 1 day of initial flights, shelters on the moon must be made or found for the crew's safety.

WEIGHTLESSNESS

The current Apollo concept accepts the condition of weightlessness for long periods of time as a necessity of or concession to vehicle weight restraints. The only known practical approach to applying artificial gravity is by rotation which in addition to the requirement of large, cumbersome vehicles constitutes in itself a psychophysiological factor disturbing to man. The fundamental physiological problems of weightlessness are the loss of muscle tone or even atrophication and similar degeneration of the cardiovascular system. Without gravity, artificial or otherwise, the only known procedure to allay these effects is to exercise by muscle manipulation and massage. Provisions for this must be available in the spacecraft and on the moon surface if one-sixth of normal gravity is not sufficient for the problem.

During the actual landing phase on the moon the times involved are short and the lack of full weight is of no consequence to the well being of the crew regardless of the vehicle used for landing. The difference then between using the basic vehicle to land or a separate landing vehicle lies only in the fact that if special equipment is required for exercise it may have to be duplicated or transferred for the latter case. Finally though, if the moon stay is only a day (26 hours is specified for Apollo), no special exercise equipment is required in the landing craft. Man has sustained such periods of weightlessness in flight or simulated in water with no incapacitating effects or marked difficulties. Thus for initial landings of short duration no duplication or transfer of special equipment is necessary and the separate landing craft does not impose added problems or weight from the standpoint of weightlessness.

REFERENCES

1. Tolson, Robert H.: Effects of Some Typical Geometrical Constraints on Lunar Trajectories. NASA TN D-938, 1961.

2. Michael, William H., Jr., and Crenshaw, Jack W.: Trajectory Considerations for Circumlunar Missions. IAS Paper No. 61-35, 1961.

3. Michael, William H., Jr., and Tolson, Robert H.: Three-Dimensional Lunar Mission Studies. NASA MEMO 6-29-59L, 1959.

4. Tolson, Robert H.: Some Considerations for Terrestial and Lunar Rendezvous Orbits. For internal distribution, 1961.

5. Queijo, M. J., and Riley, Donald R.: A Fixed-Base-Simulator Study of the Ability of a Pilot to Establish Close Orbits Around the Moon. NASA TN D-917, 1961.

6. Eggleston, John M., and Dunning, Robert S.: Analytical Evaluation of a Method of Midcourse Guidance For Rendezvous With Earth Satellities. NASA TN D-883, 1961.

7. Eggleston, John M.: Some Recent Studies on the Optimum Time to Rendezvous. Proposed TN.

8. Barker, C. L., Jr., and Straley, W. H.: Rendezvous by the Chasing Technique. ABMA Rept. DSP-TM-15-59, Oct. 1959.

9. Houbolt, John C.: Problems and Potentialities of Space Rendezvous. Paper presented at Louvecinnes, France. June 19-21, 1961.

10. Brissenden, Roy F., Burton, Bert B., Foudriat, Edwin C., and Whitten, James B.: Analog Simulation of a Pilot-Controlled Rendezvous. NASA TN D-747, 1961.

11. Foelsche, Trutz: Notes on Radiation Environment and Shielding Requirements in Space. (Presented to NASA Advisory Committee on Nuclear Systems, NASA Marshall Center, Huntsville, Alabama, May 9-10, 1961.)

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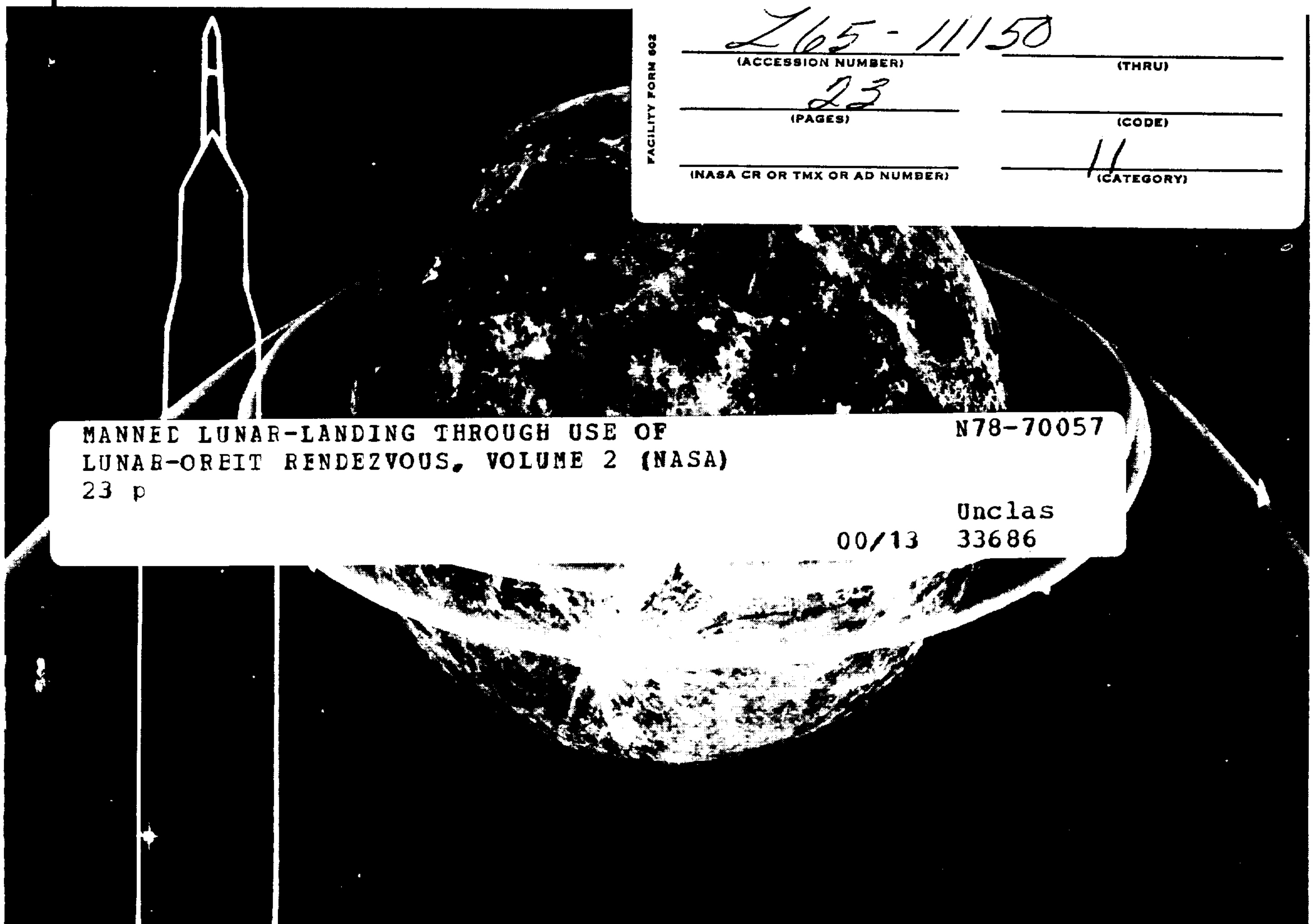
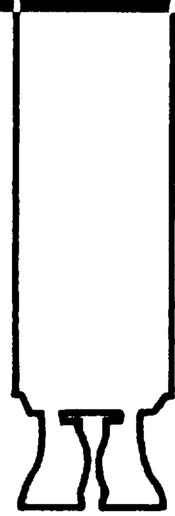
 VOLUME II





MANNED LUNAR-LANDING through use of LUNAR-ORBIT RENDEZVOUS

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PART III

A P P E N D I X

ADDITIONAL FACILITIES AND STUDIES (UNDERWAY AND PLANNED) IN SUPPORT OF A MANNED LUNAR LANDING

This appendix contains short descriptions of a number of facilities and studies not described elsewhere (both underway and planned) which will contribute to the achievement of a manned lunar landing. It will be noted that emphasis has been given in this work to proof of the thesis that man with visual and in some cases supplementary electronic sensing is capable of handling the various operations pertinent to a manned lunar mission.

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Establishment of a Lunar Orbit

Simulation study of pilot's control from hypersonic velocity to a desired lunar orbit.- Although this phase is not peculiar to the lunar-rendezvous technique for landing a man on the moon, it is included since it influences the landing, abort, and take-off phases. For a realistic study of this phase, we must establish the accuracies with which a human can establish the Ephemeris of a lunar orbit; this was part of the simulation study made.

After appropriate midcourse corrections, the space vehicle will be on a hyperbolic trajectory approaching the moon. Basically the problem of establishing a close circular orbit consists of application of retro-thrust in such a manner as to make the point of closest approach occur at a selected altitude above the lunar surface, and simultaneously modifying the velocity to the magnitude and direction required for a circular orbit. Presuming that the vehicle is in the proper orbit plane, the parameters which determine the orbit characteristics are altitude, radial velocity, and circumferential velocity. The desired circular orbit conditions are zero radial velocity, some selected altitude, and a circumferential velocity equal to circular orbital velocity at that altitude. These three parameters therefore must be measured quite carefully and displayed to the pilot, so that he can apply any required correction through his control of thrust and vehicle orientation.

In order for a pilot to control the orbit characteristics in the simulator, he was presented a display indicating the vehicle orientation and attitude rates in addition to the three orbit parameters already mentioned. The pilot was then given the task to apply correct thrust to establish an orbit.

It was assumed that a manned space vehicle was approaching the lunar surface on a hyperbolic trajectory which would have a point of closest approach at an altitude of about 56 miles and a velocity at that point of 8,500 feet per second. The pilot's task was to establish a circular orbit at a 50-mile altitude. The pilot was given control of the thrust (along the vehicle longitudinal axis) and torques about all three body axes.

The information display given to the pilot was a hodograph of the vehicle rate of descent and circumferential velocity, an altimeter, and vehicle attitude and rate meters. The general procedure used in the investigation was to permit the pilots to become familiar with the instrumentation, controls, and vehicle dynamics by flying a simple "nominal" trajectory for which the operating mode was specified. The pilots had no difficulty flying this nominal trajectory, and used only about 2 percent more fuel than theoretically required to accomplish the task as shown in figure 1. The results of the investigation showed that the pilots soon became adept at flying the simulator, and could manage "off-nominal" trajectories with little or no difficulty. See figure 2. The indicated fuel consumption generally was about 1 to 3 percent of the initial vehicle mass more than that required by use of a two-impulse Hohmann maneuver. The results of this simulator study have been published in NASA TN D-917 by Queijo and Riley.

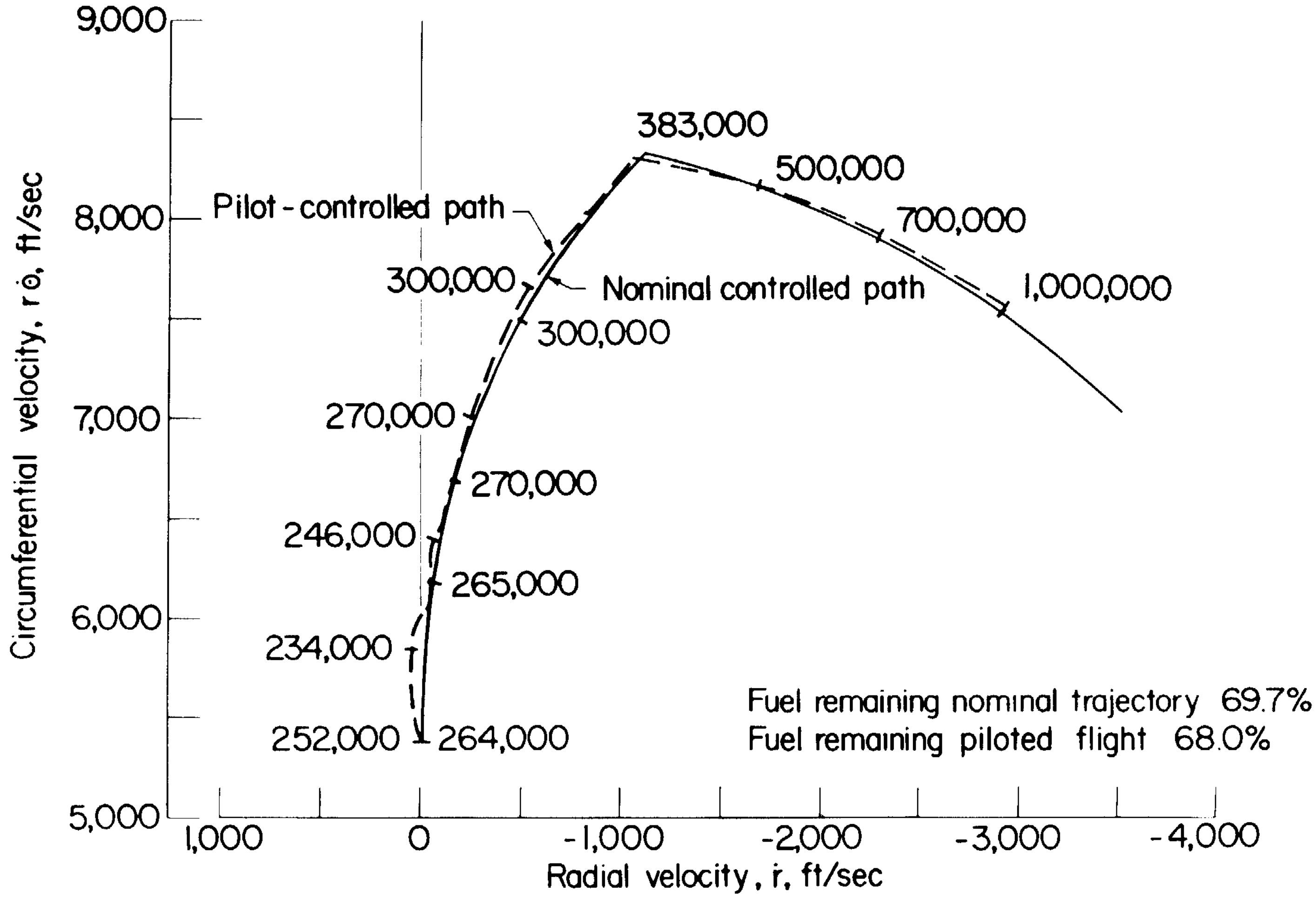
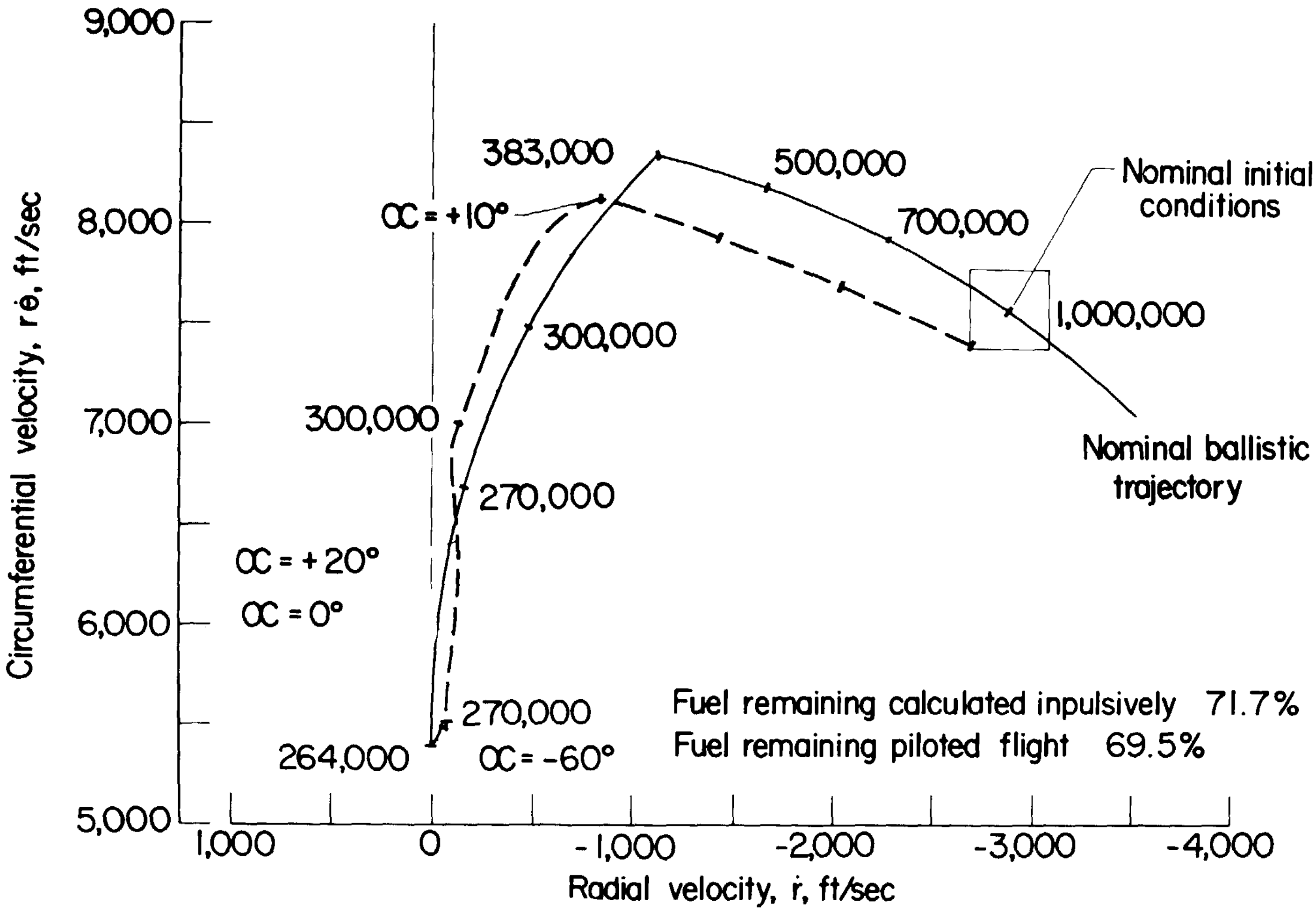


Figure 1.- Typical flight starting with nominal ballistic trajectory. L-1303



L-1302

Figure 2.- Flight starting with $r\dot{\theta}$ and $\dot{r}$ each 200 ft/sec less than nominal.

Lunar Letdown Phase

Simulation study of visual control of a vertical descent.- A preliminary study, using a visual simulation, has been completed to establish a pilot's ability to judge and control the rate of descent during a vertical approach from a lunar orbit.

The vertical descent phase of a lunar letdown to the moon's surface was controlled by the pilot in alternate periods of free fall and rocket braking. The motion in descents, from 125,000 feet to 25,000 feet and from 25,000 feet to 2,000 feet, was simulated with visual cues provided by coupling an analog computer to servo-drive an optical projector. Rate of descent was seen as growth of the projected lunarscape features. The pilot was seated in a fixed-base cockpit viewing the lunarscape as through a hatch. A photograph, figure 3, shows the equipment in operation.

Results of the tests are shown in figure 4. The pilots controlled the rate of descent relatively near the human threshold of perception of descent rate, resulting in an excessive use of fuel.

Further studies are planned utilizing a paired set of zoomar projection lenses to obtain a wider range of simulated altitude. These tests will also include visual aids to allow the pilot to measure descent rates and to predict the near-optimum time to initiate rocket firing for efficient control of the letdown maneuver.

Helicopter tests to establish a human's ability to control position during a vertical descent.- Autorotation flights were made using a helicopter descending from 10,000 feet over marshland near Wallops Island as shown in figure 5. Radar position information was then utilized to determine the cone of position control the pilot generated during his descent. One set of flights has been made and one set is scheduled for this month. Upon completion of the vertical phase, similar tests are contemplated for a slant-approach to landing technique.

Instrumented control study of a lunar letdown.- A preliminary simulation study of a pilot controlling a slant-approach to a lunar landing area was made with an all-instrument display. This study was completed primarily to obtain the initial conditions of velocity, altitude, mass, and relative position of the parent orbiting vehicle for a study of the problems of aborting the landing at any time during the approach. A future study, section 2(d), will present visual cues to the pilot to determine the minimum instrumentation requirements for this task.

The simulation maneuvers to touchdown were controlled by a pilot seated in a fixed-base cockpit with an instrumented display of altitude, vertical velocity, vehicle attitude, and flight-path velocity. The pilot accomplished the maneuvers by controlling vehicle attitude and thrust.

In control, an initial thrust period was used to establish (from a lunar orbit) an approximate ballistic path to a landing sight about 45° around the moon's circumference. Final braking could be controlled down to about 2,000 feet of the moon's surface.

Results indicate that the pilot's choice is to use continuous thrust for the final braking maneuver and to establish a hover condition. The effects of engine thrust capability and specific impulse on fuel consumption is shown in figure 6. Fuel used during hover would consume about an additional 1.5 percent of the initial mass/min.

Simulation study of a piloted lunar letdown.- After establishing an initial lunar orbit, the pilot can modify the orbit characteristics by proper application of thrust. This might be desirable in order to pass the orbit over a specific point on the lunar surface or to establish a more precise circular orbit. The next phase is the departure of the lunar lander from the mother ship, with subsequent deorbit, letdown, and landing.

A six-degree-of-freedom, fixed-base analog simulator study is currently underway to determine the ability and efficiency of pilots to control the deorbit and letdown phase of the lunar landing, and the control and display requirements. Photographs of the control console used in the initial studies are shown in figure 7. The pilot is given control of thrust along the vehicle longitudinal axis, and moment control about all three body axes. The display as used in initial tests showed altitude, vertical velocity, circumferential velocity, vehicle angular rates, and vehicle attitude.

In the actual lunar landing vehicle the altitude, radial velocity, and tangential velocity components may be measured using onboard pulse doppler radar. Vehicle attitude and angular rates can be measured by use of an inertial table or, preferably, by visual observation of the lunar surface and horizon.

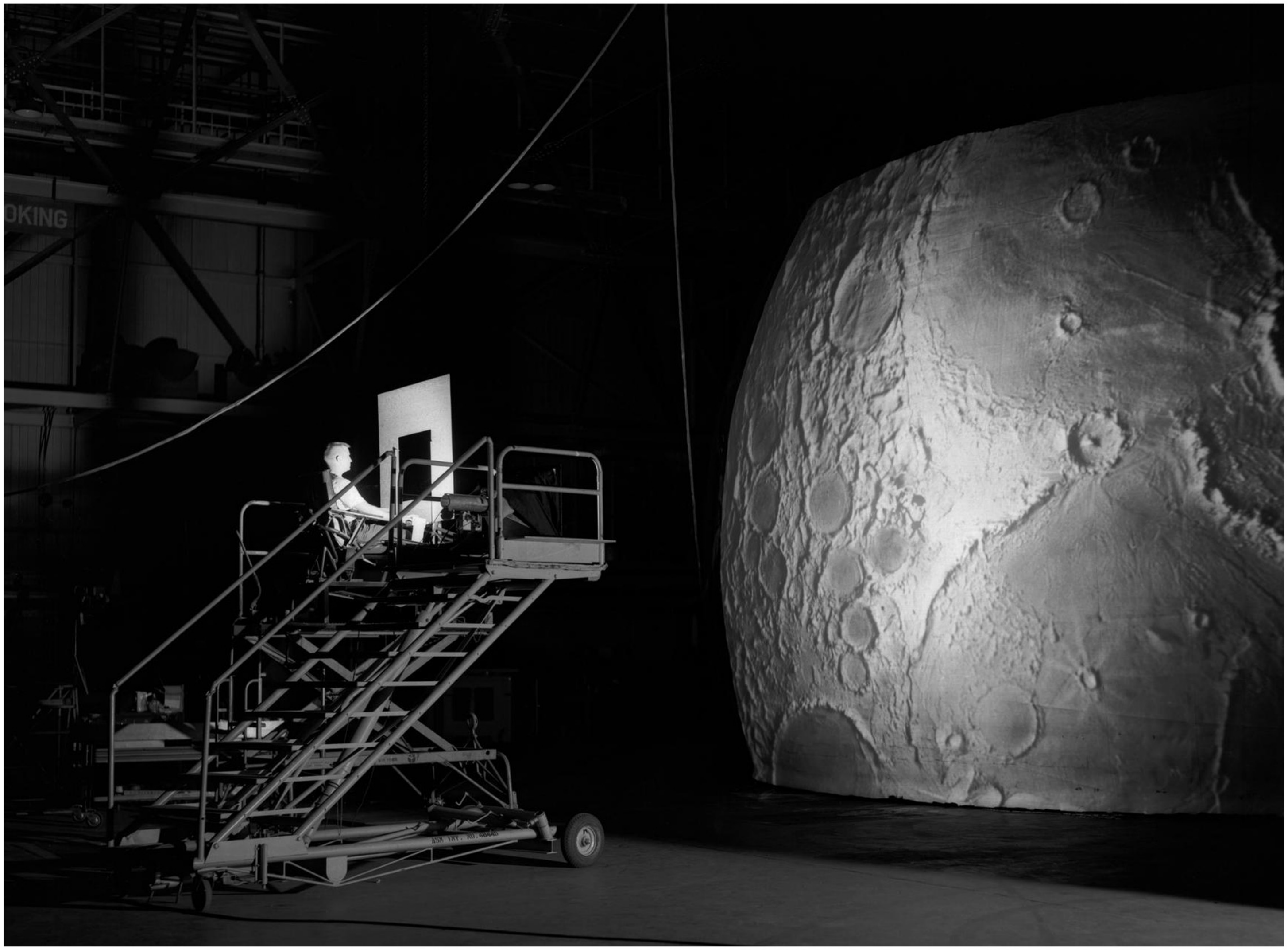


Figure 3.- Pilot-controlled vertical descent simulator. L-61-5040

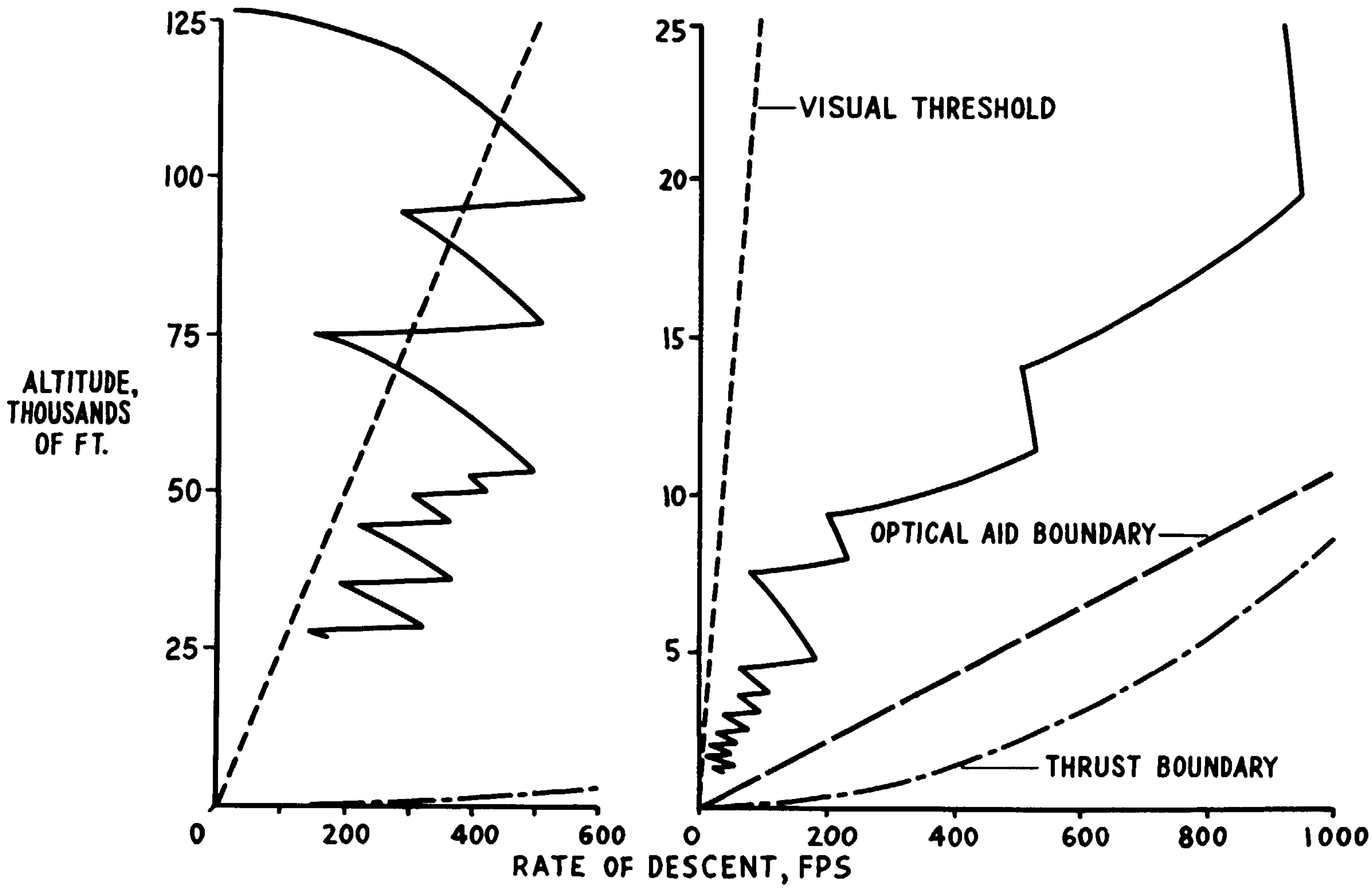


Figure 4.- Vertical descent simulator results. L-1310

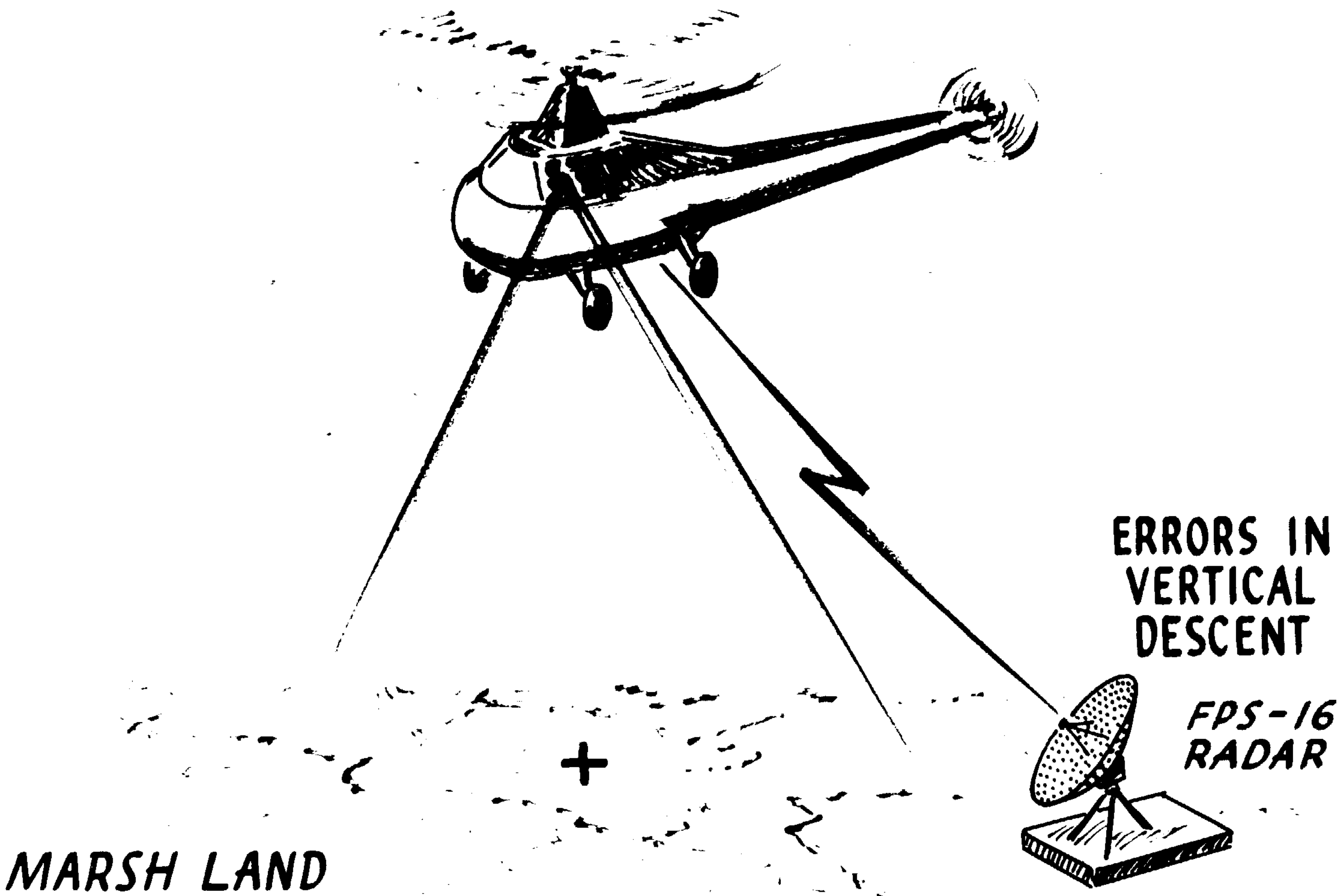


Figure 5.- Pilot-controlled vertical descent. L-1294

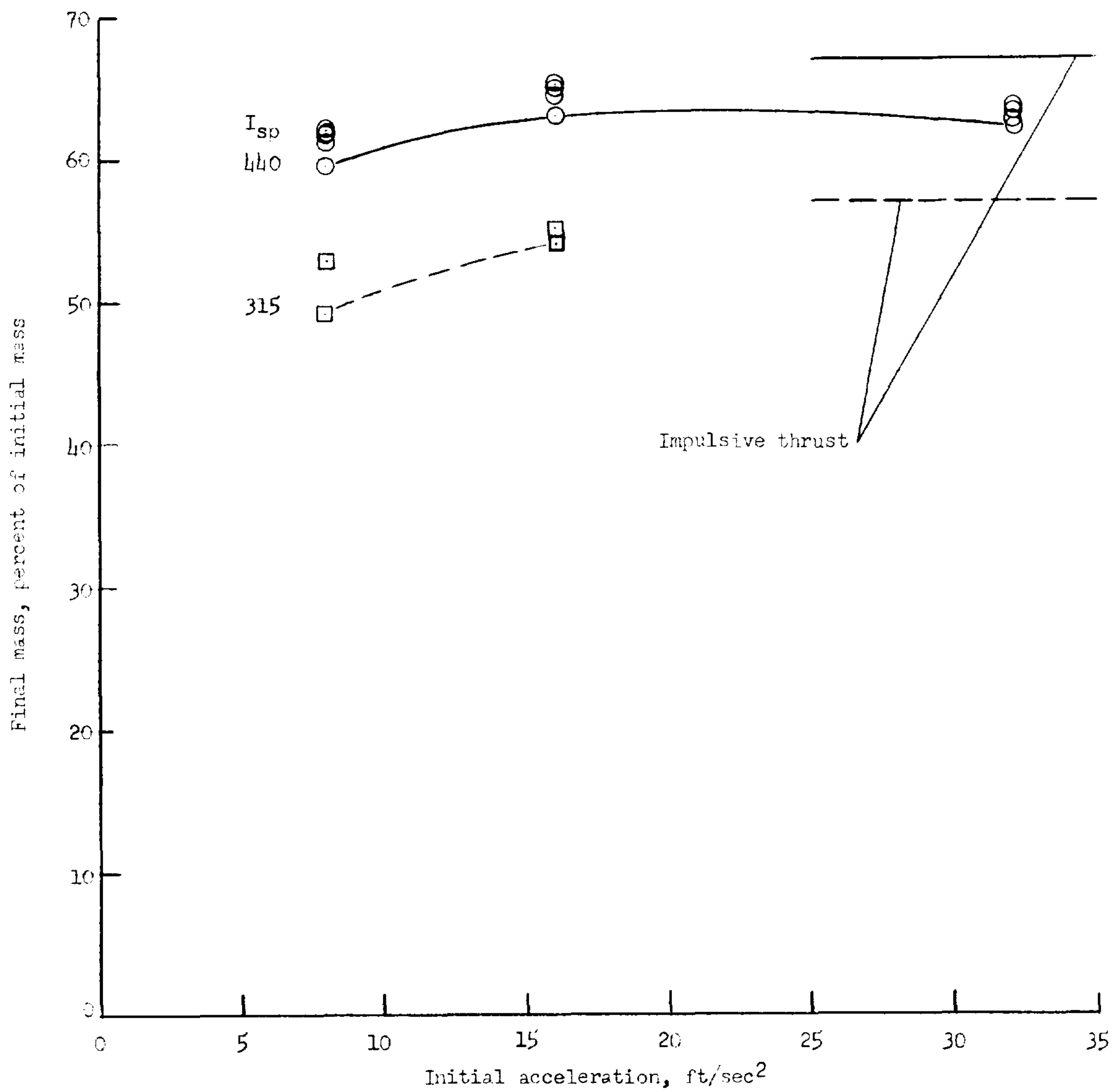


Figure 6.- Effect of thrust capability on fuel use. Landing requiring about 45° of moon's circumference. I_{sp} = 440 and 315.

(a) Instrument panel. L-61-6517

Figure 7.- Piloted lunar letdown console.

(b) Cockpit. L-61-6516

Figure 7.- Concluded.

Lunar Landing Phase

Human pilot control of a vertical landing.- To investigate (on earth) pilot control of a lunar landing, a means of supporting 5/6 of the mass must be obtained to simulate the lunar gravitational accelerations. A model servo control system is presently being built to check out a system to accomplish this. The system will be capable of supporting a man in a small (300-pound) research vehicle and with vertical descents up to 50 feet per second.

This design was made primarily as a pilot model for a full-scale research facility, but will permit preliminary research to be conducted on control effectiveness and thrust requirements for a pilot to land a rocket-powered vehicle.

An artist's drawing of the rig is shown in figure 8. Construction of this pilot model has started. Tests are contemplated to start in early spring of 1962.

Jet-blast effects during a lunar landing.- A preliminary investigation has been conducted to determine the effects of jet blast, at low ambient pressure, on a surface covered with loose particles. Tests were conducted using groups of from one to four nozzles at various cant angles and heights.

The results indicate the possibility of problems existing in this area ranging from visibility impairment to damage from impact of the surface particles with the vehicle.

A more extensive study, both theoretical and experimental, of jet-blast effects at low ambient pressure is being conducted. Both phases of the investigation utilize a single jet alined normal to a variety of ground surfaces. The theoretical and experimental studies will be compared to arrive at a scaling criterion to relate the small-scale tests to full-scale results on the lunar surface. The experimental studies will include correlary tests to determine the effect of various surface materials.

Coincident with the studies just discussed, an experimental investigation is planned to study means of minimizing jet-blast effects at low ambient pressure by use of canted multiple jets.

All of these investigations are intended to ultimately arrive at operating techniques to minimize the effects of jet blast on a lunar landing vehicle.

An artist's reproduction made from a high-speed motion picture of one of the runs is shown in figure 9.

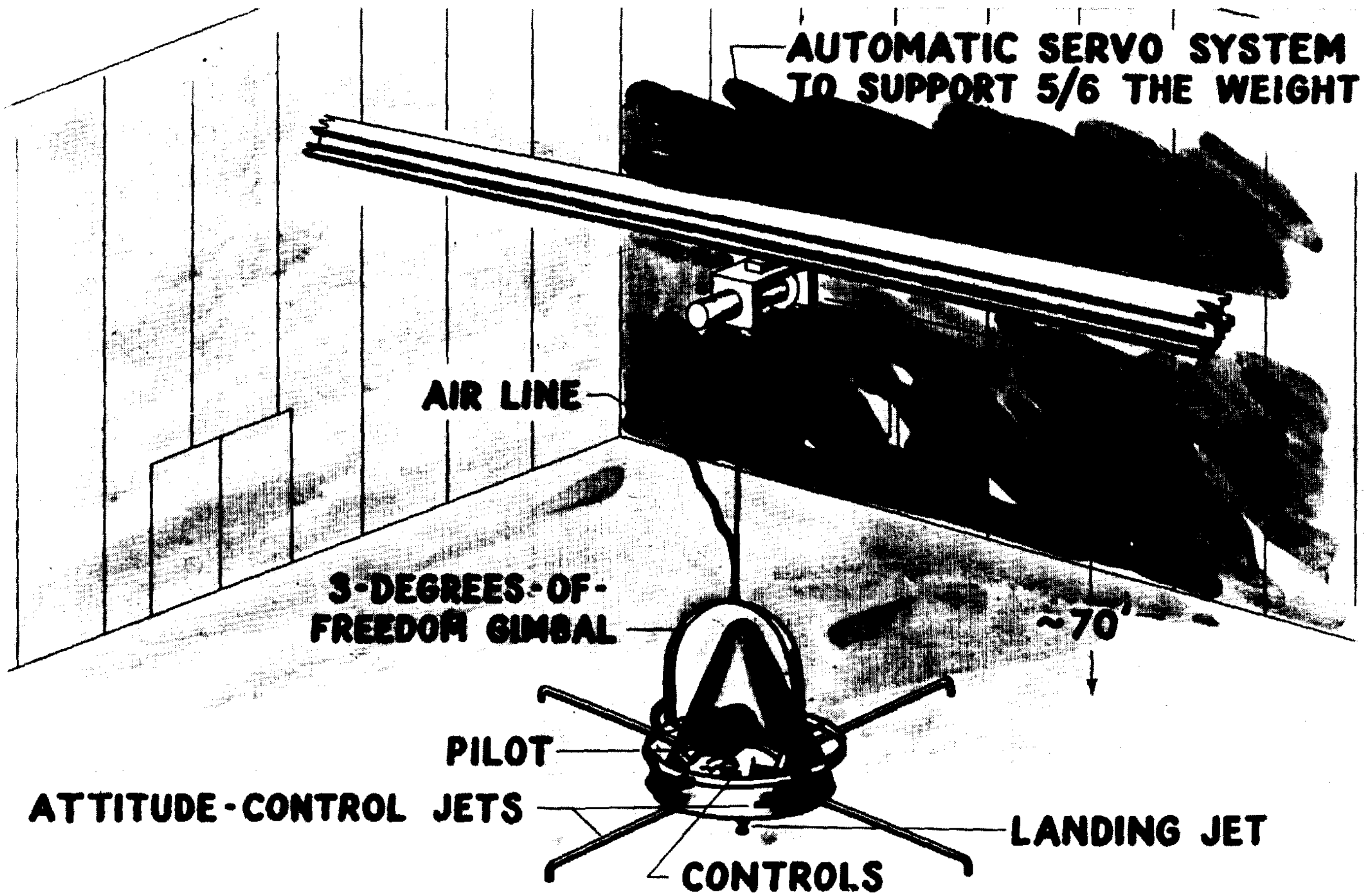


Figure 8.- Pilot model of vertical descent research facility. L-1295

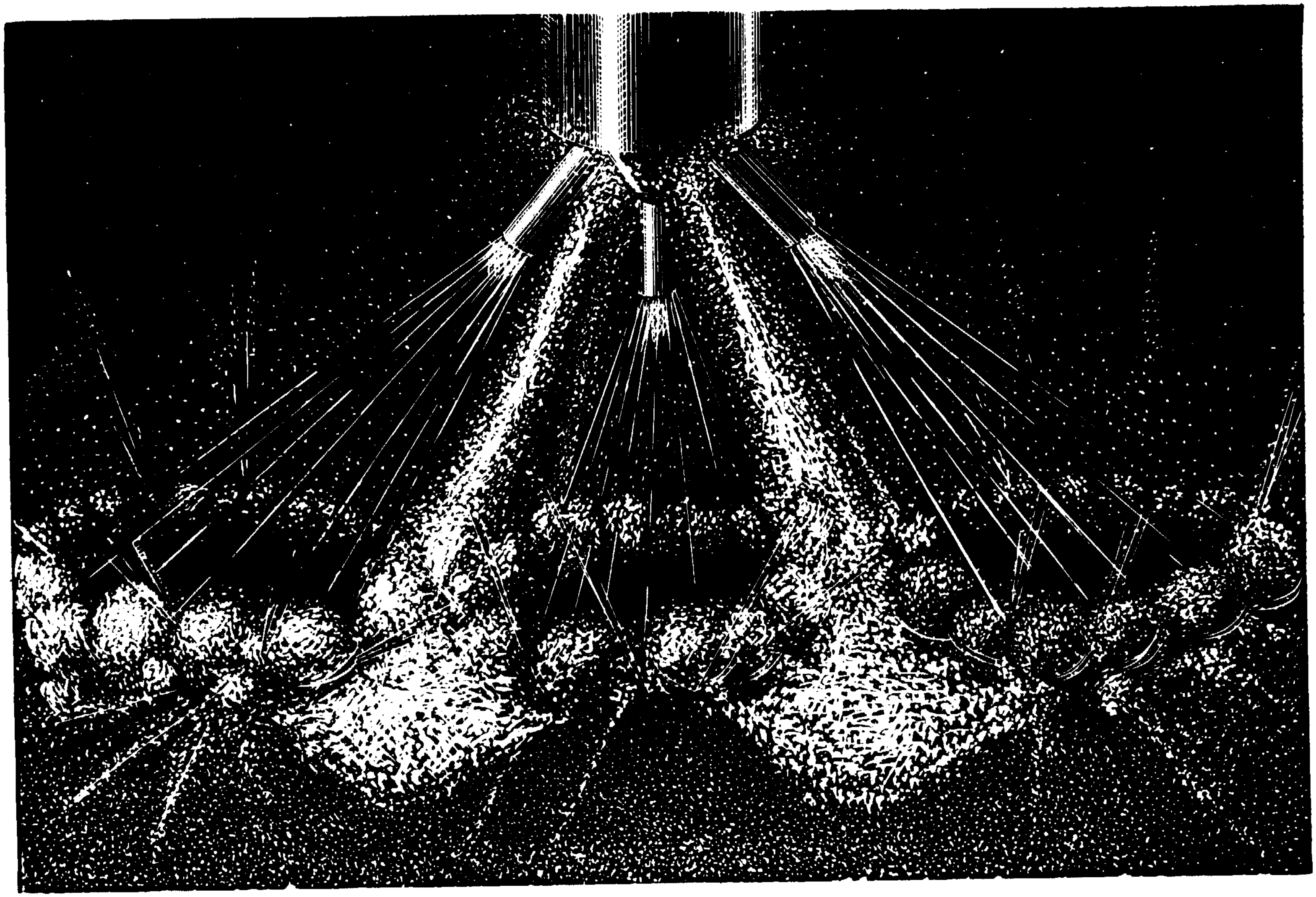


Figure 9.- Artist's reproduction of jet-blast effects.

Lunar Take-Off

Pilot control of a lunar take-off and rendezvous to a lunar orbiting station.- A simulation study of a lunar take-off, midcourse guidance, and rendezvous is to be made. This study will cover a large range of conditions such as relative positions of the vehicles at take-off, vehicle parameters, thrust levels, etc. This study will include a continuation of the abort studies described elsewhere. The pilot will utilize only visual cues for control during the initial part of the studies (instruments will be added if required).

The essential components of the simulator are shown in the composite photo of figure 10. A visual presentation of the stars and the orbiting station will be displayed by projectors. Motion cues during thrusting periods will be supplied by rotational accelerations of the 3-axis chair shown in figure 11. Motion cues during coasting periods will be supplied by a 3-axis-of-rotation projector drive unit not shown on the photo. An appropriate washout system will be incorporated to keep the pilot in a near-vertical seating position at all times.

Research is expected to begin in mid-Fall of 1961.

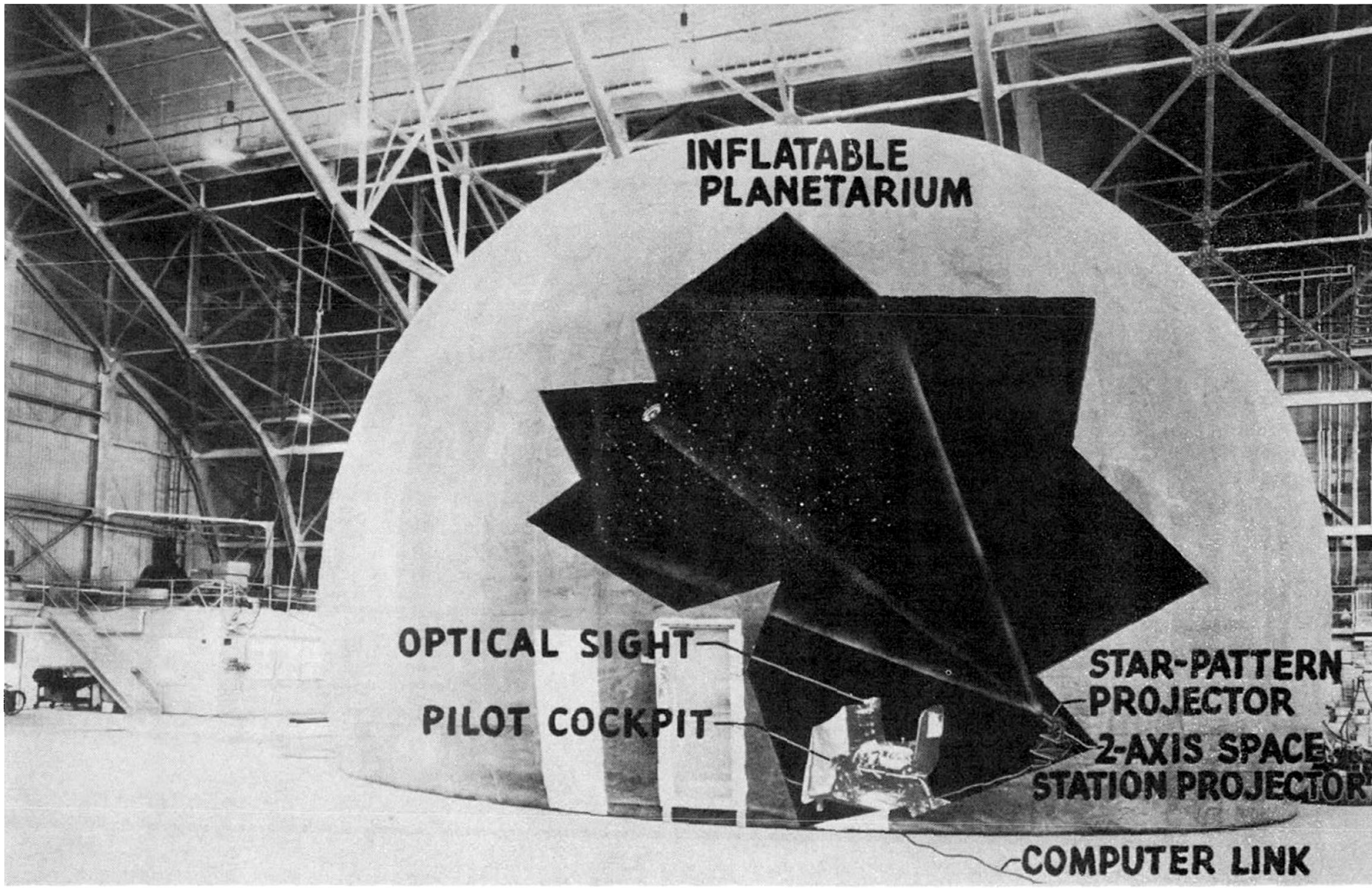


Figure 10.- Pilot-controlled lunar take-off simulator. L-1292

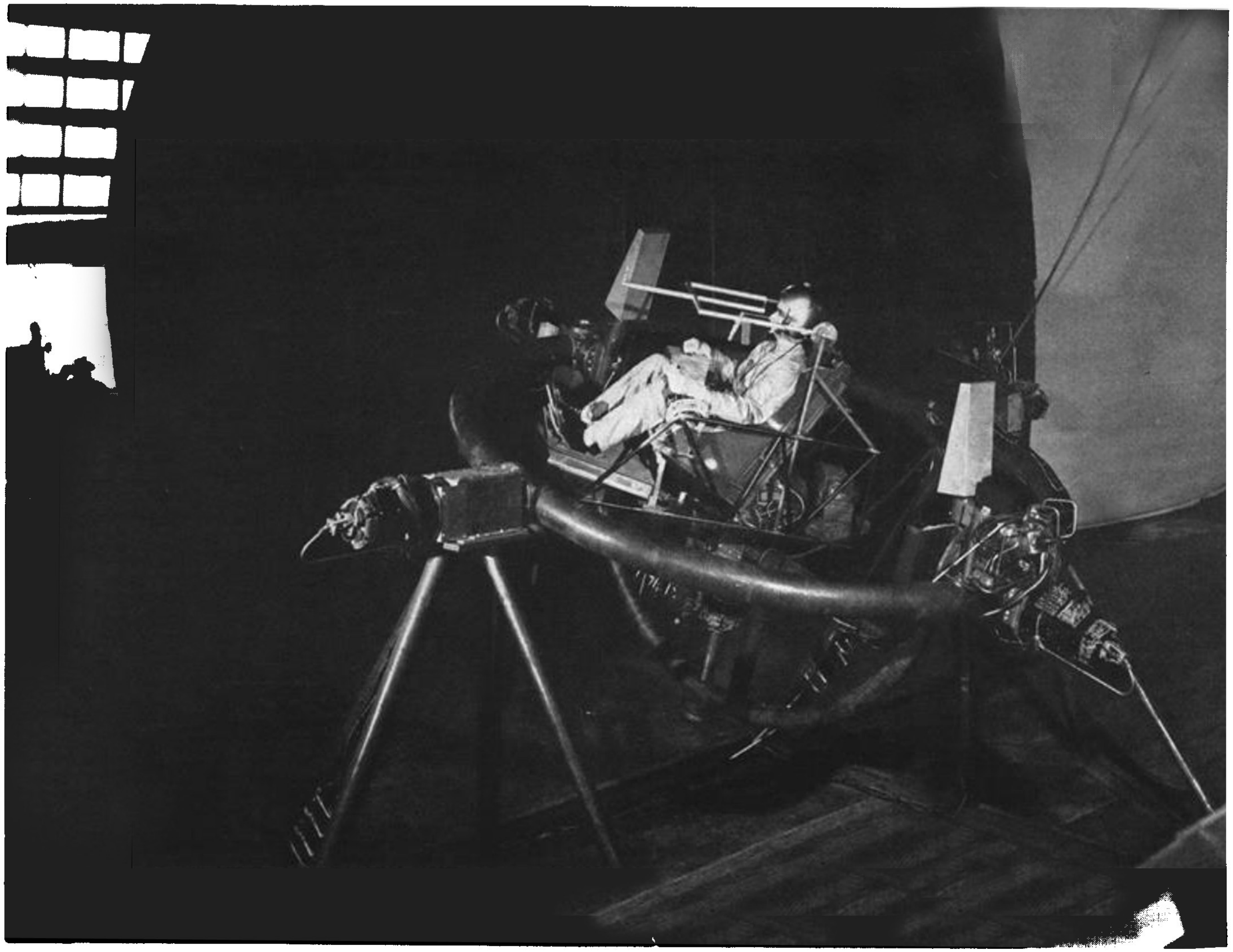


Figure 11.- Three-axis chair.

L-61-5204

Orbital Assembly

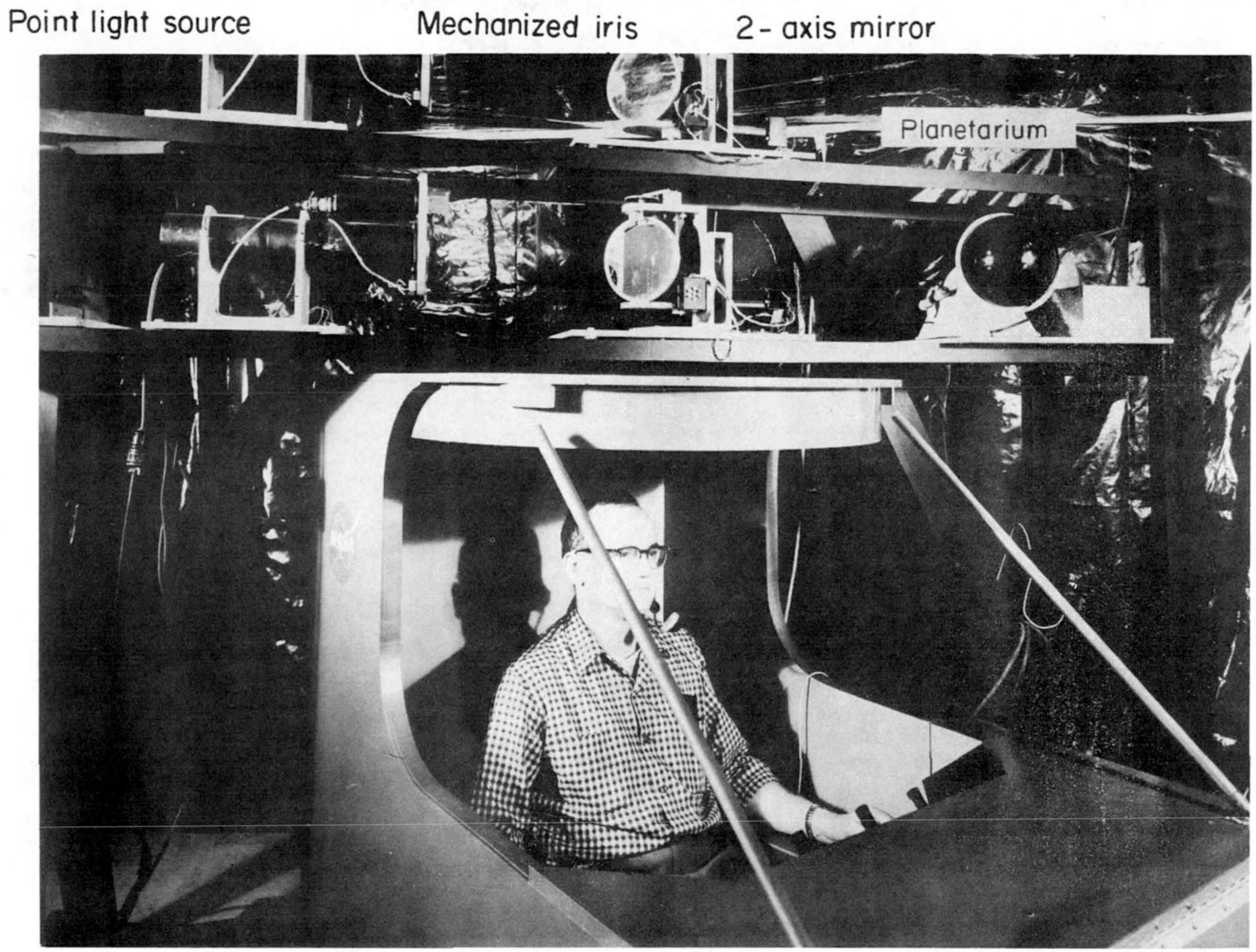
Fixed-base analog study of a pilot's performance in controlling from a distance the docking of two orbiting bodies using visual cues for guidance.- An investigation is in progress to assess a pilot's capability to perform the final docking maneuvers of two or more unmanned tanks from a remotely located manned space capsule. This fixed-base analog study primarily considers task performance by the pilot through the use of visual cues.

The results of the investigation are of interest in regard to the proposed scheme of achieving early manned lunar and interplanetary flights by utilizing the approach of constructing the final space vehicle from several payloads which are placed in near proximity and in nearly circular orbits by the use of launch vehicles now under development.

Photographs of the equipment being utilized in the investigation are shown in figure 12. The two unmanned tanks are represented by light spots which are projected on a cylindrical screen having an 8-foot radius. The projection system for each vehicle consists of a point light source, a mechanized iris which regulates the size of the light spot according to the vehicle's range, and a two-axis mirror which positions the light spot on the screen according to the equations of motion as programmed on an analog computer. Also shown in figure 12(a) are the pilot's cabin and a small planetarium used to provide a star background.

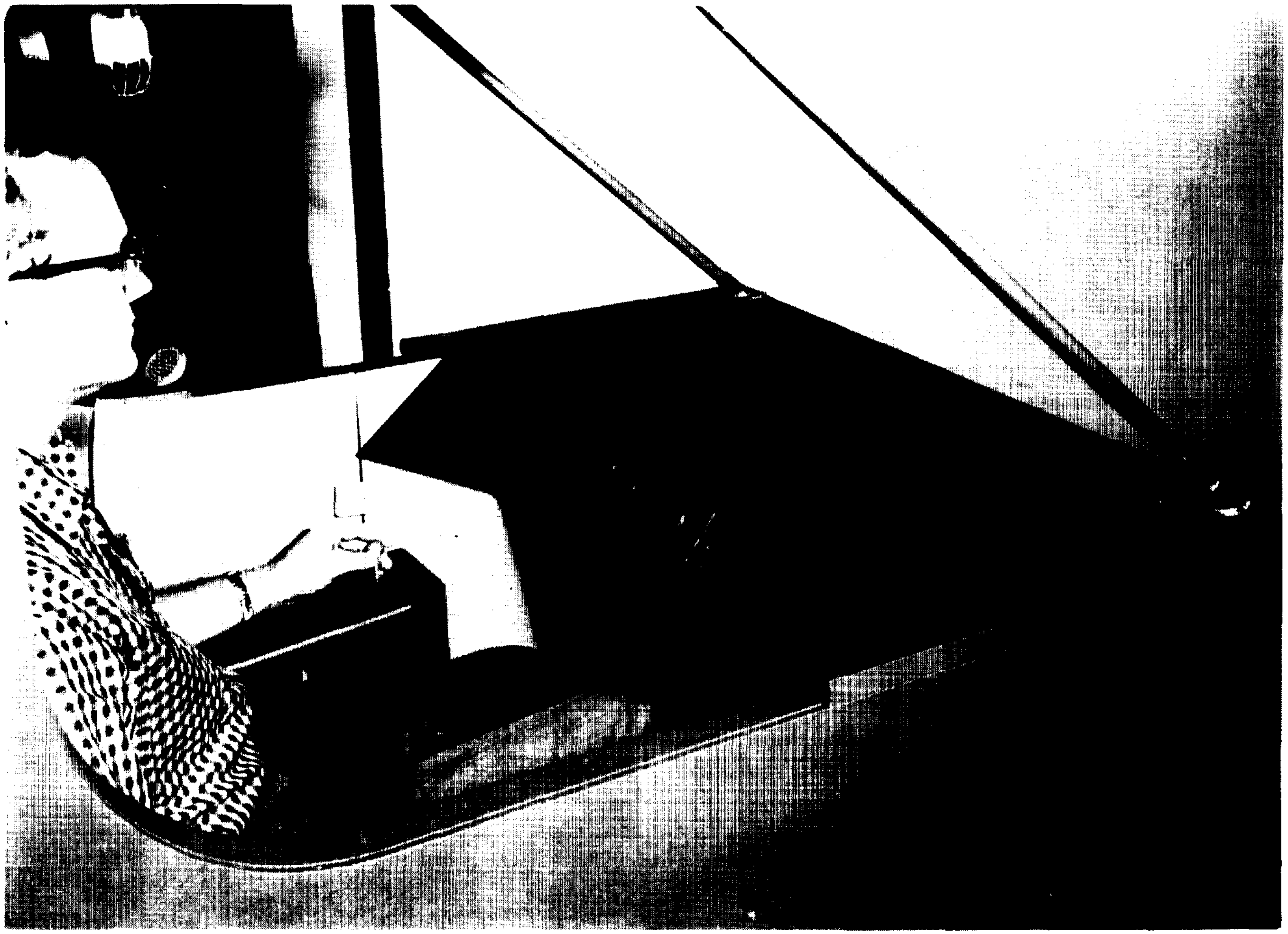
The initial conditions for this docking study consider the rendezvous maneuver to have been completed when the three vehicles are located within a 1/4-mile range of each other and to be moving with residual velocities of only a few feet a second relative to one another. The pilot is given control of his own vehicle and one unmanned tank. The task is to maneuver his own capsule and the controllable tank in such a manner that he performs the docking maneuver between the two unmanned tanks when they are located directly in front of him. Since the vehicles are all assumed attitude stabilized, the pilot controls only the translational motions of two of the vehicles. The docking maneuver is considered completed when the two light spots are located in front of the pilot, are the same size, and are tangent to each other.

Preliminary results of the investigation obtained thus far indicate that with a little practice in flying the simulator a docking maneuver can be accomplished entirely visually in about 5 to 8 minutes with contact velocities between vehicles of only several tenths of a foot a second. Based on an unmanned tank diameter of 10 feet, the above docking values were obtained with the unmanned tanks at approximately 100 to 130 feet range from the manned vehicle at initiation of the docking maneuver.



(a) Projection equipment and pilot's cabin. L-61-6513

Figure 12.- Photograph of docking simulator.



(b) Pilot's cabin and controls. L-61-6514

Figure 12.- Concluded.

NASA-Langley, 1961