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Improved Lunar Cargo and Personnel Delivery Systems

INTRODUCTION

Lunar research and exploitation will require larger quantities of cargo and more available manhours than will be required by early exploration. Freedom of movement of personnel to and from the lunar surface and maximum cargo delivery per launch are required to maintain necessary economies. These requirements coupled with the large area of the lunar surface make landings anywhere on the surface very desirable.

In order to define the optimum candidate transportation systems, NASA has directed studies through the "Improved Lunar Cargo and Personnel Delivery System Study" (ref. 1), which has the objective of detailed conceptual design of two pairs of transportation systems: early and advanced. Each pair consists of a personnel delivery system and a cargo delivery system.

In this study two principal flight modes for personnel delivery were investigated: Lunar orbit rendezvous (LOR) and direct flight. The LOR mode, as shown on the left in figure 1, is basically the same as the Apollo flight mode, with the exception that surface staytimes of up to 90 days were investigated. Staytimes in this range require that the command and service module (CSM) remain unmanned and dormant in lunar orbit for 90 days and that the lunar module (LM) remain unmanned and dormant on the lunar surface for 90 days. The delivery of three men can be achieved with this system.

The other principal personnel delivery mode under consideration has been termed "direct flight." In this mode the Earth return stage, with a command module and its service support, is landed on the lunar surface for direct return to Earth. This latter flight mode can be seen to be of considerable merit when we consider the significance of the illustration in the center of figure 1. When a landing site is located at latitudes off the lunar equator, an extended staytime results in the landing site moving out of the plane of the orbit; this requires a plane change for pickup of the LM ascent stage except at the times of intersection of the landing site and the orbit plane. The direct-flight mode eliminates the requirement for large plane changes and, hence, fuel expenditures in accomplishing the rendezvous and docking maneuvers.

Another mode of personnel delivery considered was the "pickup" method. In the pickup mode four men make the translunar flight; three men land in the LM, and one man returns to Earth. The return of the three men is accomplished by a subsequent launch, with one man performing the translunar flight and landing of an unmanned LM followed by pickup of the three men and Earth return. This personnel delivery system was not included in the final phases of the study since its advantages were outweighed by its disadvantages.

Large-cargo delivery must be accomplished by unmanned direct flight. The direct-flight mode selected was initial brake into a lunar orbit followed by descent to the landing site. The lunar parking orbit is utilized to increase

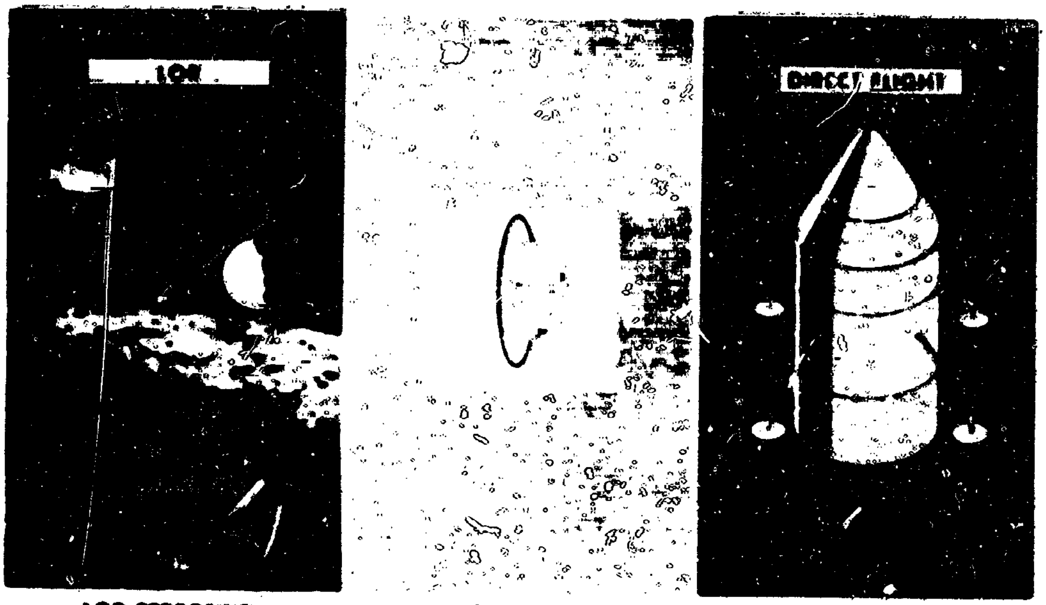


FIGURE 1.—Personnel delivery flight modes.

the accuracy of landing and safety and reliability of delivery.

Possible modes of evolution of lunar transportation systems are shown in figure 2. The development of a single-stage cargo vehicle allows evolutionary use of the stage over a long time period. This stage could be utilized for early missions by combining with an intermediate launch vehicle as the third stage, and later with the Saturn V as a fourth stage. This same stage may be stacked using one stage as a lunar-orbit braking stage and one as a landing stage for direct delivery of personnel or cargo with a 180-day dormancy capability.

The initial phases of the study program, shown in table 1, included consideration of six propellants, N_2O_4/50-50, LO_2/LH_2, LF_2/LH_2, FLOX/CH_4, OF_2/MMH, and LF_2/NH_3, in parametric investigations of LOR, direct-flight personnel delivery, and direct cargo delivery. A matrix of transportation systems was established and individual stages were configured and sized for applicable propellant loadings. Mass summaries were compiled by selecting point designs, utilizing applicable design loads, and performing relatively detailed analyses.

By use of these stage point designs, launch vehicle requirements were determined for the delivery of three, four, and six men by the LOR flight mode for staytimes of 14, 28, and 90 days. A standard velocity budget was used in these evaluations. All of the stages in set A employ the same propellants. In set B, the service module and LM descent stage were modified to correspond with the indicated propellants. For set C, only the descent stage was modified. The change in propellants for the service module, set D, represents the less difficult modification in that only one stage is affected, and structural envelopes are compatible.

The direct-flight personnel systems were evaluated for single-stage and two-stage delivery to the lunar surface. Here, too, all the stages in set A employ the same propellants. The stages making up set B have the indicated propellants in the Earth-return stages. These Earth-return stages are delivered by LO_2/LH_2-powered lower stages.

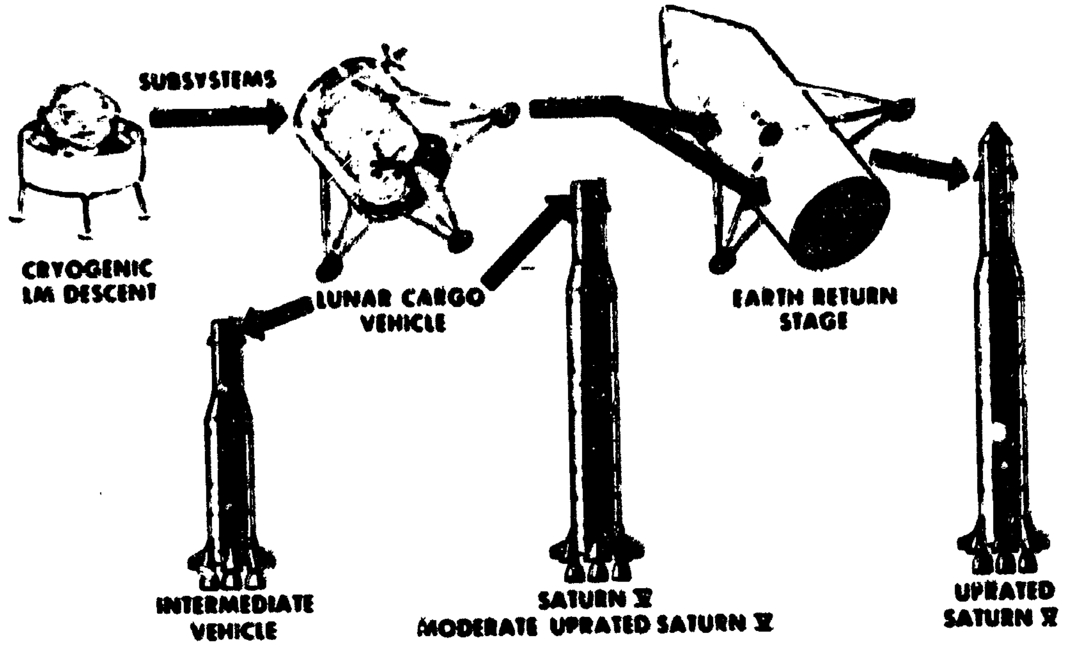


FIGURE 2.—Evolutionary usage of a multiapplication stage.

TABLE 1.—*Parametric Study Candidates*

Candidate propellants	Transportation system	Set	Stage	Resulting number	Variations	Total number cases examined
N_2O_4/50-50	LOR personnel	A	All stages		3, 4, 6 men 14, 28, 90 days.	108
		B	Service module LM descent.	2		
		C	LM descent	1		
		D	Service module	5		
LO_2/LH_2, LF_2/LH_2.	Direct personnel	A	All stages; 3 stage	11	3, 4, 6 men 14, 28, 90 days.	198
		B	Earth return; 2 stage	11		
FLOX/CH_4, OF_2/MMH, LF_2/NH_3.	Direct cargo	A	2 stage	6	3 ELV	38
		B	1 stage	6		
		C	LM truck	2		

The parametric data for the direct-flight cargo stage were iterated by assuming launch-vehicle capabilities and determining the resultant cargo delivery by use of a standard velocity budget.

The conclusions drawn from these parametric studies are shown in table 2. Note that the delivery of three men with 90-day dormancy by the LOR method may be accomplished with the present Saturn V if all of the current Apollo stages are changed to space-storable propellants or if the service module (SM) and

TABLE 2.—*Conclusions From Parametric Studies*

LOR personnel delivery—3 men, 90-day dormancy
Within Saturn V capabilities if—
All stages: space-storable propellants
Service module and LM descent: cryogenic propellants
Saturn V upratings required:
All stages N_2O_4/50-50, ~15 percent uprating
SM space-storable or cryogenic: max. 10 percent uprating
Direct personnel delivery—6 men, 180-day dormancy
All stages: N_2O_4/50-50, ~100 percent uprating
LO_2/LH_2 braking and landing stage: satisfactory (~60 percent)
Earth-return stages: LF_2/LH_2 or space storable
Direct cargo delivery
Single-stage cargo-delivery capabilities are only approximately 5 percent less than those of two-stage

LM descent are changed to cryogenic propellants. If the existing Apollo LM propellants (N_2O_4/50-50) are employed, a Saturn V uprating of approximately 15 percent is required. Changing the service module propellants results in a lesser launch-vehicle requirement.

Retaining the propellants in the present Apollo LM systems in direct-flight stages requires at least a 75-percent uprating of the Saturn V capability to deliver six men with 90 days' dormancy. The combination LO_2/LH_2 for the braking and landing stages greatly improves performance and appears to give results comparable to those of other propellant combinations. Earth-return stages require very good mass fractions, and LF_2/LH_2 or space-storable propellants appear to be attractive for this purpose.

Evaluation of the direct cargo delivery also indicated comparable performance between space-storable propellants and LO_2/LH_2. This appears to result principally from the attractive mass fractions of space-storable propellants, accompanied with moderately good specific impulses, and the excellent specific impulse of LO_2/LH_2 with less attractive mass fractions. Earth-storable propellants (N_2O_4/50-50) result in much poorer performance. An unexpected and very significant finding was that single-stage cargo delivery was comparable with two-stage delivery, which indicates that only one stage needs to be developed with only minor payload penalty (approximately 5 percent).

On the basis of these results, 14 transportation systems were selected for refined analysis, as indicated in figure 3. The selected stages and spacecraft design concepts were refined by increasing the depth of the subsystem analyses and by considering all factors influencing the masses and subsequent performance indices.

A number of factors were considered in the selection of the candidate systems for detailed analysis. An evolutionary development analysis was used to reduce the number of candidates by identifying promising "building-block" stages which exhibited the maximum potential for evolutionary growth. A propellant selection analysis was conducted to determine the most suitable fuel/oxidizer combinations for the selected building-block stages. For each potential stage, two factors were considered: (1) how the stage would be used in the early system and (2) how it could be used, within the imposed limitations, in an improved system.

The selection of propellants for the systems studied was the result of investigation of the status of engine programs, availability of propellants, safety considerations, and a number of other related factors. As a result of these investigations, the following propellants (other than present Apollo LM propellants) were selected: LO_2/LH_2 and $FLOX/CH_4$, both a cryogenic and a space-storable propellant. LF_2/LH_2 was examined in the Earth-return stage for advanced personnel delivery. From these refined analyses and a reconsideration of related factors such as engine availability, four systems were selected for conceptual design. These are enclosed in boxes in figure 3 and consist of:

(1) LOR systems for delivery of three men

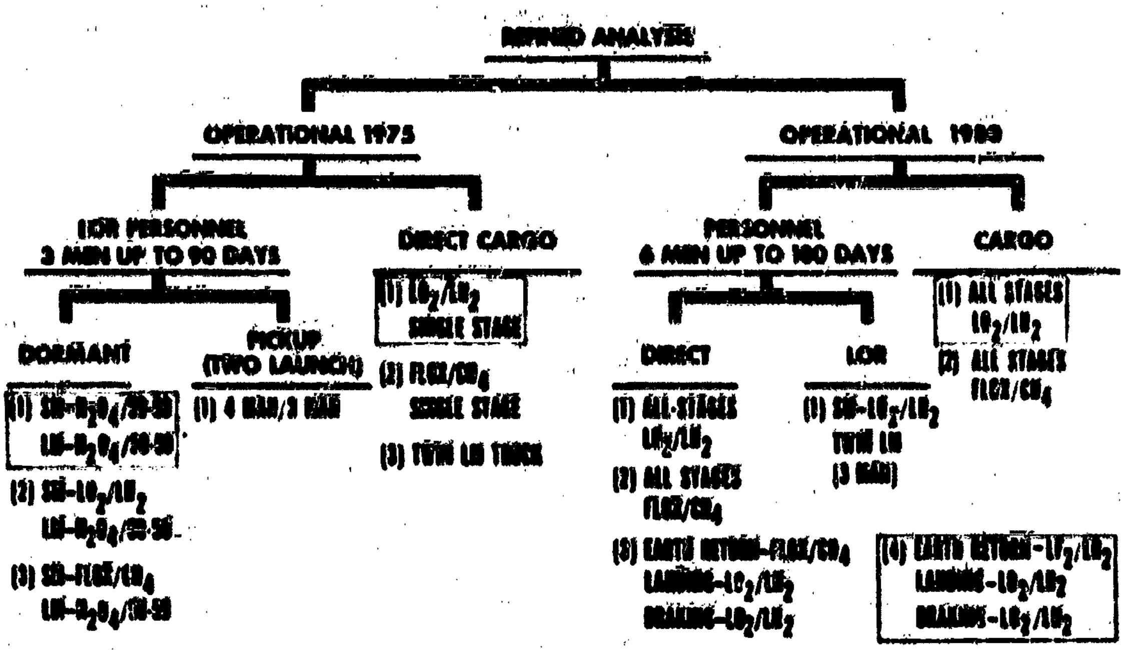


FIGURE 3.—Transportation systems selected for refined analysis.

with 90-day dormancy using N_2O_4/50-50 propellant in all stages and utilizing present Apollo LM derivatives.

(2) Direct lunar cargo stage consisting of a single LO_2/LH_2 stage.

(3) Direct personnel system for the delivery of six men with a dormancy of 180 days.

(4) Direct cargo system utilizing the braking and landing stage of the direct personnel system.

PERSONNEL DELIVERY—OPERATIONAL 1975

The primary purpose of the detailed investigation of personnel delivery systems was to establish launch-vehicle requirements to permit sizing of the direct cargo stages. Baseline weight stagements and design data for the standard Apollo system were provided by NASA. The CSM was examined for configuration changes required to allow for an unmanned 90-day dormancy in lunar orbit and for delivery of a three-man LM.

A basic assumption in achieving the 90-day dormancy was that the subsystems could be developed to satisfy the required lifetime with no appreciable increase in present equipment weight. This appears feasible, since there is a minimum of rotating equipment in the CSM. Modifications to the command module are relatively minor as shown in figure 4. The major factor is qualification of equipment for the 90-day operating period. The command-module internal pressure is maintained at 0.5 psia during the dormant period.

The major modification to the service module is in the electrical power system. The fuel-cell reactant requirement of 2150 pounds requires utilization of bay 1 for the storage of hydrogen. Development programs are in progress to increase the lifetime of fuel cells and to allow shutdown and restart. For these reasons, only three fuel cells were considered necessary, with one operating at reduced power during the dormancy period.

The primary propulsion propellants are increased by the mission requirements, and storage can be accomplished by increasing the length of the propellant tanks. The plumbing arrangement remains basically unchanged. During the

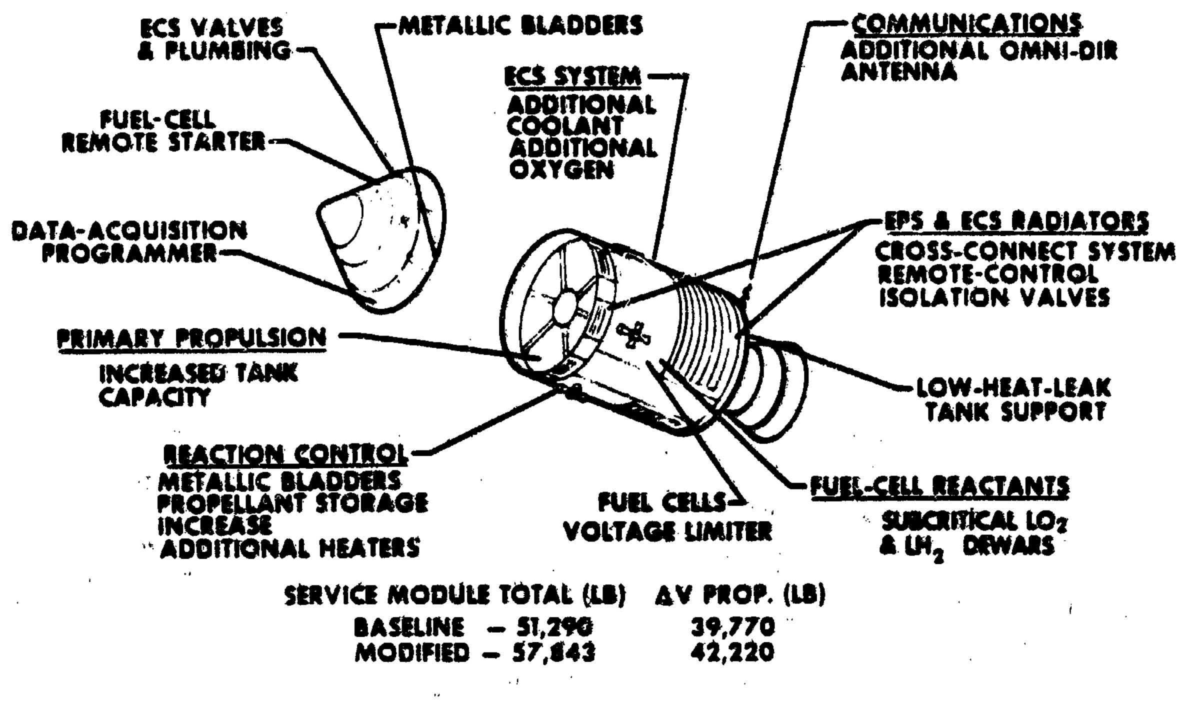


FIGURE 4.—Summary of modifications to CSM.

90-day orbital dormancy of the CSM, the systems essentially constitute an unmanned lunar satellite; it is required only that the vehicle subsystems be protected. The approach is summarized in figure 5.

Spin stabilization was found to be the best method of assuring maintenance of required radiator temperatures. The selected mode at this time entails maintaining the vehicle longitudinal axis nominally perpendicular to the vehicle sunline and allows a deviation of up to ± 20°, half-cone angle. Nulling the 20°, half-cone angle is accomplished by a two-impulse maneuver that restores the vehicle centerline to the original angular momentum vector. Since the deviation is principally a result of fuel sloshing, the analyses indicated that, at the low spin rates examined, the propellant required for 90 days increases as the spin rate is increased. A spin rate of 0.3 to 0.5 rpm has tentatively been selected which requires some 150 pounds of reaction-control propellant.

With the exception of the thermal-control system, the internal measuring unit is shut down during the dormancy period. Gyros are employed for attitude determination. The S-band command receiver is left on, together with the antenna steering circuit. Earth communications and/or control are established on command or by programmer.

Attitude control would be required only for correction of spin stabilization. Heater power can be supplied continually. In the event of a nozzle freezeup, heater power can be increased to restore the nozzle prior to use.

As previously mentioned, only one fuel cell is required during the dormant period. This allows switching of cells, and therefore lower lifetime requirement per cell. Freezing of radiators is marginal in the lunar shadow, and some deliberate heat addition could be necessary.

Modifications required for the LM ascent stage are principally related to provisions for the third man, as shown in figure 6. They are:

(1) *Crew provisions*.—An additional suit,

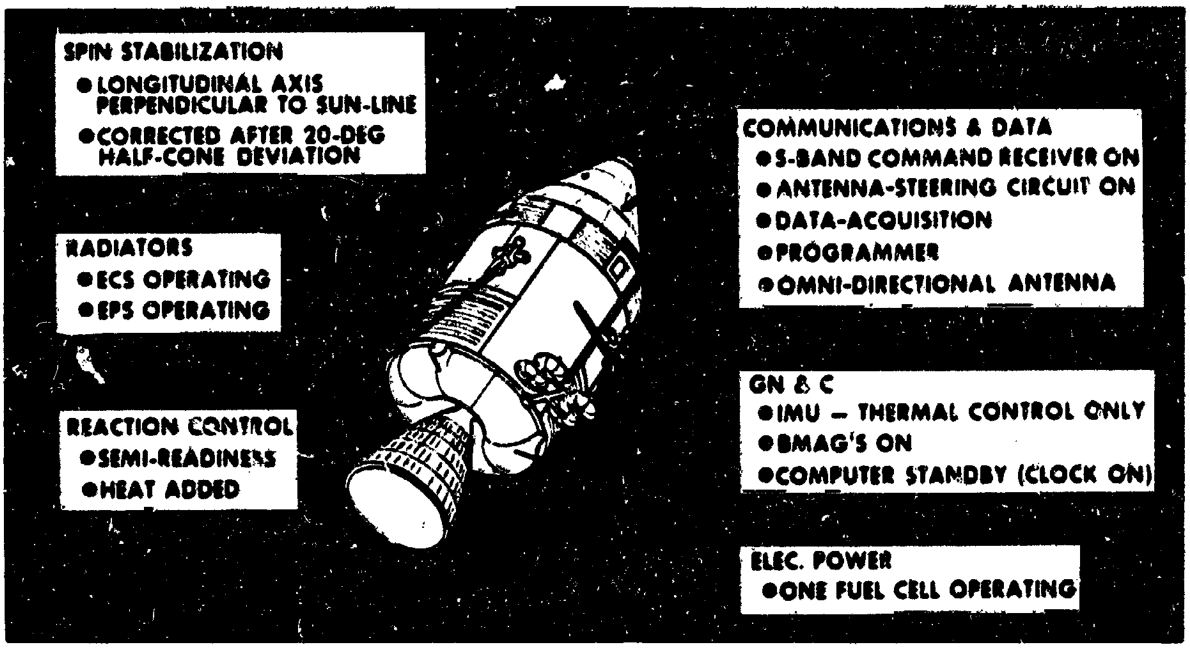


FIGURE 5.—Orbital dormancy of CSM.

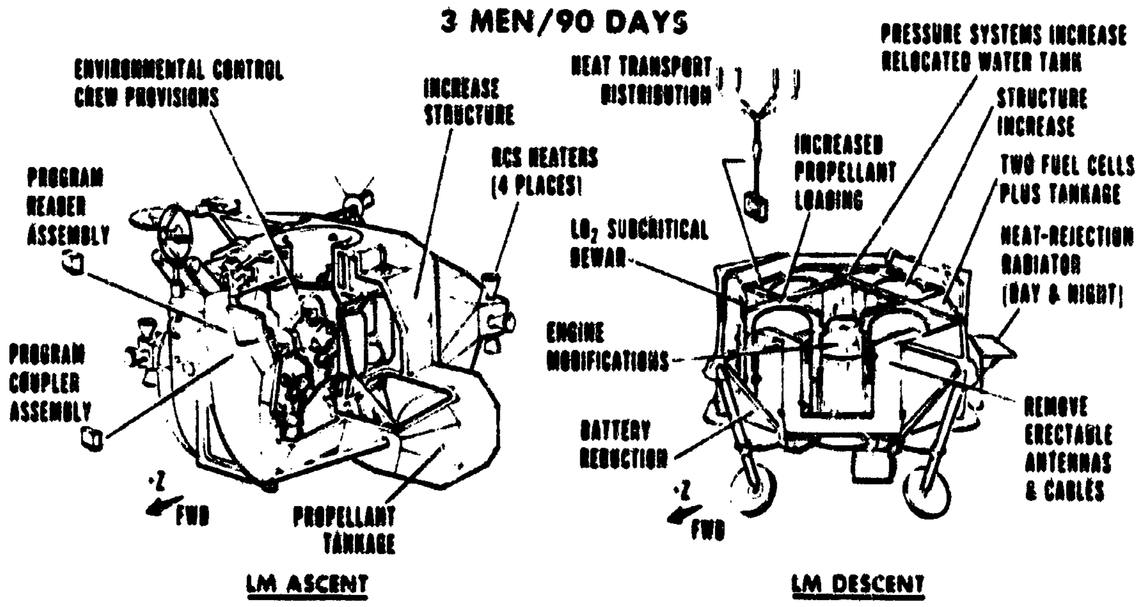


FIGURE 6.—Summary of modifications to LM for three men with 90-day dormancy.

planetary life-support system, restraints, and other equipment must be provided for the third crewman.

(2) *Environmental control subsystem.*—A controlled cabin pressure of 0.5 psia must be maintained during the quiescent period to reduce

deterioration of materials. Heat rejection can be accomplished with a radiator during the quiescent period. The radiator would be relatively small (15 ft^2), being sized to prevent freezing during lunar night. The water evaporator would be used for heat rejection during occupancy. Water and oxygen requirements are not significantly affected.

(3) *Electrical power subsystem.*—The ascent energy requirements should not require significant change. Batteries would be used.

(4) *Reaction control subsystem.*—The existing subsystem can be used; propellant lines near the nozzles will freeze unless purged. Heaters should be provided for this region for warmup prior to use.

(5) *Structures and propulsion.*—Additional meteoroid shielding may be necessary. The propellant loading must be increased to approximately 5700 pounds.

The modifications of the LM descent stage are also indicated in figure 6 and consist principally of:

(1) *Electrical subsystem.*—Batteries would be used for descent. Fuel cells must be added for the quiescent period. Only one is required, since the power requirements are lower than the capacity of single cell; however, two were included for redundancy. The cabin can be heated by a reflux condenser that draws heat from the fuel cells.

(2) *Structures and propulsion.*—The propellant loading must be increased to approximately 20 600 pounds. This could be accomplished by employing ellipsoidal tank bulkheads and increasing the length of the cylindrical section of the tank.

The launch-vehicle capability requirements were established by use of a standard velocity budget supplied by NASA Marshall Space Flight Center; the composition is shown in table 3. All expendables necessary for a normal 14-day lunar mission were considered left on board the CSM in all performance evaluations. However, it was considered realistic to assume that expendables associated with the 90-day dormancy would be either consumed or dropped prior to leaving lunar orbit, and this was considered in the evaluation of the launch-vehicle requirement.

TABLE 3.—*Summary of LOR Personnel Delivery Mission Requirements for Operational 1975 Personnel System*

Requirement	Weight, lb
Translunar injected weight	112 200
SLA jettison	3 850
Midcourse propellant	730
Lunar orbit insert propellant	29 318
Total weight in lunar orbit	78 300
Total weight LM with men	38 380
LM rescue propellant	2 489
Weight at lunar departure	35 660
Inject propellant	9 578
Midcourse propellant	103
Final burnout weight	25 980

CARGO DELIVERY—OPERATIONAL 1975

The early cargo stage was designed in considerable detail. Maximum use was made of existing hardware and techniques that are within the present state of the art. Minor sacrifices in performance were made where necessary to utilize proven systems and techniques to keep development costs minimal. The availability of the RL-10 engine, which could be modified to provide a suitable engine for direct cargo delivery, makes LO_2/LH_2 stages very attractive.

In developing the mission profile for the time period contemplated, achievement of the highest possible landing accuracy with minimum degradation of payload was considered to be of prime importance. Use of a lunar parking orbit contributes appreciably to the accuracy of a landing; Earth tracking is of maximum effectiveness when several orbits are made prior to lunar descent. The Apollo flight mode involves braking into a lunar orbit at approximately 80 nautical miles. The LM then performs a Hohmann transfer down to an elevation of approximately 50 000 feet, where the engines are restarted for final descent.

Several objectives were considered in selecting a flight mode. These include:

(1) Use of a lunar orbit

(2) A velocity budget not differing greatly from that of Apollo

(3) Minimizing number of burns

(4) Effective line of sight to landing point at initiation of burn or shortly thereafter

Minimizing the number of burns results in a significant weight saving of propellant. In addition, a number of operational problems could be eliminated if the 1-hour Hohmann transfer were omitted.

The selected flight mode is indicated in figure 7. Lunar-orbit injection is into a 100 000-foot orbit. Error analyses indicate that this maneuver can be performed with a maximum altitude error (3 sigma) of approximately 16 000 feet. Descent from 100 000 feet after several orbits is accomplished by use of a single burn. The early cargo stage is above the horizon at initiation of burn. A 10° line of sight to the landing point is achieved at an altitude of approximately 85 000 feet (using the two 20 000-pound-thrust RL–10 engines). The resulting velocity budget is slightly lower than the Apollo budget and produces comparable performance.

The selected early cargo stage concept as shown in figure 8 presents several attractive features:

(1) Maximum utilization of existing hardware

(2) Structural efficiency

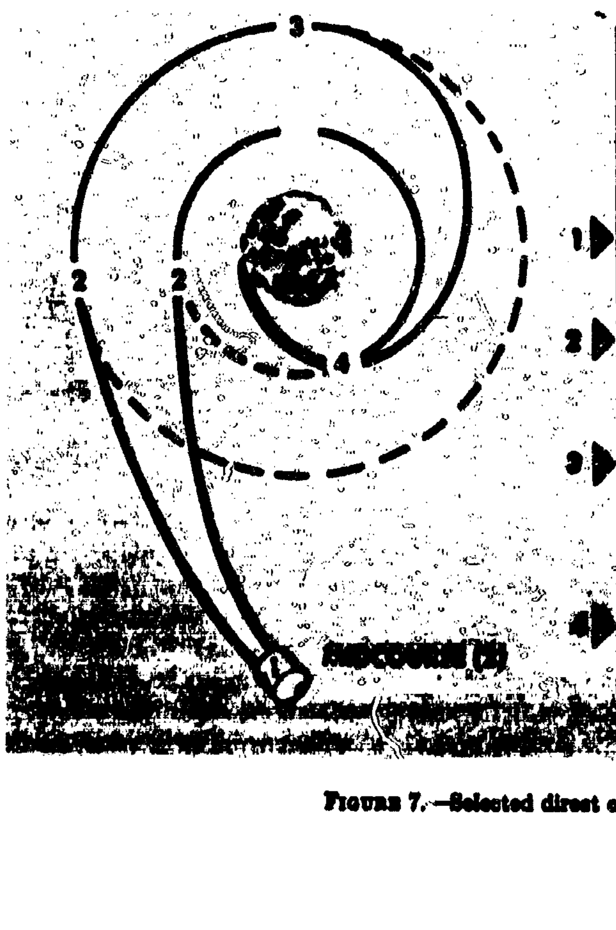
(3) Simplicity

(4) Accessibility

(5) Flexibility

The stage has been designed to utilize a modified RL–10 engine with either a 57:1 area ratio or a 150:1 area ratio to allow growth in engine performance.

The external shell is made of eight honeycomb panels of equal size. The shell is reinforced with internal rings which serve the multipurposes of external shell stiffening, assistance in propellant tank support, and landing-gear backup. The meteoroid shield is located



	STANDARD APOLLO m/sec (ft/sec)	SELECTED m/sec (ft/sec)
1 MIDCOURSES	21 (68)	21 (68)
2 LUNAR ORBIT INJECTION	972 (3,189) 80 NM	984 (3,227) 100,000 FT
3 DESCENT HOHMANN TRANSFER	30 (97)	
4 DESCENT	2,203 (7,228) FINAL DESCENT	2,171 (7,121) SINGLE BURN

FIGURE 7.—Selected direct cargo delivery flight mode.

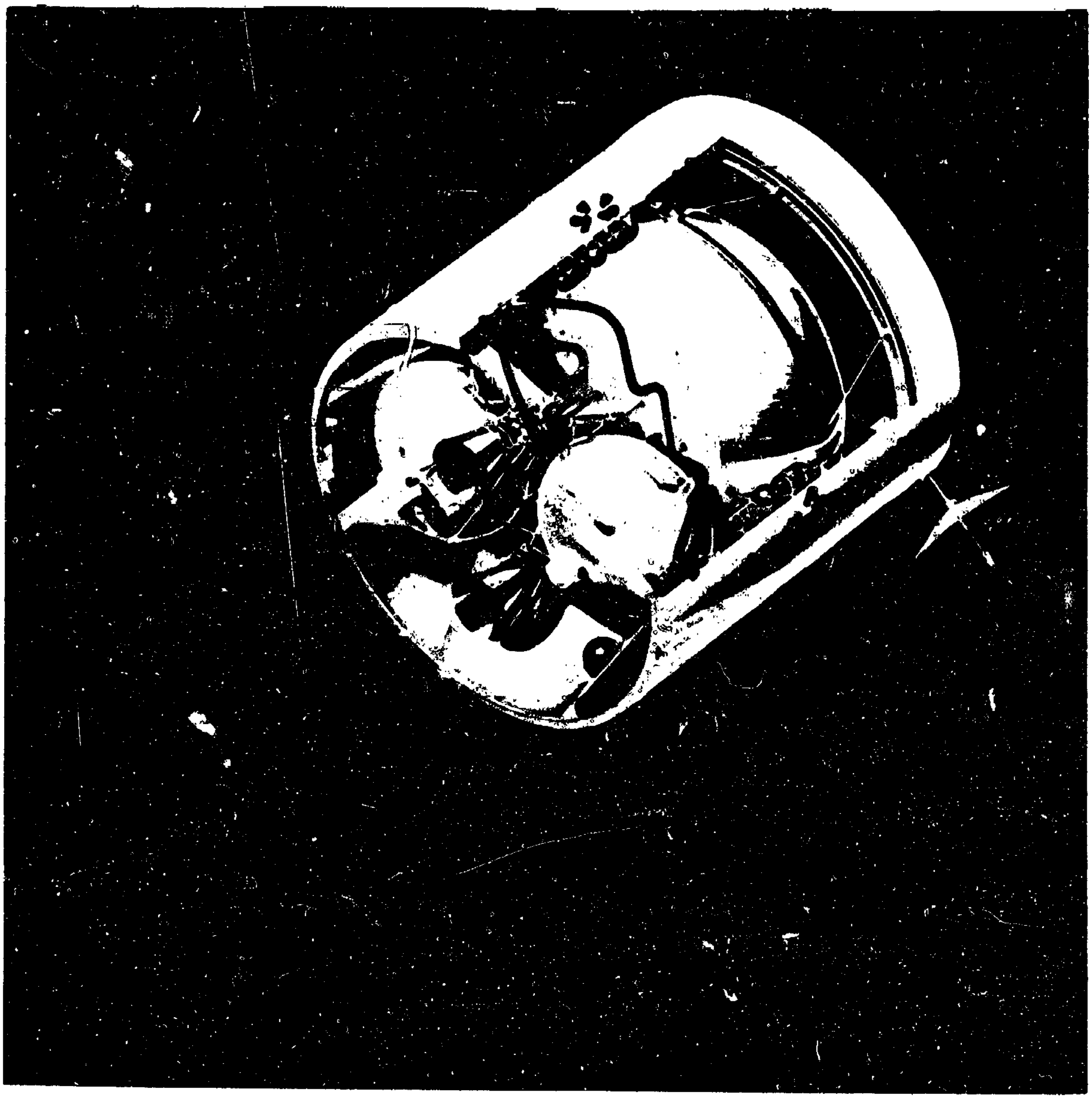


FIGURE 8.—Early direct cargo stage concept.

inside the shell and is attached to the inner edge of the rings.

The liquid-hydrogen and liquid-oxygen tanks are of 2219 aluminum and are supported with fiber-glass struts. The struts are attached to Y-rings in the tanks with most of the weight in the system concentrated in the liquid-oxygen tanks. The selected support system distributes the load effectively and efficiently to the shell.

Each propellant tank is insulated with multilayer insulation consisting of thin, crinkled, double-aluminized Mylar radiation shields with Tissuglas spacers and is applied in blankets. Areas which might leak gas into the insulation are covered with fiber-glass structure which is vented past the insulation. The thermal protection system is sufficiently effective to permit an entire translunar flight without propellant boiloff.

The reaction control system is located in four modules in the region between the liquid-hydrogen and liquid-oxygen tanks. The space-qualified modules from the Apollo service module are used.

The Saturn V instrument unit is capable of providing the major portion of the astrionic functions if it is attached to the cargo stage and taken to the lunar surface. The alternative to this is integrated stage astrionics. By using the Saturn V instrument unit, however, costs can be appreciably reduced, though the payload loss is approximately 1000 pounds more than with integrated astrionics. Additional equipment must be added to supplement the instrument unit. The landing radar is the same as that used on the LM and is located in the aft region.

The landing gear is relatively small, considering the vehicle size. It is folded on the exterior of the vehicle during launch and is covered by a fairing.

In figure 9, some of the existing hardware applications are shown:

(1) The current RL-10 engine can be modified to the required capabilities. This was a major factor in the selection of LO_2/LH_2 as propellant.

(2) As previously discussed, the modification and use of the Saturn instrument unit is attractive, since no major developments are necessary. The Apollo fuel cells and reactant storage tankage could be utilized.

(3) The flow-rate and pressure requirements of the components in the pressurization system and the propellant feed system are compatible with valves and regulators that have been used on other stages utilizing the RL-10 engine.

(4) The thermal-protection system has been tested on full-size tanks and found to be satisfactory. Lockheed has made sufficient tests to assure that the insulation system is flight qualified.

Typical results from the landing-gear stability studies are presented in figure 10. The stability angle is defined in the illustration as the point where the overturning moment exceeds the restoring moment. Note that the vehicles become unstable rather rapidly with decreasing landing-gear radius.

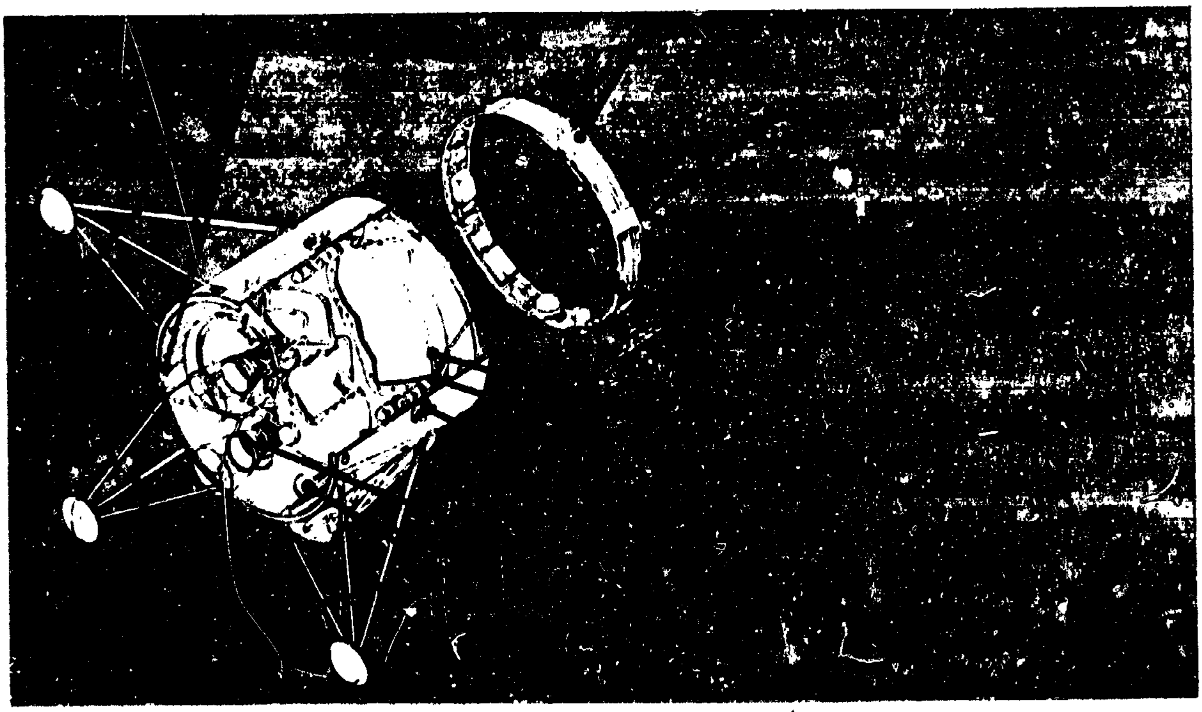


FIGURE 9.—Existing hardware utilization.

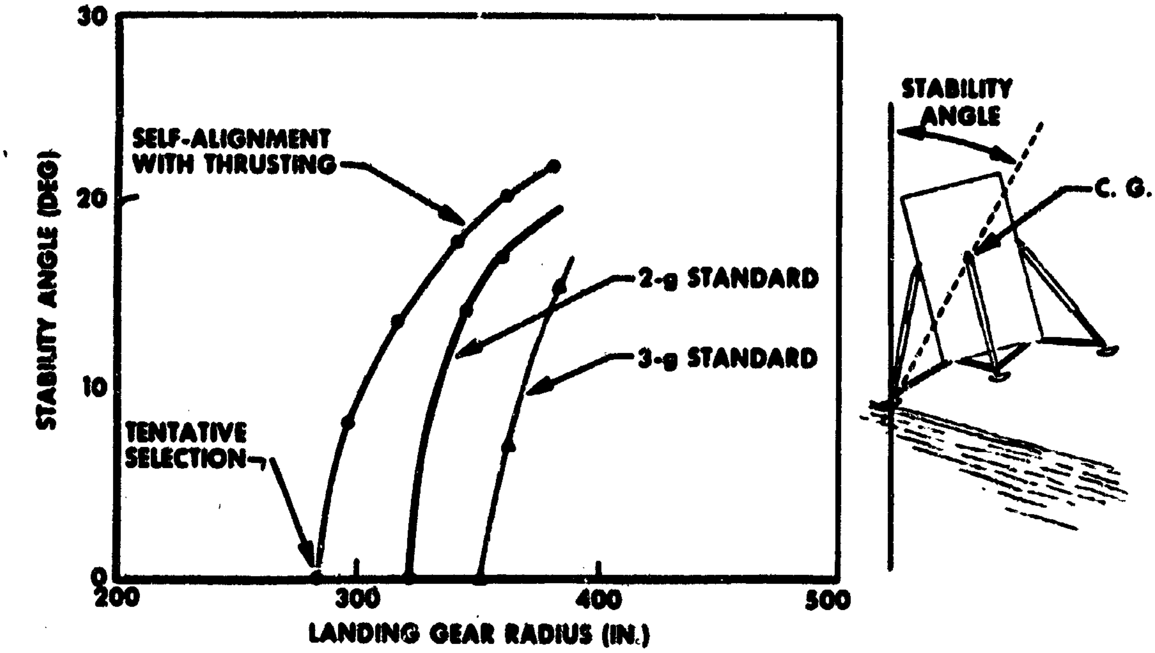


FIGURE 10.—Typical landing gear stability results.

The landing-gear radius selected during the study was 280 inches. The design incorporates a self-alining gear that strokes with almost zero load if only one leg touches until two opposing legs touch; all legs are then stiffened, and honeycomb crushing begins. The addition of reaction control thrusting downward after primary-engine cutoff assists in assuring stability. These active methods could result in additional reduction in landing-gear weight and radius in future stages.

Several observations from the studies made to date are of interest:

(1) An increase in the height of the center of gravity does not rapidly affect the stability. (Example: for an increase of 20 inches, stability angle increases by only 3°.)

(2) The lower the crushing g in the main strut, the greater the stability for a given radius.

(3) Increasing the tension-carrying capacity of the secondary struts appears to decrease stability.

(4) Horizontal velocity may be the largest contributor to instability.

Experience in evaluating thermal-protection systems and pressurization has indicated that simplified methods (such as summing insulation and boiloff to obtain minimum weight) are not sufficient for the evaluation of a high-performance insulation system for multiburn missions. An evaluation technique was tailored for this particular mission.

Detailed thermal analyses have been performed on the stage design by using computer techniques to obtain the heat input as a function of the insulation weight and throughout the mission time. Nominal thermal insulation conductivities were varied; in addition, the computer program considered variation with temperature. The resulting information is used to produce a weight index (or weight penalty) which consists of—

(1) Insulation weight
(2) Vented propellant
(3) Tank weight
(4) Pressure gas quantity
(5) Pressure gas subsystem
(6) Residual gas

An example of the information which was obtained is shown in figure 11. In the illustra-

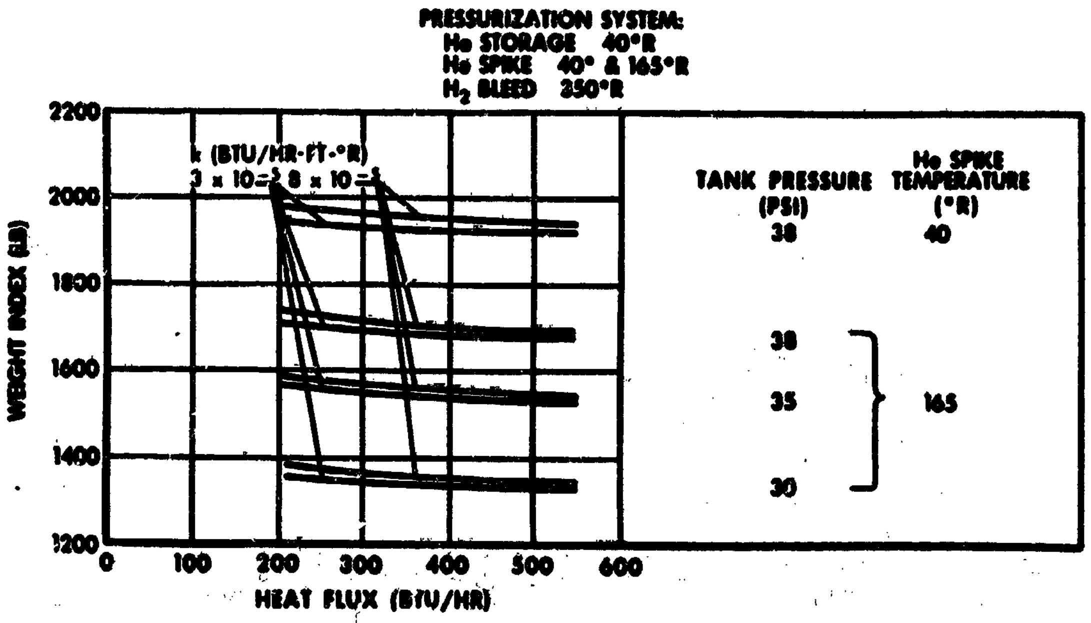


FIGURE 11.—Typical insulation/pressurization optimization results for LH_2 propellant tank.

tion, helium is used to prepressurize the liquid-hydrogen tank, and gaseous-hydrogen engine bleed is used for run pressurization of the liquid hydrogen. Estimates of stratification during ground hold and ascent have indicated that a minimum tank pressure of 35 to 40 psia may be required during the period when ullage pressure relief is not possible. Also, using the tankhead start of the RL–10 makes it desirable to have a higher tank pressure for better performance. These factors force the tank pressure up so that minimums do not occur in the curves; in addition, the effects of insulation thermal conductivity are relatively small. Heat input to the propellant from pressurization gas is on the same order of magnitude as heat input through the insulation. Pressurization gas weight is reduced by an increase in tank vapor pressure because the required pressurant weight is only the amount necessary to increase the pressure from the vapor pressure to the operating pressure.

As a result, an insulation thickness of approximately 0.3 inch is now considered to be a reasonable value. Subsequent evaluations and possibly a different pressurization system may increase this thickness. If the engine NPSP could be held from 2 to 4 psia, the insulation thermal conductivity can be degraded and still give satisfactory performance with the pressurization system under the ground rules for the data shown here.

A number of terminal guidance possibilities exist; these include:

(1) Lunar surface beacon
(2) Optical comparison (area correlation)
(3) Radar comparison (area correlation)
(4) Tracking and updating

A beacon-assisted landing is most advantageous, but the questions concerning delivery and placement of the beacon raise difficulties. If the beacon is not available from a previous landing, it may be delivered by a separate launch or by being landed from the orbiting early cargo stage. The location of the beacon must be established from Earth tracking. A beacon tracker will be added to the guidance and navigation system of the early cargo stage and will furnish relative range, range rate, and angular rate of the line of sight to the beacon. This information would be used to update the inertial system. Altitude and lunar-surface velocity data from landing radar will complete the system.

The use of automatic image (area) correlation offers a possible technique for precision landing. This guidance technique requires previously obtained photographs of the lunar surface surrounding the selected landing site. The reference photographs are stored in the correlator, and during final descent optical images are obtained in real time and compared with the reference images to determine vehicle position. Vehicle guidance would be inertial, with position correction used to improve terminal accuracy.

The same area correlation technique may also be used with a radar imaging sensor. Reference images would be constructed from previous radar mappings of the lunar surface. Two problems accompany this approach: (1) radar reference images may not be available, and (2) low resolution and low frame rates inherent in the radar system would reduce guidance accuracy.

The reduction in landing-point dispersion through the use of an altimeter and beacon tracker is clearly indicated by the 3-sigma dispersion shown in table 4. The altimeter function would be provided by the landing-radar FM/CW beam, which has an altitude-measuring capability of from 10 to 25 000 feet. As shown, the beacon tracker may be used to provide high-accuracy location guidance. However, a lunar-surface velocity sensor is necessary to insure a soft and stable landing. This sensor is provided by the landing radar FM/CM three-beam velocity sensor.

The unified instrumentation unit (UIU) will be located atop the early cargo stage as shown in figure 12. It will consist of the basic instrument unit (in operational configuration) with modifications and additions of existing, flight-proven components. The UIU will be capable of performing all required astrionics functions from Earth launch to soft lunar landing. Thus, the UIU will perform all guidance and navigation, stabilization and control, measurements and telemetry, command and communications, sequencing, and inflight checkout functions during all phases of the mission; that is, during launch, injection, midcourse corrections, braking, and soft lunar landing. The philosophy is to use the basic set of inertial sensing and computing equipment provided by the operational instrument unit, and to add auxiliary sensing equipment as required by the various phases of the mission.

The additional equipment shown in figure 12 would be utilized as follows:

(1) *Guidance and control:*

(*a*) Solar sensor: OAO or Surveyor type—provides a vehicle inertial reference direction.

(*b*) Star tracker: OAO or Surveyor type—provides a second vehicle inertial reference direction.

(*c*) Horizon sensor (lunar): determines direction of lunar local vertical as the

TABLE 4.—*Terminal Guidance Error Comparison*

Mode	3σ landing dispersion,[a] m (ft)		
	Downrange	Cross-range	Altitude
IMU only	820 (2700)	1460 (4800)	1400 (4600)
IMU with altimeter	820 (2700)	1460 (4800)	~0
IMU with altimeter and beacon	[b] ~150 (~500)	[b] ~150 (~500)	~0

[a] Top value given in meters; bottom value in parentheses, in feet.
[b] Principally error in beacon location from Earth tracking.

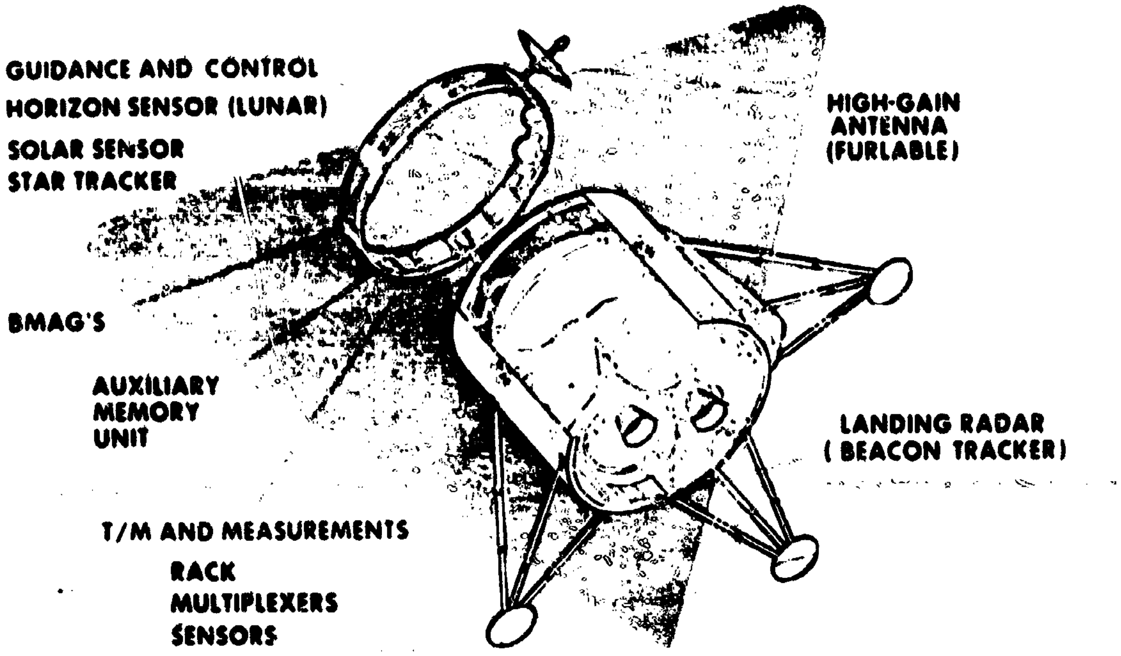


FIGURE 12.—Selected astrionic approach.

vehicle approaches the Moon. This is required for position updating using either a beacon tracker, derived data, or area correlation data.

(*d*) Body-mounted attitude gyros (BMAG): SCM type—provides an attitude reference for midcourse and terminal attitude orientation when the ST-124 inertial platform is caged.

(*e*) Auxiliary memory unit (AMU): used to provide additional program storage for the launch-vehicle digital computer.

(*f*) Landing radar (LR): Ryan LM type—provides altitude and velocity inputs to the LVDC for the terminal descent phase from about 50,000 feet to touchdown.

(2) *Telemetry and measurements.*—Measuring racks, a multiplexer and a remote submultiplexer are required to accommodate the increase in the number of measurements to be made, specifically of the parameters that would be monitored on the early cargo stage. These items are used on the basic Saturn instrument unit.

(3) *Communications.*—A high-gain, steerable S-band antenna is required for communication with Earth at midcourse and lunar distances. A Lockheed 6-foot parabolic reflector antenna that is furlable, with two-axis steering, is the recommended antenna.

PERSONNEL AND CARGO DELIVERY—1980

In the time period of 1980 and beyond, it is a reasonable assumption that Earth-launch-vehicle capabilities will have appreciably increased, and delivery of personnel by the direct-flight mode should be possible.

The direct personnel mode, shown in figure 13, employs the command module and an Earth-return stage which can be supported by a service adapter. The Earth-return stage has been designed to employ LF_2/LH_2 propellant.

For comparison purposes, two approaches were taken. An optimized system was evaluated on the basis of minimum launch-vehicle capability, as shown in the illustration. The second mode considered was based on using the early cargo stage as a braking and landing stage. Since the stage is required to sustain

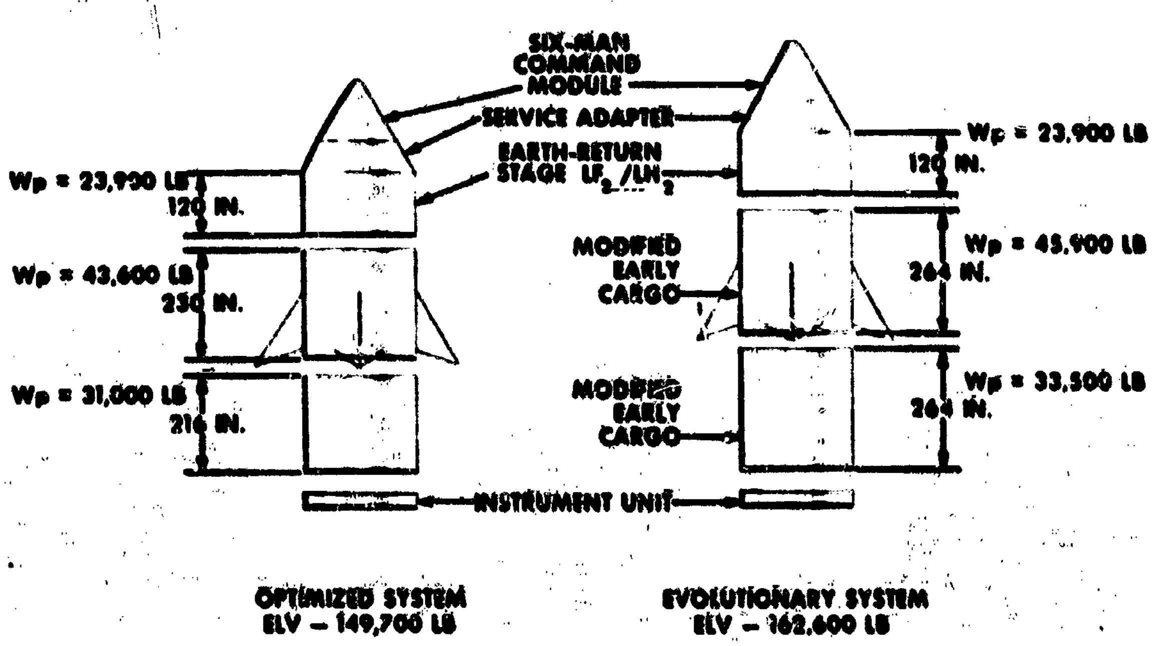


FIGURE 13.—Direct personnel delivery concepts.

more axial loading and larger bending moments, the shell weight is increased, even though attitude control can be removed from the breaking stage.

The studies indicate that the direct command module, which must accommodate six men, can be constructed from the components of the present Apollo command module. Further, Lockheed-sponsored studies indicate that this could be accomplished using refurbished command modules. In order to provide volume for the extra three men, the approach was to lengthen the pressurized cabin, thus increasing the diameter of the command module. The service adapter between the Earth-return stage and the command module houses the fuel cells and reactant supply, environmental control system (including radiators and expendables), S-band deployable antenna with associated electronics, and vehicle status sensing instrumentation.

The Earth-return stage remains on the lunar surface from the time of touchdown to time of departure. Its function is to provide direct return to Earth, with or without an intermediate lunar parking orbit. A lunar parking orbit was assumed in the studies reported here. The basic stage design adopted was identical with that previously discussed for the early cargo systems. The vehicle has been designed with two liquid-fluorine tanks and two liquid-hydrogen tanks. The tanks are supported between the shell and honeycomb shear panel crossbeams by fiber-glass struts. Since thermal protection is of major importance, the tanks are insulated with multilayer insulation in the same manner as for the early cargo stage.

The thermal protection system should be optimized so as to contribute to increasing the net vehicle performance capability. Propellant boiloff, a component of the thermal protection system, is a weight that must be landed on the lunar surface, but does not have to be lifted off from the surface. Therefore, in tradeoff studies, optimum insulation thickness trades off in a different manner than does boiloff, and a seemingly large boiloff does not cause penalties as large as might be indicated. In the case of fluorine, used in this particular mission, no venting is required if the bulk liquid is occasionally mixed.

One concept for achieving lower heat rates by utilizing a solar reflective shield is illustrated in figure 14. The shield was designed to be placed at an angle of 45° to the vertical vehicle. The thermal-analyzer program was run with and without the shield. An $\alpha/\epsilon = 0.30/0.94$ of the vehicle surface without the shield was utilized; this corresponded to that of an ultra-

SHIELDING EFFECTS

	WITHOUT SHIELD		WITH SHIELD	
	LH_2	LF_2	LH_2	LF_2
INSULATION THICKNESS (IN.)	6.5	3.7	6.0	3.0
BOILOFF WT (LB)	[illegible]	0	242	0
INS WT (LB)	262	100	246	84
VAPOR PRESSURE RISE (PSI)	10	10	10	10

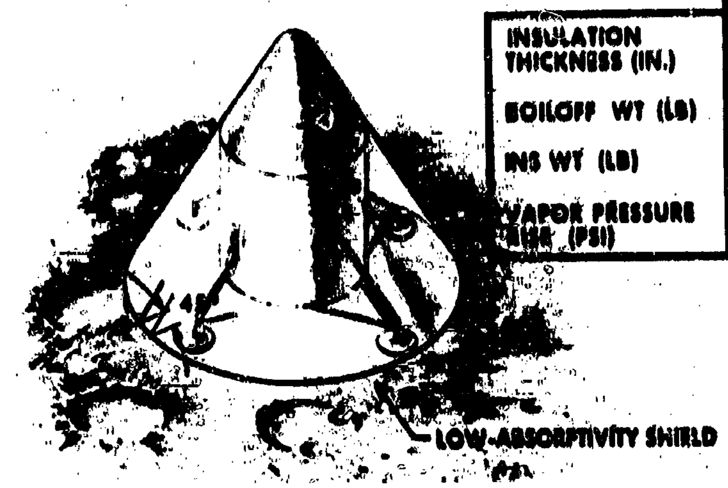


FIGURE 14.—Thermal analysis of Earth-return stage.

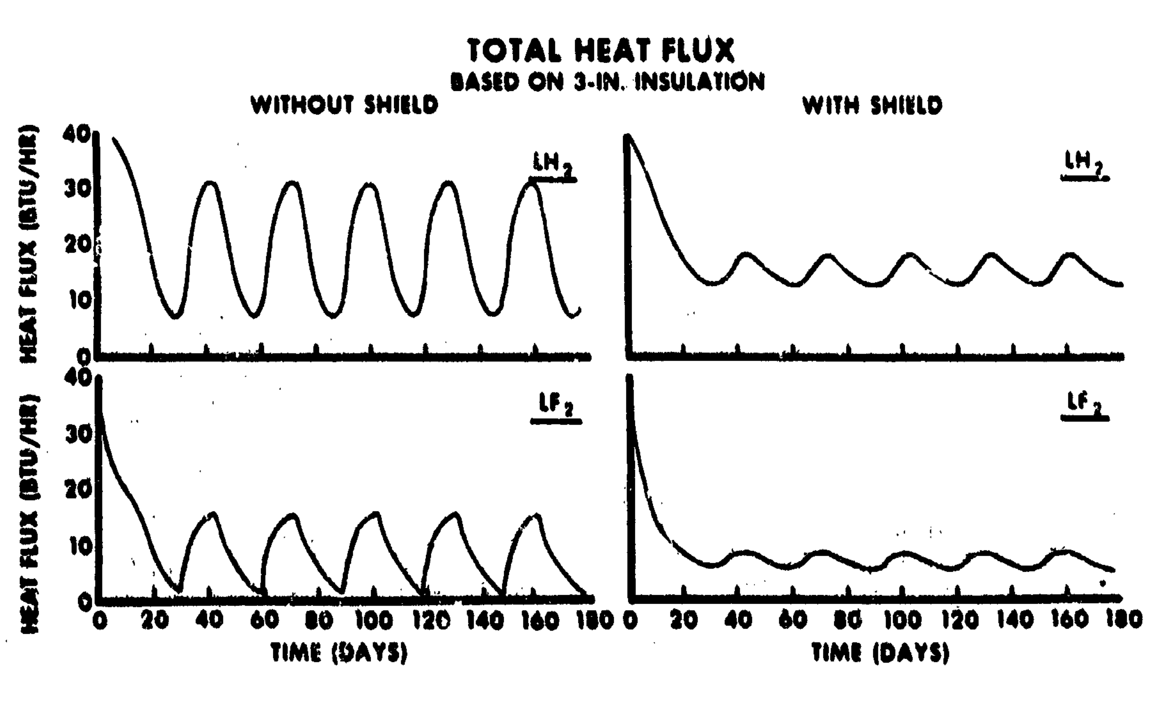


FIGURE 15.—Earth-return stage heat flux variation on lunar surface.

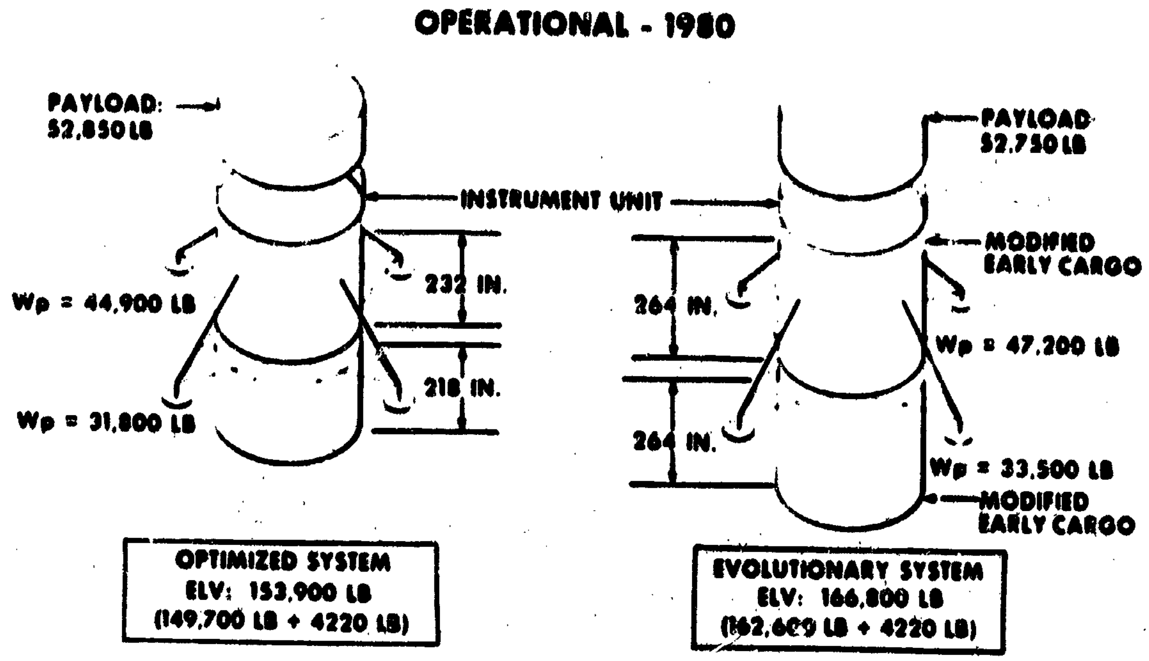


FIGURE 16.—Advanced two-stage cargo delivery.

violet degraded thermal point. With the shield, a value of $\alpha=0.94$ was used on the vehicle, and an $\alpha/\epsilon=0.06/0.4$ was used for the shield. Use of the shield would result in a reduction of the total heat flux by more than 20 percent. The effects on the heat flux produced by a shield on the Earth-return stage are shown in figure 15. The shield tends to dampen the amplitude of the heat flux, but the integrated total heat input is lowered by only about 20 percent.

In this study the personnel delivery system launch-vehicle capability has been used as the basis for determination of the cargo delivery launch-vehicle capability; when this is done these capabilities may be adjusted, depending on the location of the instrument unit. In the evaluation of the advanced cargo delivery, the instrument unit was moved to the forward end of the cargo stage, and a corresponding adjustment in capability was made as presented in figure 16.

As explained in the discussion of the advanced personnel delivery systems, the optimized-system and evolutionary-system approaches were taken in the evaluation.

TABLE 5.—*Summary of Transportation System Performance*

Item	Weight, lb
Personnel (operational 1975); 3 men, 90-day dormancy	
Launch-vehicle requirement	112 200
Cargo (operational 1975); single stage	
Based on ELV	112 200
Delivered payload	35 140
ΔV propellant	60 000
Personnel (operational 1980); 6 men, 180-day dormancy	
Launch-vehicle requirement:	
Optimized	149 700
Evolutionary	162 600
Cargo (operational 1980)	
Delivered payload:	
Optimized	52 850
Evolutionary	52 750

SUMMARY

The performances of the transportation systems are presented in table 5. The early personnel delivery requirement appears to be consistent with expected capabilities of early Saturn V product improvement. The resulting cargo delivery capability is very significant. Each of these results may be considered to be conservative, and performance in excess of that shown can be expected.

One of the most important considerations shown by the study is that of the simplicity and ease of development of the early cargo stage. This follows primarily from the fact that the subsystems previously discussed are developed or are in an advanced state of development.

Finally, the study has provided a logical approach to obtaining a direct cargo delivery capability by 1975, with growth potential to a direct personnel delivery system. The payload and volume requirements for lunar exploration and exploitation can be supplied by an economical upper stage for which existing hardware and technology are available at very modest cost.

REFERENCE

ANON.: Improved Lunar Cargo and Personnel Delivery System Study. Contract NAS 8-21000, Lockheed Missiles & Space Co.