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NASA Contract NAS 5-302

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME IV. ON-BOARD PROPULSION
Book 1 — Text and Appendix P-C

Program Manager: Dr. G. R. Arthur
Project Engineer: H. L. Bloom

Prepared for:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Contract NAS 5-302

May 15, 1961

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requires an additional 200 lb for the pressurization system. A further source of weight increase, 100 lb, is attributable to the extra thrust chambers.

Figure I-4-6 shows a summary of the range of payload capabilities of various propellant combinations as calculated by the Aerojet-General Corporation. The figure is based upon a mission velocity of 7500 ft/sec and assumes a pressure-fed system operating at 100 psia chamber pressure with an expansion ratio of 40:1. It indicates that payloads in excess of 7000 lb can be obtained only with the high-energy cryogenic combinations or with the more advanced storable systems using light metal hydrides or slurries. Sufficient experience with these latter propellants does not exist to recommend them for use on manned vehicles in the time span under consideration.

In Figure I-4-7, payload as a function of mission velocity is shown for the hydrogen/oxygen propellant combination. Values are given for several weights at boost burnout.

On the basis of the parametric studies outlined here, it is quite evident that the total requirements of the APOLLO mission can be best satisfied by the application of a pressurized hydrogen-oxygen propulsion system.

4.2 SYSTEM SELECTED

4.2.1 Key Features

The on-board propulsion system for APOLLO has been selected to meet the basic mission requirements of safety, reliability, and performance. From the parametric and design studies, it appears evident that the on-board propulsion system can meet these objectives and provide both instant abort impulse for super-orbital return as well as the necessary velocity increment for lunar orbit and return. Use of a pressure-fed liquid hydrogen/liquid oxygen rocket engine provides the requisite high performance, yet permits attainment of the design objectives of reliability and safety.

The Aerojet-General on-board propulsion system, AJ-10-133, satisfactorily meets the requirements for the APOLLO mission and has been selected for the recommended vehicle. Other propulsion systems have significant advantages in specific areas, and are discussed in subsequent sections of this report.

Key features of the selected Aerojet-General AJ-10-133 Propulsion System are tabulated in Table I-4-IV.

TABLE I-4-IV. KEY FEATURES OF APOLLO MAIN PROPULSION SYSTEM (AEROJET-GENERAL AJ-10-133)

a.	Designed specifically for manned space flight
b.	High performance (I_{sp} = 430 sec) at low chamber pressure (65 psia)
c.	Safe, Reliable
d.	Versatile, potential growth
e.	Simple, reliable, pressure-fed propellants
f.	Single powerplant for all maneuvers
g.	New, super-insulation (SI-4) permits sealed storage for fourteen days
h.	Simple, proven ablative thrust chambers
i.	Redundant thrust chambers and critical components
j.	Proven pressurization system(s)
k.	Reliable ignition (4 igniters per chamber + O_3F_2 for hypergolicity)
l.	H_2/O_2 propellants are safe, nondetonable, nontoxic, noncorrosive, readily available to the engine
m.	Compatible with space environment
n.	Fuel energy management system
o.	Instant readiness for super-orbital abort (24,000 lb thrust)

These key features are discussed below.

a. The proposed APOLLO engine is designed specifically for manned space flight and incorporates existing technology and components where applicable. The propulsion system can thereby be built up as an integrated system to meet the vehicle requirements of safety, reliability, and performance rather than attempting to compromise the APOLLO to existing engines which are neither designed nor qualified for manned space flight.

b. The engine will provide the requisite high performance (I_{sp} = 430 sec) with liquid H_2/ liquid O_2 at a low chamber pressure of 65 psia which facilitates safety and reliability.

c. Using representative numbers for unit reliability (see Paragraph 4.2.2), the probability of providing safe propulsion throughout the mission should be at least 0.978 and the probability of achieving a successful mission of approximately 0.954.

d. The single-propulsion package has both versatility and progressive growth capacity. The basic vehicle for 1963 weighs 18,000 lb at booster burnout for a payload of 7940 lb, and if boosted to lunar trajectory velocity, provides for a velocity increment of 7500 ft/sec, or provides sufficient propulsion to carry itself to escape and at a velocity of at least 6000 ft/sec after escape. Undertanking the propellants permits reduction of the vehicle weight to 14,025 lb which can be carried to escape by the Saturn C-2 or orbited by the Saturn C-1 with sufficient velocity for super-orbital abort. Thus, the complete powerplant can be checked out early in the program under actual operating conditions and the propulsion impulse increased for later lunar flights. Growth of the powerplant can be readily achieved, so that by 1966 the vehicle should be capable of achieving the lunar orbit mission.

e. The APOLLO propulsion features the simplicity and reliability of a pressurized-fed system. Such a system is inherently simple, should be available at an early date at low cost but with high performance. Pump-fed systems have been compared in many configurations but cannot better the payload-carrying capacity achievable with H_2/O_2 at 65 psia in a vacuum. Further, pumped engines are complex and require considerable conditioning for proper engine starts. This means that storage of LH_2/LO_2 would be difficult and inefficient with a significant weight of propellants lost in cooling down the engine to reach temperature equilibrium during starts. In addition, throttling to reduced thrust on a single chamber is quite feasible.

f. The single powerplant is capable of providing all necessary velocity increments during the APOLLO mission. This includes midcourse corrections, lunar orbit and de-orbit, and any other required maneuvers.

g. Use of the new super-insulation, such as SI-4, readily permits storage of liquid H_2/ liquid O_2 for the fourteen-day mission without excessive pressure use or the need for venting a most difficult task in a zero-gravity trajectory.

h. The simplicity and proven reliability of the ablative thrust chambers should greatly enhance the over-all system reliability. At the low chamber pressures of 65 psia, it is easily within the state-of-the-art for any single chamber to operate for the entire burning time of 546 seconds. Without cooling passages, start and shutdown can be quite rapid, (about one second) with minimum loss of H_2. Further, these chambers avoid potential leakage areas associated with regeneratively cooled chambers. With the ablative cooled chamber walls, operation of a single chamber at partial thrust is facilitated, and starts can be made at low flow, if desired, to settle the propellants.

i. Multiple redundant thrust chambers, tanks, and critical valves ensure high reliability and safety. For illustrative purposes, if chambers have a demonstrated reliability as low as 89 percent, two units raise the reliability to 98.74 percent and four units raise reliability to 99.99 percent. This in turn, should lower the cost and speed development, since for this illustration it is only necessary to demonstrate a chamber reliability of 89 percent. This can be done with only a few engine tests. As chamber reliability rises with development, it would probably be advantageous to consider the use of two chambers with double the thrust.

j. The proposed engine utilizes the proven pressurization system developed by Aerojet-General for the Hylas engine under AF Contract 04(611)-5170. An alternate pressurization system (which is probably mutually compatible with the proposed Hylas system) is designated VaPak by Aerojet and should provide a "belt and suspenders" redundancy for feeding propellants. The initial pressurization system, out to lunar orbit, will be the Hylas type, pressurizing the liquid O_2 with heated H and the liquid H_2 with heated gaseous H_2. The return from the Moon could be achieved with either the Hylas or the VaPak system — either of which should be adequate for pressure feed. Thus, if the "belt" fails, the "suspenders" can still keep the pants up.

k. Demonstrated reliable ignition of the H_2/O_2 using four surface-gap spark plugs in each of the chambers should further enhance reliability. Further ignition safety may be incorporated by using O_3F_2 which has been shown to produce hypergolicity of these propellants by Temple Research Institute. As little as 0.05 percent O_3F_2 has reliably produced ignition when in solution with the liquid O_2 in small thrust chambers. More research is needed on O_3F_2

l. The propellants selected (O_2 and H_2) are safe, nonexplosive, nontoxic, noncorrosive, and are readily available. Excellent experience is available from over a decade of testing, handling, and storage. The propellants are compatible logistically with the upper stages of Saturn, and are daily being handled safely on a tonnage basis.

m. The proposed system is compatible with space environment. The natural vacuum of space facilitates storage and permits operation of the thrust chamber at high performance with low chamber pressure. Protection and redundancy of components provide safety in the space environment of radiation and meteorites. The ablative chambers and radiation cooled skirts are fairly resistant or unsensitive to meteorite puncture.

n. A fuel energy management system is provided for conservation and best utilization of remaining propellants, particularly in the event of a malfunction. Further, there is the possibility of manually monitoring the utilization of propellants to assure minimum residual propellants.

o. The pressurized system will be in readiness during boost so that superorbital orbits can be effected with rapid (one second) application of full, 24,000-lb thrust.

Other features of the selected system are described in the Aerojet report, Appendix P-A. Specific examples of other possible advantages include use of the heated H_2 alone for attitude control (I_{sp} of H_2 gas is 200 seconds at 270 degrees R), use of the O_2 for breathing in an emergency, use of the H_2/O_2 for the fuel cells in an emergency, and possible use of the settling jet for small corrections in ΔV or for precise impulse termination.

4.2.2 Main Propulsion System Design

4.2.2.1 GENERAL

The basic propulsion system selected for discussion here is the Aerojet-General AJ10-133 system described in the Model Specification in Appendix P-A, Aerojet's section. This engine, shown in Figure I-4-8, is designed to be available for flight in 1963, and may be used with either the D-2 direct re-entry vehicle or the R-3 lenticular vehicle. Gross weight at boost termination, if propellants are completely loaded, is 18,000 lbs for the D-2 vehicle.

Performance with these two vehicles is shown in Table I-4-V. In each case, the total weight exceeds the allowable weight of 15,000 lbs which the Saturn can boost to escape. For purposes of this discussion, the analysis will be confined to performance of this powerplant with the D-2 vehicle, although obviously the same reasoning would apply to the R-3.

4.2.2.2 D-2 PROPULSION PERFORMANCE AND WEIGHT

The actual weight and performance with the D-2 are shown on Table I-4-VI. For a vehicle weight of 18,000 lb at boost termination, a payload of 7940 lb may be given a velocity increment of 8450 ft/sec. Part of this propulsion (1440 ft/sec) can be used for escape, leaving a capability of over 6000 ft/sec for maneuvering after escape or super-orbital abort. Or, the powerplant is capable of giving the stage a velocity of 7500 ft/sec with 5 percent reserve, 3 percent outage.

During the 1963 period, the basic AJ10-133 powerplant will be available for earth-orbital and near-space missions. The propellant may be undertanked as illustrated in Table I-4-VI to provide the basic 15,000 lb which the Saturn can boost to escape, thus reducing the total available velocity to 4840 ft/sec. A combination of reducing

Note: In discussing the AJ10-133 APOLLO powerplant, it has appeared appropriate to restate much of the material prepared by the Aerojet-General Corporation. The attempt has been made to bring this material into sharper focus, but this has necessitated repeating some of their material. Specific credit is given where possible, and reference is made to Appendix P-A of this report for more complete details of the AJ10-133 propulsion system.

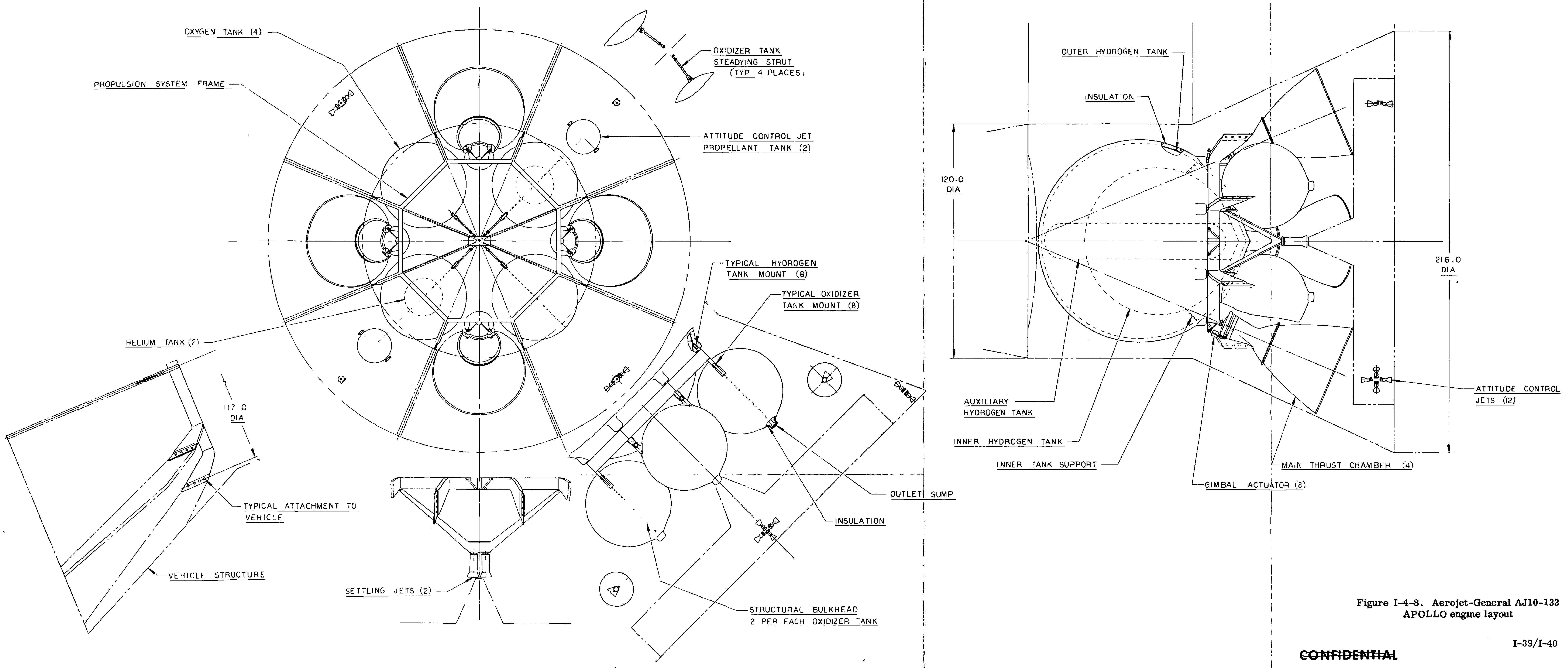


Figure I-4-8. Aerojet-General AJ10-133 APOLLO engine layout

TABLE I-4-V. LIQUID OXYGEN/LIQUID HYDROGEN 5 PERCENT RESERVE

	D-2 Re-entry Vehicle	R-3 Lenticular Vehicle
PENALTY WEIGHTS, lb		
Adapter	326	200
Small Separation Rockets	43	---
Large Separation Rockets	54 (335)[1]	---
Abort Rockets	552 (1829)[1]	407 (1128)[1]
Total Penalty Weight	975	607
Payload Weight	7940	9025
Propulsion System Weight (5% Reserve Propellants)	10,060	10,060
Useful Weight at Boost Termination	18,000	19,085
Total Weight on Pad	20,520	20,413
Total ΔV 5% Reserve (Stage Velocity) ft/sec	7500	7000
Total ΔV 3% Ullage No Reserves, ft/sec No Ullage	8450	7660
Mission ΔV (After Escape) ft/sec (5% Reserves, 3% Ullage)	6060	5200
Mission ΔV (After Escape) ft/sec (No Reserves, No Ullage)	7010	5860
[1] Total on the Pad Weight		

TABLE I-4-VI. SUMMARY OF APOLLO D-2 PROPULSION WEIGHTS AND PERFORMANCE

	Weight, lb			
	1963 System Under-tanked	D - 2 Vehicle Basic Proposed System	1966 System D - 2A Vehicle	D - 2X Vehicle
Vehicle Weight @ Boost Burnout	14,025	18,000	19,300	25,600
Total Vehicle Weight on Pad	16,545	20,520	21,833	28,133
Payload Weight	7940	7940	7000	7983
Propulsion System Weight (Incl. att Contr , sep. rockets)	6085	10,060	12,417	17,734
Propulsion Fixed Weight (Incl. gas, att cont. units)	1684	1684	1725	1900
Burnout Weight (No Reserve or Ullage)	9858	9741	8842	10,000
Available Propellants Weight*	4167 (tanks not full)	8142 (tanks filled)	10,458	15,600
	Δ V, ft/sec			
Maximum Δ V** (assuming use of reserves)	4840	8450	10,820	13,600
Δ V Used to achieve escape	None	1440	2050	3740
Δ V after escape w/5% res., 3% outage	3820	6060	7500	7500
Δ V after escape, no reserve	4840	7010	8770	9860
Δ V of stage with 5% res., 3% outage		7500	9550	11,240

* Does not include attitude control propellants

** Calculation for H_2/O_2, I_{sp} = 430 sec the payload (from 7983 to 7384 lb) and undertanking provides a capsule in the 15,000-lb class which has the capability of 5600-ft/sec velocity for super-orbital abort.

Thus, the basic powerplant can be proven, along with the APOLLO capsule in numerous missions prior to cis-lunar flights. The curve in Figure I-4-9 illustrates the range of ΔV achievable with this powerplant by undertanking the propellants. For example, the complete APOLLO vehicle could be launched with the Titan II vehicle at a weight of approximately 12,000 lb and provide a ΔV of 2600 ft/sec to help get the APOLLO capsule into a low earth-orbit and de-orbit. This would permit an early test of the capsule and propulsion system.

Improvements and weight reductions, available during this period, should permit reduction of the payload and powerplant specific weights so that by 1966 the D-2A vehicle should be realizable. This vehicle is illustrated in column 3 of Table I-4-VI and would have a vehicle weight at boost burnout of 19,300 lb for a payload weight of 7000 lb. This vehicle would then be capable (in 1966 when the C-2 booster was available) of propelling itself out to the Moon, orbiting and de-orbiting, and returning to the earth.

The D-2X vehicle represents a backup for the consideration of how the 1963 payload of 7983 lb could be orbited around the Moon and returned. Here, with today's state-of-the-art, this mission can be accomplished, but with a vehicle weight of 25,600 lb at boost completion.

4.2.2.3 ENGINE DESCRIPTION

The propulsion package for the D-2 configuration will utilize existing technology and components, where suitable, to provide a simple, reliable, high-performance rocket engine system. Selection of a pressurized propellant-fed system facilitates achievement of these goals by means of simple, uncooled, ablative thrust chambers similar to those developed by Aerojet General under Contract AF 04(611)-5170.

The configuration selected was determined by the thrust level required and envelope requirements. Super-orbital abort maneuvering necessitates a thrust of 24,000 lb for an average acceleration of about 2g's.

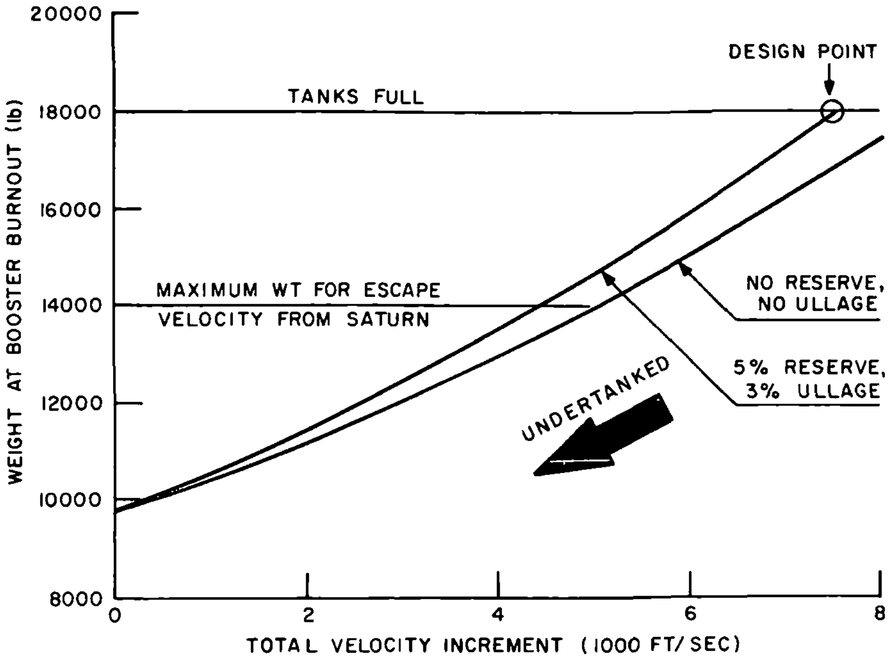


Figure I-4-9. D-2 vehicle performance payload = 7940 lb

Four individual chambers of 6000-lb thrust were selected, each capable of providing the necessary space maneuvers alone. All four chambers are fired for 24,000 lb thrust required in super-orbital abort. Two chambers of 12,000-lb thrust, throttleable to 6000 lb, would fulfill the same mission but exceed the available length. Therefore, the proposed engine is designed around the four chambers which provide an excellent reliability with high redundancy.

A summary of the 1963 engine dry weights is shown in Table I-4-VII and over-all system weights in Table I-4-VIII. Reduction of the super-orbital abort thrust to 12,000 lb would allow savings of approximately 300 lb in engine weight.

The four main thrust chambers are canted at 23 degrees to align thrust with the center of gravity, and each may be gimballed 5 degrees in any direction to follow center-of-gravity travel. This thrust is applied at four places on an octagonal ring which forms the main structural member of the propulsion system. All components except the attitude control system are mounted on the octagonal ring or on substructures attached to it. Thus, the principal components of the propulsion system are integrated in an assembly that may be acceptance tested, transported, and installed in the vehicle

TABLE I-4-VII. REVISED AEROJET-GENERAL PROPULSION SYSTEM NOMINAL DRY WEIGHT SUMMARY

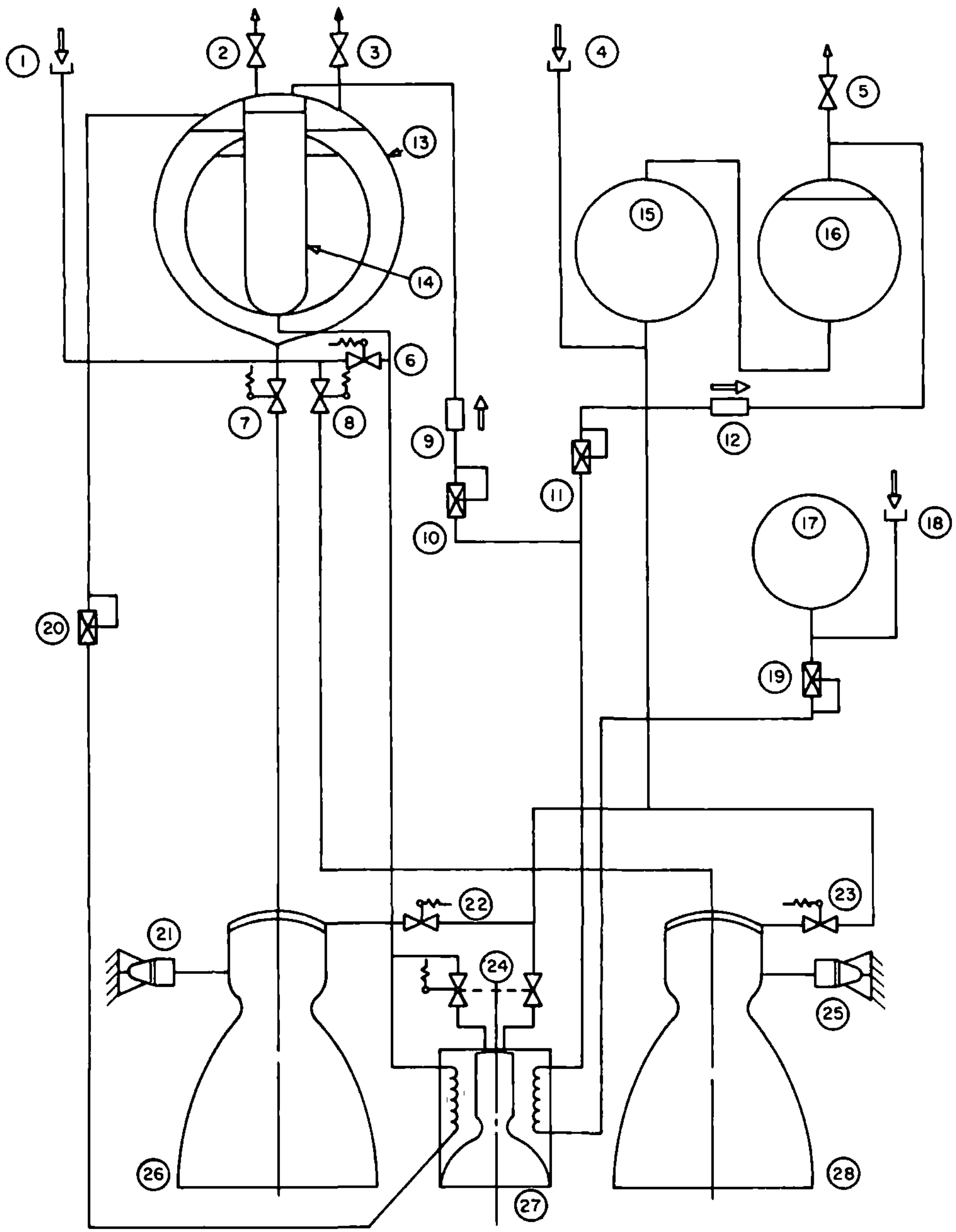
	Unit Dry Weight, lb	Number Required	Total Dry Weight, lb
Fuel tank, outer, with insulation and mounts	335	1	340
Fuel tank, inner, with support cone	80	1	80
Fuel tank, auxilliary	22	1	22
Oxidizer tank, with insulation and mounts	54	4	216
Helium tank and supports	54	2	108
Thrust chamber assembly with propellant valves and gimbal actuators	128	4	512
Settling jets	10	2	20
Structure	121	1	121
Lines, fittings, valves, electrical	42	1	42
Attitude control thrust units	1	12	12
Attitude control tankage	25	1	25
Total Dry Weight, lb			1498

TABLE I-4-VIII. AEROJET-GENERAL APOLLO D-2 PROPULSION SYSTEM LOADED WEIGHT

		1963 System
Powerplant Weight Summation		
Propellant		8,376 lbs.
Outbound midcourse	361	
Orbit maneuvers	7,562	
Inbound midcourse	219	
Attitude control	234	
Other Fluids		143
Fuel used for pressurization	120	
Helium	23	
System Dry Weight (including attitude control units)		1,498
Small Separation Rockets		43
Total Loaded Weight		10,060 lbs.

without disassembly or other operations which might disturb its proven operability. This same assembly may be left behind as a unit during the launch abort escape maneuver.

Envelope and heat transfer considerations dictate the use of a single spherical or near spherical hydrogen tank. To minimize length and remain within the specified envelope, the oxidizer was divided into four tanks spaced between the thrust chambers. This basic configuration is shown in detail in Figure I-4-8. A schematic of one-half of the propulsion system is shown in Figure I-4-10. The aft support structure is separated and left with the boost vehicle to leave the chambers free and to prevent impingement of the exhaust upon the aft skirt of the vehicle. Although not required for single thrust chamber operation, it may be necessary to provide a flame shield to restrict base recirculation and heating when all four thrust chambers are in operation. A suggested



Item	Description
1	*Fuel Fill Disconnect*
2	Auxiliary Fuel Tank Vent Valve
3	Fuel Tank Vent Valve
4	Oxidizer Fill Disconnect
5	Oxidizer Tank Vent Valve
6	Auxiliary Fuel Hydrogen Fill Valve
7	(1) Thrust Chamber Fuel Valve
8	(3) Thrust Chamber Fuel Valve
9	Auxiliary Fuel Pressurization Check Valve
10	Auxiliary Fuel Pressure Regulator
11	Oxidizer Pressure Regulator
12	Oxidizer Pressurization Check Valve
13	(1) Fuel Tank
14	Auxiliary Fuel Tank
15	Oxidizer Tank
16	Oxidizer Tank
17	(1) Helium Tank
18	Helium Fill Valve
19	Helium Pressure Regulator
20	Fuel Tank Pressure Regulator
21	(1) Thrust Vector Control Actuator's (2 Req'd)
22	(1) Thrust Chamber Oxidizer Valve
23	(3) Thrust Chamber Oxidizer Valve
24	Settling Jet Bipropellant Valve
25	(3) Thrust Vector Control Actuators (2 Req'd)
26	(1) Thrust Chamber
27	Settling Jet
28	(3) Thrust Chamber

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Figure I-4-10. Lunar insertion propulsion system schematic installation is shown in Figure I-4-11. Although ineffective as a radiation shield for the propellant tanks (which are already shielded), this device might offer some meteoroid protection and limit radiation to the supporting structure.

Using the method described in the Aerojet appendix, a parametric study was performed by Aerojet General to select the optimum levels of thrust chamber pressure, expansion area ratio, and propellant mixture ratio. The results of this analysis, based upon utilization of a Hylas Type pressurization system and four ablative thrust chambers, is presented in Figures I-4-12, I-4-13, and I-4-14. Figure I-4-12 (at a propellant mixture ratio of 5.0) shows the optimum thrust-chamber pressure to be a function of expansion area ratio with a nominal value of 70 psia at a 40:1 expansion. Optimum expansion area ratio, as shown by Figure I-4-13, is in the 40:1 to 50:1 range with little advantage for values over 40. Figure I-4-14 indicates optimum propellant mixture ratio to be just under 5.0. Selection of the Hylas design point of 65 psia chamber pressure, 40:1 expansion area ratio, and 5.0 mixture ratio as indicated on the curves (and at which considerable design and experimental work have been performed) represents almost exactly the optimum operating condition. Packaging considerations indicated that it was necessary to reduce the expansion area ratio to 35. Figure I-4-13 shows, however, that this does not result in an appreciable weight penalty.

4.2.2.4 DETAILED DESIGN FEATURES

4.2.2.4.1 Thrust Chamber Assembly

The Aerojet thrust chamber assembly consists of an ablative cooled combustion chamber and nozzle bolted to a lightweight aluminum injector. Thrust mounts and propellant valves are attached directly to the injector. The nozzle will be radiation-cooled between the area ratio of approximately 3:1 and the exit area ratio of 35. Table I-4-IX summarizes the AJ10-133 thrust chamber data and performance. Initial phases of development of the combustion chamber were completed during the Hylas program.

The Aerojet combustion chamber is constructed of an ablative liner, a thin layer of insulation, and a high-strength overwrap. This provides the high thermal resistance and the high strength needed for a lightweight design. The first 12 in of the ablative

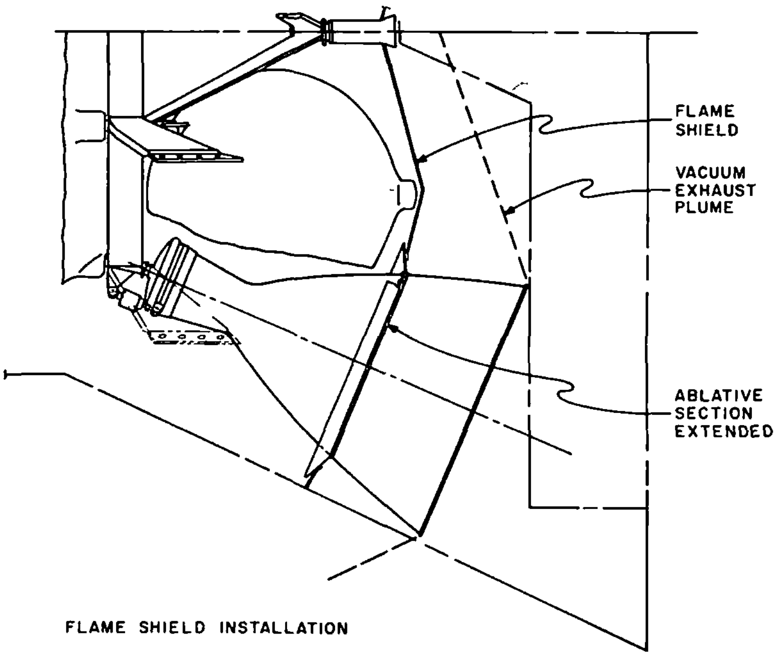

Figure I-4-11. Possible flame shield installation for AJ10-133 engine

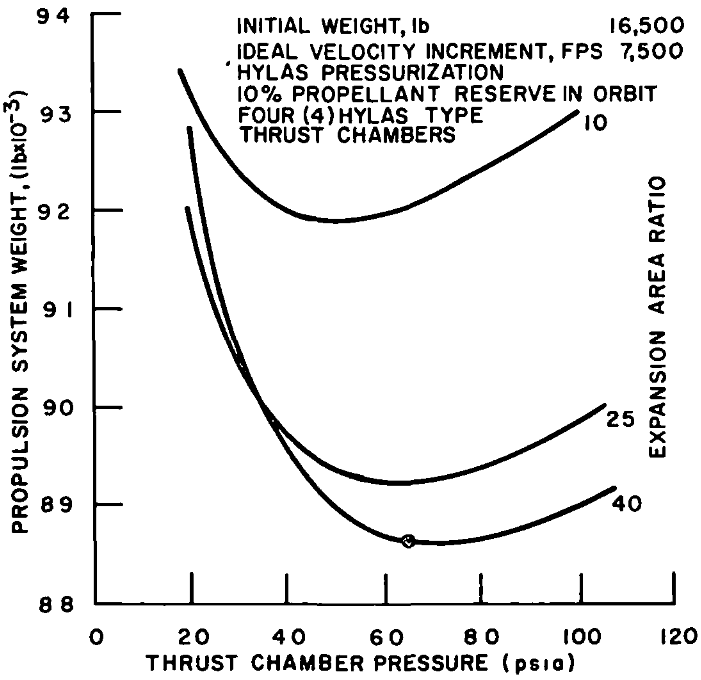

Figure I-4-12. Effect of thrust chamber pressure of propulsion system weight

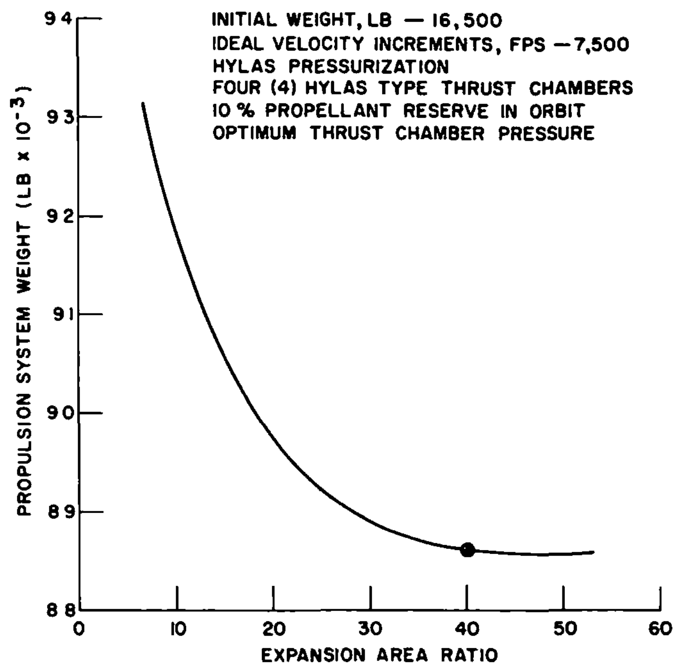

Figure I-4-13. Effect of expansion area ratio on propulsion system weight

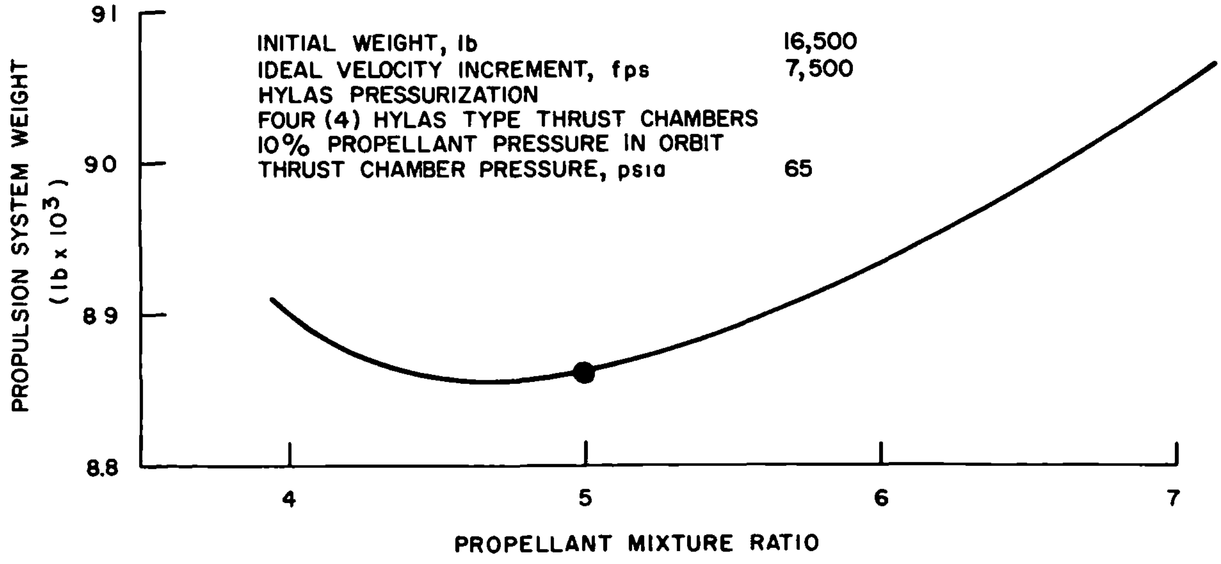

Figure I-4-14. Effect of propellant mixture ratio on propulsion system weight

TABLE I-4-IX. THRUST CHAMBER DATA

	Single Chamber	Four Chambers
OPERATING CONDITIONS		
Thrust (vacuum), lb	6,000	24,000
Propellants	LO_2/LH_2	LO_2/LH_2
Chamber Pressure, psia	65	65
Propellant Flow Rate, lbm/sec	13.95	55.8
Mixture Ratio	5:1	5:1
Expansion Area Ratio	35:1	35:1
Specific Impulse (vacuum), sec	430	430
Maximum Total Duration of Full Thrust, sec	546	137
DIMENSIONAL DATA		
Overall Length, in.	61.3	-
Exit (outside) Diameter, in.	50.0	-
Throat (inside) Diameter, in.	8.05	-
Contraction Ratio	2:1	-
MATERIALS		
Injector	Aluminum	-
Combustion Chamber	Ablative Plastic Fiberglass wrapped	-
Expansion Nozzle	Titanium	-

liner is composed of phenolic-impregnated asbestos fibers, edge-wrapped with a 60-degree orientation to gas flow. The area from 12 in below the injector to 5 in below the throat is 60-degree edge-wrapped Refrasil (phenolic-impregnated quartz fibers), and the nozzle portion from 5 in below the throat to an area ratio of 3:1 is the same type asbestos wrap as the upper chamber. A thin wrap of tangentially oriented phenolic-impregnated asbestos is used on the outside of the Refrasil portion for insulation. The high-strength overwrap of the entire assembly is composed of glass cloth for longitudinal strength and circumferential-wound glass filaments for hoop stress. The glass wrap is bonded with epoxy resin. The high thermal resistance of the ablative liner, plus the asbestos insulation behind the Refrasil, isolates the outer wrap and permits it to be used at moderate temperatures where strength is high. The use of nonmetallic materials at moderate temperatures (300 F) in vacuum conditions for periods of 30 days has been shown to be no problem.* Specimens subject to these conditions have shown a 1-2 percent decrease in ablative material weight and a very slight loss in flexural strength. Similar control specimens subject to the same temperature history but at sea level pressures show similar changes in properties substantiating the theory that with chain polymers the temperature rather than the vacuum is the rate controlling factor and the process is one of pyrolysis rather than evaporation or sublimation.

Following shutdown of an ablative thrust chamber after a long-duration run, the chamber will continue to ablate until it cools below the ablation temperature. The method of Appendix P-A shows that this required approximately 30 seconds and, for the Aerojet chamber, will result in a char depth growth of approximately 10 percent. Thus, any reasonable number of restarts can be designed for by selecting a suitable thickness of ablative material. The chamber recommended for this application is capable of up to 17 firings. On short duration runs such as may be required for course corrections, the heat sink capability of the chamber may not be exceeded and the ablation process not started. See Appendix P-A.

* Research and Development on Components for Pressure-Fed Liquid Oxygen-Liquid Hydrogen Upper Stage Propulsion Systems, Report No. 1933 (Final) Contract AF 0416 (616) - 5170, Aerojet General Corporation, Azusa, Calif.

The ablative material will be terminated at an area ratio of 3:1, and a radiation-cooled metallic skirt will be attached through a bolt-on flange. The mass of the flange is sufficient to avoid an excessive temperature rise with the resultant bonding problems. A trapped O-ring seal is used to provide for convenient assembly of the thrust chamber and nozzle at the launching or test site.

When more than one thrust chamber of the cluster is in operation, cross radiation between nozzle expansion skirts will take place raising the skirt temperature. The most critical condition exists on the portion of the nozzle nearest the vehicle centerline when all four thrust chambers are in operation. Due to the relatively wide spacing of the thrust chambers and the fact that the exhaust plume is transparent to radiation from the nozzle skirt, the solid angle viewed by a nozzle element at this location is reduced by only 19 percent. The resulting 5-percent rise in temperature is readily compensated for in the design.

Test firings at Aerojet of radiation-cooled nozzle extensions with clusters of 1/16-in holes drilled at area ratios of approximately 10, 15, and 25 have been conducted to verify that skirt integrity will be maintained in the event of a meteoroid puncture. Post fire examination of the skirts after tests of 30 seconds duration at a chamber pressure of 150 psia revealed no apparent growth of the holes.

Two injector configurations are envisioned by Aerojet for the experimental phase, one is a conventional, concentric-ring, shower-head design, and the other is a design containing a multiplicity of rosettes in a face lined with ablative material. In both designs, intermanifold welds are minimized, and rapid breakup of the oxidizer is emphasized. This latter operation has been shown experimentally to be the key factor in achieving high performance with LO_2/LH_2 propellants. A simple "mono-ball" structure is used for thrust take out. This design permits easy accessibility for servicing.

Ignition is accomplished in the Aerojet chambers by four surface-gap spark plugs located around the periphery of the injector. These plugs are positioned such that the injector film cooling will protect them during steady-state operation. During the starting sequence, a 0.1 sec oxidizer lead is programmed to provide oxidizer in the area of the plugs at the time fuel flow starts. Tests have proven this lead time to be adequate for ignition to occur before the fuel film blankets the plugs. This system has been developed by Aerojet and proved in over 30 firings on Titan-size hardware using LO_2/LH_2.

Estimated start and shutdown transients of the AJ10-133 engine are given respectively in Figures I-4-15 and I-4-16. The start transients, as shown in this curve, are based on pressurized tanks. For initial runs of the system when the ullage is small, pre-pressurization of the tanks can be accomplished in 1 to 2 seconds. This would be the situation in the event of a super orbital abort. Later runs, where the tank ullage is high, might require several seconds pressurization time.

Figure I-4-17 shows the degradation in performance associated with short-duration runs due to the inefficiency of the start and shutdown transients. These data are based upon an average of several Aerojet Hydra-Hylas test runs which indicate an effective specific impulse of 340 sec (corrected to vacuum) during the start and shutdown periods.

4.2.2.4.2 Pressurization System

For propellant pressurization, the AJ10-133 system utilizes hydrogen to pressurize the fuel and helium to pressurize the oxidizer. This system has four principal components: An auxiliary fuel tank, a helium-sphere, a heat exchanger, and a settling rocket. The design parameters used have all been verified by the Hylas test program in over 40 expulsion tests.

To provide a positive pressure differential between the supply of pressurization fluid and the fuel tank, a pressurized auxiliary tank is used. Hydrogen is stored as a liquid in the auxiliary fuel tank to keep the volume and weight of the tank to a minimum. This is accomplished by submerging the auxiliary tank in the main fuel tank, which also saves space and eliminates the need for insulation. The liquid hydrogen is supplied to the heat exchanger by helium pressurization of the auxiliary fuel tank. The use of helium for this application does not present any problems, because the density of the helium at the design temperature and pressure (38 R, 185 psia) is less than the density of liquid hydrogen under the same conditions.

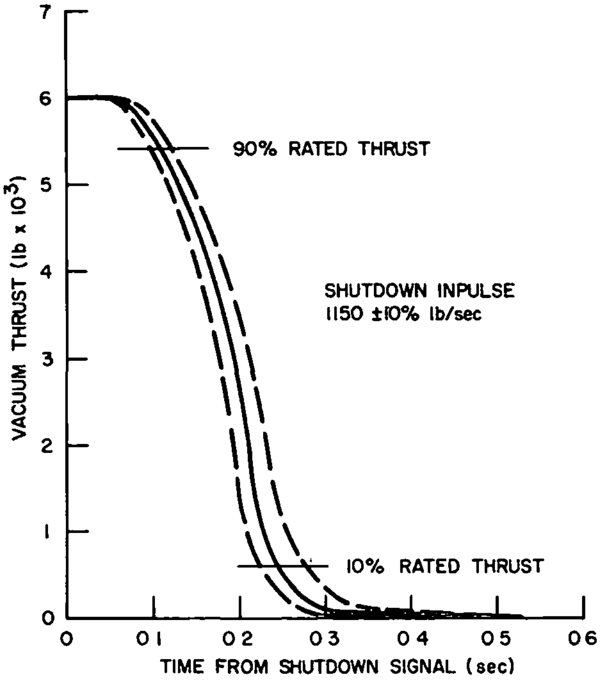

Figure I-4-15. Estimated start transient

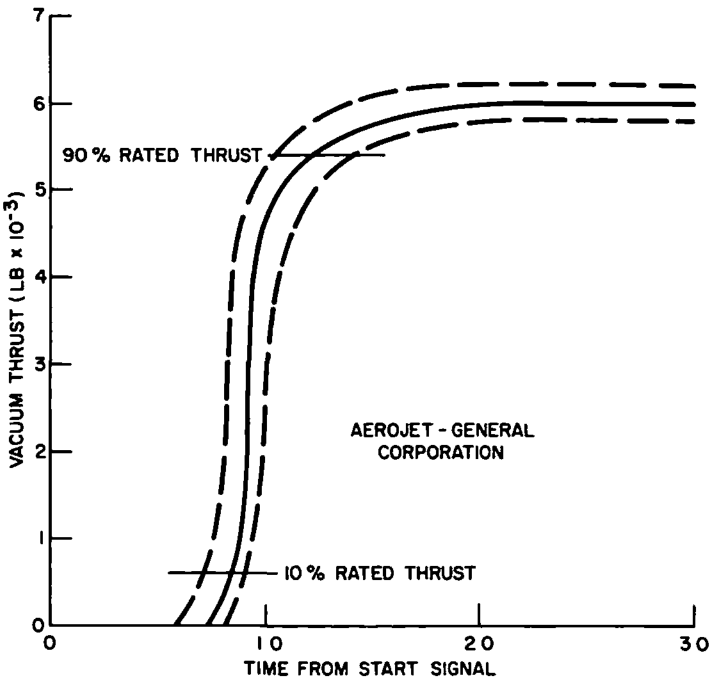

Figure I-4-16. Estimated shutdown transient

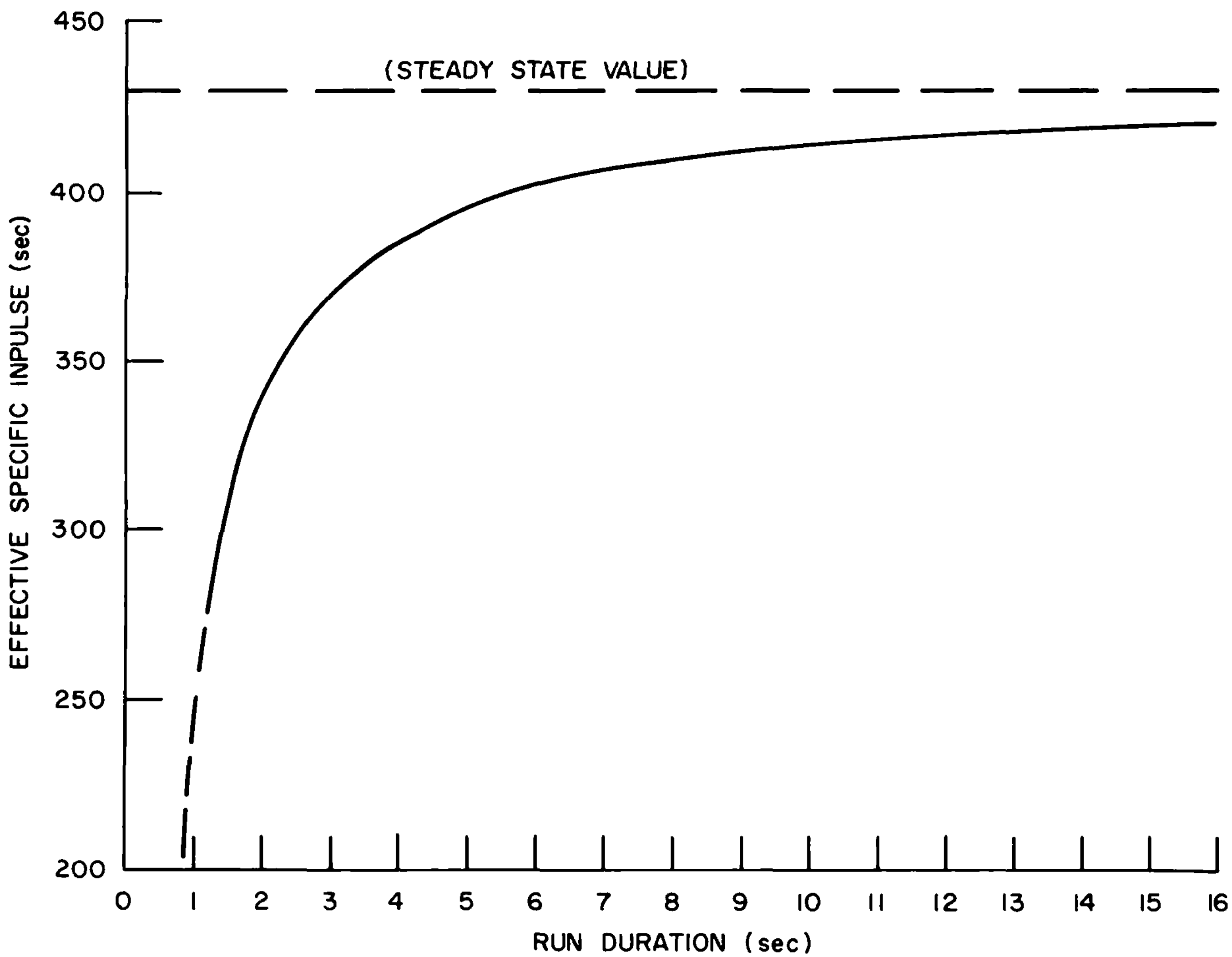

Figure I-4-17. Effective Specific impulse for short duration (course correction) firings

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4.2.2.4.3 Tankage and Structure

A titanium alloy was chosen for the liquid hydrogen tanks which consist of an outer spherical tank, an inner spherical tank, and a cylindrical auxiliary tank supplying hydrogen for pressurization. The alloy (A110-AT) may be readily formed and welded, has a high strength/density ratio without heat treat, and has good impact strength at -423 F.

Titanium is not proposed for use in the liquid oxygen tanks because of questionable compatibility. Previous experience has indicated such usage might be hazardous. Instead, a heat treatable aluminum alloy 6061, is used for the liquid oxygen tanks. Two hemispheres are fabricated and heat treated to the T6 condition. The helium storage sphere is installed, and the hemispheres are inert arc gas welded together. Wall thickness at the girth weld is great enough to reduce stress below yield in the heat-affected area. A heat-treatable alloy was used instead of depending on work-hardening for high yield strength because of the several bosses and attachments which may conveniently be welded-on before heat treat. A material with a higher strength/density ratio, such as heat treated AM350 or 17-7 PH, was not used because the tank wall thickness is already at the minimum for handling loads with the aluminum.

The two helium storage spheres for the main propellants are fabricated from AM350, heat-treated to a room temperature yield strength of 135,000 psi. The two helium tanks immersed in liquid oxygen have a yield strength of 190,000 psi. This material is compatible with the oxygen, may be welded and machined before heat treat, has a high strength/density ratio especially at cryogenic temperatures, and has sufficient ductibility at the temperature of liquid oxygen. A summary of tank data is included in Table I-4-X.

The octagonal ring which constitutes the principal member of the propulsion system frame is supported by eight attachments to the vehicle structure. The ring, in turn, supports the hydrogen tanks, the four oxygen tanks, the four main thrust-chambers and a sub-frame on which the settling jets are mounted. The frame utilizes box-beam construction and is fabricated from 7075-T6 aluminum alloy sheet and extrusions. Its weight is 121 lb including all attachments.

TABLE I-4-X. TABULATED TANK DATA

Tank	Diameter, in.	Shape	Operating Pressure psia	Material	Wall Thickness in.	Number Used and Total Dry Weight, lb.
Hydrogen #1	100	Spherical	100	Ti A110-AT	0.025	1, 151
Hydrogen #2	75	Spherical	100	Ti A110-AT	0.020	1, 66
Hydrogen Aux.	18	Cylindrical (92 in. long)	200	Ti A110-AT	0.020	1, 22
Oxygen	44	Spherical	100	A1 6061-T6	0.033	4, 142
N_2O_4/A-50	17.5	Spherical	200	A1 6061-T6	0.040	2, 10
Helium	18	Spherical	4,000	Steel Am 350	0.150	2, 104
				Total Tank weight, lb (dry)		495

The use of eight points of attachment to the vehicle structure, a relatively high number, is compatible with the number of components which it supports and is structurally sound. The eight attachments provide good load distribution in the vehicle and low bending moments in the octagonal ring which is essentially an assembly of eight simple beams. At each corner of the octagon, a short radial beam spans the distance to the corresponding attachment point to the vehicle. The attachment points are located on a 117-in diameter circle. Each attachment transmits a maximum shear load of 6,000 lb and a maximum moment of 87,000 in-lb to the vehicle.

The applicable tank data are tabulated in Table I-4-X.

4.2.2.4.4 Thrust Vector Control Actuators

The AJ-10-133 engine uses thrust vector control actuators to allow thrust vector alignment through the vehicle center of gravity. Previous studies indicate that an electric motor servo mechanism with a ball-screw actuator is suitable for operation at very low temperatures such as are encountered in an O_2/H_2 system. Work is in progress at Aerojet on actuators for similar applications. Therefore, their use is considered feasible here.

4.2.2.5 COMPONENT STATUS SUMMARY

A brief description and status summary of major components are presented in Table I-4-XI.

4.2.2.6 MALFUNCTION DETECTION AND SEQUENCER UNIT

4.2.2.6.1 Purpose

The malfunction detection and sequencer distributes electrical power to control operation of the motors and engines. It can be designed using state-of-the-art principles similar to those used in the Malfunction Detection System for the Dyna-Soar engines presently being designed and the XLR91 (Titan State II) Airborne Sequencer.

Engine parameters can be monitored to detect incipient engine failure. These parameters will be used as criteria for engine shutdown and also initiate redundant equipment start up.

TABLE I-4-XI. SUMMARY OF COMPONENT STATUS AT AEROJET-GENERAL

Component	Description	Remarks
Combustion Chamber	Ablative cooled low pressure	Feasibility and design criteria established by Aerojet under AF 33(616)-7401 and AF 04(611)-5170 (Hydra-Hylas)
Expansion Nozzle	Radiation cooled titanium	Demonstrated on simulated altitude tests and five flights of Ablestar vehicle. Puncture tests showed skirt insensitive to meteoroid damage.
Injector	Concentric ring showerhead	High performance demonstrated by Aerojet on Hydra-Hylas test program.
Propellant Valves	Butterfly type	Scaled down versions of 1-203990 fuel valve and 1-203650 oxidizer valve used on Titan and approximately 55 oxygen hydrogen firings
Ignition System	Surface gap spark plugs	Use of 1 to 4 spark plugs demonstrated by Aerojet on Hydra-Hylas test program. Research indicates 0.05% O_3F_2 (wt % in O_2) may promote reliable hypergolic ignition.
Thrust Vector Actuators	Electric motor and ball jack-screw	Since high gimbal rates are not required, a simple system is possible utilizing commercially available parts.
Settling Jet-Heat Exchanger	Low pressure, low mixture	Performance not critical. Design criteria established on Titan and Hydra-Hylas programs. 1-234290 fuel and 1-234476 oxidizer (Titan gas generator valves) valves can be used.
Tankage	Spherical, conventional	Good background at Aerojet in flight weight pressurized propellant tanks. Experience and facilities from Able, Delta, Ablestar, Aerobee, and others applicable.
Structure	Box beam or tube, conventional	Standard structural frame techniques used.
Gas Pressure Regulators	Commercial products	Hadley P/N 10998 (hydrogen) and Skyvalve (helium) P/N R0102-8P performed satisfactorily in 40 pressurization system tests.

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Logic and timing for all phases of the flight except re-entry spin control can be contained in this unit. Temperature control of this device which may contain semiconductors could be obtained by installation in the mission module or by installation on the vehicle wall.

4.2.2.6.2 Electrical Power Requirements

Power input can originate from a single or dual (redundant) source. For operation of one thrust chamber, approximately 2 amp at 28 vdc and 5 amp at 115 v, 400 cps are required. For super orbital abort, approximately 4 amp at 28 vdc and 13 amp at 115 v, 400 cps are required. The duration of these requirements is only about half that for the normal mission. Attitude control thrust chamber valves require 0.5 amp each at 28 vdc. Since there are twelve such valves (any six of which could be operated at one time), up to 3 amp at 28 vdc could be used.

If minimum energy consumption is desired, the thrust chamber igniters may be turned off after ignition.

4.2.3 Propulsion System Operation

The selected system would use the following sequence of events: *

4.2.3.1 NORMAL

(1) Escape and high dynamic pressure separation rockets are jettisoned, four at first-stage burnout, six at second-stage burnout, and two at third-stage burnout.

(2) After boost, but before midcourse correction, attitude is automatically corrected by the attitude control system (ACS) which functions as needed for the duration of the flight up to re-entry unless over-ridden by the pilot

(3) Fuel gages and all tank pressures are checked to ensure that the propellant system is normal.

* Where engine restart is involved, the steps for restart are omitted for simplicity.

(4) All engines are checked for continuity at this time. (This may not be necessary, but is suggested for consideration.)

(5) All other checks of vehicle normalcy (electrical power supply for engine, etc.) are made at this time.

(6) #(1) Thrust vector control actuator is energized to move the engine to nominal firing attitude to minimize "kick" at start-up.

(7) Propellants are settled with the (1) settling jets. Thrust chamber igniters are turned on.

(8) #(1) Thrust chamber fuel and oxidizer valves are opened.

(9) The propellants ignite and burn, and thrust is supplied until the guidance computer determines that the velocity vector is correct.

(10) #(1) Thrust chamber fuel and oxidizer valves are closed.

(11) #(1) Thrust chamber igniters are turned off.

(12) During coast to the vicinity of the Moon, Steps 3, 4, and 5 are repeated as necessary.

(13) The engine is again fired using the procedure in Steps 6, 7, 8, 9, 10, and 11 to accomplish lunar insertion.

(14) The No. 1, lower, outer, LH_2 tank is vented to space to prevent pressure buildup and inner tank collapse.

(15) During the stay in orbit around the Moon, Steps 3, 4, and 5 are again repeated.

(16) The No. 2 engine is fired using a procedure similar to Steps 6, 7, 8, 9, and 10, except that No. 2 hardware is used to accomplish the lunar exit maneuver.

(17) During the coast back to the vicinity of the earth, Steps 2, 3, 4, and 5 are repeated for the final midcourse maneuver.

(18) The No. 2 engine is again fired to accomplish the midcourse correction. The main propulsion module is disconnected.

(19) At the correct point in relation to the earth, the separation rockets are fired to separate the re-entry vehicle from the spent spacecraft.

(20) Any spin of the re-entry vehicle is automatically corrected by the re-entry spin-control jets.

4.2.3.2 ABORT DURING BOOST OR ON THE PAD

(1) The booster malfunction detection system detects a booster malfunction necessitating abort.

(2) The propulsion system attachment bolts are fired.

(3) Aerodynamic drag on the lower section of the skirt separates the vehicle and main propulsion.

(4) The eight solid rocket abort motors are fired.

(5) The APOLLO vehicle (less main propulsion) is accelerated away from the Saturn booster for two seconds.

(6) The spacecraft aft shell is separated.

(7) The high dynamic pressure separation rockets are fired to separate the re-entry vehicle from the spacecraft.

(8) Any spin of the re-entry vehicle is automatically corrected by the re-entry spin control jets.

4.2.3.3 SUPER-ORBITAL ABORT

(1) After separation from the third stage, super-orbital abort is possible if immediate return to earth is required. This can occur any time after orbital velocity is achieved during third-stage burning. If the booster is not separated, the first step is to fire the two remaining abort rockets to escape from the third stage. The main propulsion module is retained in this situation.

(2) A decision is supplied on the most appropriate super-orbital abort maneuver.

(3) Initiate super-orbital abort.

(4) The attitude control orients the vehicle to the proper attitude.

(5) Steps 6, 7, 8, 9, and 10 of paragraph 4.2.3.1 are automatically performed except that engines 1, 2, 3, and 4 fire simultaneously to produce a 24,000-lb thrust.

(6) The vehicle is deflected and heads toward the atmosphere.

(7) Steps 18 and 19 are performed to accomplish re-entry.

4.2.3.4 ABORT AFTER BOOST BUT BEFORE LUNAR INSERTION

(1) During steps 1 through 7 of the normal sequence of events (paragraph 4.3.2.1), an uncorrectable situation is discovered. Procedure is normal except redundant equipment is used.

(2) The pilot decides to abort the attempted circumlunar mission and make a free return to earth (cislunar mission).

(3) If there is no danger of impacting the Moon, midcourse correction is delayed until after apogee is attained.

(4) Normal sequence is resumed starting with Step 16.

4.2.3.5 MALFUNCTION AFTER LUNAR INSERTION

(1) Procedure is Normal using redundant equipment.

4.2.4 Space Storage of Propellants

Perhaps the key to successful utilization of cryogenic, high-energy propellants is the successful storage and expulsion during the 14-day mission. Heat leaking into the propellants must be minimized by minimizing tank surface area and using good insulation (such as Linde SI-4 plus utilization of the vacuum of space), suitable tank supports, and an adequate pressurization system. Propellant tank venting to relieve the pressure built up by this heat is difficult to achieve for this mission and wastes propellant energy. Proper design and insulation of tanks should minimize the total pressure which can be kept well below 100 psi. Since minimum gage problems dictate that walls will stand at least 100 psi, it is not planned to vent the cryogenic propellant tanks during the mission.

Available space permits storage of hydrogen in a single exposed sphere. This provides minimum weight for the largest volume tank and minimum surface area to insulate. The liquid oxygen will be stored in four individual spheres.

Highly efficient lightweight insulations are commercially available which are suitable for space storage. Linde Type SI-4 has been tentatively selected as being representative of the multiple-radiation-shield type of insulation. It consists of 40 to 80 layers of aluminum foil per inch separated by submicron glass fiber paper. When the pressure of the insulating space is at 1 micron of mercury or less, the insulation has very low thermal conductivity. It has no structural strength, but will support its own weight under considerable vibration and shock loading. The aerodynamic shield used to stabilize the spacecraft during the early abort phase will serve to protect the insulation during boost. Sufficient studies have been conducted to conclude that this insulation will be adequate for the mission without a severe weight penalty. However, a detailed study is needed to determine the optimum insulation thickness. A thickness of 2 in on the hydrogen tank and 1/2 in on the oxygen tanks was selected for the preliminary design. The heat transfer rates and weights of insulation are shown on Figures I-4-18 and I-4-19. For these curves, it was conservatively assumed that the outer layer of insulation was at 530 F. The resulting heat transfer rates are probably somewhat high, since the outer layer of insulation will face other cold propellant tanks and structures as well as the warm outer skin of the vehicle.

The structural design of a vehicle using O_2/H_2 propellants has a great effect in determining the adequacy of the vehicle for space storage. Even if a highly effective insulation is used to reduce the amount of external energy absorbed, heat conduction through structural members can negate the effect of this insulation. Also, the structural members can serve as easy paths for heat from the various internal sources such as the payload and guidance and control units.

A common method of reducing the heat transfer to cryogenic fluids is to suspend the tanks on long, highly stressed tensile members. Because of the specific design requirements of the APOLLO, it is not possible to use tensile members without imposing a severe weight penalty. Therefore, a "heat barrier" system is used which employs the principle of a series of stacked plates, forming a laminated, multiple-contact

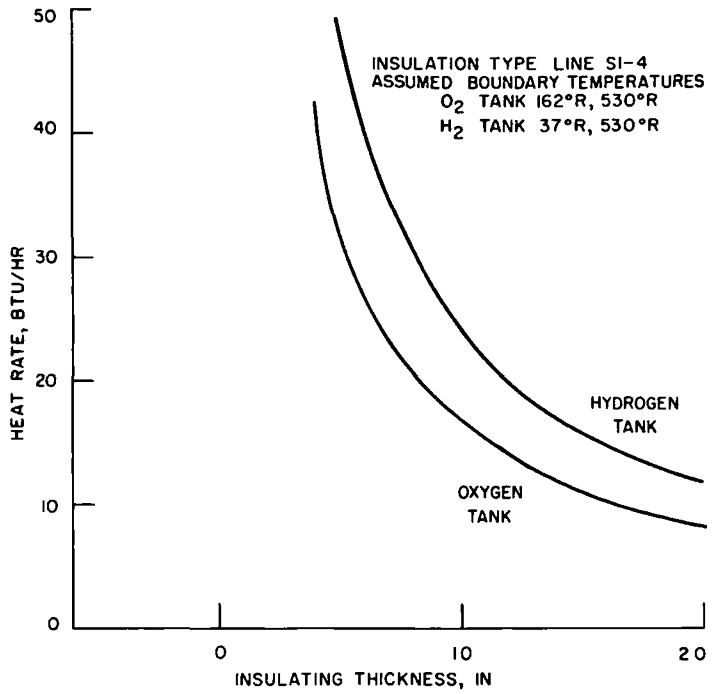

Figure I-4-18. Heat transfer rate vs insulation thickness

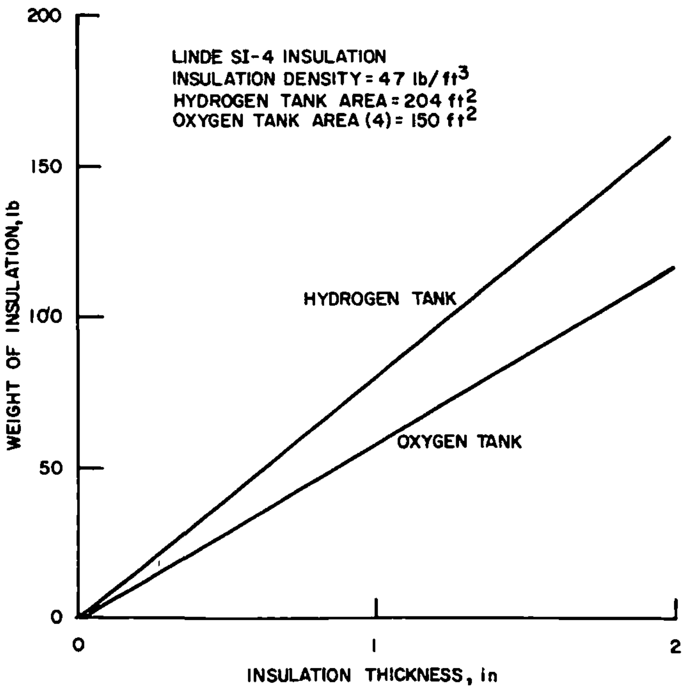

Figure I-4-19. Insulation weight vs thickness compression support member. The effectiveness of this principle has been demonstrated * The thermal resistance of the gap between two pieces of metal pressed together increases the thermal resistance of the member without reducing its compressive strength. The resistance of the gap may be further increased by dusting the plates with manganese dioxide, or by placing layers of Micarta between the metal plates. The actual configuration selected for the preliminary design utilizes a metal strip tightly rolled into a coil. A typical curve of the heat current through a member of this type is shown in Figure I-4-20. Since there is no load on the coil during the coast periods, the heat transfer will be low during these periods.

After the heat transferred to the tanks has been minimized, three methods of storage are possible: Storage in an unvented tank with a refrigerator to reliquefy the propellant boiloff, storage in a vented tank, and storage in unvented tanks, allowing the temperature and pressure of the propellants to rise. Storage by refrigeration was considered by Aerojet briefly and found to be undesirable for the low heat rates and short storage times of the APOLLO vehicle. Therefore, this method was not considered further.

The simplest way of storing cryogenic propellants is to utilize the heat capacity of the propellants by allowing the temperature and hence the vapor pressure to rise. By utilizing this method, the problem of venting the propellants in a gravity-free condition is circumvented, and no additional propellants must be carried along to compensate for losses due to venting. However, a decrease in density and stratification of the propellants may occur with diffusion and/or conduction of energy into the propellants being the main mechanism of heat transfer. At high rates of heat transfer, a vapor envelope may tend to form resulting in a reduced heat capacity of the storage system for a given pressure limit of the tank, because the bulk temperature of the fluid will not rise uniformly with that of the gas. The vapor pressure of the fluid would then be below the tank pressure. However, the vapor envelope itself would form a heat barrier which would reduce the rate of heat transfer to the tanks. Even without stratification or formation of a vapor envelope, the propellants for the lunar mission return trip midcourse corrections will undergo a considerable vapor pressure rise. This may be attributed to the small mass and hence low heat capacity of the propellants required. The tank

* Heat Conduction Through Insulating Supports in Very Low Temperature Equipment, R.P. Mikesell and R.B. Scott, Journal of Research, NBS Research Paper #2726, Vol. 57, No. 6, dtd Dec 1956

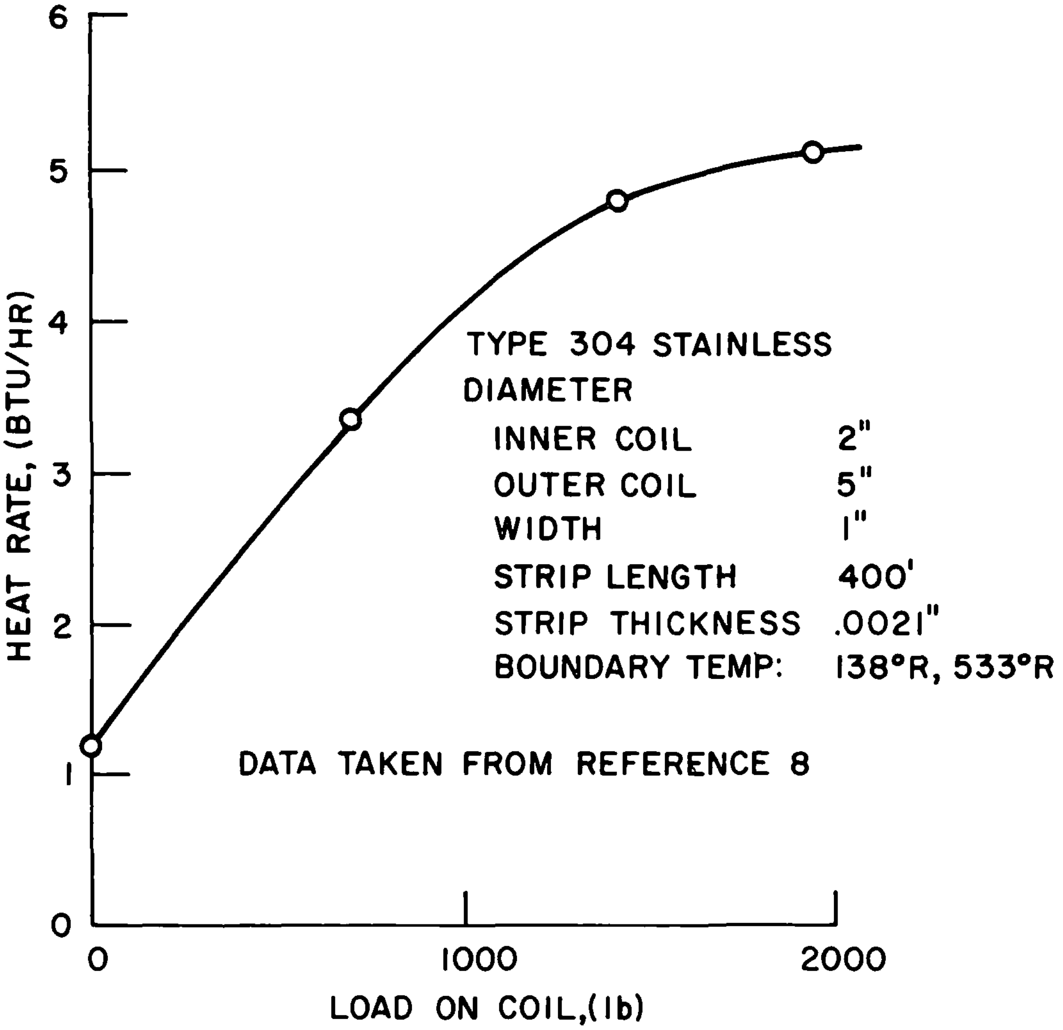

Figure I-4-20. Heat rate vs load on laminated support pressure schedules and temperature for a Hylas-type pressurization system have been calculated for a storage heat of 20 Btu/hr into the hydrogen tank and 50 Btu/hr into the oxygen tanks. The results are shown in Figures I-4-21 and I-4-22.

Operation of the Hylas-type pressurization system is described in Appendix P-A. Since propellant density is a function of temperature, the densities of both propellants will decrease during the storage period. This will cause a decrease in propellant flow rates and a shift in thrust chamber mixture ratio. The calculated mixture ratio is shown in Figure I-4-23. The rise in temperature and pressure of the propellants after a firing is due to the heat added by the pressurizing gas. It was assumed that after each firing sufficient time existed for the pressurizing gas and the remaining liquid to come to thermal equilibrium. If this does not occur, less heat will be absorbed by the liquid, and less shift in mixture ratio will result. Some form of flow-regulating device could be used to maintain the mixture ratio at a preselected value. However, its use degrades system reliability, and it is felt that a more realistic approach is to let the mixture ratio vary and accept the small degradation in performance.

An alternate pressurization system for return from space uses the vapor pressure for self expulsion of the propellants. This provides a type of redundancy in this critical area of pressurization.

The tank pressure history for a VaPak type pressurization system is shown in Figure I-4-24. The operation of the system is described in Appendix P-A. In this system, the energy to expel the propellants is obtained from the heat stored in the propellants, by allowing the propellant temperature (and hence vapor pressure) to drop during the run. The pressure drops during firing have been computed and compared with values* determined by Linde and shown in Figure I-4-25. Testing is currently being conducted to substantiate the computed values. The tank pressure and propellant density variations during a firing result in a larger shift in mixture ratio than in a Hylas-type system where tank pressure throughout a firing remains constant. The mixture ratio variation for the lunar mission is shown in Figure I-4-26, and the resulting specific

* Pressure Phenomena During Transfer of Saturated Cyrogenic Fluids, J.M. Canty, presented at 1960 Cyrogenic Engineering Conference, Linde Company, Division of Union Carbide Corp., New York, N.Y.

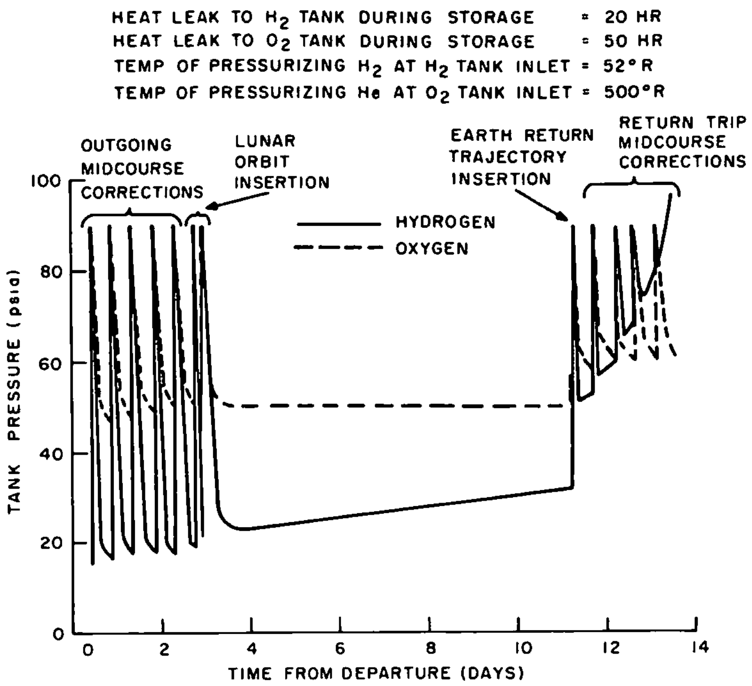

Figure I-4-21. Tank pressure vs time from departure - hylas system

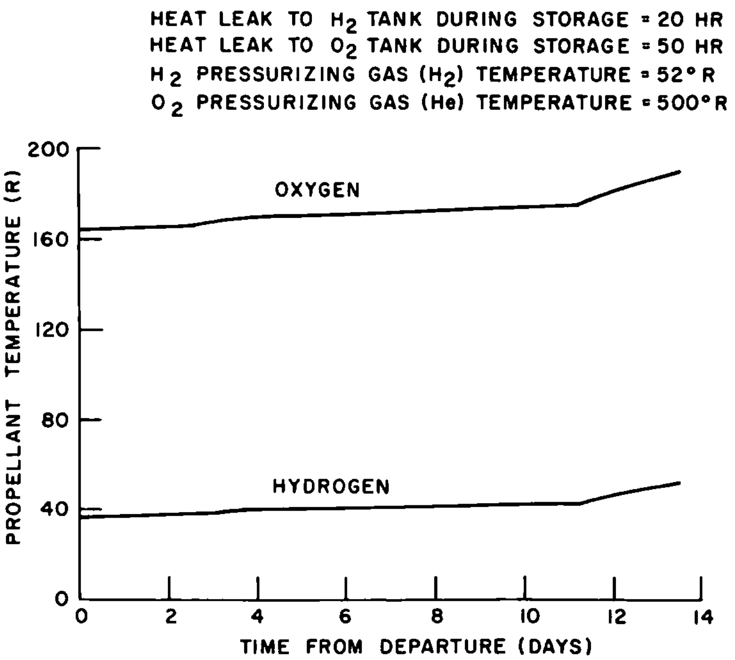

Figure I-4-22. Propellant temperature vs time from departure - hylas system

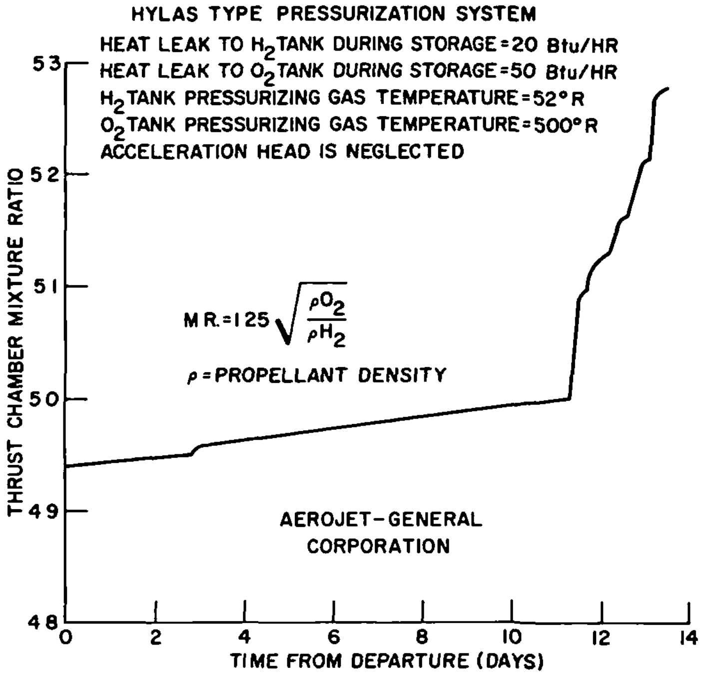

Figure I-4-23. Propellant mixture ratio vs time from departure - hylas system

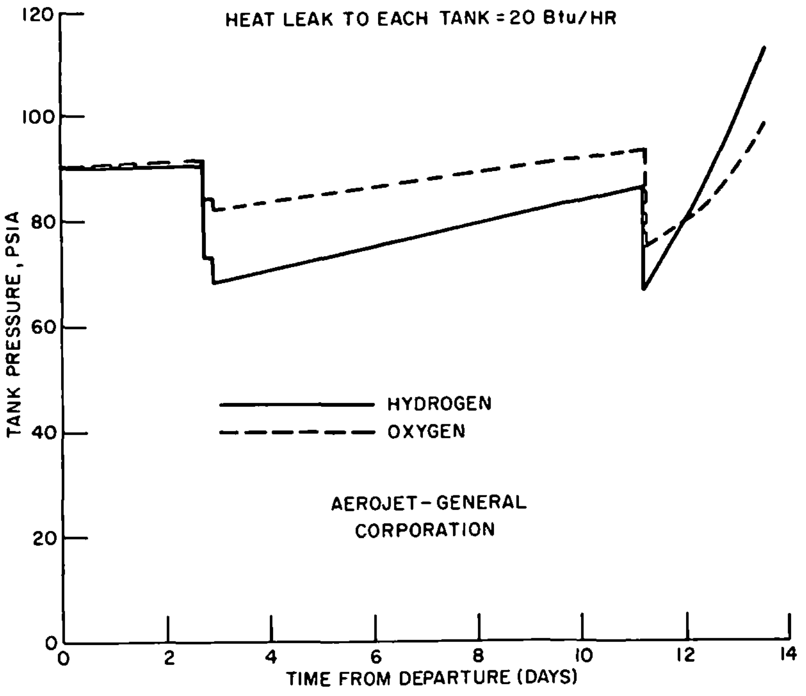

Figure I-4-24. Tank pressure vs time from departure - VaPAK system

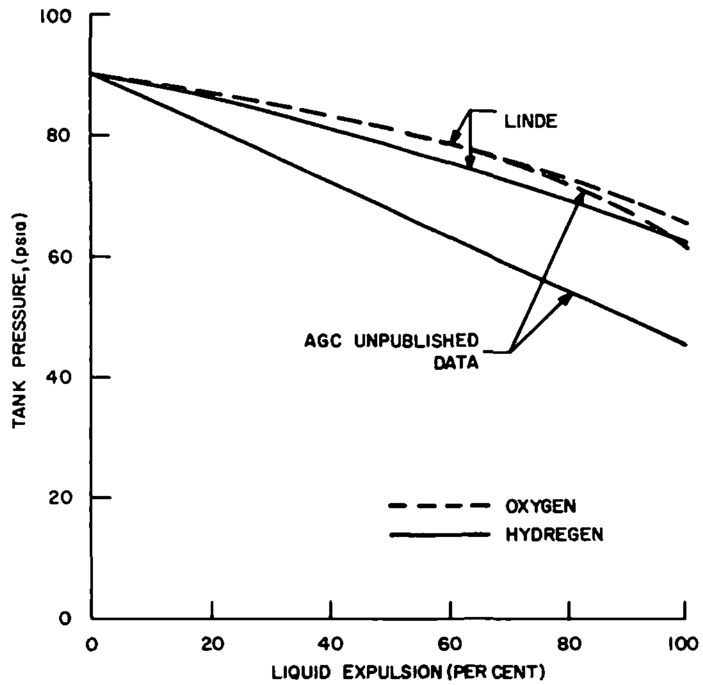

Figure I-4-25. Pressure decay comparison for VaPAK system

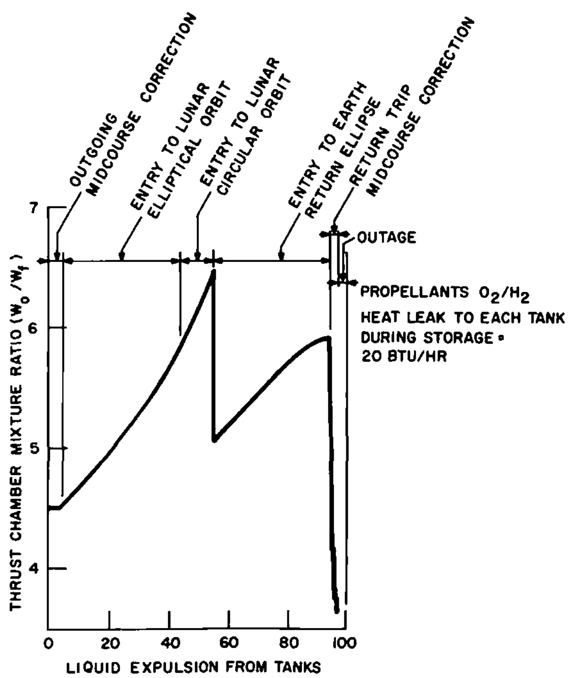

Figure I-4-26. Mixture ratio for APOLLO lunar mission with VaPAK pressurization

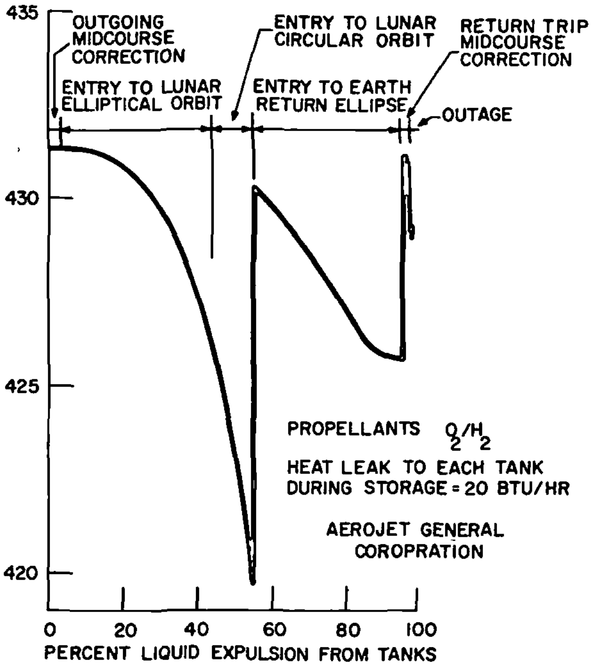

Figure I-4-27. Specific impulse for APOLLO lunar mission with VāPAK pressurization system impulse variation is shown in Figure I-4-27. It may be necessary to use a flow-regulating device in the VaPak system. However, the system is inherently reliable because of its simplicity, and the heat leak to the tanks will be low because no pressurization plumbing or auxiliary equipment is required.

4.2.5 Reliability and Safety Apportionment for AJ-10-133 Engine

Reliability is defined as the probability that the propulsion system will operate successfully, so that orbit about the Moon and return to earth is possible.

The system considered here is an integrated liquid rocket system using solid rockets for abort and separation maneuvers.

The only way the mission can be accomplished is to have no failure during the boost phase. After boost, one engine failure of each lunar maneuvering pair and one tank failure can be survived. Failure of the remaining tank cannot be survived.

After lunar orbit insertion, one engine failure can be survived, but no failure of the remaining tanks is permissible. If the malfunction detection system fails when it is needed, the mission fails unless the pilot and observers on earth can be used as a redundant malfunction detection system. The malfunction detection system could be of either the fail-run or fail-safe type. If the fail-run type is used, the malfunction detection system would shut the No. 1 engine down if it detected a "self" failure.

Assumptions made in the analysis that follows are that failure of a tank or engine does not induce failure in another tank or engine and that sufficient reserve propellant is available to make up for wastage during startup of a faulty engine.

Table I-4-XII shows a list of estimated reliability values for the system components under consideration. These values were estimated from previous experience on various programs and constitute a very conservative estimation when compared with currently advertised values. Data for solid rockets were developed along lines described in Appendix P-A. Data for liquid rockets were based on "most similar" TITAN data, as were the studies in Appendix P-A. Where restart is involved, weighting factors were used as developed in Appendix P-A.

TABLE I-4-XII. ESTIMATED RELIABILITY OF COMPONENTS

Symbols and Assigned Values		P	1-P
P_{B1}	Reliability of Booster 1st Stage	-	-
P_{B2}	Reliability of Booster 2nd Stage	-	-
P_{B3}	Reliability of Booster 3rd Stage	-	-
P_{SO}	Probability that Super-Orbital Abort will not be Required	-	-
P_{E1A}	Reliability of (1)* and (3) Engines for Super-Orbital Abort	0.99025	0.00975
P_{E2A}	Reliability of (2) and (4) Engines for Super-Orbital Abort	0.99025	0.00975
P_{E1M}	Reliability of (1) and (3) Engines for First Mid-course Correction**	0.97250	0.02750
P_{T1M}	Reliability of Tank for First Midcourse Correction	1.0000	-
P_{E2M}	Reliability of (2) and (4) Engines for Second Mid-course Correction*	0.98230	0.0177
P_{T2M}	Reliability of Tank for Second Midcourse Correction	1.0000	-
P_{E1L}	Reliability of (1) and (3) Engines for Lunar Insertion***	0.98770	0.01230
P_{T1L}	Reliability of Tank for Lunar Insertion	1.0000	-
P_{E2L}	Reliability of (2) and (4) Engines for Lunar Exit	0.99473	0.00787
P_{T2L}	Reliability of tank for Lunar Exit	1.0000	-
P_S	Reliability of Re-entry Vehicle Separation Rockets (Solid)	0.9950	0.005
P_{SC}	Reliability of Re-entry Vehicle Spin Control System	0.9923	0.0077
P_A	Reliability of Abort Rockets (Solid)	0.9950	0.005
P_V	Reliability of Attitude Control	1.0000	-
P_M	Reliability of Malfunction Detection System	0.999	0.001

* Numbers in parenthesis refer to engine position on the aft end of the spacecraft.

** Includes 5 starts.

*** Includes 2 starts.

Tankage in the system under consideration is partially redundant as regards safety. However, since the tankage is not wholly redundant, a generous safety margin should be used in the design, especially for H_2 tanks. Reliability of this component will undoubtedly be very high. For the purpose of this study, it will be taken to be 1.00000. The malfunction detection system can be expected to have a reliability of 0.99900. This represents a 50 percent failure-rate reduction over the system under development for Dyna-Soar. By use of these reliability values, the results shown in Tables I-4-12 through I-4-15 were obtained. Attitude control reliability is taken as 0.99900.

The sequencing device for this system would have about the same reliability as the engine sequencer on TITAN Stage II, 0.99900. Malfunction detection and other sequencing is taken as 0.99900.

Table I-4-13 develops the reliability in terms of success in accomplishing the mission with no failures at any phase. The expected reliability is 0.91787 if the booster works properly. Table I-4-13 develops the enhancement due to redundancy possible with the selected configuration. A twofold reduction in failure rate is obtained by the use of redundancy Probability of completing the mission is approximately 0.95454.

Safety, the most important consideration is developed in Table I-4-14. Since booster reliability is not known this value cannot be exactly evaluated. However, numerical values of safety have been developed for each of the three possible booster stage failures and superorbital abort. Table I-4-XV gives values of safety for various booster reliabilities. Safety after a successful boost phase is 0.97801. The 1966 system would be somewhat improved.

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TABLE I-4-XIII. RELIABILITY IN TERMS OF SUCCESS IN ACCOMPLISHING THE MISSION
(Including The Effect Redundancy)

Phase	Description	Probability of Success in Phase	Symbol or Calculated Value For Phase	Cumulative
1	Booster Stage 1	$P_1 = P_{B1}$	P_{B1}	P_{B1}
2	Booster Stage 2	$P_2 = P_{B2}$	P_{B2}	P_{B2}
3	Booster Stage 3	$P_3 = P_{B3}$	P_{B3}	$P_{B1} \cdot P_{B2} \cdot P_{B3}$
4	Super Orbital	$P_4 = P_{SO}$	P_{SO}	$P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
5	First Midcourse Correction	$P_5 = \left[1-(1-P_{E1M})^2\right] \cdot P_{T1M} \cdot P_V$	0.99385	$0.99385\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
6	Lunar Insertion	$P_6 = P_{E1L}\left[1+(1-P_{E1L})P_{E1M}\right]P_{T1L}\ P_V$	0.98215	$0.97608\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
7	Lunar Exit	$P_7 = \left[1-(1-P_{E2L})^2\right]P_{T2L}\ P_V$	0.99390	$0.97015\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
8	Second Midcourse Correction	$P_8 = P_{E2M}\left[1+(1-P_{E2M})P_{E2L}\right]P_{T2M}\ P_V$	0.98442	$0.95504\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
9	Re-entry Vehicle Separation	$P_9 = \left[1-(1-P_S)^2\right]$	0.999975	$0.95502\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
10	Re-entry Vehicle Spin Control	$P_{10} = \left[1-(1-P_{SC})^2\right]$	0.999410	$0.95454\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$

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TABLE I-4-XIV. SAFETY IN TERMS OF SUCCESSFUL RETURNS
(Following Uncorrectable Malfunction at any Phase)

Phase	Mode of Safe Return	[Cumulative Probability of Success Thru Preceding Phase]	[Probability of Single Malfunction This Phase]	[Probability of Success in Return Following Failure in Phase] =	Calculated Value	
		Actual Value Table III				
1-10	Normal Mission	0.95445 $P_{B_1} P_{B_2} P_{B_3} P_{SO}$			0.95454	$P_{B_1}P_{B_2}P_{B_3}P_{SO}$
1	Booster Stage 1 requires 7 of 8 Abort Rockets, 2 of 2 Large Separation Rockets	1.0000	$1-P_{B_1}$	$P_A{}^7\ (8-7P_A)(P_S)^2\ P_{10}$	0.98171	$(1-P_{B_1})$
2	Booster Stage 2 requires 3 of 4 Abort Rockets, 1 of 2 Separation Rockets	P_{B_1}	$1-P_{B_2}$	$(P_A)^4\ (1-(1-P_S)^2)\ P_{10}$	0.97998	$P_{B_1}\ (1-P_{B_2})$
3	Booster Stage 3	$P_{B_1}\ P_{B_2}$	$1-P_{B_3}$	$(P_A)^2\ (1-(1-P_S)^2)\ P_{10}$	0.98963	$P_{B_1}\ P_{B_2}\ (1-P_{B_3})$
4	Super Orbital Abort use all Engines	$P_{B_1}\ P_{B_2}P_{B_3}$	$1-P_{SO}$	$(P_{E_{1A}})^2(P_{E_{2A}})^2(P_{T_1})(P_{T_2})P_9P_{10}$	0.96227	$P_{B_1}P_{B_2}P_{B_3}(1-P_{SO})$
5	First Midcourse Correction If (1) and (3) engines fail Return via Cislunar Path	$P_{B_1}P_{B_2}P_{B_3}P_{SO}$	$1-P_5$	$P_MP_8P_9P_{10}$	0 00604	$P_{B_1} \cdot P_{B_2} \cdot P_{B_3} \cdot P_{SO}$
	If tank fails abort and return via cislunar path	$P_{B_1}P_{B_2}P_{B_3}P_{SO}$	$1-P_{TM}$	$P_MP_8P_9P_{10}$	0*	
6	Lunar Insertion If (1) & (3) engines fail return via cislunar path	0.99385 $P_{B_1}P_{B_2}P_{B_3}$	$1-P_6$	$P_MP_8P_9P_{10}$	.01743	$P_{B_1}P_{B_2}P_{B_3}$
	If tank fails, return via cislunar path	0.99385 $P_{B_1}P_{B_2}P_{B_3}$	$1-P_{T_{1L}}$	$P_MP_8P_9P_{10}$	0*	
				TOTAL	0.97801	$P_{B_1}P_{B_2}P_{B_3}P_{SO}$ +

0.98171$(1-P_{B_1})$+0.97998 $P_{B_1}\ (1-P_{B_2})$ +
0.98963 $P_{B_1}\ P_{B_2}\ (1-P_{B_3})$ + 0.96227
$P_{B_1}\ P_{B_2}\ P_{B_3}\ (1-P_{SO})$

* Since $P_{T_{1M}} = 1$

TABLE I-4-XV. RELIABILITY AND SAFETY SUMMARY CALCULATED FOR VARIOUS VALUES OF BOOSTER AND SUPER ORBITAL RELIABILITY

	Numerical Values For P_{B_1}, P_{B_2}, P_{B_3}, and P_{SO}				
	0.6	0.7	0.8	0.9	0.99
Safety (in Successful Return following failure in any Phase - Table IV)	0.9802	0.9784	0.9790	0.9783	0.9780