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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

GENERAL ELECTRIC

MISSILE AND SPACE VEHICLE DEPARTMENT

A Department Of The Defense Electronics Division

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LIQUID ROCKET PROPULSION SYSTEMS FOR THE GENERAL ELECTRIC COMPANY APOLLO VEHICLE

Volume I - Final Design Study Report

TR 3808A

May, 1961

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Preliminary Design

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PROPRIETARY INFORMATION

RMD ADM-148

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I. INTRODUCTION

This report describes the Thiokol Chemical Corporation, Reaction Motors Division, approach to design, development, quality assurance testing, and delivery of the NASA Apollo manned spacecraft powerplant and attitude control systems. The design study is being conducted for the Missile and Space Vehicle Department of General Electric Company. The phase "is being conducted" is used purposely, because this report reflects the results of considerable work conducted during the last year on many aspects of spacecraft powerplants and indicates a determination to continue and expand the Divisions's activities in this field.

Thiokol-RMD firmly believes that the single most important element in the operation of manned spacecraft is man himself. The rigorous application of manned ratings and manned safety concepts to the design of the propulsion system is not merely a desirable appendage to the program but actually must constitute a basic philosophy guiding overall and detail system design at every stage in the effort.

The Thiokol-RMD XLR99 engine currently in use in the X-15 research aircraft is the latest product in a line of RMD man-rated rocket engines, amply testifying to RMD's ability to translate a keen awareness of manned rating concepts into a practical system. The same well developed skills and procedures used in the design of the XLR99 have been employed to establish the system design and approaches described for the Apollo application. Table I shows a history of RMD rocket powered aircraft flights.

For the Apollo application, the manned rating concepts established for manned rocket craft operating within the earth's atmosphere have been extended to cover operations in space. To the concept of insuring integrity of the combustion chamber has been added the requirement for propulsion availability. This is expressed in the following statement of propulsion system requirements for flight safety in space.

Even with powerplant spacecraft system reliabilities approaching 100%, malfunctions may occur in the powerplant, in the powerplant propellant and

TABLE I

MISSION RELIABILITY RECORD

RMD ROCKET ENGINE SYSTEMS

Aircraft	Total Flights Made	Successful Missions	Overall Mission Reliability (%)	Mission Failures Due to Engine (%)	Failures Due to Aircraft (%)	Experienced Engine Reliability (%)
X-1 Series	191	173	90.58	7.33	2.09	92.67
D-558-2	158	145	91.20	4.8	4.0	95.2
XF-91	10	10	100	0	0*	100
X-15 with XLR11-RM-5 Engine	29	27	93	3.45	3.45	96.55
X-15 with XLR99-RM-1 Engine	5	5	100	0	0	100

* Aircraft saved by rocket engine when jet engine failed

No loss of vehicle or any human life experienced from engine malfunction control system or possibly in its environmental control system. Under such conditions, the powerplant must present no hazard to the spacecraft with which it is intimately associated and must remain operable or be capable of becoming operable within the limitations of on-board repair capability. In the performance of certain missions, (for instance, lunar landing) propulsive effort must be insured positively at all times; this may require redundancy or use of alternate propulsion devices.

An additional major propulsion system factor in the area of flight safety in space is propellant availability. It is essential that a propellant management system be incorporated in the spacecraft so that adequate ΔV capability will be assured at all times for a safe return to earth. This item is considered beyond the scope of the propulsion system study except insofar as propellant supply or flowrate indicators are to be incorporated into the propulsion system or as an emergency interconnection capability of propellant tanks of the several systems is incorporated into the propulsion system design.

The consequences of incorporating manned safe features into the propulsion system will be discussed in detail in the following sections. It is interesting to note that providing manned safe features will directly contribute to increased reliability of operation of the spacecraft.

The broad purpose of the current study is to conduct analytical and design work necessary to develop a propulsion system concept. This system concept, in turn, is consistent with the limits established by the General Electric Company's Statement of Work 730-A-12 and subsequent requirements. In general, the study should provide a basis for defining realistic powerplant requirements and capabilities for the Apollo mission. More specifically, the following goals were established:

- 1963 propulsion system design
- 1966 propulsion system design
- Vehicle capabilities
- Suggested vehicle changes and resulting performance improvements
- Areas for fruitful future work
- Establishment of preliminary specifications for manned spacecraft rocket powerplants.

For the past year Thiokol-RMD has been studying spacecraft propulsion - the effect on payloads of changing the propellant combinations and of varying the chamber pressure, area ratio, and other system parameters. We have developed and tested a series of flight weight, radiation cooled, bipropellant attitude control rockets, ranging in thrust from 0.5 to 10 lb using the earth storable hypergol bipropellant combination N_2O_4/MMH and N_2O_4/UDMH. We have continued the development of vortex thrust chamber injectors over the thrust range from 30 to 10,000 lb also using earth storable hypergols. We have completed a series of technical reports on spacecraft powerplants (including analytical and experimental data) which has culminated in the award of the Surveyor spacecraft vernier propulsion system development work to the Reaction Motors Division. The main retro rocket development work for the Surveyor has been awarded to the Thiokol-Elkton Division. This joint award is a direct outgrowth of the close and continuing relationship both in studies and presentations of the liquid and solid divisions of the Thiokol Chemical Corporation.

The background of analytical and experimental work described above implies a growing awareness of the problems associated with space propulsion and a competency to participate in development programs. The personnel and the technology developed in these efforts have been utilized in the present study and will continue to participate in future work.

In setting the requirements for the Apollo on-board propulsion system study, the General Electric Company requested Thiokol-RMD to perform a detailed design study for either a pressurized hydrogen-oxygen system or a pressurized storable system. The design parameters further specified a 16,000 lb vehicle for 1963 and a 15,000 lb vehicle for 1966, anticipating that the propulsion system for the latter vehicle would reflect possible weight savings and performance increases available with an additional three years of research and development.

A pressurized storable propellant system approach has been chosen by Thiokol-RMD. Table II summarizes the design features of the system.

TABLE II

DESIGN FEATURES, ON-BOARD PROPULSION SYSTEMS

On-Board Propulsion System	Propellants	Thrust	Thrust Chambers	Pressurization System
1963	N_2O_4/MMH	24K	4 at 6K	Helium Gas (stored as gas)
1966	OF_2/MMH	24K	2 at 6K-12K	Helium Gas (stored as liquid)

The unique feature of this proposal is the upgrading of vehicle performance (payload, ΔV) possible through the substitution of oxygen bifluoride (OF_2) for nitrogen tetroxide (N_2O_4). The technology needed to design and develop a helium gas pressurized N_2O_4/MMH system is currently known at Thiokol-RMD. This proposed 1963 system will meet the requirements of an Apollo circumlunar mission, i.e., it will supply adequate ΔV to provide for in-space abort of the mission.

The three year period available for research in storage and combustion of OF_2 prior to engine system development testing promises every success in this area. The resulting 1966 powerplant will, we feel sure, meet the requirements of the Apollo lunar orbiting mission.

Thiokol-RMD has selected the OF_2/MMH propellant combination in preference to oxygen hydrogen because its performance (vehicle-wise) is superior, its storability and handling is simplified, and because it is hypergolic, thereby eliminating the requirement for an ignition system, simplifying the engine, and improving reliability. OF_2 is considered superior to F_2 because it offers the same payload and hypergolicity advantages without the compatibility disadvantages.

For the spacecraft attitude control system and re-entry vehicle roll control system, Thiokol-RMD has selected the use of the earth storable hypergolic propellant combination N_2O_4/MMH. As noted previously, this propellant combination has been exhaustively tested by RMD over a range of thrust of 0.5 to 500 lb and (using the very similar N_2O_4/UDMH combination) the 1300 and 10,000 lb thrust levels. Testing has covered both pulsing operation and throttling sea level and high altitude firings, radiation, and regenerative cooling. The inescapable conclusion is that N_2O_4/MMH (or UDMH for nonregeneratively cooled designs) provides an ideal "control rocket" propellant combination. It is anticipated that only a relatively straightforward development program will be required for these two systems. The Thiokol-RMD program for development of the Surveyor spacecraft vernier rocket system will provide much of the detail system and component technology required for these two Apollo spacecraft systems.

II. MANNED SAFETY AND RELIABILITY

A. MANNED SAFETY IN SPACE

Man's well-being in space depends on numerous factors involving his care, feeding, and housing. But, overshadowing these essentials is controlled mobility (ΔV capability) which provides the means for reaching his destination and his return.

It is of paramount importance that the means for achieving this ΔV capability (on-board propulsion) be of such a nature as to positively insure man's safety in space flight at all times.

1. Differences between Rocket Powered Aircraft and Spacecraft Safety

Until the present time, man's flight has taken place within the protective blanket of the earth's atmosphere. His positive assurance of safety in flight required only that his on-board propulsion shut down noncatastrophically in case of malfunction. Safe return to earth was available by simple gliding or, in extreme cases, by parachuting.

The requirement for aircraft rocket safety, which automatically insures flight safety, was set forth in MIL-E-5149, a general specification for aircraft rockets. This specification states that under any single condition of malfunction of the rocket powerplant, or in certain cases of malfunction of the powerplant propellant supply, the powerplant will shut down or react in a safe manner without creating a hazard to the aircraft.

It follows that runout or loss of propellants in an aircraft mission is equivalent to a safe rocket shutdown, and flight safety will continue to be maintained by gliding to earth.

However, in the case of spacecraft operations, the availability of propellants and a propulsion system to utilize them to provide the ΔV requirements for guidance, propulsion, and retropropulsion efforts are generally mandatory.

For example, for a manned landing on airless planetary bodies such as the moon, failure of a powerplant or runout of propellants during the landing maneuver would be disastrous. An inherently safe system requires:

(1) assurance that fuel is adequate prior to committing the vehicle to the landing

(2) powerplant redundancy.

The lunar takeoff for return to earth requires similar provisions.

As another case, consider re-entry into the earth's atmosphere. To insure survival, a relatively tight re-entry corridor must be reached - this will necessitate operation of guidance rockets. Failure of these rockets is likely to result in disaster.

As a final case, consider more extensive space operations. Assume a single malfunction has occurred involving the loss of propellant through an incompletely closed propellant valve. The consequence of this loss may be inadequate propellant for guidance or propulsion. Either situation could prove disastrous. The remedy in this simple case is to have a tank valve in series with the thrust chamber prop valve to insure positive shutoff. Also, a provision for a signal to indicate valve malfunction should be provided, indicating need for a repair operation. As another related instance, consider an improperly operating mixture ratio control resulting in excessive fuel consumption during some main propulsive effort. The propellant remaining may be inadequate for safe completion of the balance of the mission, unless mission abort were carried out immediately. The remedy here is simple - a propellant loading or consumption indicator system which signals inadequate supply for the mission completion.

There are two important points to be recognized in this space flight safety area. First, even with powerplant spacecraft system reliabilities approaching 100%, malfunctions may occur in the powerplant, in the powerplant propellant and control system, or possibly in its environmental control system. Under such conditions, the powerplant must present no hazard to the spacecraft with which it is intimately associated and must remain operable or be capable of becoming operable within the limitations of on-board repair capability. In the performance of certain missions, (for instance, lunar landing) propulsive effort must be insured positively at all times; this may require redundancy or use of alternate propulsion devices. Second, propellant availability must be monitored continuously to insure adequate ΔV capacity for safe return to earth or space base. Any plan for estimating propellant requirements for flight safety must allow for the variation of trip time with ΔV since the consumption of food and other vital necessities may be a controlling factor in establishing the maximum allowable trip time.

It may be seen from the foregoing discussion that flight safety in space (from the viewpoint of propulsion) may be divided into two major areas, on-board propulsion safety and redundancy, and propellant management.

a. On-Board Propulsion Safety and Redundancy

It is apparent that on-board propulsion safety requires a safe shutdown or a safe reaction (preferably continued operation) in the event of a malfunction in the propellant plant, its supply, or control system. For space applications, a safe shutdown may be defined as one that does not cause hazard to the spacecraft, or one that does not eliminate the possibility of in-space repair of the system.

In its XLR99 powerplant for the X-15 rocket plane, Thiokol-RMD has provided a propulsion system which has clearly demonstrated its safe reaction (usually a harmless shutdown) under numerous types of single malfunctions or propellant supply malfunctions. Indeed, the XLR99 is sufficiently flexible to permit attempts at restart following manual reset of the electric circuitry after such nonrecurring malfunctions as a sticking relay.

The broad philosophy behind the XLR99, which is applicable generally to restartable engines employing nonhypergols, is to prevent by practical means the accumulation of quantities of propellants in any zone of the engine sufficient to damage the engine under conditions of deflagration burning. To accomplish this, an in-series igniter system, a purge system, and redundancy of certain components were provided in the engine system design. It was possible to assure, even on paper, that the engine system would be safe.

For hypergolic propellants, the engine system may be simplified. Hypergols generally provide chemical safety because their reactivity is sufficiently high to preclude the accumulative destructive quantities of unburned propellants in the combustion chamber. Nevertheless, in larger size engines the possibility of injector passageway cross contamination exists, and provision must be made to eliminate this possibility by bleed gas, purge, or face shutoff valves.

Where pressurized engines are used, the possibilities for malfunction are reduced as compared to the turborockets, i.e., the reliability may be somewhat greater. However, safety considerations (i.e., what happens when something does go wrong) are still of significance.

The application of powerplant redundancy may take several forms:

(1) Common feed system - redundant chambers

(2) Redundant feed system - common chambers

(3) Redundant feed system - redundant chambers

Generally, redundancy implies carrying along extra components which means extra weight. Alternately, redundancy may, for example, take the form of three chambers used at 100% of design rating for the design mission; but if one fails, the other two operate at 150% of design as an emergency rating.

Redundancy may also take the form of an alternate powerplant, say a solid rocket, which provides an emergency ΔV capacity sufficient to place the spacecraft in an emergency or life saving orbit.

b. Propellant Management

Associated with the accomplishment of any space mission - no matter how simple or complex - is an overall ΔV requirement. This ΔV requirement is composed of individual increments covering midcourse correction, retro propulsion, re-entry guidance, etc. The actual mission ΔV will depend to a large extent on the particular launch date, trip time, and thrust levels employed for orbit change. Generally, shorter trip times require higher ΔV, as do launches at other than selected optimum times.

At the start of any mission it is reasonable to assume that sufficient propellants will be available to provide the basic ΔV capability for the design mission plus a reserve allowance. If all goes well, the reserve will still be available when the mission is complete. The problem that concerns us here is what happens if all does not go well.

As previously discussed, under conceivable conditions of malfunction, propellant consumption during guidance rocket or prime propulsion rocket firing may be greater than intended. Where cryogenic propellants are employed, malfunction of the thermal-control system (whether it be vehicle orientation control or tank surface absorptivity-emissivity ratio control) will result in excessive propellant loss due to boil-off. No matter what the cause is, malfunction sensing devices are required to warn of the propellant loss. In addition, on-board facilities must be available to plot a strategy for a safe return to earth or space base, considering the remaining ΔV capability of the vehicle. This ΔV capability assessment will automatically account for continual drain on the propellant supply if the malfunction condition cannot be corrected.

Nothing in the foregoing considerations should be construed as advocating abandonment of lifeboat concepts which will provide the ultimate escape capability. However, the cost and complexity of projected manned space missions warrants serious consideration of building into spacecraft propulsion (and the spacecraft in general for that matter) features which enhance the probability of completing the mission or an alternate mission in spite of malfunction. An example of this, in the aircraft field which was previously discussed, is the Thiokol-RMD XLR99 engine which will permit restart attempts after malfunction shutdowns thereby giving the pilot a second chance (actually up to four additional chances) to fire the engine and salvage the mission.

2. Achieving Manned Safety in Space

The Reaction Motors Division of Thiokol Chemical Corporation has developed and fabricated rocket engine systems for manned application over the past fifteen years and has developed philosophies, techniques, and methods of approach to achieve manned safety in rocket engine systems which have become basic criteria for the industry.

Thiokol-RMD rocket engines have powered the Bell X-1, Bell X-1A, Douglas 558 series, Republic XF-91, and the North American X-15. In each case manned safety was a primary design requirement. The outstanding record of these engines is given in Table I. It is interesting to note that the reliability of these engine systems far surpasses the reliabilities of the current ballistic missile system rocket engines. Impressive is the fact that there has never been a loss of vehicle or human life from any engine malfunction. This clearly indicates the effectiveness of the philosophies, techniques, and methods of Thiokol-RMD in the field of manned safety.

The achievement of manned safety requires extensive study and analysis (and implementation of the results) of two fundamental areas of hazards in rocket engine systems: chemical safety (critical quantity) and system malfunction safety.

a. Chemical Safety

Propellant accumulation or entrapment is a major deterrent to manned safety in rocket engine systems. An engine system must be designed to prevent combustible propellants from accumulating anywhere within the system under all operating or malfunction conditions. In addition, a certain "critical quantity" of propellants must never be allowed to accumulate within a combustion device prior to ignition. A "critical quantity" is that amount of propellant which if ignited and burned (deflagration basis) would produce a rupturing pressure peak. To achieve chemical safety, considerable study and investigation must be undertaken. Such items as ignition delay, propellant feed times, and injector volumes must be analyzed and design corrections made accordingly.

For the Apollo systems, chemical safety will be achieved through the use of hypergolic propellants and injector face shutoff valves which would eliminate the need for special ignition and purge systems.

b. Safety Under System Malfunction Conditions

Any rocket engine system designed for manned application must undergo a malfunction analysis. A malfunction analysis is a methodical and systematic determination of all potential hazardous conditions that might occur under any conceivable type of system malfunction.

In a malfunction analysis each individual component of the engine system, in turn, is assumed to fail in the energized and de-energized position during each operational phase of the engine system (including transients) and the resulting response of the engine system analyzed. Any potential hazards to the other systems, the vehicle, or the crew are eliminated by system design changes.

To guarantee that the safety requirements of the engine system and mission have been met, applicable malfunction tests will be established. The tests will be designed to demonstrate operational characteristics when subjected to the assumed conditions of malfunction. Testing is concentrated on conditions of malfunction which represent potential hazards, or depend upon series operation of other components for detection of the potential hazard, and those which initiate safety circuit action. In the case where multiple malfunctions represent potential hazards, the probability of the occurrence of such multiple malfunctions is analyzed. If the probability of occurrence is significant, then such multiple malfunction conditions will be eliminated from the engine system and vehicle.

A sample malfunction analysis as applied to several components selected from the re-entry roll control pressurization package is included as Appendix C.

B. RELIABILITY

Reliability of a propulsion system is defined as the probability that there will be no failure of any component or combination of components to operate within design performance limits (at the time required and for the length of time required under the environmental conditions associated with its use) which would ultimately reflect in the failure of the propulsion unit to perform within its performance limits.

With reliability defined as a probability, it is possible to obtain values representative of component, subsystem, and propulsion system reliability by following statistical theory in determining the tests that should be made and the method of analyzing data, and by observing the fundamental requirements of quality control and quality assurance. These values can never be stated with certainty, but only with a probability, i.e., a degree of assurance that any given standard of reliability has been achieved.

Based on actual flight test data (394 flights with five aircraft types) the rocket engine system for manned aircraft developed by this contractor have demonstrated a reliability of greater than 95% at a 95% confidence level. We therefore predict a reliability for the Apollo spacecraft propulsion system described in this report greater than this value.

A reliability program should not be thought of as a separate and remote effort within the development program, but should be considered an intimate part of the development program. Achieving reliability objectives requires an integrated effort of everyone concerned in the various phases of the development program, from early design to flight testing.

III. SUMMARY OF PROPULSION SYSTEM REQUIREMENTS

The propulsion system design requirements established by the General Electric Company's MSVD are given below:

A. VEHICLE WEIGHT

Gross Vehicle Weight, lb (after separation from booster)	15,715 (1963) 14,715 (1966)

B. ATTITUDE CONTROL (Overall Vehicle)

Total Impulse, lb-sec	60,000
Maximum Number of Starts	3,000
Maximum Single Impulse, lb-sec	200
Unit Thrust, lb	3
Number of Units	12
Location of Units	above numbers calculated on the basis of 9 foot lever arms.

C. ROLL CONTROL (Re-Entry Vehicle)

Total Impulse, lb-sec	7,000
Maximum Number of Starts	500
Maximum Single Impulse, lb-sec	100
Unit Thrust, lb	18
Number of Units	4
Location of Units	above numbers calculated on the basis of 3 foot lever arms.

D. PROPULSION DESIGN PARAMETER

Number of Thrust Chambers	4
Gimbal Angle	±5° (any direction)
Fuel Tank Compartments	2*
Oxidizer Tank Compartments (may be separate spheres)	4*
Propellant Reserve	10% by weight
Residual Propellant (outage)	5%
Boil-Off Allowance	
a. Assuming non-vented pressurized tanks	0%
b. Assuming a pumped system	as required

E. MID-COURSE CORRECTION (Outbound)

Outbound ΔV, ft/sec	250
Minimum g	0.25
Maximum g	1.5
Number of Starts (maximum)	5

---

* The effect of the additional valves required for tank compartmenting should be considered. The basic question is whether the additional complexity reduces the reliability level to the point where it may be better to risk a tank puncture.

F. ENTERING LUNAR ORBIT

Required ΔV, ft/sec	3,500
Minimum g	0.25
Maximum g	1.5
Number of Starts	2**

G. LEAVING LUNAR ORBIT

Required ΔV, ft/sec	3,500
Minimum g	0.33
Maximum g	2 (approximately)
Number of Starts (maximum)	2

H. MID-COURSE CORRECTION (Inbound)

Inbound ΔV, ft/sec	250
Minimum g	0.5 (approximately)
Maximum g	3 (approximately)
Number of Starts (maximum)	5

I. GENERAL

Mission Duration	14 days
Space Available on Apollo Vehicle for Propulsion	See Figure 1.

** It may be necessary to obtain small values of ΔV after the second firing. Data are requested for the thrust duration available by operating off of tank pressure.

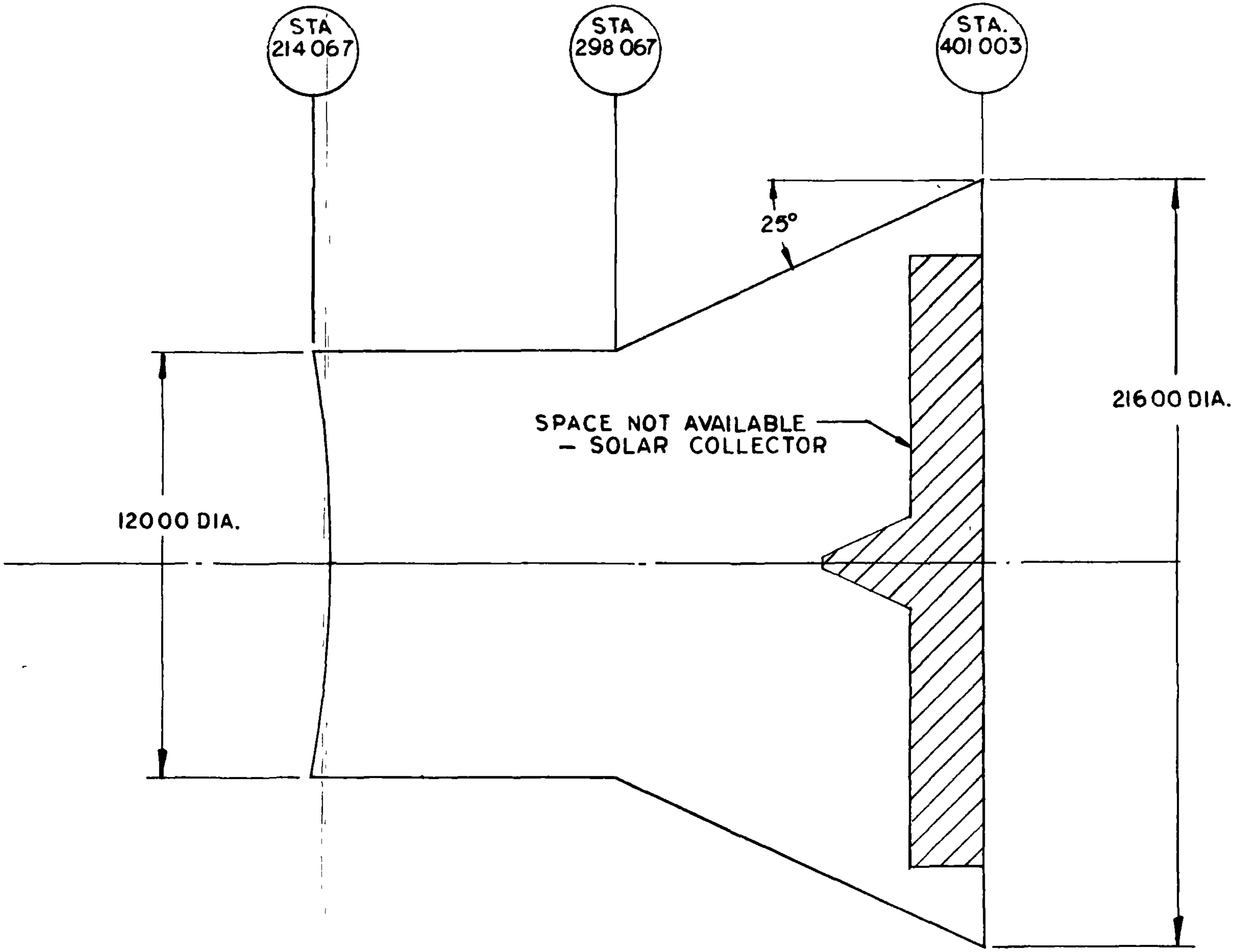


Figure 1. Space Available on Apollo for Propulsion

IV. RE-ENTRY ROLL CONTROL AND MAIN ATTITUDE CONTROL

A. DESCRIPTION OF SYSTEMS

The proposed re-entry roll control and main attitude control systems use bipropellant control rockets of a configuration currently under development at Thiokol-RMD. The storable hypergolic propellants are supplied to the control rockets by pressurizing the propellant tanks with helium. The pressurization package components are basically an extension of the system now being developed at Thiokol-RMD for the NASA Surveyor spacecraft vernier powerplant system. These proposed systems combine reliable hypergolic space ignition with current state of the art technology.

1. Main Attitude Control System

The main attitude control system consists of a pressurizing gas source, pressure regulating stage, propellant tanks, and twelve thrust chamber and propellant valve assemblies. (Refer ahead to Figure 4.) The twelve thrust chambers are mounted to produce six pairs to provide pitch, yaw, and roll control of the spacecraft.

Pressurizing gas for the attitude control system is taken off downstream of the main helium regulating stage, regulated again to the required tank pressure, and supplied to the attitude control propellant tanks. Two parallel pressurizing circuits are used to provide redundancy. The two tanks are fabricated from titanium and contain Teflon bladders to insure positive expulsion of the propellants. The twelve thrust chambers themselves provide redundancy since attitude control can be maintained despite the failure of several chambers. These thrust chambers are arranged in four groups of three chambers each. (Refer ahead to Figures 8 and 9.)

The 1963 system will use cold stored helium gas as a pressurant source. The 1966 system will be improved by the utilization of stored liquid helium which is converted on demand in a heat exchanger to the gas. This system offers a considerable savings in weight without introducing an unduly complex control system. The weight savings of the 1966 liquid helium storage system over that of the 1963 gaseous helium storage system is greater than 100 lb.*

---

* It should be remembered that this system supplies pressurant for both the main on-board propulsion and main attitude control systems.

Performance of the 1966 main attitude control system is further improved by replacing the N_2O_4 with OF_2. This results in an overall weight savings of 54 lb of propellant.

2. Re-Entry Roll Control System

The re-entry roll control system is a separate, self-contained unit consisting of a helium pressurizing gas storage sphere, pressure regulating stage, propellant tanks, and eight thrust chamber and propellant valve assemblies. (See Figure 2.) The eight thrust chambers are mounted to provide four pairs of thrust units for roll control.

One system is used in both the 1963 and 1966 propulsion systems, with N_2O_4/MMH as propellants. The stored helium gas is supplied to the propellant tanks through two parallel, redundant pressure regulating circuits. The propellant tanks are small titanium spheres containing Teflon bladders for positive expulsion. Eight thrust chambers are provided, mounted in four pairs to insure complete redundancy in case of chamber failures. Because of the location of the propellant tanks within the manned re-entry capsule and the negligible weight savings to be realized, it was decided not to upgrade the 1966 roll control system to OF_2.

SYSTEM PARAMETERS AND WEIGHTS

Main Attitude Control System	1963 System	1966 System
Propellants	N_2O_4/MMH	OF_2/MMH
Pressurization System	Helium Gas (stored as gas)	Helium Gas (stored as liquid)
Vacuum Specific Impulse (minimum)	310 sec	374 sec
Operating Temp Limits	20-80F	20-80F
Oxidizer to Fuel Ratio (bw)	2 4	2 5
Area Ratio	100	100
Pressurant Storage Pressure	3000 psia	3000 psia and 500 psia
Propellant Tank Pressure	150 psia	150 psia
Chamber Pressure	40 psia	40 psia
Total Propellant Weight	390 lb	336 lb

SYSTEM PARAMETERS AND WEIGHTS (cont)

Main Attitude Control System	1963 System	1966 System
Usable Propellant for		
Attitude Control	193.6 lb	160 5 lb
Velocity Increment Reserve*	170.0 lb	150 0 lb
10% Reserve	19 4 lb	16.0 lb
Total Loaded System Wt**	424 lb	368 lb

Re-Entry Roll Control System	1963 and 1966 Systems
Propellants	N_2O_4/MMH
Pressurization System	Cold Stored Helium Gas
Vacuum Specific Impulse (minimum)	310 sec
Operating Temp Limits	20-80F
Oxidizer to Fuel Ratio (bw)	2 4
Area Ratio	100
Pressurant Storage Pressure	3000 psia
Propellant Tank Pressure	150 psia
Chamber Pressure	40 psia
Total Propellant Weight	25 4 lb
Usable Propellant for	
Roll Control	22 6 lb
10% Reserve	2.3 lb
Total Loaded System Weight	66 8 lb

A detailed weight summary of all of the systems is given in Appendix A

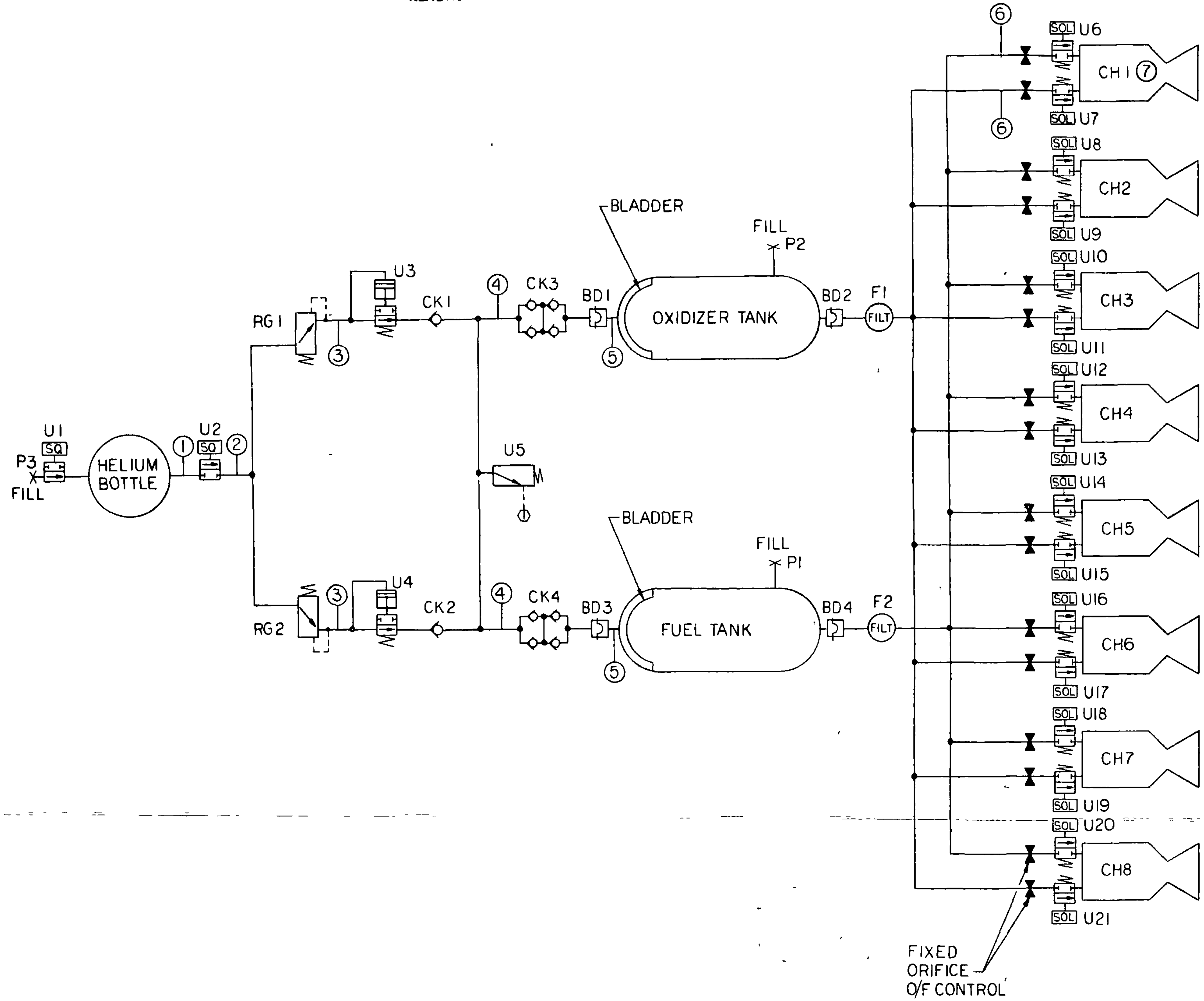
---

* This reserve is provided to permit a final velocity increment of 250 ft/sec on the inbound flight should all the main propellants be used up Cross-manifolding is provided to operate one of the 6K chambers from the attitude control propellant tankage Refer to Figure 4

** Since the main on-board propulsion system and attitude control system have a common helium source, the weights of these items are not included here

B. SYSTEM SCHEMATICS AND SEQUENCE OF OPERATIONS

Figures 2 and 3 present the fluid and electrical schematic drawings of the re-entry roll control system. The pressure budget is given in Table III. The sequence of operation can be found in the Model Specification, Appendix B.



ITEM	DESCRIPTION
U1	DUAL SQUIB OPERATED, HELIUM FILL VALVE
U2	DUAL SQUIB OPERATED, PRESSURIZING VALVE
U3, U4	PRESSURE OPERATED, REGULATOR MALF VALVE
U5	HELIUM RELIEF VALVE
U6-U21	SOLENOID OPERATED, PROPELLANT VALVE
CH1-CH4	ROLL CONTROL THRUST CHAMBER
CH5-CH8	REDUNDANT ROLL CONTROL THRUST CHAMBER
RG1, RG2	HELIUM PRESSURE REGULATOR
CK1, CK2	CHECK VALVE
CK3, CK4	QUAD. CHECK VALVE
BD1 BD4	FUEL TANK & OX TANK BURST DISCS
P1 - P3	FILL PLUGS
F1	OXIDIZER FILTER
F2	FUEL FILTER

1 HELIUM OVERBOARD DRAIN

Figure 2. Re-Entry Roll Control System Schematic (Dwg 52212)

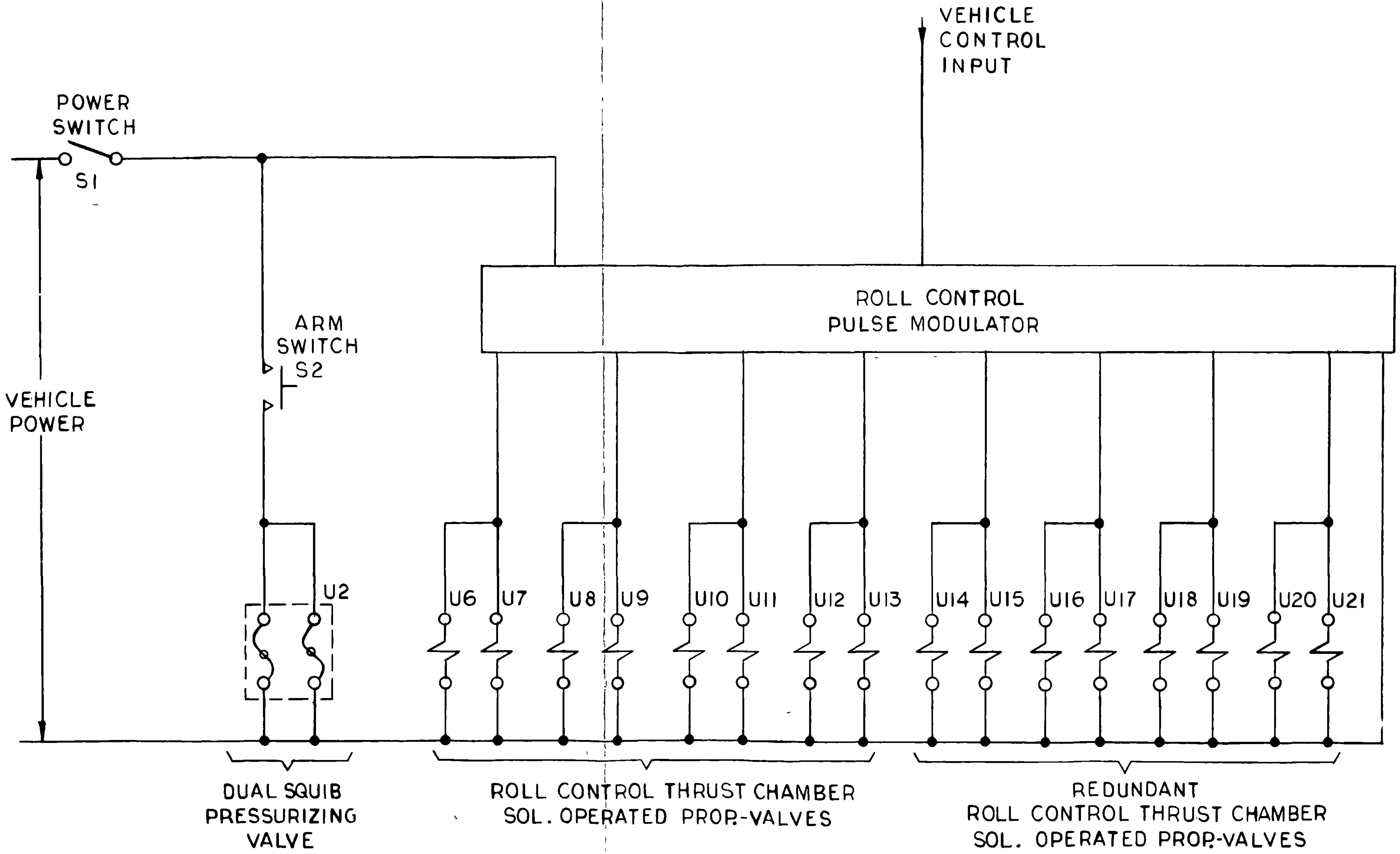


Figure 3. Re-Entry Roll Control System Electrical Schematic (Dwg 52213)

TABLE III

PRESSURE BUDGET RE-ENTRY ROLL CONTROL SYSTEM
(1963)

Station*	Description	Pressure (psia)
1	Helium Storage	3000-2900
2	Pressurizing Valve Exit	2900-2800
3	Regulator Exit	180
4	Propellant Tank Check Valve Inlet	170
5	Propellant Tank	150
6	Prop Valve Inlet	140
7	Chamber	40

* Stations refer to Figure 2.

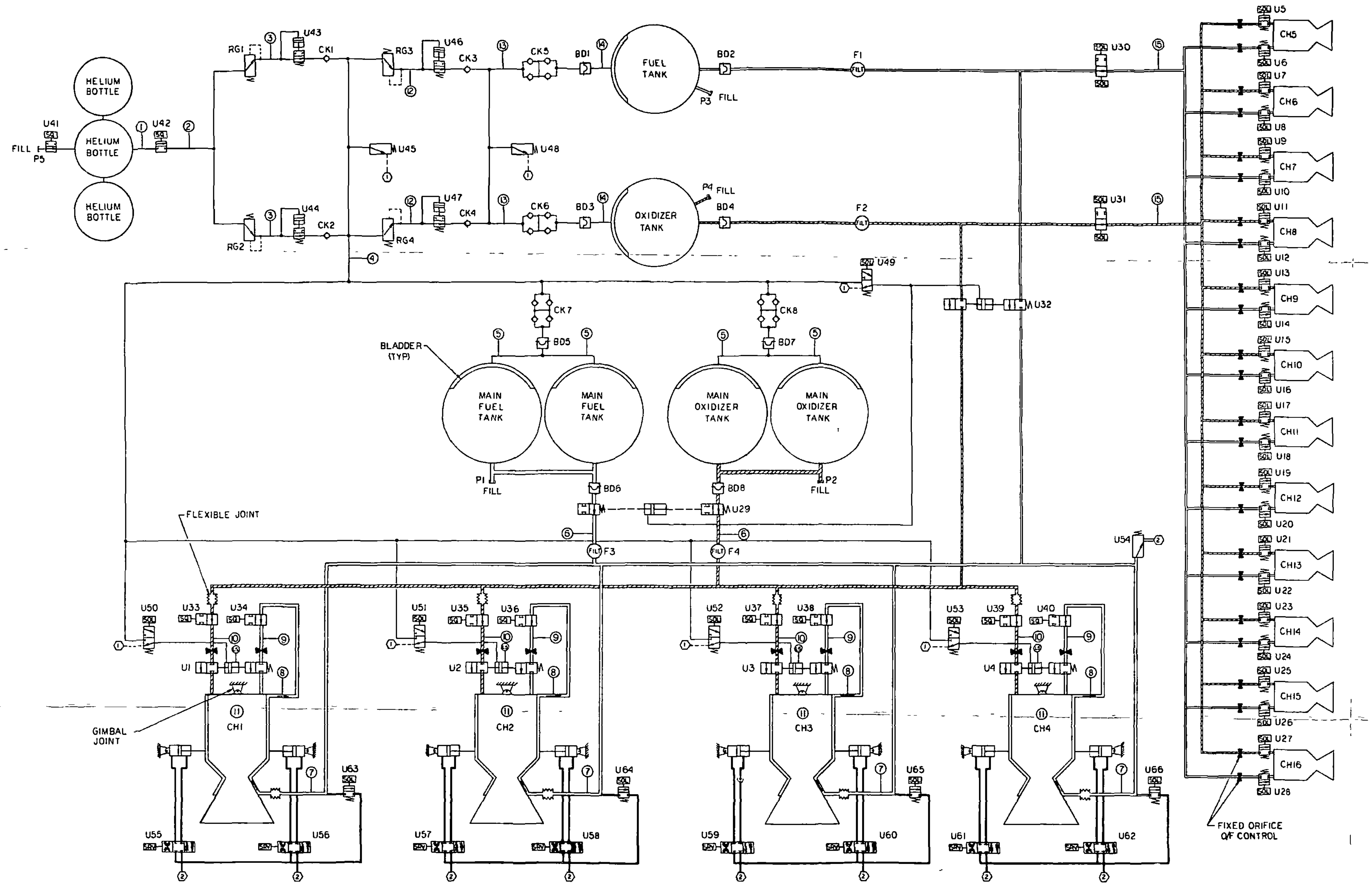
Figures 4 and 5 are fluid and electrical schematic drawings of the main propulsion system for 1963 and show the main attitude control system. This system is linked to the main on-board propulsion system through the main helium regulators and pressure relief valves and helium storage bottles. The sequence of operation can be found in the Model Specification, Appendix B. The pressure budget is given in Table IV.

C. THRUST CHAMBER DESIGNS

The thrust chamber designs proposed for both the 3 lb thrust main and 18 lb thrust re-entry roll control rockets are radiation cooled units utilizing conical spray injector nozzles similar to the types used in current Thiokol-RMD pulse rocket programs. Figure 6 illustrates the design and shows the mounting of a group of three of the main attitude control rockets. Figure 7 shows the design of the re-entry roll control rocket. Both engines have a nozzle area ratio of 100 and a design chamber pressure of 40 psia. Calculated design and performance data are given in Table V. Both the 3 lb and 18 lb thrust chambers will be capable of pulsed or steady state operation.

Heat transfer calculations indicate that complete radiation cooling of both the 3 lb and 18 lb thrust control rocket chambers is feasible in space. The maximum equilibrium wall temperature will be approximately 3000F in the throat. The heat transfer coefficient decreases with an increase in diameter and since both the chamber and exit cone have diameters greater than the throat, they will stabilize at a lower temperature. Considering two dimensional heat transfer, heat will be conducted axially in the wall thus reducing the local hot area in the throat.

A molybdenum-titanium alloy was selected for the chamber and injector body material because of its good high temperature characteristics and its compatibility with the propellants. To protect the "gas side" surfaces from oxidation, a coating of aluminum oxide will be applied to the internal surfaces. Rokide A was selected for this application because its coefficient of thermal expansion is close to that of the molybdenum alloy, therefore lessening the possibility of the Rokide cracking and stripping from the walls. Thiokol-RMD has had considerable experience in advancing the technology of flame sprayed ceramic coatings and has used aluminum oxide coatings on rocket engines from the small (1/2 lb) micro-rockets to the large (50K) XLR99 rocket engine. Because of our advances in the state of the art and our development of new and unique equipment for flame spraying, we can design for aluminum oxide protective coatings with complete confidence. The exterior surfaces are



ITEM	DESCRIPTION
CH1-CH4	MAIN THRUST CHAMBER, GIMBALED
CH5-CH16	ATTITUDE CONTROL THRUST CHAMBER
U1-U4	MAIN PROP VALVE, PNEU. OPERATED
U5-U28	ATTITUDE CONTL. PROP VALVE SOLENOID OPERATED
U29	MAIN PROPELLANT TANK SHUTOFF VALVE, PNEU OPERATED
U30,U31	ATTITUDE CONTROL SYS SHUTOFF VALVE, SOLENOID OPER.
U32	CROSS FEED PROPELLANT VALVE PNEU OPERATED
U33 U40	MAIN CHAMBER MALF SHUTOFF VALVE DUAL SQUIB OPER.
U41	HELIUM FILL VALVE, DUAL SQUIB OPERATED
U42	PRESSURIZING VALVE, DUAL SQUIB OPERATED
U43 U44	MAIN HE REGULATOR MALF SHUTOFF VALVE PNEU. OPER
U45	MAIN HELIUM PRESSURE RELIEF VALVE
U46 U47	ATTITUDE CONT'L HE REGULATOR MALF VALVE, PNEU OPER
U48	ATTITUDE CONT'L PRESSURE RELIEF VALVE
U49	MAIN PROP SHUTOFF & CROSSFEED VALVE PILOT VALVE, SOL OPER.
U50-U53	MAIN PROP VALVE PILOT VALVE SOLENOID OPERATED
U55-U62	MAIN THRUST CHAMBER GIMBAL VALVE SERVO OPERATED
CK1-CK4	CHECK VALVE, SINGLE
CK5-CK8	CHECK VALVE, QUAD
BD1-BD4	ATTITUDE CONT'L PROPELLANT TANK BURST DISC
BD5-BD8	MAIN PROPELLANT TANK BURST DISC
F1 F2	ATTITUDE CONT'L PROPELLANT FILTER
F3,F4	MAIN PROPELLANT FILTER
P1 P2	MAIN PROPELLANT TANK FILL PLUG
P3,P4	ATTITUDE CONT L PROPELLANT TANK FILL PLUG
P5	HELIUM FILL PLUG
U54	MAIN FUEL MANIFOLD RELIEF VALVE
U63-U66	SERVO FEED VALVE SOLENOID OPERATED
RG1-RG4	HELIUM PRESSURE REGULATOR

Figure 4. Main Propulsion System Schematic, 1963 (Dwg 52214)

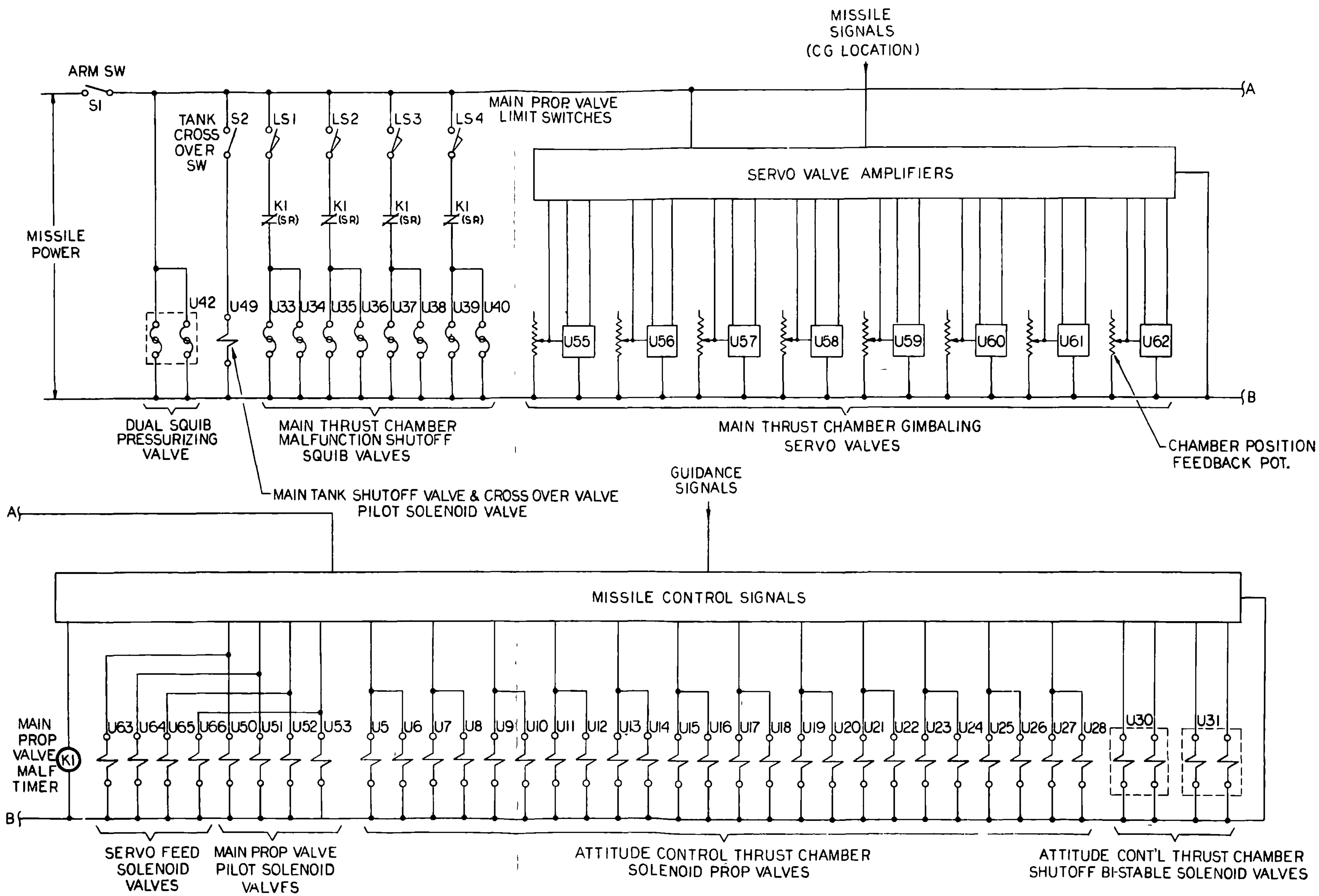


Figure 5. Main Propulsion System Electrical Schematic, 1963 (Dwg 52215)

TABLE IV

PRESSURE BUDGET (1963) MAIN PROPULSION SYSTEM

Station*	Description	Pressure (psia)
1	Helium Storage	3000-540
2	Pressurizing Valve Exit	2990-530
3	First Stage Regulator Exit	430
4	Second Stage Regulator Entrance	420
5	Main Propellant Tanks	390
6	Main Propellant Tank Exit	380
7	Main Chamber Cooling Jacket Inlet	365
8	Main Chamber Cooling Jacket Exit	310
9	Fuel Prop Valve Inlet	300
10	Main Oxidizer Prop Valve Inlet	355
11	Chamber	200
12	Second Stage Regulator Exit	190
13	Attitude Control Tank Check Valve Entrance	180
14	Attitude Control Tank	150
15	Attitude Control Prop Valve Inlet	140
16	Attitude Control Chamber	40

* Stations refer to Figure 4.

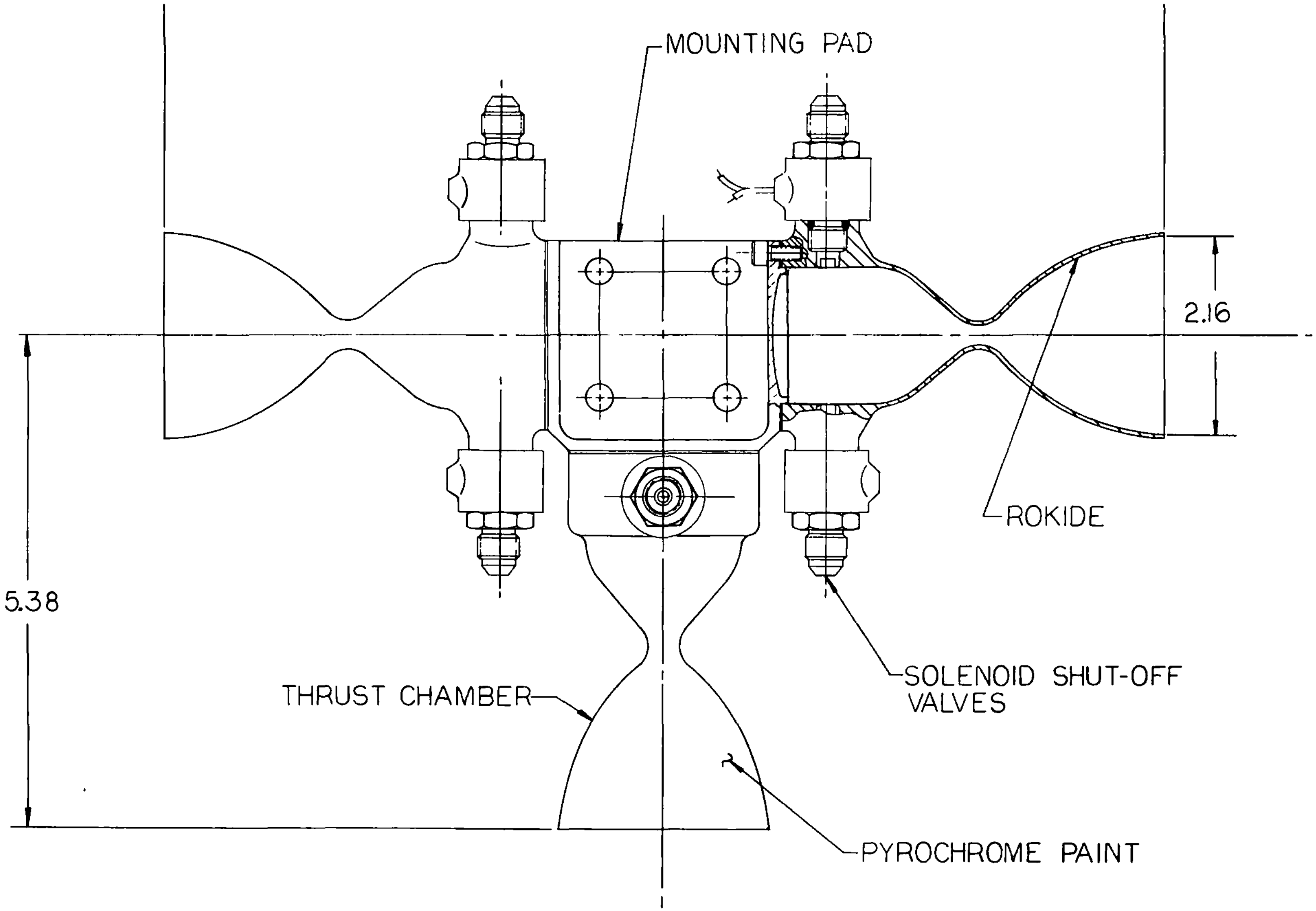


Figure 6. Main Attitude Control Rockets (Group of Three) (Dwg 52098)

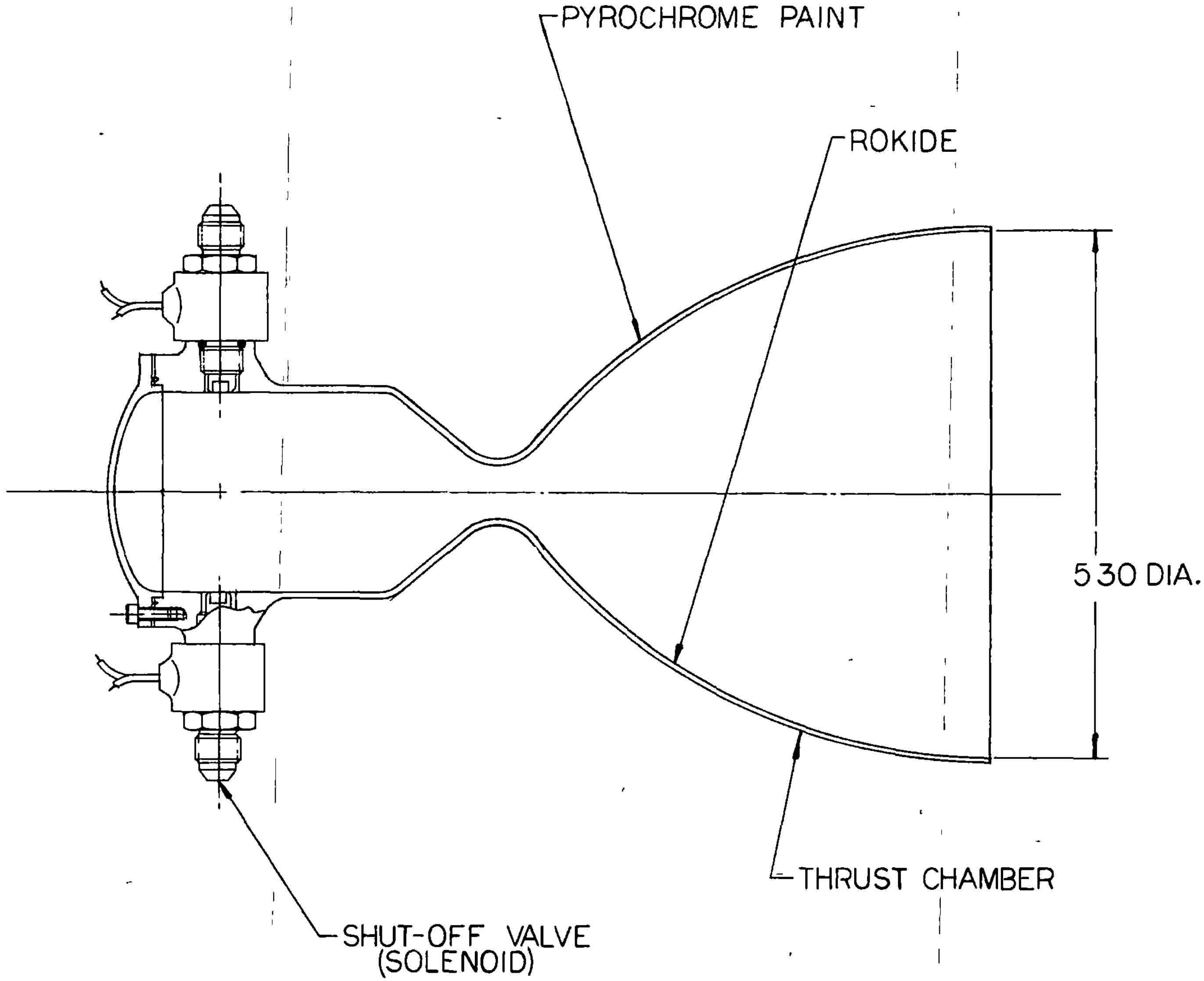


Figure 7. Re-Entry Roll Control Rocket (Dwg 52100)

TABLE V

PERFORMANCE AND DESIGN DATA

		Attitude Control System	
	Units	Main	Re-Entry
Thrust, F	lb	3	18
Chamber Pressure, P_c	psia	40	40
Mixture Ratio, O/F	---	2.4	2.4
Expansion Area Ratio, A_e/A_t	---	100	100
Contraction Area Ratio, A_c/A_t	---	30.8	14.25
Specific Impulse, I_{sp}	$\frac{lb_f\ sec}{lb_m}$	310 (min)	310 (min)
Theoretical Characteristic Velocity, C^*_t	ft/sec	5500	5500
Thrust Coefficient, $C_{f_{vac}}$	---	2.04	2.04
Avg Specific Heat Ratio	---	1.14	1.14
Total Flowrate, $\dot{W}_T$	lb/sec	0.00967	0.0581
Fuel Flowrate, $\dot{W}_f$	lb/sec	0.00284	0.0171
Oxidizer Flowrate, $\dot{W}_{ox}$	lb/sec	0.00683	0.041
Throat Diameter, D_t	in.	0.216	0.53
Exit Diameter, D_e	in.	2.16	5.3
Chamber Diameter, D_c	in.	1.2	2.0
Chamber Temp, T_c	^oF	3920 (at 89% C*)	3920 (at 89% C*)
Equilibrium Wall Temp, T_w	^oF	3000 (max)	3000 (max)
Characteristic Length, L*	in.	37	37

painted with a sodium silicate type paint containing a chromic oxide pigment to provide a surface with high emissivity. A commercially available paint of this type is Pyrochrome, manufactured by Preferred Utilities, Inc., New York City. In the heat transfer calculations, a value of the product of shape factor and emissivity of 1.0 was used. Since Pyrochrome paint is known to have an emissivity of approximately 0.9 and the shape factor for the nozzle can be on the order of 1.2 to 1.4, the design is conservative. The Pyrochrome paint is good for applications up to 4000F. The spray injection nozzles operate at a pressure drop of approximately 50 psia of water at 0.0057 and 0.030 lb/sec for the oxidizer and 0.0035 and 0.020 lb/sec for the fuel main and re-entry nozzles, respectively. The conical spray angle is 45 degrees. These nozzles will be AISI 300 series stainless steel. The nozzles will be fitted into the outlet port of the solenoid valves to make an integral assembly with the injector head. The oxidizer and fuel solenoid valves are the normally closed coaxial type with response times of 3.5 to 5 milliseconds. This type of arrangement was used successfully in the Thiokol-RMD Micro-Rocket Program and has been demonstrated to be satisfactory for low flowrate applications.

The main attitude control rockets will be arranged in four groups of three and attached to the missile as shown in Figure 8. Such an arrangement allows for all the couples required for roll, pitch, and yaw control. Figure 9 indicates the rocket combinations required for each maneuver. The letters correspond to Figure 8. The grouping of three rockets to an assembly was selected to allow for easy replacement of single or whole units to reduce the cost of fabrication by providing identical assemblies. The solenoid valves will be protected from the near absolute zero temperatures of space by electric heating pads or other electrical resistance devices.

The re-entry roll control rockets are arranged in four groups of two and are attached to the command module at a six foot diameter (Figure 10). These rockets are shielded from excessive aerodynamic heating during re-entry. The solenoid valves will also be protected from the re-entry heating conditions. However, there is sufficient exposure to permit radiation cooling.

D. RELATED EXPERIENCE

Thiokol Chemical Corporation, Reaction Motors Division, has been engaged in a company sponsored program to test and evaluate low thrust, bipropellant micro-pulse rockets with the following objectives in view: first, to demonstrate the feasibility of a low thrust bipropellant pulse rocket; second, to evaluate experimentally its response characteristics; and third, to uncover problem areas to be investigated in subsequent test programs.

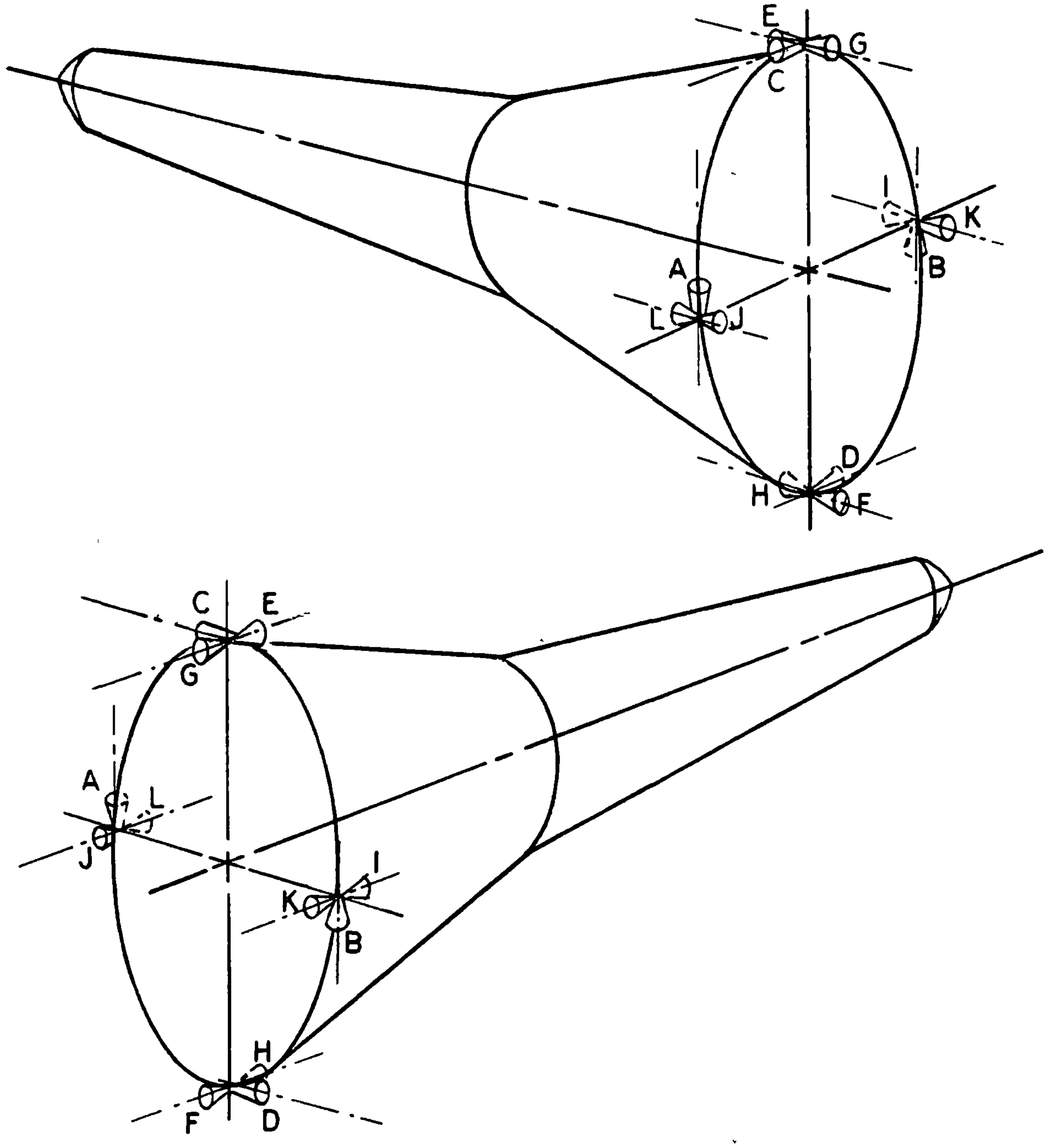


Figure 8. Main Attitude Control Rockets (Four Groups of Three)

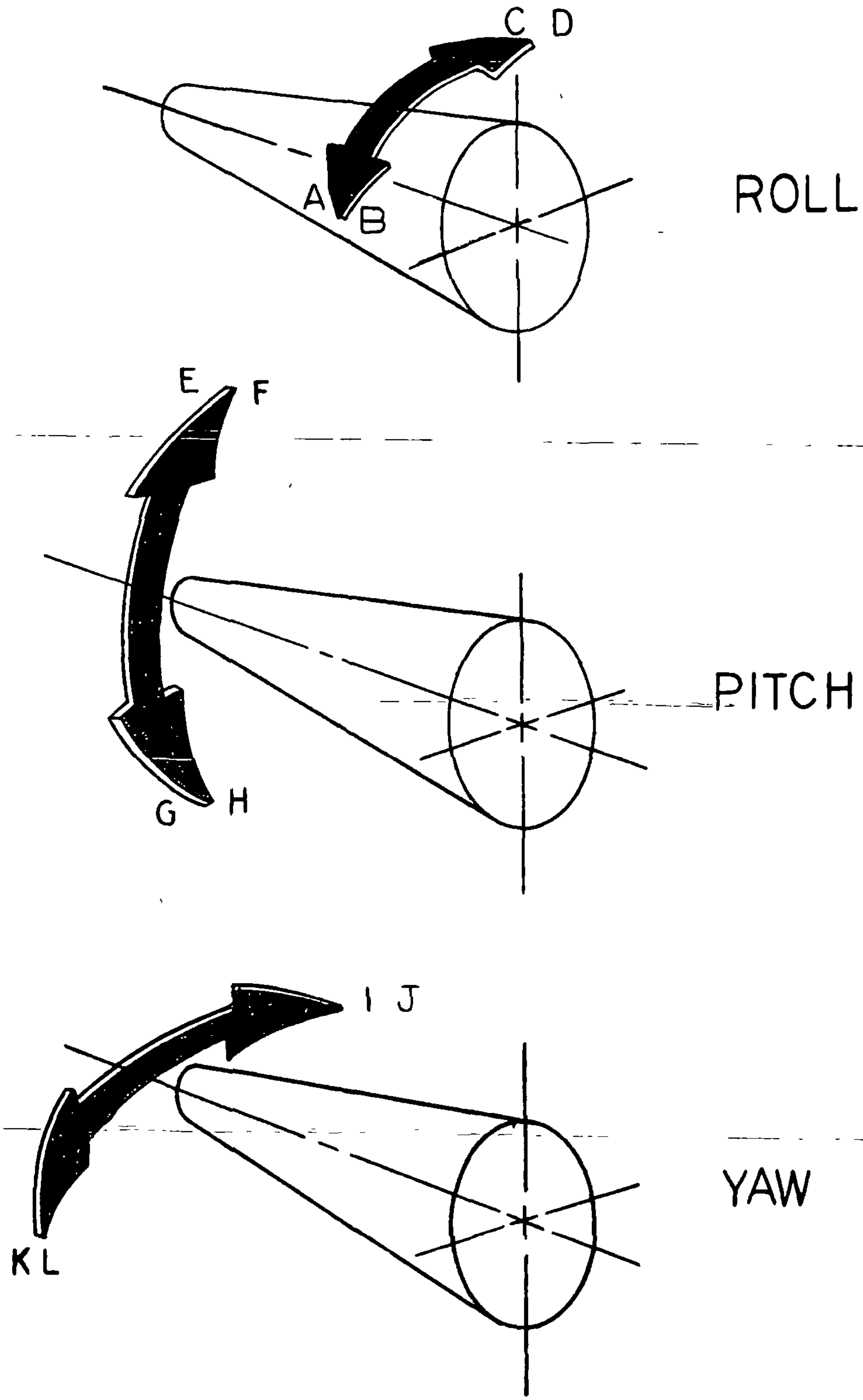


Figure 9. Rocket Combinations Required for Roll, Pitch, and Yaw Control

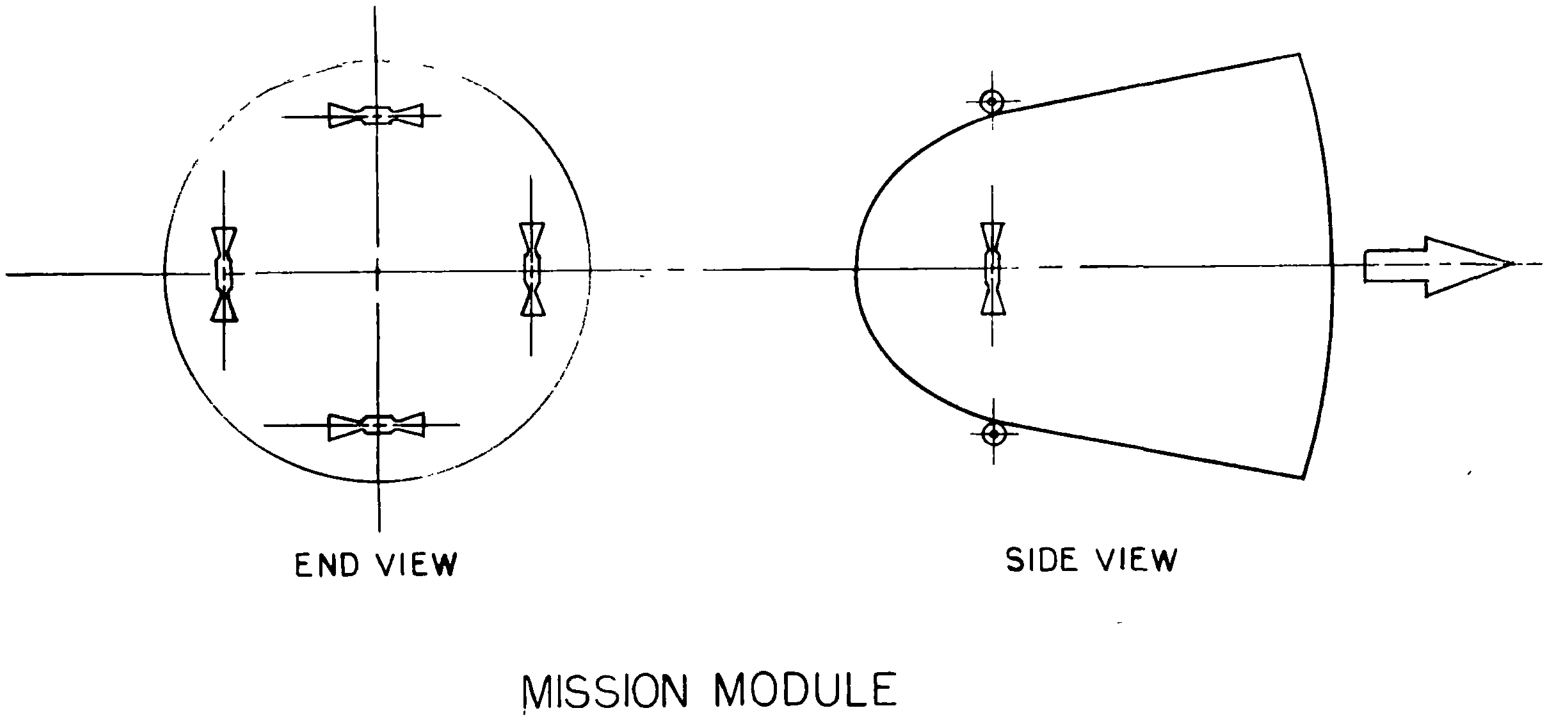


Figure 10. Re-Entry Roll Control Rockets (Four Groups of Two)

Thiokol-RMD work with the micro-pulse engines is discussed in detail in reports TPR 89 and TPR 93. The following paragraphs present a brief discussion of the engine designs and test work conducted.

Initial pulse testing was performed with an uncooled, bipropellant workhorse engine of stainless steel construction housing a graphite liner which delivered a nominal thrust of 3 lb at sea level. The injectors were 60 degree spray nozzles of a type manufactured by the Delavan Corporation for oil burner applications and were mounted to produce a one-on-one impinging conical spray pattern. This design is illustrated in Figure 11. It is equipped with solenoid actuated propellant control valves. Two storable, hypergolic propellant combinations were used: nitrogen tetroxide (N_2O_4) with unsymmetrical dimethylhydrazine (UDMH), and N_2O_4 with monomethylhydrazine (MMH). A total of 240 pulsing runs were made at various duty cycles with pulse durations ranging from 2 seconds down to 8 milliseconds. Cycling rates up to 22.5 pulses per second were achieved. The first 85 runs were made with N_2O_4 and UDMH, the remainder with N_2O_4 and MMH. Of the 155 runs with the latter combination, 133 were at sea level ambient pressure and 22 at simulated altitudes from 53,000 ft to 104,000 ft.

Subsequent tests were also conducted with low thrust (1/2 lb) micro-rockets made of pure molybdenum utilizing N_2O_4/MMH. This chamber was designed for operation at extremely low chamber pressure, approximately 7 psia, with injection pressure drops of 1 to 3 psia. In the series of tests conducted two problems became apparent. Ignition was unreliable and performance was particularly low. Careful review of test data indicated these faults probably resulted from partial gasification of liquid propellants in the injection system under the simulated attitude conditions of 90,000 ft.

Testing was then conducted on the medium chamber pressure unit illustrated in Figure 12. This unit was also fabricated from pure molybdenum. Performance of this unit was generally satisfactory. The C* values obtained were slightly low (approximately 4900 ft/sec). This is believed to be a result of operation in these tests at other than optimum mixture ratios for this propellant combination. Ignition was reliable at simulated altitudes of 90,000 ft. The pulse wave form was generally satisfactory with a rise time of approximately 28 milliseconds. The key problem area associated with the 1/2 lb micro-rocket was that of injector clogging. Chamber testing and injector flow calibrations were often interrupted due to clogging of the spray nozzles, even when exceptional caution was taken with filtration.

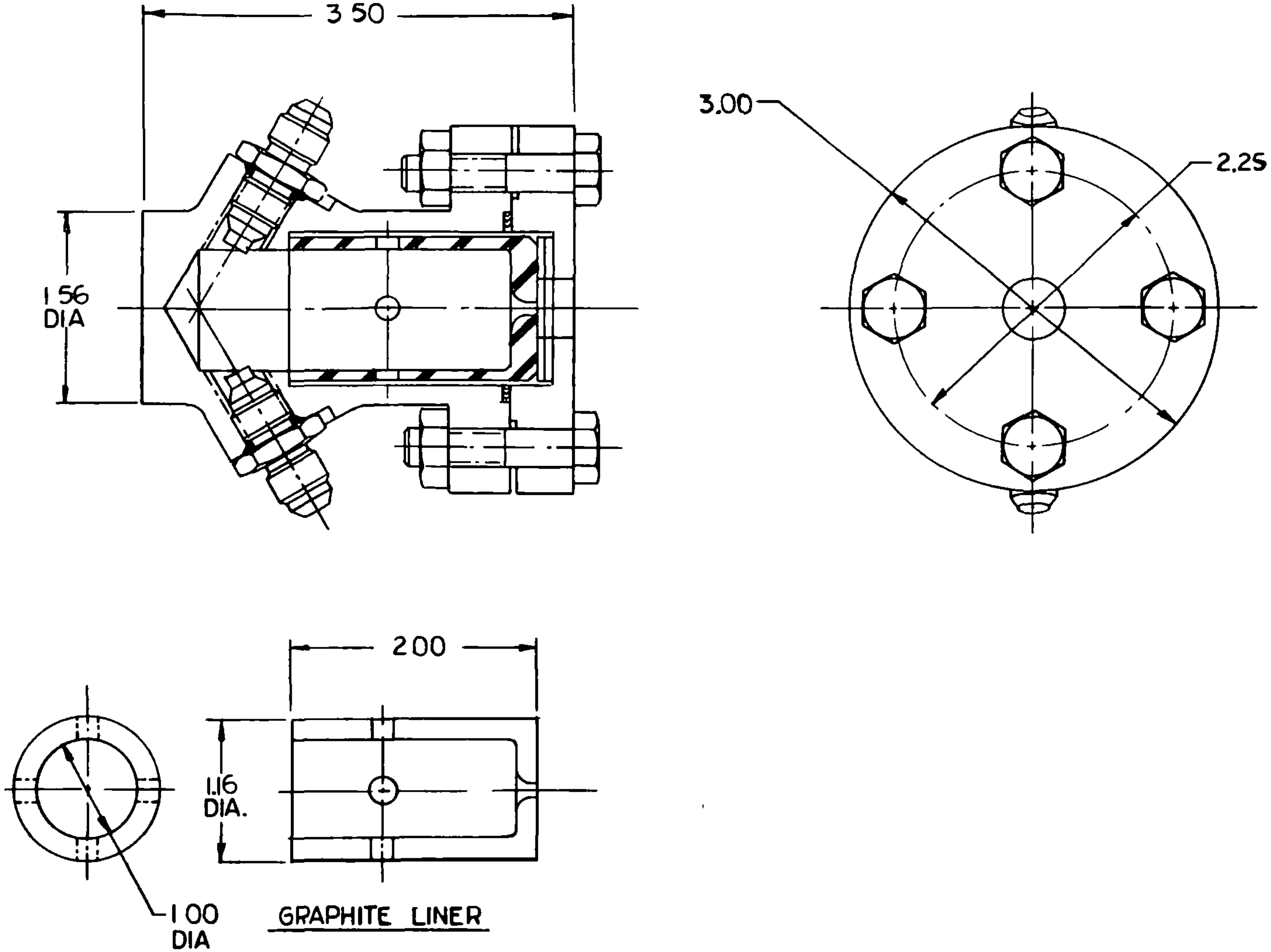


Figure 11. Micro-Pulse Thrust Chamber (Dwg 52005)

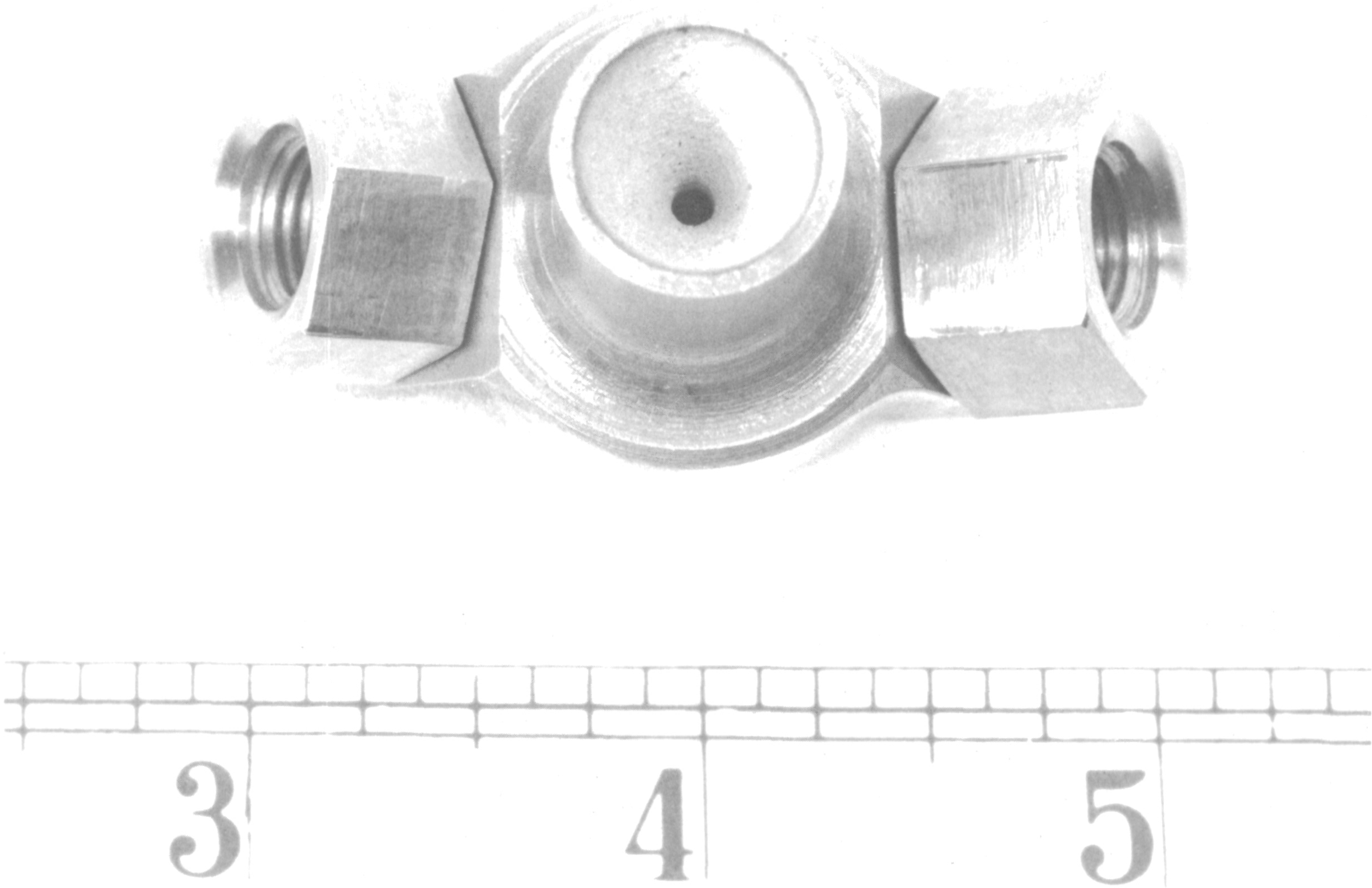


Figure 12. View of Medium Pressure 1/2 lb Chamber (2113-2)

To overcome this difficulty the medium chamber pressure unit was equipped with the Thiokol-RMD self-cleaning spray nozzles for injectors. Performance, altitude ignition, pulse wave form, and overall system reliability were satisfactorily demonstrated. Filters were not installed in the system during this testing and no difficulty was encountered with injector clogging; therefore, it is concluded that the self-cleaning feature was satisfactorily demonstrated.

A test program was also conducted to successfully operate in a vacuum a flight type 10 lb thrust, radiation cooled unit at 50% pulse duty cycle for 30 minutes and also for three ten-second steady state firings. These objectives were successfully met with the design configuration shown in Figure 13 consisting of a 75% molybdenum 25% tungsten chamber with a graphite nozzle. Propellants were N_2O_4/MMH. A trace of a typical firing is shown in Figure 14.

This preliminary program has established the feasibility of the Thiokol-RMD bipropellant pulse rocket beyond any doubt. Reliable, safe, rapid pulsing was accomplished. The effects on pulse response of such parameters as control valve design, thrust chamber L*, chamber pressure, and altitude environment were evaluated.

Thiokol-RMD is conducting a test program to study the propellant combination monomethylhydrazine-oxygen difluoride (MMH-OF_2) at low pressure (15 psia) in a 5 lb thrust, radiation cooled, attitude control rocket. Figure 15 is a photograph of this control rocket and Figure 16 is a sketch which indicates two types of injection schemes. One is a radial injector and the other is an inclined forward injector. An aft inclined injector (similar to Figure 11) was considered for this design but was rejected because the solenoid valves would be in "view" of the chamber and would overheat due to the greater amount of thermal radiaton from this chamber. This change in injector design was necessary because the 5 lb OF_2 chamber is designed to operate at an equilibrium temperature of 3000F whereas the micro-pulse engines were designed for only a 2000F equilibrium wall temperature. The chamber is fabricated from molybdenum.

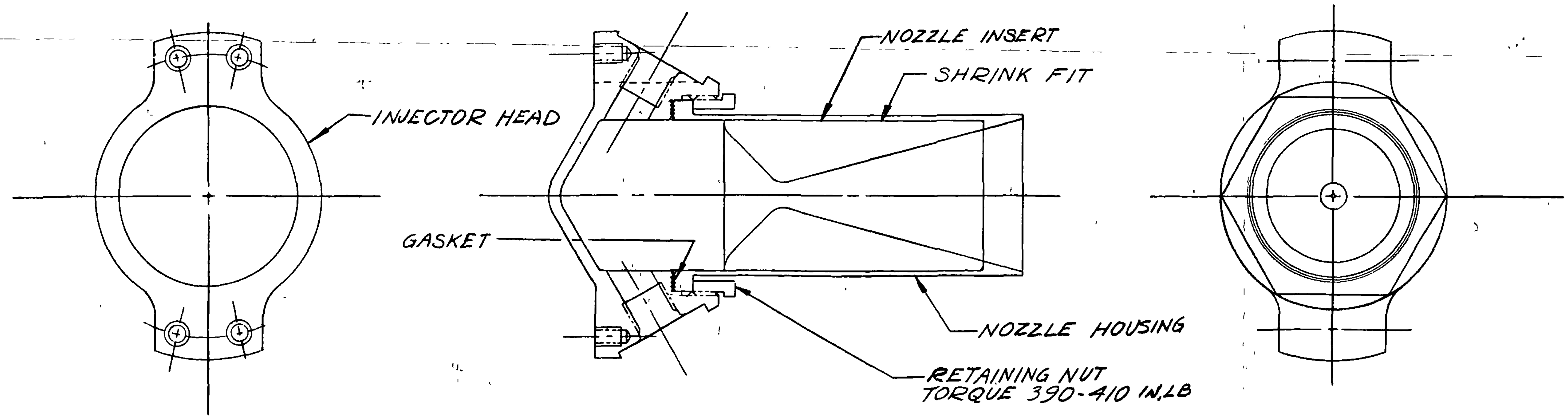


Figure 13. Flight Type 10 lb Thrust Bipropellant Chamber (Dwg 51981)

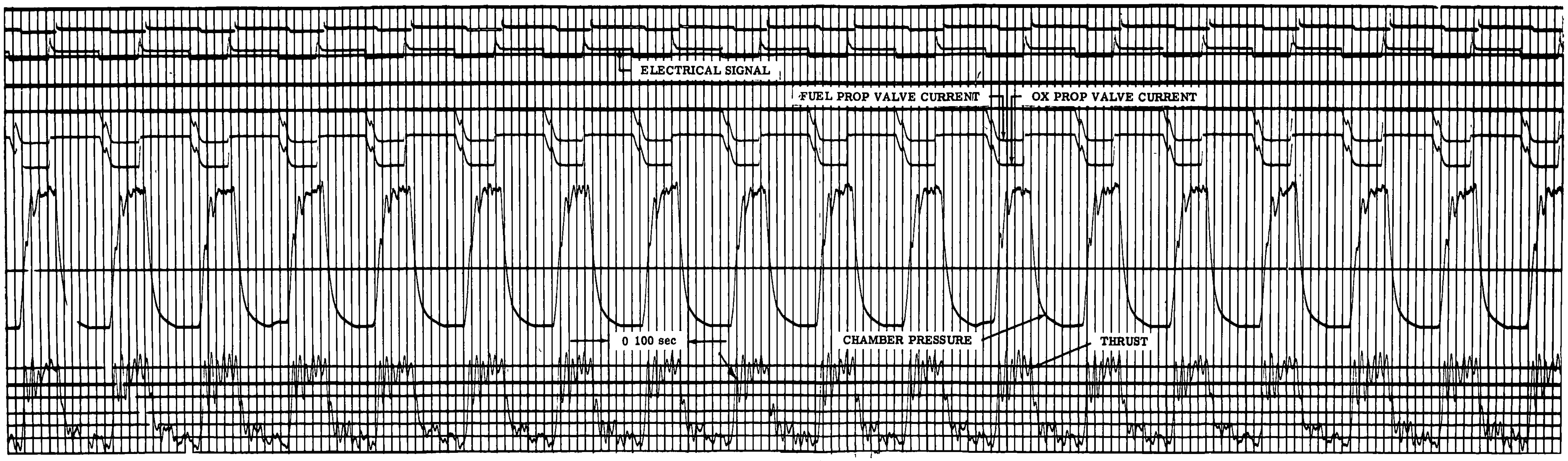


Figure 14. Typical Traces of 10 lb Thrust Unit, Run No. 2AX3487

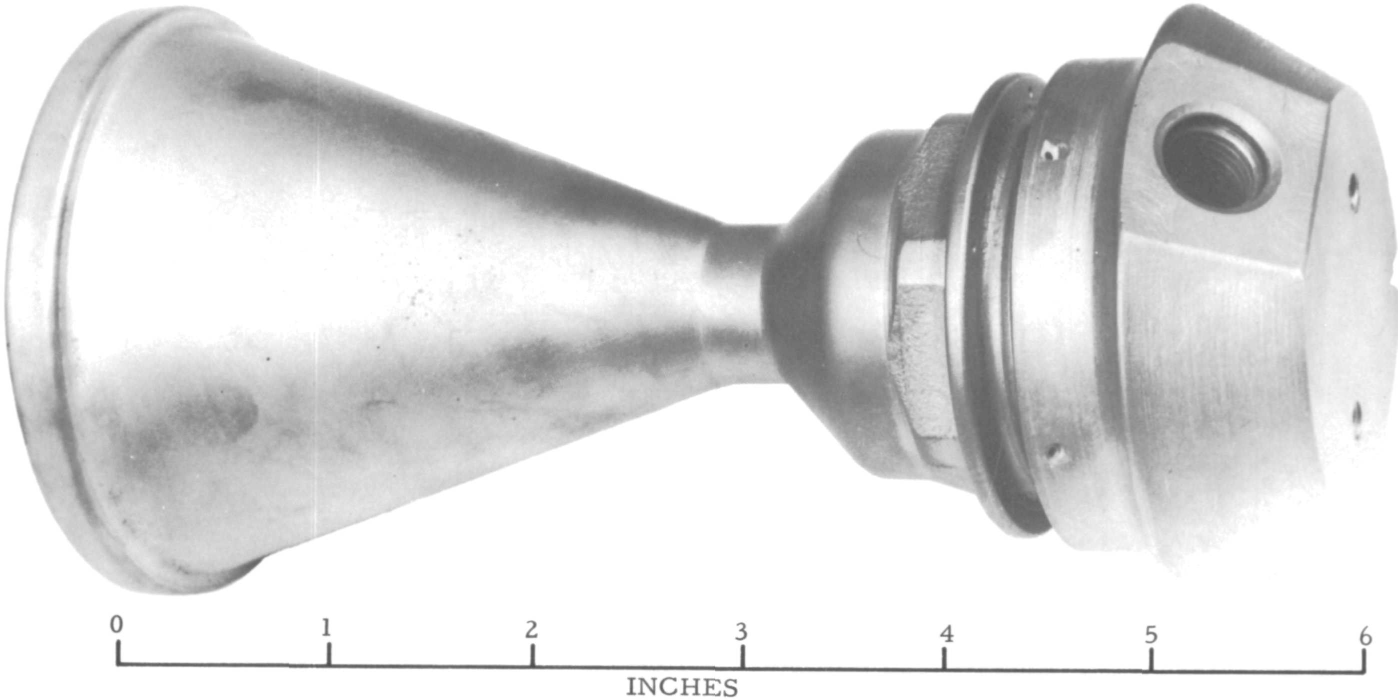


Figure 15. Attitude Control Rocket (2093-21)

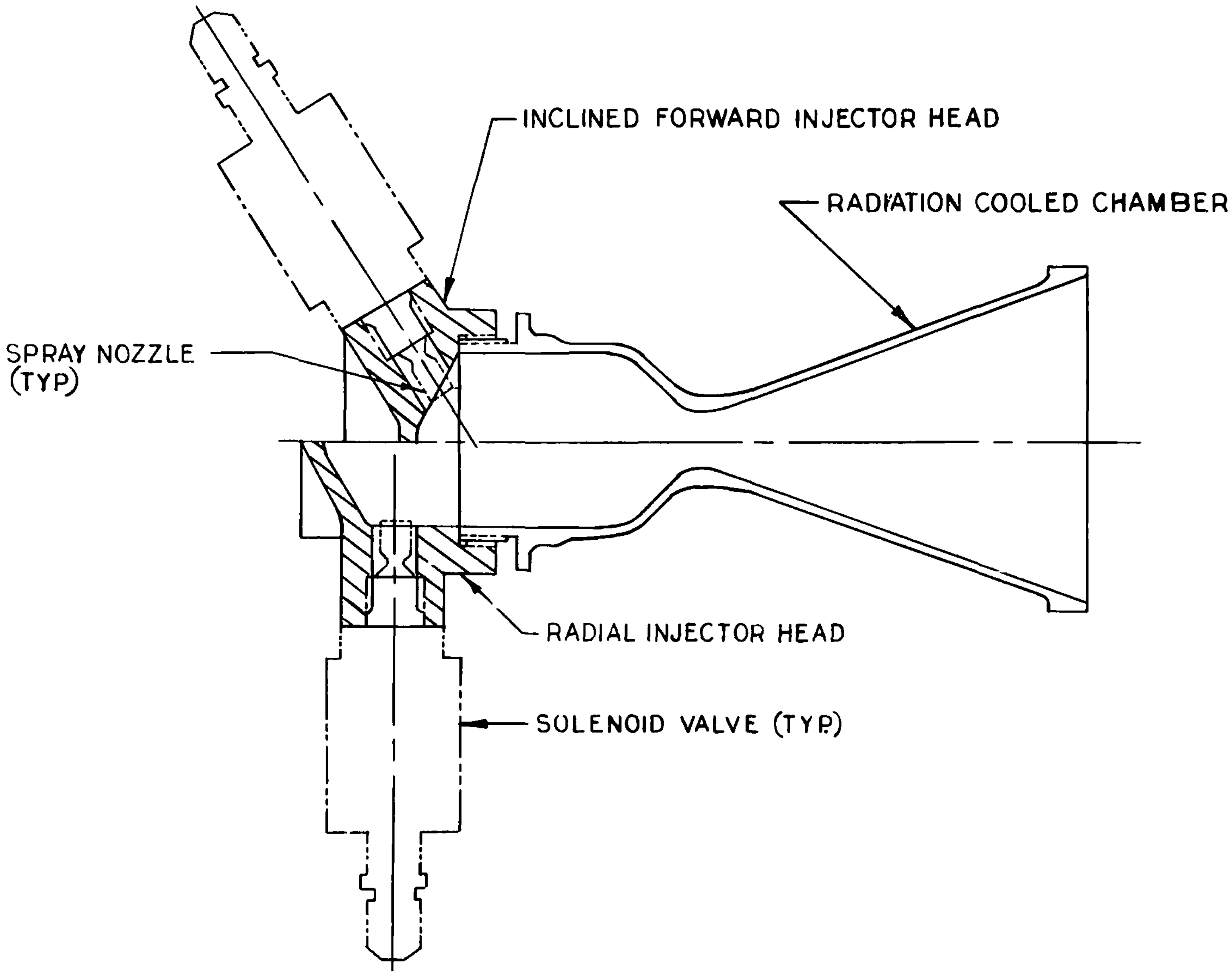


Figure 16. OF_2/MMH Space Motor, Showing Two Types of Injection Head Arrangements

V. MAIN ON-BOARD PROPULSION

A DESCRIPTION OF SYSTEMS

The main on-board propulsion systems proposed by Thiokol-RMD are helium pressurized propellant feed systems utilizing storable hypergolic propellants. The injectors are of a configuration currently under development on several programs at Thiokol-RMD. Other key design features (for example, the cooled, ceramic throat insert and the ablative chamber linings) are under active development or are incorporated in production engines. These designs are discussed in detail in a subsequent section.

The 1963 system will utilize cold stored helium gas to pressurize positive expulsion propellant tanks. Expulsion will be by flexible bladders. The storable hypergolic propellant combination chosen for this system is N_2O_4/MMH. Gimbaling of the thrust chambers will be achieved by servo operated hydraulic supply valves utilizing the pressurized fuel supply as the hydraulic fluid. The design of this system is based on the parameters established by the General Electric Company. A layout of the system is shown in Figure 17, which shows the propulsion system supported by a conical structure. This cone carries the basic structural weight and transmits the thrust loads to the vehicle. It further acts as a flameshield and as a meteorite shield against meteoritic particles entering from the rear of the propulsion module. An important advantage of this powerplant is its relatively compact arrangement. A reduction in the length of the propulsion module of 54 inches is possible. At 52 lb/ft, this provides a weight savings of 234 pounds.* Detailed discussions of the various components in this system will appear in subsequent sections.

A summary of the vehicle performance with this powerplant follows.

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* The 234 pounds represents weight saved in the basic vehicle structure which would represent an increase in useful payload.

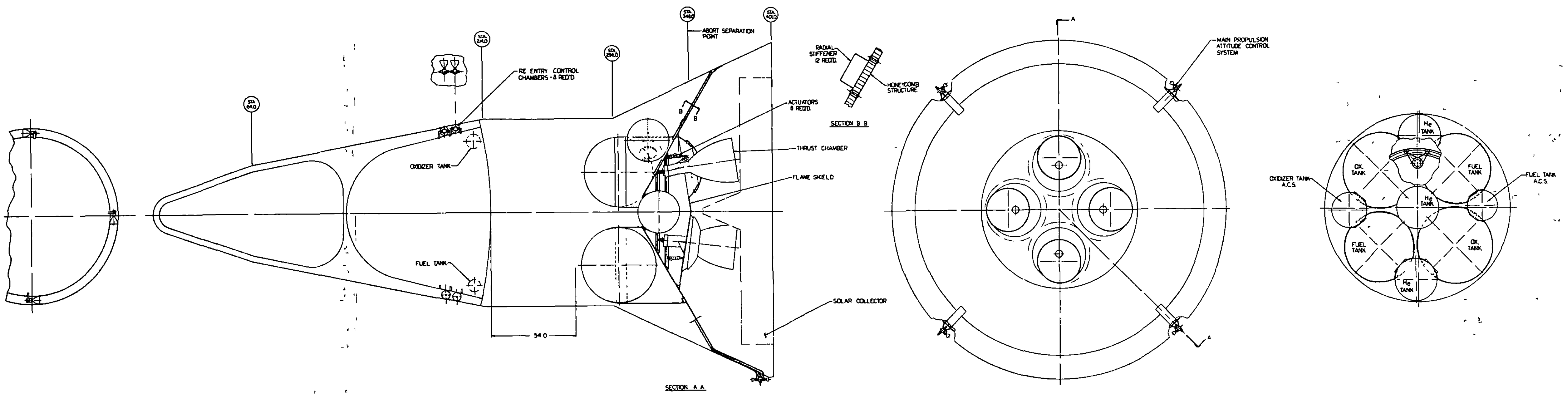

Figure 17. Main On-Board Propulsion System, 1963 (Dwg 52210)

Initial Vehicle Weight, lb	Propellant Reserve	ΔV ft/sec	Payload and Structure Weight, lb*†
15,715	10%	7500	5544
15,715	5%	7500	5942
15,000	10%	5600	6861
15,000	5%	5600	7181

The 1966 system will be upgraded in performance by the substitution of OF_2 for the N_2O_4 and a pressurizing system utilizing helium gas stored as a liquid. These two items result in a sharp decrease in powerplant loaded weight, providing for increased payloads. The liquid helium is converted to gas in a heat exchanger wrapped around the main propellant manifolds. Expulsion will be by flexible bladders, probably Teflon or a metallic design in the case of the OF_2. Gimbaling, propellant injection, and other features are essentially similar to the earlier system except that it is planned to use two chambers rather than four. These chambers will have nozzle extensions of molybdenum which will be radiation cooled. Stepped thrust will be utilized to provide the 6K and 24K flexibility, i.e., each chamber will operate at 6K or 12K. A layout of the system is shown in Figure 18.

Prior to operation (at separation from the booster) the aft conical structure would be removed from the vehicle to permit the chambers to radiate heat into space and to provide the required angle for gimbaling.

A summary of the vehicle performance with this powerplant is given below.

Initial Vehicle Weight, lb	Propellant Reserve	ΔV ft/sec	Payload and Structure Weight, lb*
14,715	10%	7500	6381
14,715	5%	7500	6705

*The payload and structure weight is the difference between the loaded powerplant weight (including all main propellants, attitude and re-entry roll control propellants, and all helium gas) and the initial vehicle weight.
†This does not include the 234 lb weight savings possible as a result of shortening the propulsion module.

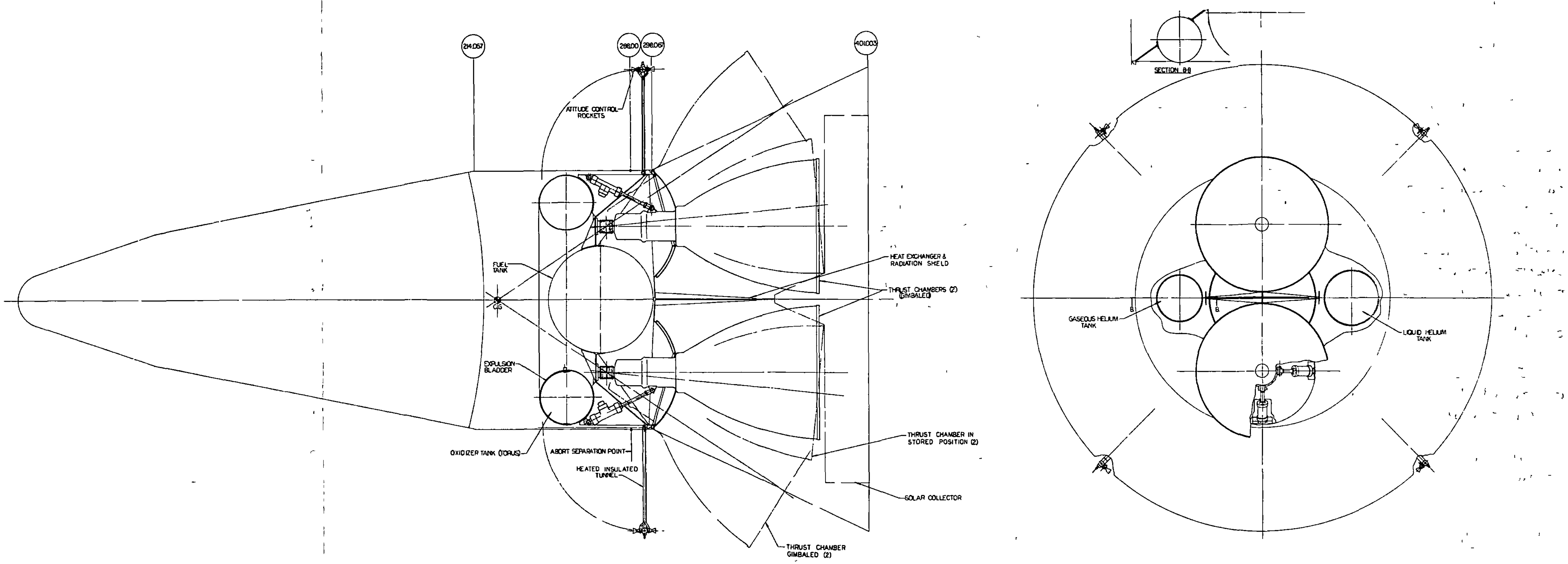

Figure 18. Lunar Powerplant Stepped Thrust Chamber (Dwg 52272)

SYSTEM PARAMETERS AND WEIGHTS

Item	1963 System	1966 System
Propellants	N_2O_4/MMH	OF_2/MMH
Pressurization System	Helium Gas (stored as a gas)	Helium Gas (stored as a liquid)
Operating Temperature Limits	20-80 F	20-80 F
Oxidizer to Fuel Ratio (bw)	2.1	2.5
Vacuum Specific Impulse (minimum)	319 sec	403 sec
Area Ratio	40	100
Pressurant Storage Pressure	3000 psia	500 psia (liquid) 3000 psia (gaseous)
Propellant Tank Pressure	390 psia	390 psia
Chamber Pressure	200 psia	200 psia
Total Propellant Weight	8931 lb	7127 lb
Usable Propellant for:		
Velocity Increments*	7960 lb	6290 lb
10% Reserve	796 lb	629 lb
Total Loaded System Weight**	10,171 lb	8334 lb

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* A velocity increment of 7250 ft/sec was used to calculate required propellant weights. A further increment of 250 ft/sec is provided for in the main attitude control propellant supply.

** These figures include helium supply and tankage for the main attitude control and re-entry roll control systems. The weight of the structure to support the propulsion system and tie it into the vehicle skin is not included since this originally was a part of the vehicle.

A detailed weight summary of all of the propulsion systems is given in Appendix A.

An alternate to the 1963 system is presented in Figure 19. This system employs four fixed 6K thrust chambers, having molybdenum radiation cooled nozzle extensions with an area ratio of 100. Prior to operation (at separation from the booster) the aft conical structure would be removed from the vehicle to permit radiation from the nozzle extensions outward into space. The main attitude control rockets would be relocated but, functionally, this system and the re-entry roll control system would remain the same.

This alternate system is designed to provide space abort capability only. Utilizing the N_2O_4/MMH propellant combination it would provide a simple, reliable powerplant based on current Thiokol-RMD technology that could readily be developed for circumlunar flight by 1963.

A summary of the vehicle performance with this powerplant is given below.

Initial Vehicle Weight, lb	Propellant Reserve	ΔV ft/sec	Payload and Structure Weight, lb*
15,715	10%	5600	7416
15,715	5%	5600	7737
15,000	10%	5600	7020
15,000	5%	5600	7327

An alternate to the 1966 system is shown in Figure 20. This system employs a reverse flow nozzle fed by eight individual combustors. A discussion of the thrust chamber concept is given in Part E, Thrust Chamber Design, of this section. This alternate system utilizes OF_2/MMH and a hybrid chemical gas generation system for pressurizing the propellant tanks. The increased payload capability over that of the 1963 system results from the large savings in propellant weight resulting from the increased performance of OF_2/MMH and the reduction in system weight resulting from

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* The payload and structure weight is the difference between the loaded powerplant weight (including all main propellants, attitude and roll control propellants, and helium gas) and the initial vehicle weight.

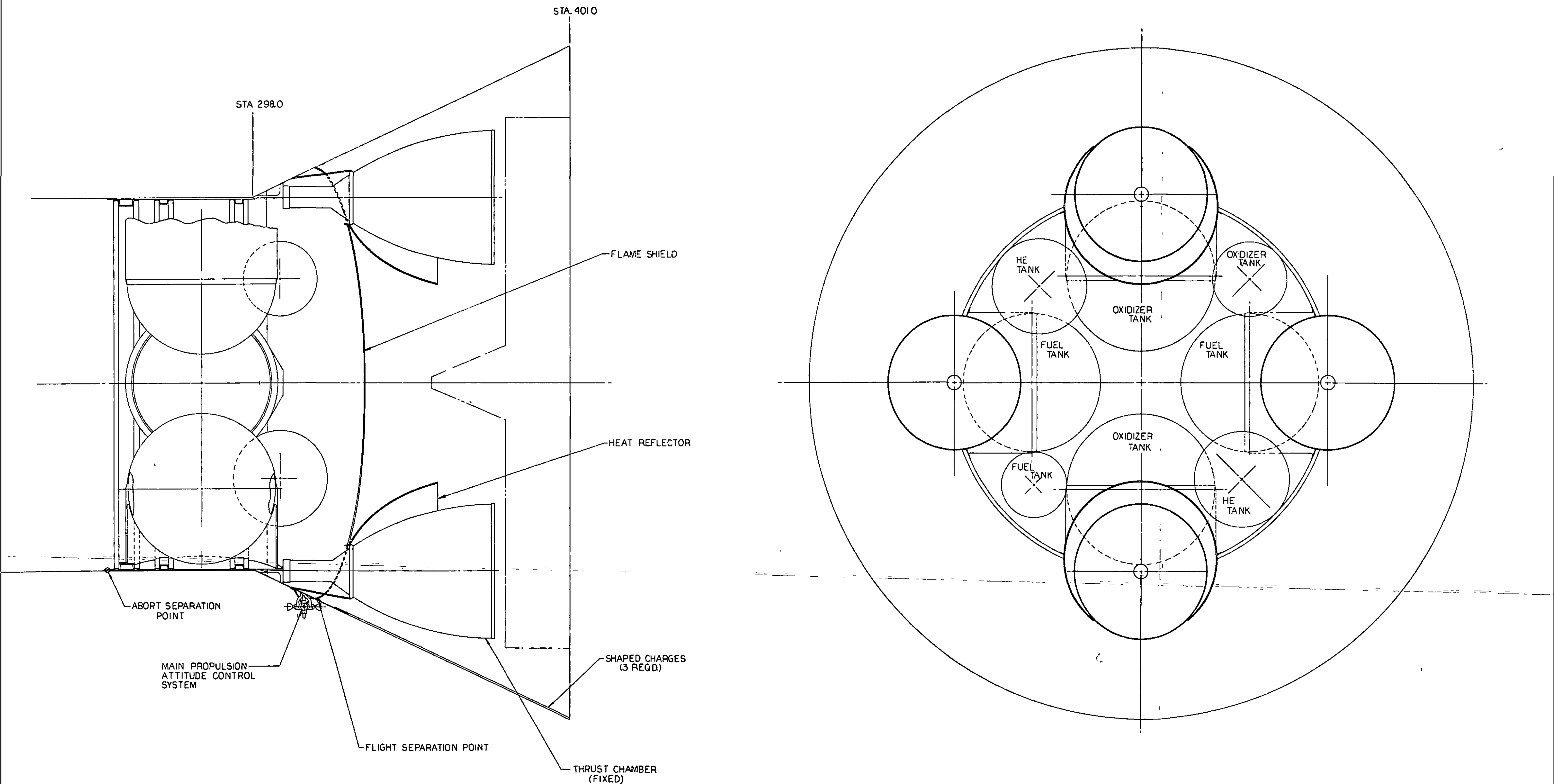

Figure 19. Main On-Board Propulsion Alternate System, 1963 (Dwg 52228)

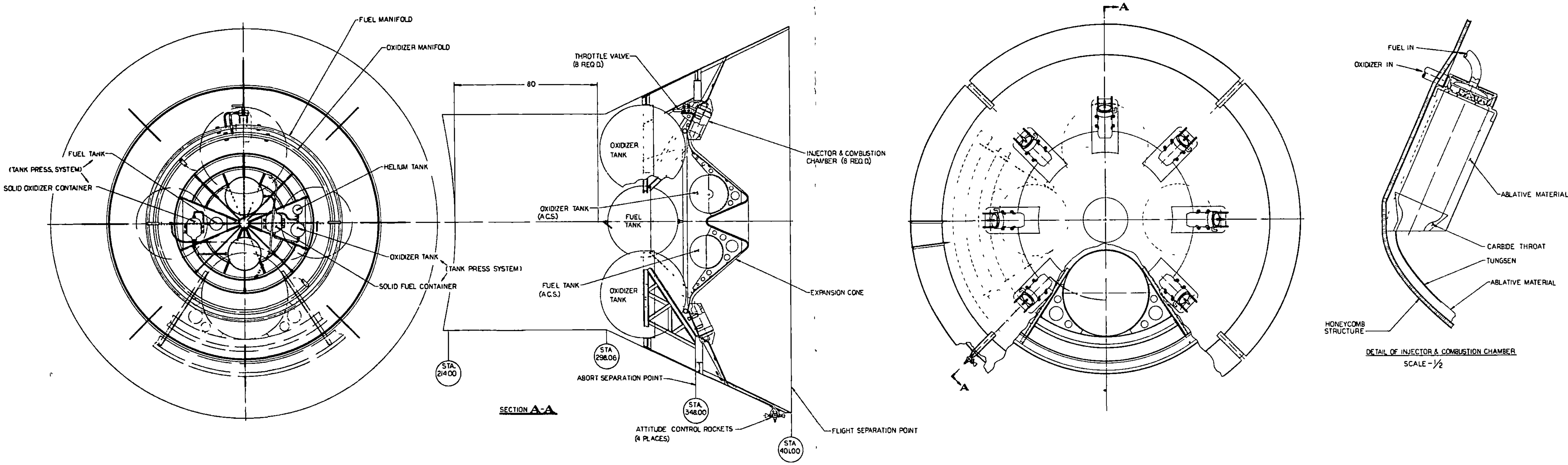

Figure 20. Main On-Board Propulsion Alternate System, 1966 (Dwg 52209)

the more efficient pressurizing system. It is recognized that the development of the pressurizing system must be carried out under the applied research program during the first year of the 1966 program However, the system offers a weight savings of more than 100 pounds over the helium gas (stored as gas) pressurization system and is competitive with the liquid helium storage system. The wide experience in solid fuels and oxidizers and hybrid propulsion systems of Thiokol Chemical Corporation will be utilized here.

A third important advantage of this alternate system is its extreme compactness Referring to Figure 20, a shortening of the ten foot diameter cylindrical skin of the propulsion module of some 80 inches is possible. At 52 lb/ft, this means an additional weight savings of 350 lb, and a means of shortening the overall vehicle length by almost seven feet.

A summary of the vehicle performance with this powerplant is given below.

Initial Vehicle Weight, lb	Propellant Reserve	ΔV ft/sec	Payload and Structure Weight, lb*
14,715	10%	7500	6347
14,715	5%	7500	6662

Figure 21 illustrates a second advanced nozzle concept applied to the 1966 system, a radial outflow nozzle. Although a preliminary estimate of the weight of this system indicates that it will be heavier than the other two 1966 systems, it offers the following advantages:

Compact design - a reduction in vehicle length of 118 inches, resulting in a reduction in vehicle structural weight of well over 500 pounds.

Higher performance - illustrated is a design with an area ratio of 100 Combustion research and detail design promise to reduce weight and permit an increase in area ratio.

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* The payload and structure weight is the difference between the loaded powerplant weight (including all main propellants, attitude and roll control propellants, and propellant tank pressurant weights) and the initial vehicle weight. It does not include the 350 lb weight savings possible as a result of shortening the structure.

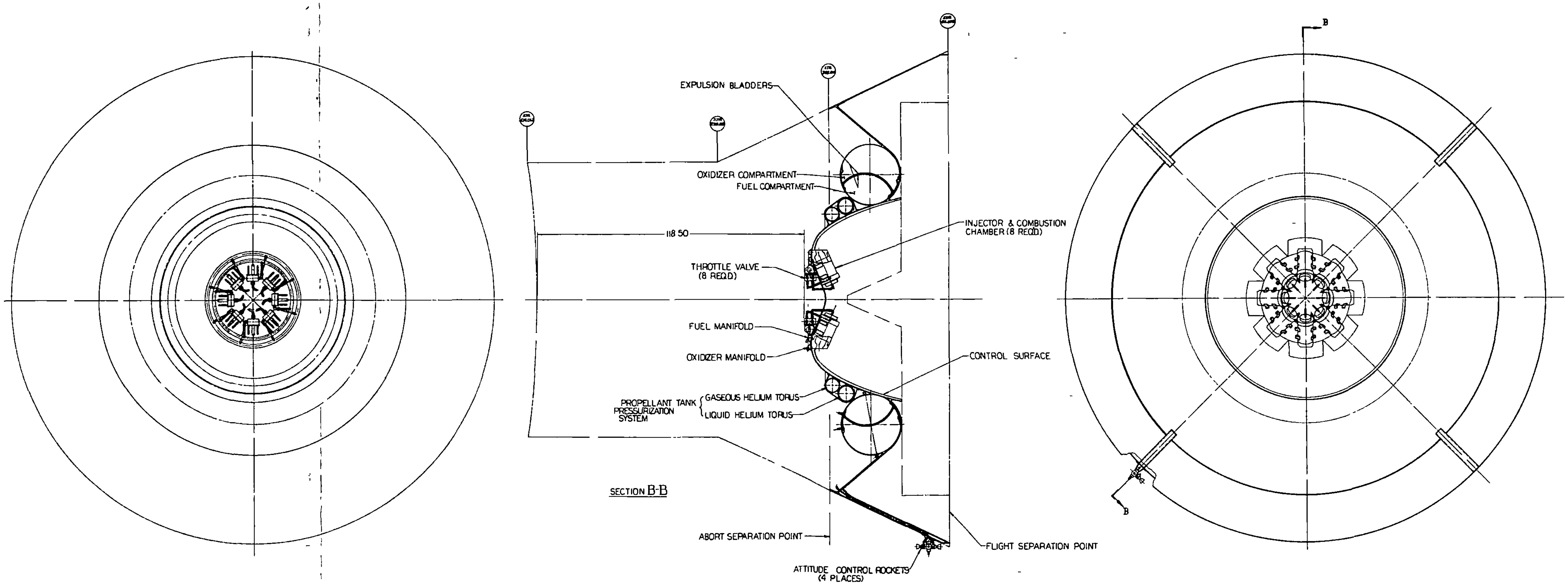


Figure 21. Lunar Powerplant, Radial Chambers Installation (Dwg 52273)

It should be remembered that these two advanced nozzle concepts are preliminary in nature. They stand to benefit from the experience gained during several years of applied research and development.

B SYSTEM SCHEMATICS AND SEQUENCE OF OPERATIONS

Figures 4 and 5 present the fluid and electrical schematic drawings of the 1963 main on-board propulsion system. The pressure budget is given in Table IV and the sequence of operation is outlined in the Model Specification, Appendix B.

Figures 22 and 23 present the fluid and electrical schematic drawings of the 1966 main on-board propulsion system. The sequence of operation is given separately in Appendix D.

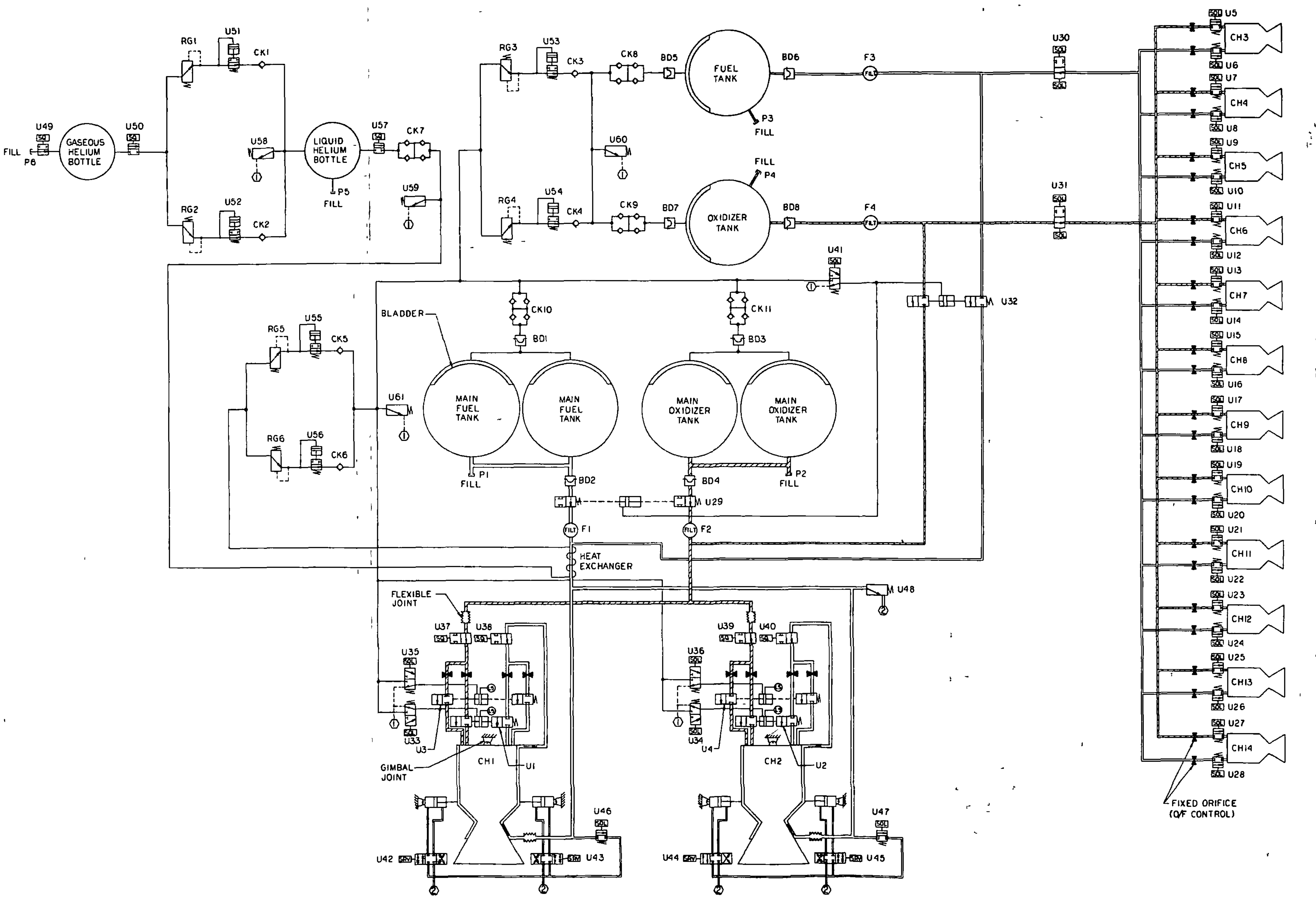
C. PROPELLANT SELECTION

An analysis was performed covering a number of earth storable and cryogenic propellant combinations for a typical Apollo mission. The variation in payload and vehicle structure weight as a function of velocity increment is shown in Figures 24 and 25. Figure 25 illustrates the division of the propellant systems into three performance groups: earth storable hypergols, cryogenics and BN systems, and the fluorinated oxidizers.

The earth storable hypergols offer hypergolic ignition, less complex handling requirements, and an advanced state of development. To employ the increased payload capacity of the other propellant groups requires low temperature propellant storage and development of techniques for handling and, in some cases, ignition.

Of particular interest in Figure 24 is the high performance of N_2O_4/MMH in the earth storable hypergols group, and OF_2/MMH in the fluorinated oxidizers groups. A propulsion system developed for N_2O_4/MMH can be upgraded considerably in performance by the substitution of OF_2 for the N_2O_4

At optimum O/F, both propellant combinations have about the same bulk densities. Also, both oxidizers are compatible with several common tankage materials. For the case of a fixed velocity increment, the reduced propellant weight of the OF_2/MMH system could be directly loaded in a tankage system designed for N_2O_4/MMH with only a small penalty in excess tank weight involved.



ITEM	DESCRIPTION
CH1 CH2	MAIN THRUST CHAMBER GIMBALED
CH3-CH14	ATTITUDE CONTROL THRUST CHAMBER
U1 U2	1ST STAGE MAIN PROP VALVE, PNEU OPERATED
U3 U4	2ND STAGE MAIN PROP VALVE, PNEU OPERATED
U5-U28	ATTITUDE CONTROL PROP VALVE, SOLENOID OPERATED
U29	MAIN PROPELLANT TANK SHUTOFF VALVE, PNEU OPERATED
U30,U31	ATTITUDE CONTROL SYSTEM SHUTOFF VALVE, SOL OPERATED
U32	CROSSFEED PROPELLANT VALVE SOLENOID OPERATED
U33,U34	1ST. STAGE MAIN PROP VALVE PILOT VALVE SOL OPERATED
U35,U36	2ND STAGE MAIN PROP VALVE PILOT VALVE, SOL OPERATED
U37-U40	MAIN CHAMBER MALF SHUTOFF VALVE SQUIB OPERATED
U41	MAIN PROP TANK SHUTOFF VALVE & CROSSOVER VALVE PILOT VALVE
U42-U45	MAIN CHAMBER GIMBAL VALVE, SERVO OPERATED
U46 U47	SERVO FEED VALVE, SOLENOID OPERATED'
U48	MAIN FUEL MANIFOLD RELIEF VALVE
U49	GASEOUS HELIUM FILL VALVE DUAL SQUIB OPERATED
U50	PRESSURIZING VALVE DUAL SQUIB OPERATED
U51-U56	HELIUM REGULATOR MALF SHUTOFF VALVE PNEU OPER
U57	HELIUM SUPPLY VALVE, DUAL SQUIB OPERATED
U58-U61	HELIUM PRESSURE RELIEF VALVE
RG1-RG6	PRESSURIZING GAS REGULATOR
CK1-CK6	CHECK VALVE, SINGLE
CK7-CK11	CHECK VALVE, QUAD
BD1-BD4	MAIN PROPELLANT TANK BURST DISC
BD5-BD8	ATTITUDE CONT'L. PROPELLANT TANK BURST DISC
F1-F4	PROPELLANT FILTER
P1-P6	FILL PLUGS

Figure 22. Main On-Board Propulsion System Schematic, 1966

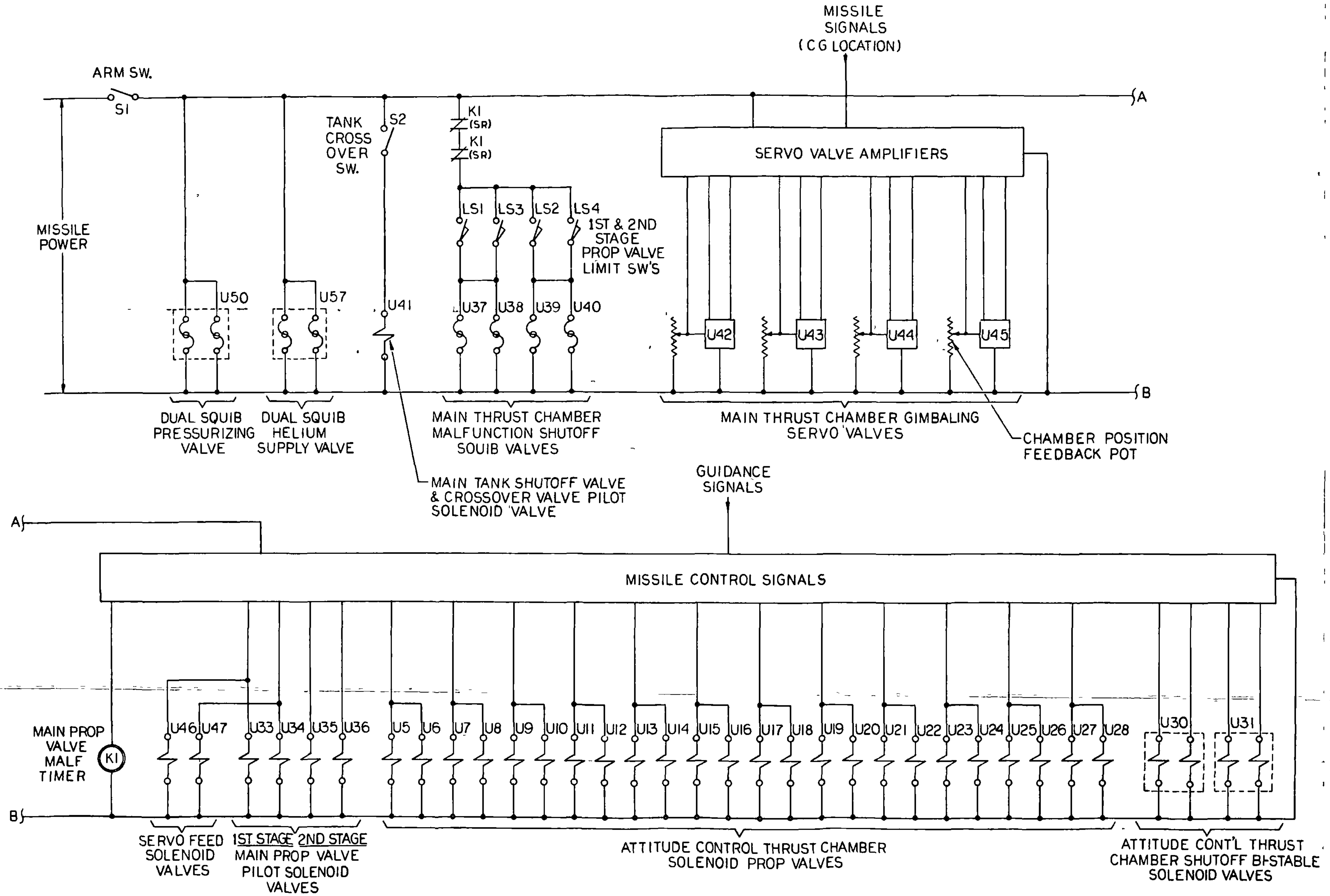


Figure 23. Main On-Board Propulsion System Electrical Schematic, 1966

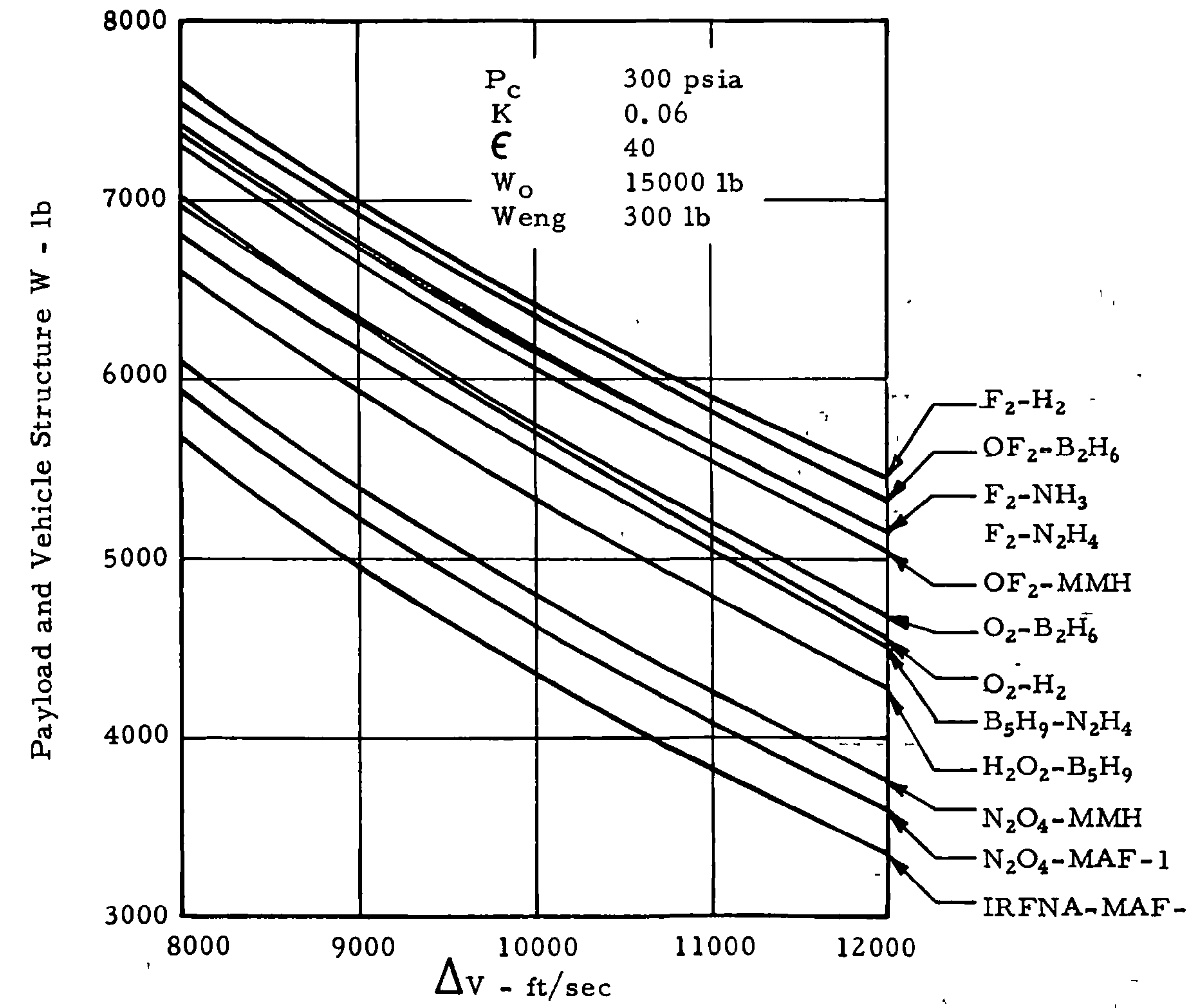


Figure 24. Weight vs Velocity Increment (Apollo)

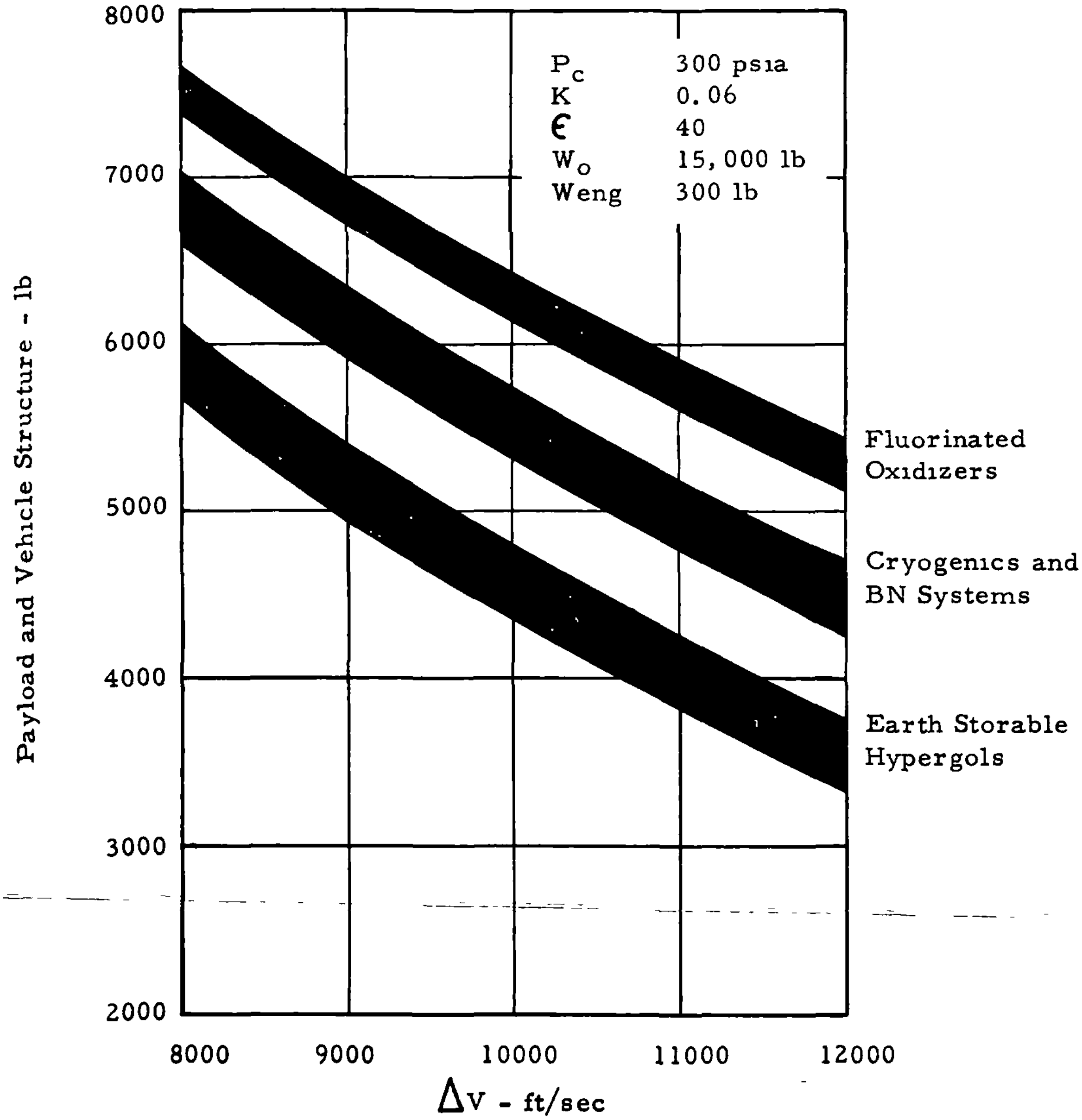


Figure 25. Weight vs Velocity Increment (Apollo)

The use of the nitrogen tetroxide system would permit early testing and flight of the vehicle, taking full advantage of technology which exists now at Thiokol-RMD The change to OF_2 at a later date would permit realization of the very high payload capabilities of the fluorinated oxidizers.

For missions involving extended periods of time in space, the effects of boil-off losses on the performance of cryogenic systems are large. To minimize these losses, particularly with the liquid hydrogen systems, consideration must be given to tank insulation, vehicle orientation, and outer surface absorptivity and emissivity characteristics.

Satisfactory storage of the earth storable hypergols can be achieved by selection of tank surface finishes alone. The entire vehicle (command, mission, and propulsion module) will be maintained at substantially the same temperature. Under these conditions the storage of OF_2 can be achieved with a minimum of insulation weight; for example, for the Apollo system, only about five pounds is required.

A study was made of four candidate propellant systems for Apollo. In each case the propellant, tankage, and insulation weights were carefully calculated to provide a valid comparison. With these data the payload and vehicle structure weights were determined after calculating the powerplant weights. The results are shown in Figure 26. The payload capability of OF_2/MMH is competitive with O_2/H_2 and F_2/NH_3. If the weight savings from shortening the vehicle is considered, a payload advantage of 300 pounds occurs over that of O_2/H_2. This figure illustrates the advantages of the OF_2/MMH system.

- Competitive payload
- Hypergolic ignition - increased reliability in space
- Improved space storability
- Simplified handling characteristics
- Growth potential - payload increase of N_2O_4/MMH is 13%.

D. DESIGN TRADE-OFFS

The principal trade-offs considered to date were chamber pressure and area ratio. Several studies on optimum chamber pressure have been completed recently at Thiokol-RMD. These studies have shown that, for pressurized engine systems, the optimum chamber pressure falls between 100 and 200 psia. The cryogenic systems optimize at the lower end of the scale,

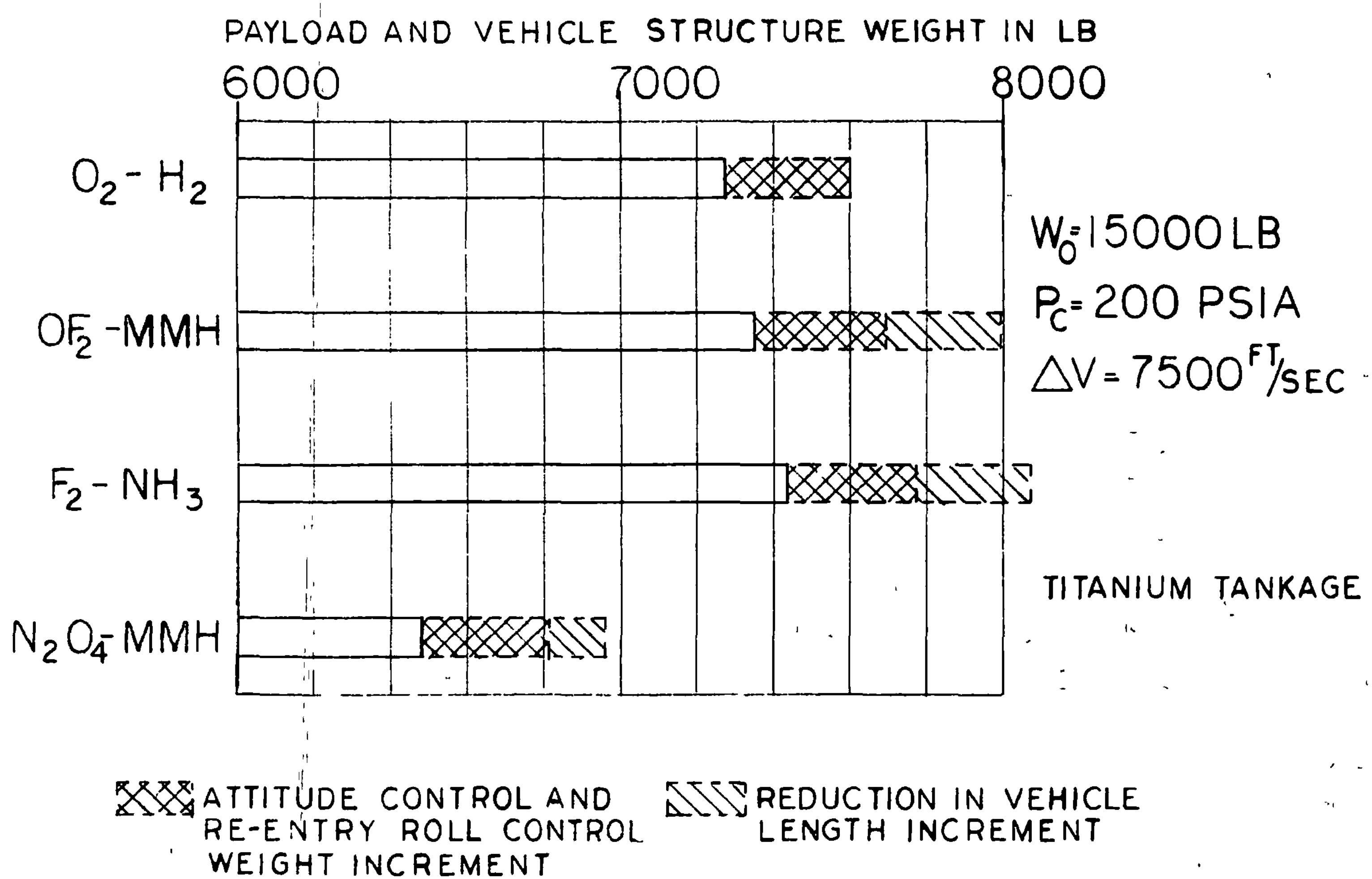


Figure 26. Comparative Propellant Performance

the storable hypergols at the higher chamber pressures. The results of one of these studies are shown in Figure 27

This study was done for a 10,000 lb thrust exo-atmospheric pressurized engine. The area ratio was held constant at 40. Propellants were pentaborane/hydrazine. The minimum loaded powerplant weight occurs at or just below 200 psia chamber pressure. On the basis of these studies the chamber pressure for the Apollo vehicle was chosen initially as 200 psia.

An area ratio trade-off was performed for the 1963 Apollo system. The study was made for a vehicle weight of 15,715 pounds and a velocity increment of 7500 ft/sec. Figure 28 shows that maximum payload occurs at an area ratio between 50 and 60. However, the gain in payload beyond 40 is small. This latter value was chosen as representing the maximum payload while reducing development costs and design complexity.

The optimum area ratio is smaller than anticipated, originally, it was planned to operate at an area ratio of 100. The reason for this is the fact that only one thrust chamber out of four operates during a normal mission. The performance gain in going to a higher area ratio is, therefore, enjoyed by only one chamber The weight penalty of the larger nozzle, however, is paid by all four. The net result of this is an optimum area ratio somewhat lower, near 50.

The overall vehicle performance for the two principal propulsion systems is given in Figure 29. For several initial gross loaded weights, the variation in payload and structural weight as a function of velocity increment is presented. These curves represent minimum vehicle performance. All of the dry weights and propellants associated with the main attitude and re-entry roll control systems (including mounting brackets, etc.) are included in the loaded powerplant weight. No allowance has been made for increase in payload as a result of reduction of vehicle length This figure graphically illustrates the payload improvement possible through upgrading the 1963 system. The OF_2/MMH system provides a 26% increase in payload and structural weight when compared on a constant vehicle weight basis.

E. THRUST CHAMBER DESIGN

The design of the main on-board propulsion system for the 1963 version of the Apollo vehicle embodies concepts which are well established in the background of the Reaction Motors Division of the Thiokol Chemical Corporation.

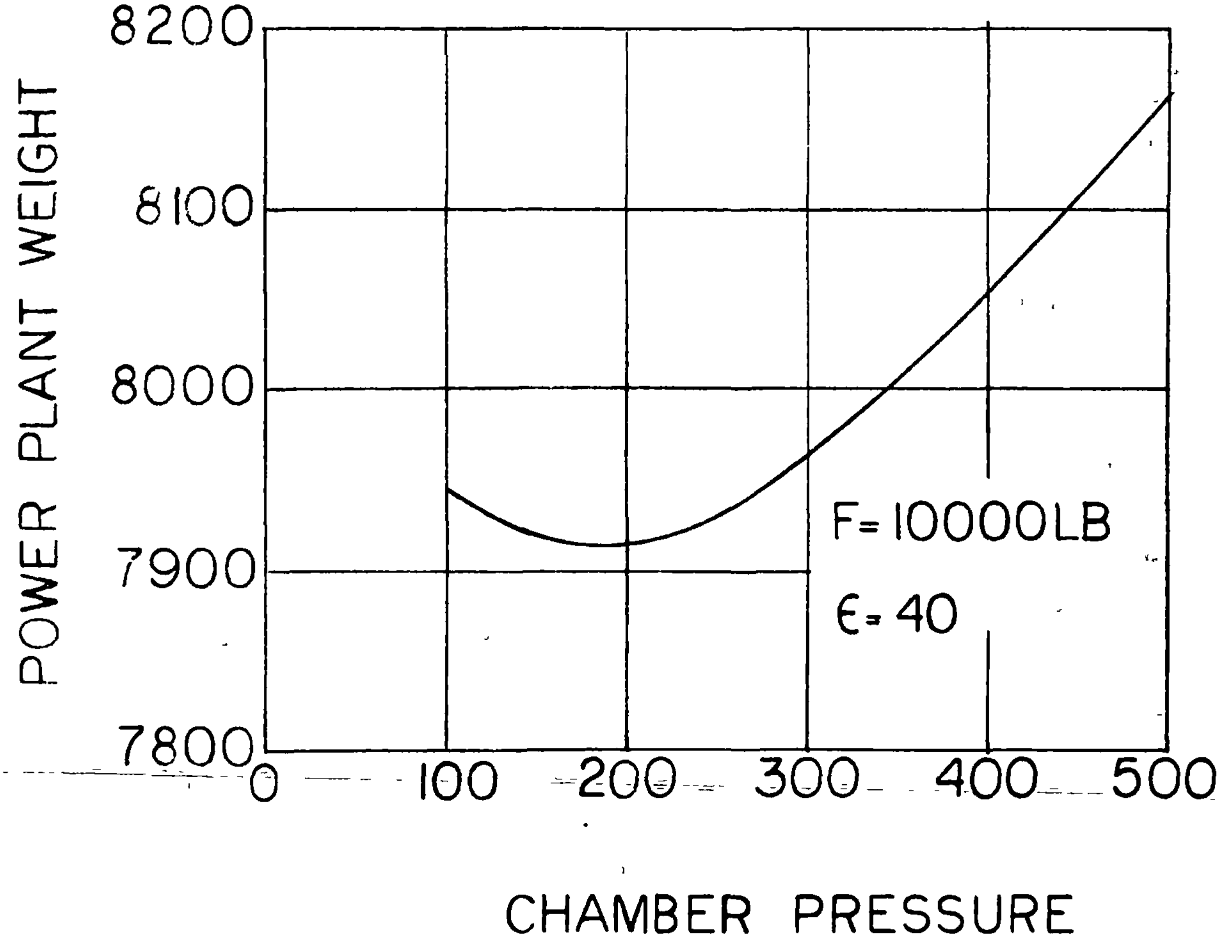


Figure 27. Chamber Pressure Trade-Off

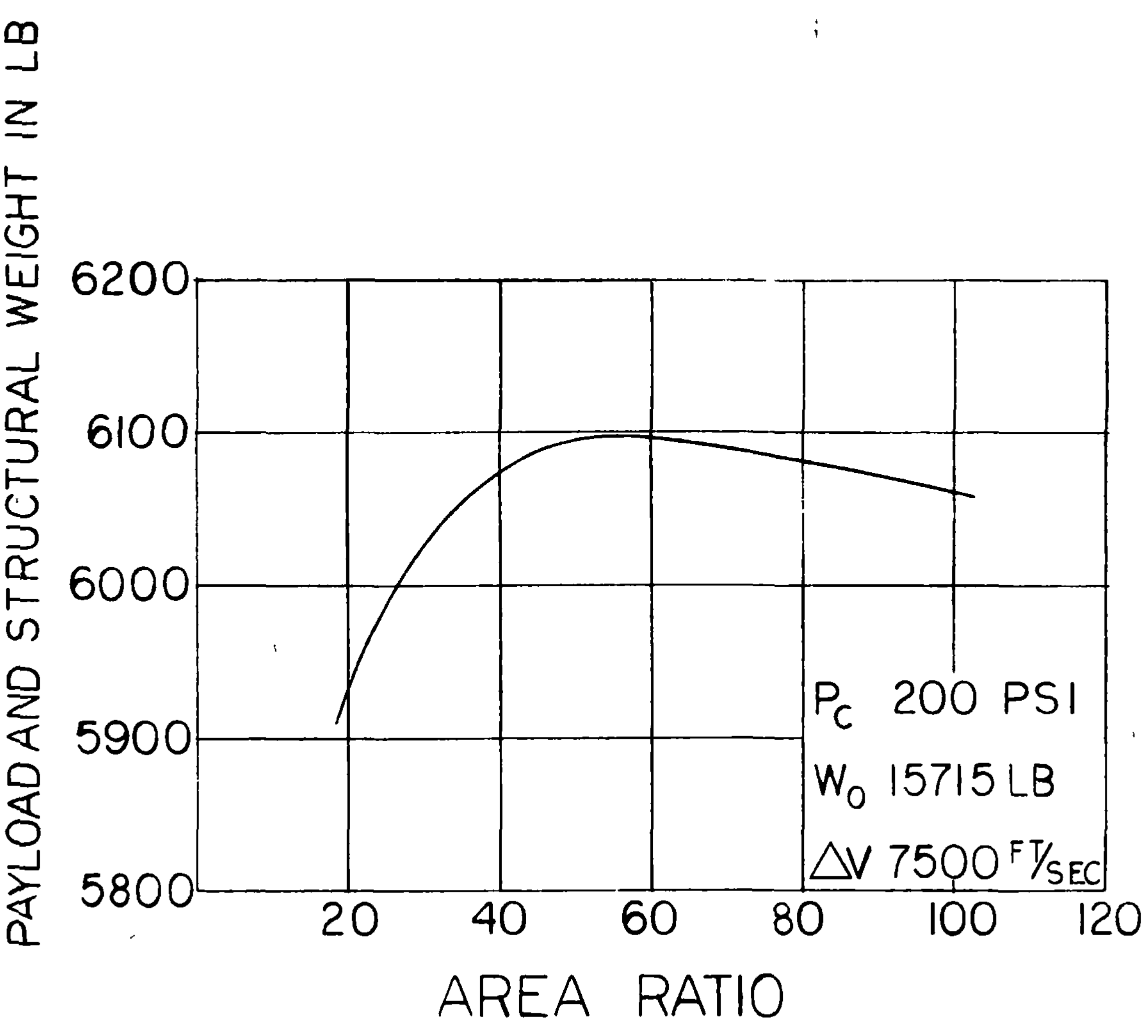


Figure 28. Area Ratio Trade-Off

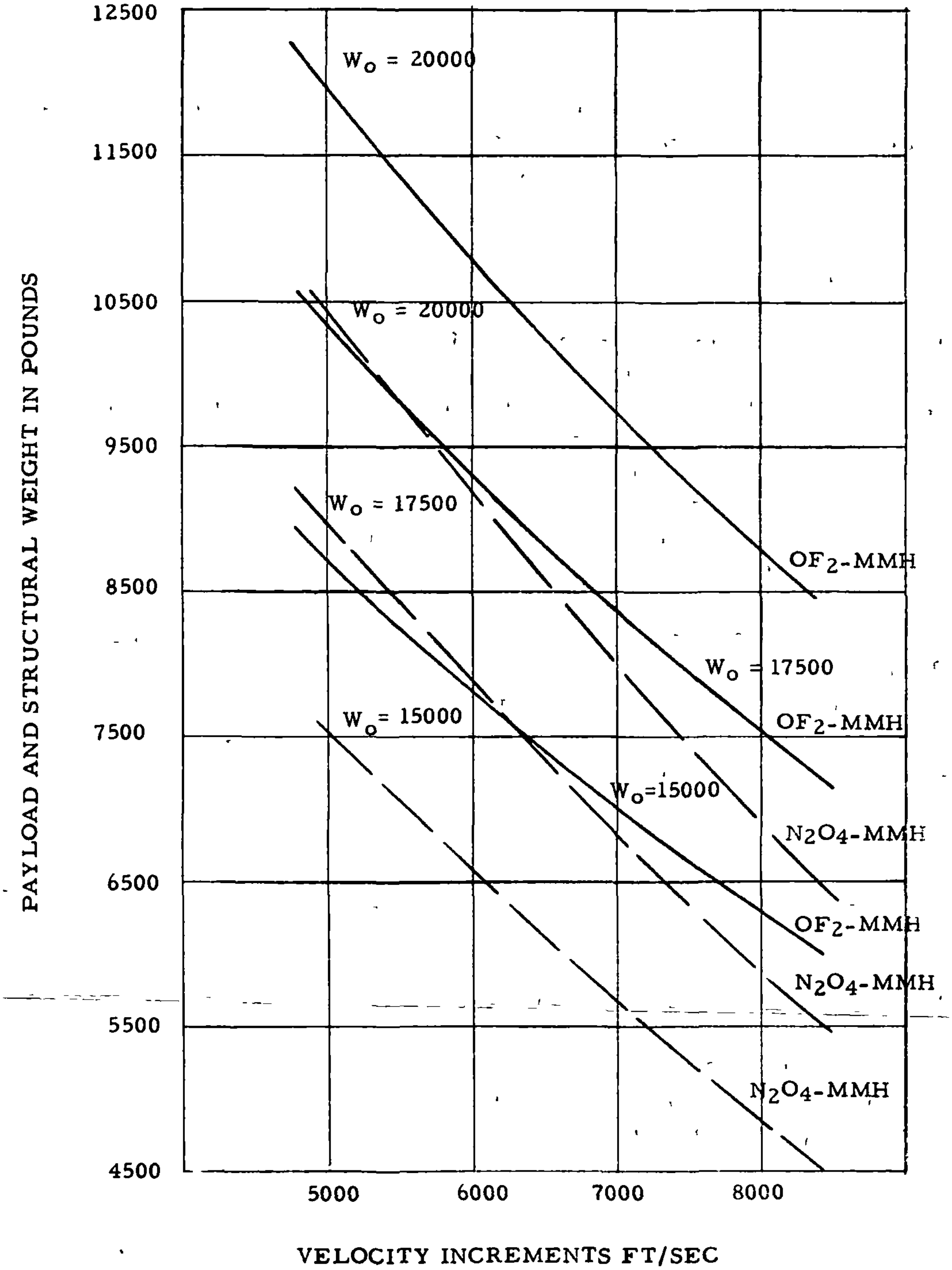


Figure 29. Propulsion System Performance

Certain of these features are unique to Thiokol and have been developed and brought to a high state of perfection by the efforts of our engineering people

1. Injector

The thrust chamber, shown in Figure 30, one of the four of the system, incorporates a liquid propellant injection concept which originated within Thiokol-RMD and upon which we have been working for the past five years. This injector is known as the Vortex type.

Vortex injectors for rocket engines have been extensively studied and tested at Thiokol-RMD under the sponsorship of the Corporation, NASA, the U.S. Air Force, and the U.S. Army. In the vortex system, one propellant is injected tangentially through the periphery of the combustion chamber, inducing vortex flow. The other propellant is injected radially and is entrained by the swirling flow while undergoing the physical and chemical processes leading to ignition and subsequent combustion. The vortex flow causes velocity, and the pressure gradient along the axis of the vortex gives rise to recirculation of the flow and induces turbulent mixing of incoming propellants and hot combustion products.

Complicated injectors are not necessary in vortex combustion chambers. The velocity gradients and hot gas recirculation serve most of the function of breaking up, vaporizing, and mixing propellants. This injector type has proven itself to be simple, reliable, and capable of developing high performance with the propellants proposed for this application. This type of injector has been operated at thrust levels up to 10,000 lb at Thiokol-RMD with performances above 97% of theoretical characteristic velocity

A summary of these performance data for vortex combustion chambers is given in Table VI. Figures 31 and 32 show the vortex injector and thrust chamber for the 1300 lb thrust unit The vortex injector and thrust chamber assembly for the 10,000 lb thrust unit are shown in Figures 33 and 34.

An interesting feature of this injector is that the heat transfer rates to the cooling jacket are less than those which could normally be expected under similar operating conditions with the conventional impinging stream injector

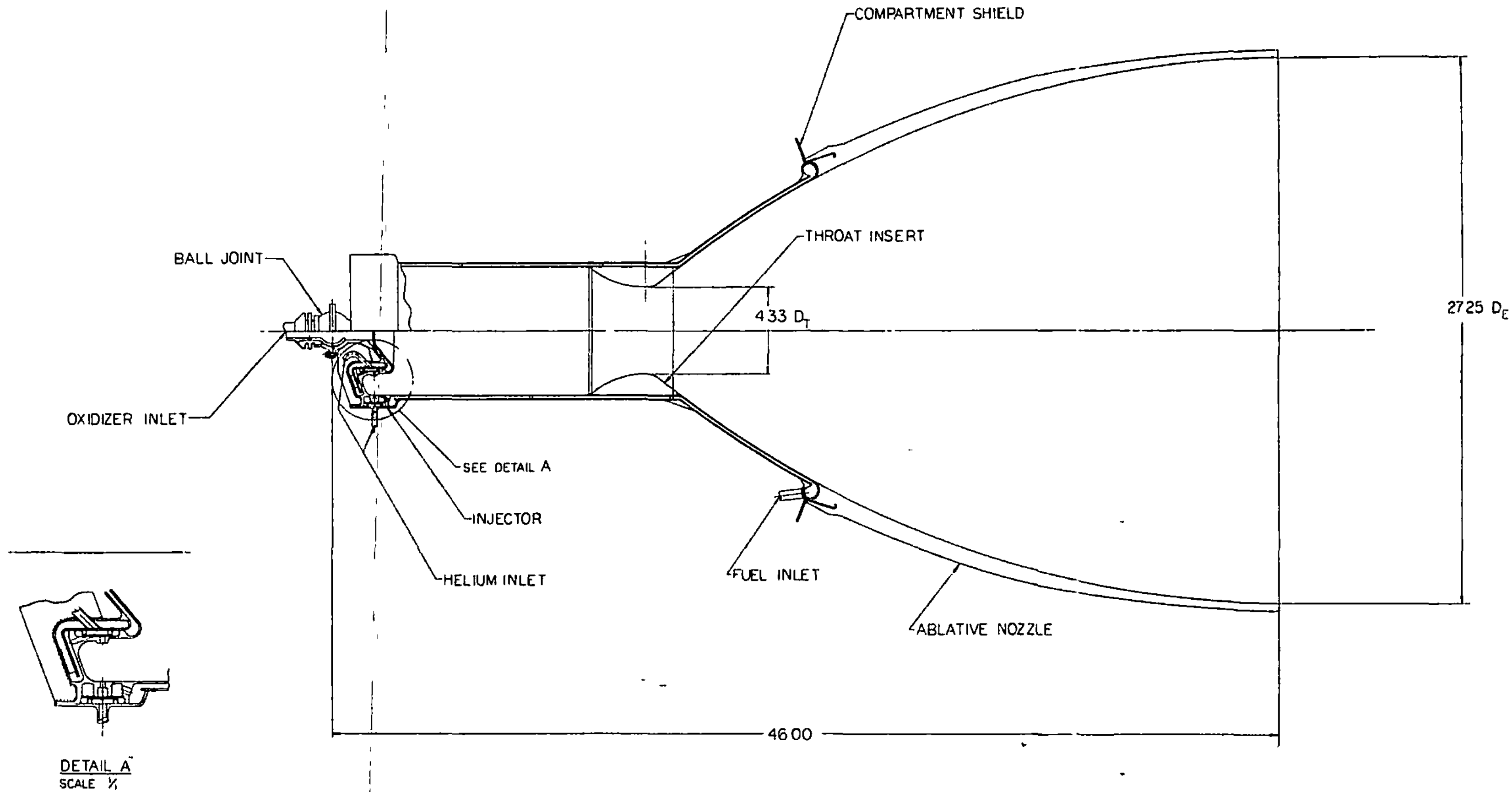


Figure 30. Thrust Chamber, 1963 Main On-Board Propulsion System (Dwg 52223)

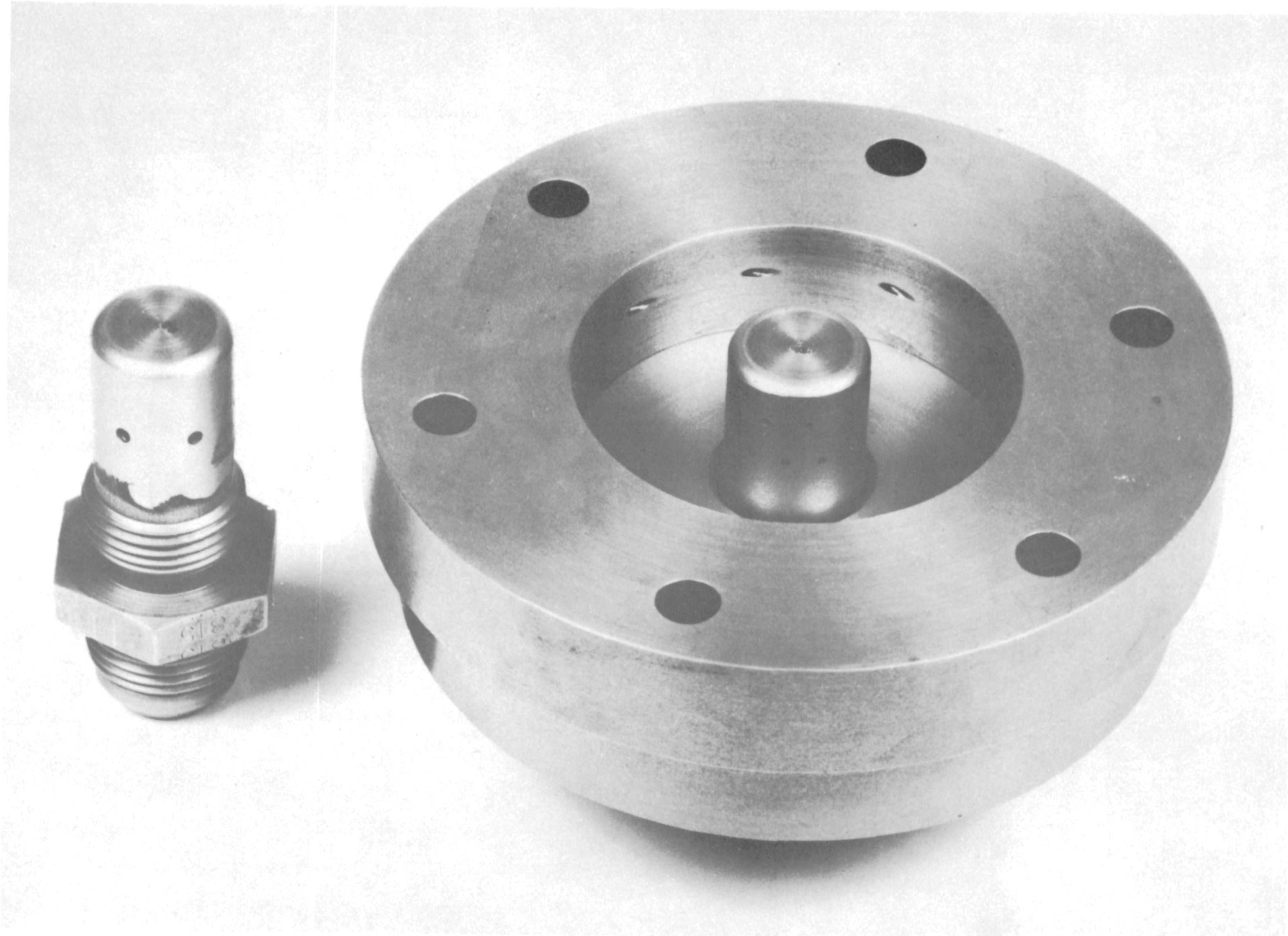


Figure 31. Vortex Injector, 1300 lb Thrust Unit (1165-31)

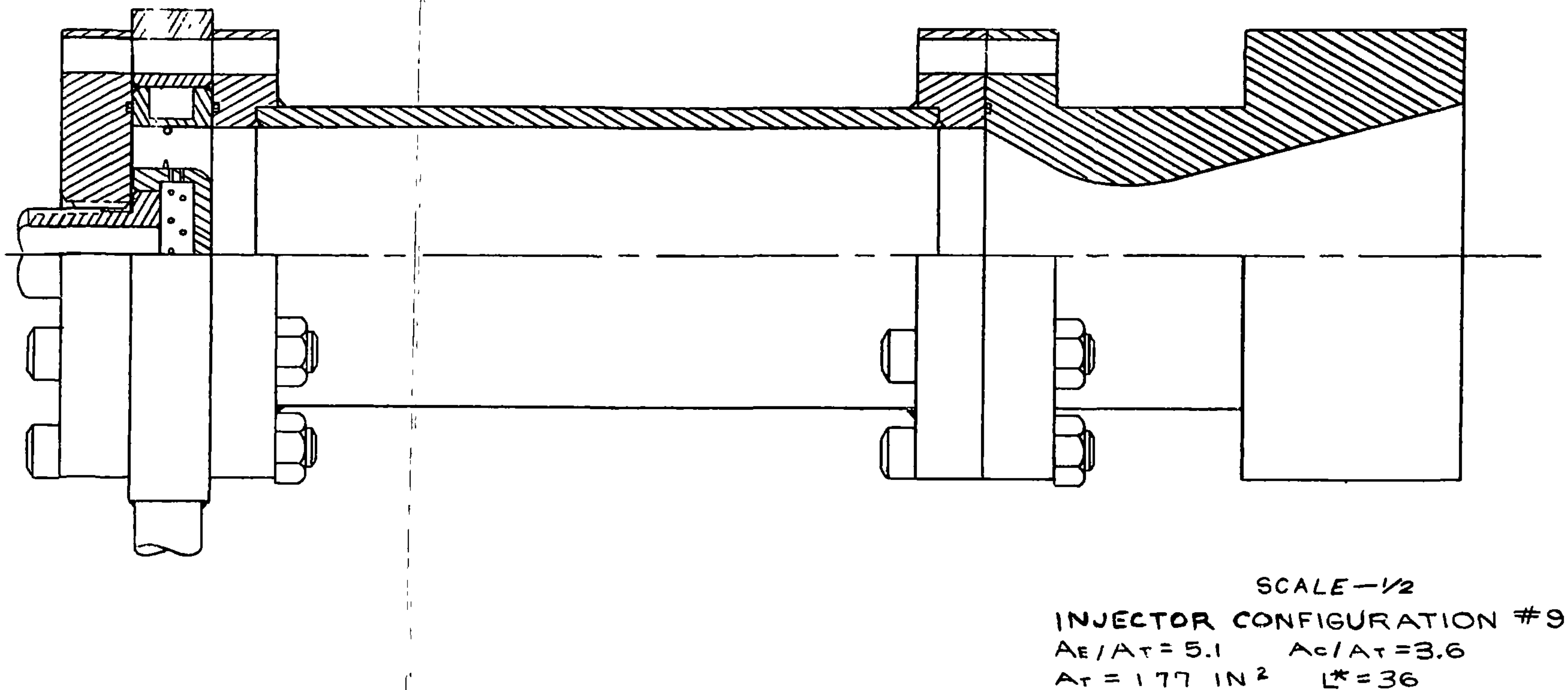


Figure 32. Liquid-Liquid 1300 lb Vortex Thrust Chamber (72-490)

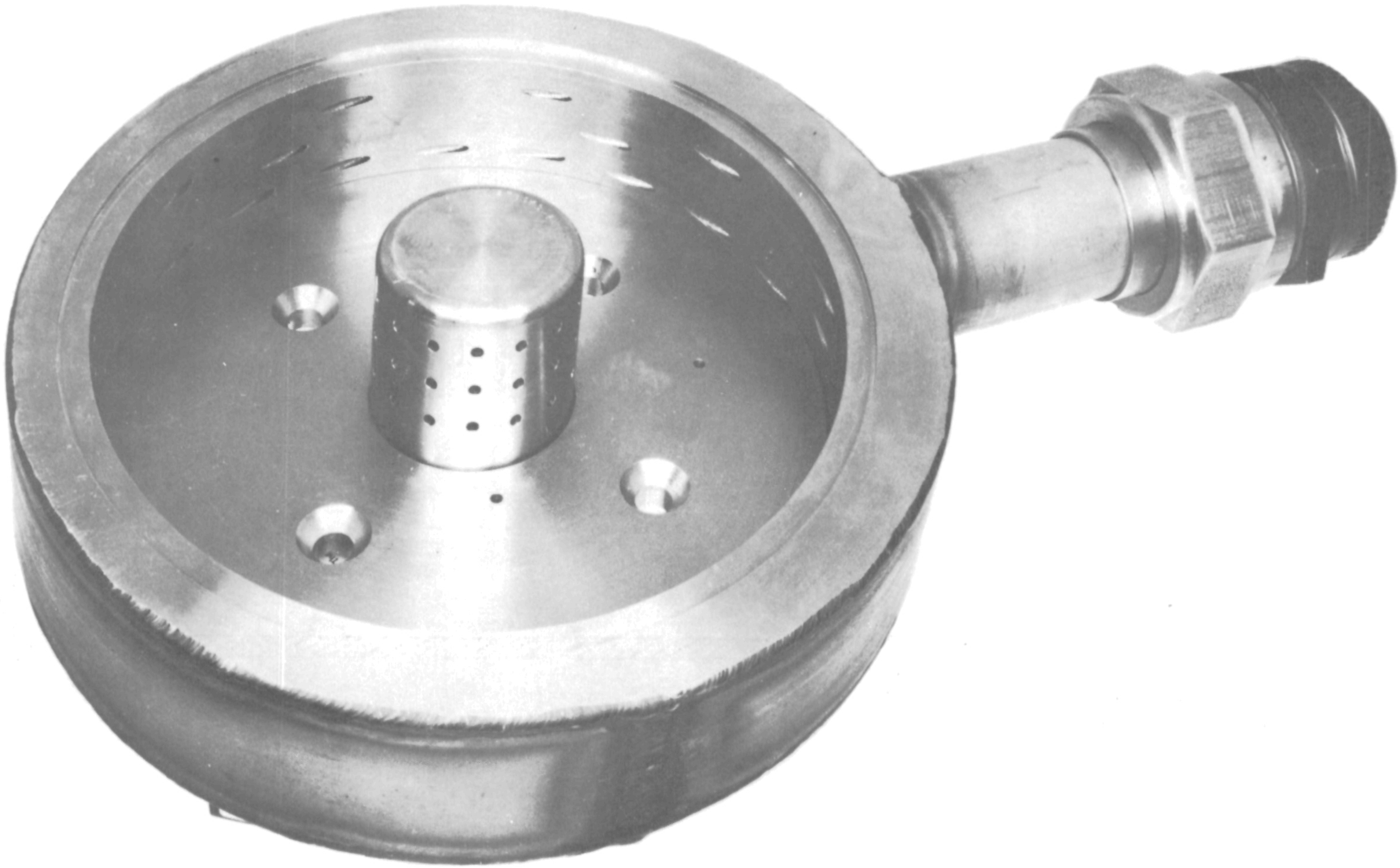


Figure 33. Vortex Injector, 10,000 lb Thrust Unit (2036-5)

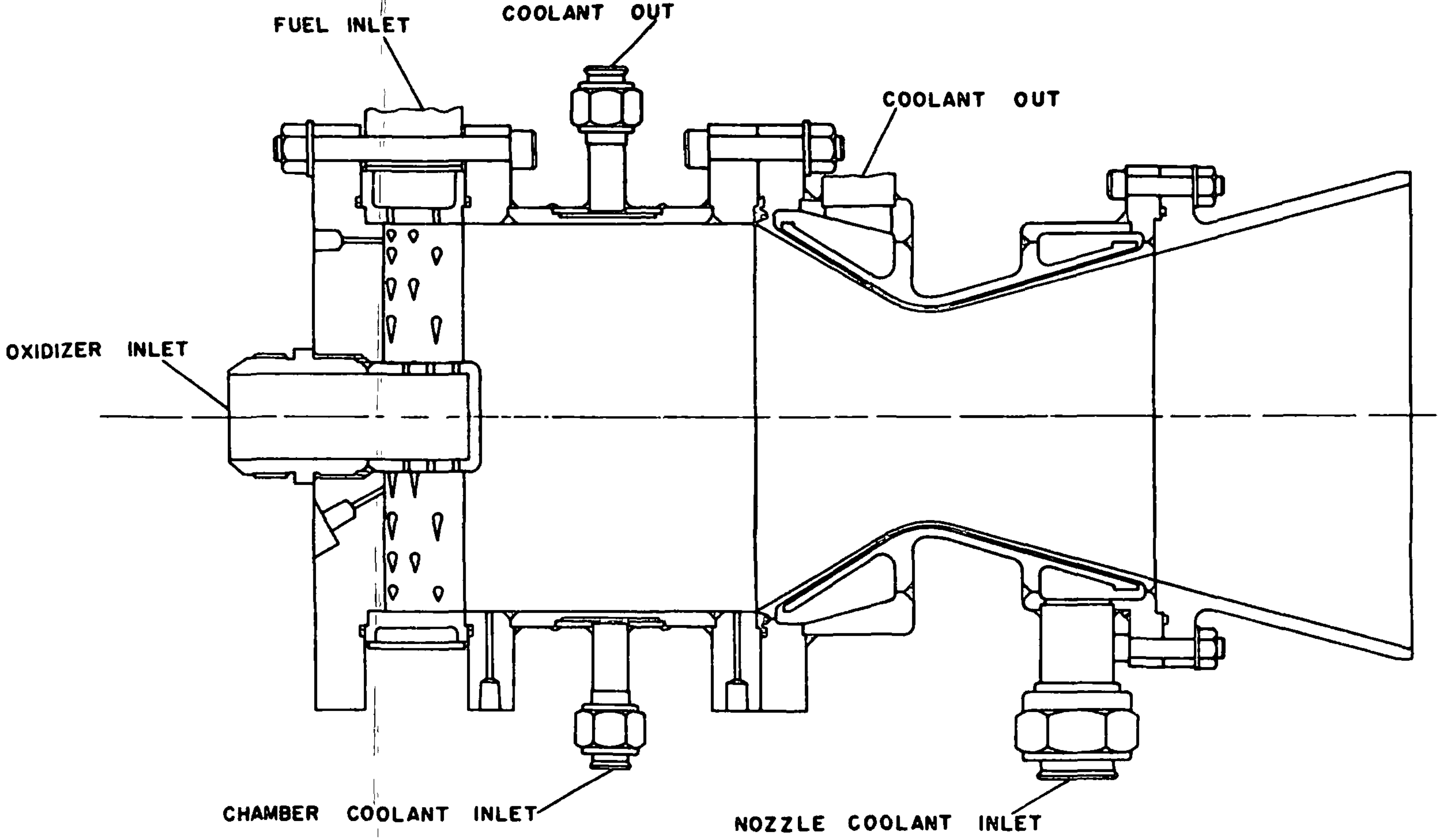


Figure 34. 10,000 lb Vortex Thrust Chamber, All-Liquid Injection (72-491)

TABLE VI

PERFORMANCE DATA SUMMARY
FOR VORTEX COMBUSTION CHAMBERS

Propellants Oxidizer	Fuel	Nom Thrust lb	Chamber Press psia	Exp C* ft/sec	Percent of Theo C*
N_2O_4	MMH	400	300	5400	96
N_2O_4	UDMH	1300	400	5560	98
N_2O_4	UDMH	10,000	400	5370	95

* Shifting Equilibrium

It is theoretically feasible for a vortex system to operate without provision for regenerative cooling, the Thiokol-RMD vortex igniters in the XLR99 engine is one example. Flame does not extend to the wall in vortex combustion since the wall is protected by a sheath of relatively cool, unburned propellants. Lowered heat transfer rates are always attractive in a rocket engine especially in a low pressure application with the problem of reduced saturation temperatures and the associated restrictions upon the ability of the coolant to absorb heat.

To insure safe reliable operation where numerable restarts are required, the vortex injector for this manned rocket application utilizes face shutoff valves. These valves terminate propellant injection immediately on the upstream entrance of the injection orifices of both the fuel and the oxidizer. This is illustrated in the enlarged sectional view in Figure 30. The face shutoff valves are basically simple flexible plastic boots which are forced against the inlets of the propellant injection orifices by helium gas pressure. This principle has been successfully employed in a commercial flow control (Flex-Flo) valve. Through close control of the injected propellant quantities, it is possible to use the concept of chemical safety to insure that destructive malfunction conditions do not exist. Chemical safety means that under no circumstances shall there be in the chamber a quantity of propellants greater than that which can safely be combusted in a deflagration process without destruction of the chamber. With the face shutoff valves this criterion can be adhered to quite easily.

Testing at Thiokol-RMD has established that the combustion of vortex injector engines is stable. Further, it has been shown both analytically and experimentally that longitudinal shock waves and detonation fronts are readily attenuated in a swirling gas with radial velocity gradients. This is significant especially where high reliability is required to insure safe performance. Vortex combustion has shown itself capable of operating reliably and stably over a relatively wide range of mixture ratios. Thiokol-RMD igniter experience with lox/NH_3, as well as extensive tests with hypergols, have supported this.

Recirculation of hot gases in the vortex combustion chamber promotes flame stabilization and complete burning of the fuel in a comparatively small volume. Theoretically, the available stay time in a vortex chamber is unusually long, even within a small physical volume, since the fuel can rotate in the vortex under controlled burning conditions for as long as necessary to complete combustion.

2. Combustion Chamber Design

The barrel of the thrust chamber, as well as a small portion of the nozzle expansion section, is cooled regeneratively with the fuel, monomethylhydrazine. Beyond an area ratio of 12.6 an uncooled expansion nozzle is used. The cylindrical barrel is constructed from two concentric shells between which a helical coolant passage is formed by wrapping a sheet metal strip about the inner shell. This practice is well developed at Thiokol-RMD and requires no additional development work.

Heat transfer analysis indicates that the 6000 lb thrust chamber can be cooled with monomethylhydrazine to an area ratio of 12.6, without excessive bulk rise or high coolant velocities, at the combustion chamber pressure of 200 psia.

The summary of the heat transfer for the cooled portion is shown in Table VII.

TABLE VII

HEAT TRANSFER SUMMARY

Station	Heat Flux Btu/sec-in^2	Bulk Temp (Inlet 120 F)	Wall Temp Twg ^{o}F	Velocity ft/sec	Cooling Safety Margin	Cooling Press Drop psi
Chamber	2.30	325	690	40	1.20	51
Throat	2.00	201	3300*	22	1.74	
Exit	.50	120	540	10	5.0	0

* Cooled Ceramic Insert

To obtain the required velocities, a helical passage is necessary. On the barrel, the height of the passage is 0 10 in. and the helix angle is varied to increase the velocity from 22 ft/sec to 40 ft/sec at the injector end. From the nozzle exit manifold to the throat section the velocity is increased from 10 ft/sec to 22 ft/sec. The passage height is kept constant at 0.10 in. The pressure drop for this helical passage is 51 psi.

A significant feature of this design is the use of a cooled ceramic throat insert in place of either a fully uncooled throat or a fully regenerative cooled throat. This design, which has been successfully employed in our engines for a number of years, utilizes a ceramic with high thermal conductivity, high melting point, and good thermal shock resistance. Data at elevated temperatures (3000-3500F) and reduced pressures (10^{-5} atm) indicate that decomposition of this material in space will not present a problem. By the use of this design approach in the throat it is possible to minimize any problems of throat burnout which often accompany initial chamber development.

High reliability can be anticipated since it has been demonstrated in our YLR48 engine program that ceramic throat life, both in time and cycles of operation, can exceed the requirements of this program. In the YLR48 engine program, which employed a much higher chamber pressure (600 psia) than is planned here, firings on the order of 15 times were common without nozzle deterioration. Through this approach a satisfactory design of the main on-board propulsion chamber can be assured with a high degree of reliability.

3. Nozzle

The nozzle extension consists of a plastic, ablatively cooled extension section which covers the area ratio range of 12.6 to 40:1. Trade-off studies (Part D) indicated that, in this installation, optimum payload would be obtained in the range of area ratio from 40 to 50:1. This extension is formed from a Thiokol developed material (Ablatalite) which was specifically engineered to meet the requirements for uncooled nozzle extensions and uncooled combustion chambers. Radiation cooling of the extensions was considered. However, for this application (considering the close proximity of the extensions to each other and to the surrounding structure) it was decided that the radiant heat interchange between the nozzles would cause excessive surface temperatures. In addition, the nozzles are shrouded by the outer structure and therefore cannot "see" space to permit efficient radiation of heat away from the hardware.

Ablatalite, in contrast to typical materials developed by non-rocket engine manufacturers, is designed and engineered to fulfill the requirements of a rocket engine. This material has features which make it very attractive for use in such an application. The use of ablation type materials in restartable rocket engine applications has often been avoided, partly because of the difficulty arising from the gross deterioration of the material after shutdown due to a buildup of a thick char layer. This char layer is removed upon the start of a subsequent firing cycle, thus requiring additional material allowances and imposing weight penalties on the design. The char layer results from the heat soak from molten ablative material developed during the prior firing. The heat content of this layer is transmitted by the high conductivity of the conventional plastic materials into the backup layers. This heat soak results in the gross charring after shutdown. Ablatalite has a very low thermal conductivity, approximately one half that of corresponding materials currently used in nozzle construction. The thermal conductivity of Ablatalite along with Durez, Haveg, and Taylor nylon phenolic is shown on Figure 35. Because of the low thermal conductivity of Ablatalite, the charring is insignificant compared to those of other represented materials. An Ablatalite liner tested in a thrust chamber at 500 psia chamber pressure at Thiokol-RMD lost less than 0.002 in./sec in 27 sec of operation in a thrust chamber employing a vortex injector with nitrogen tetroxide/hydrazine propellants. A thin adherent char layer approximately 0.030 inch thick was produced. In this test the char level of the Ablatalite was approximately 1/32 of an inch whereas with a quartz-phenolic or Refrasil phenolic, the char layer may build up to approximately 3/16 of an inch.

The results of additional tests to substantiate the selection of Ablatalite for the nozzle extension are shown in Figure 36 on which are plotted temperatures measured at the back surface of samples of Durez, Haveg, Taylor phenolics, and Ablatalite. These samples were 3/8 of an inch thick. An oxyacetylene torch was applied, providing a temperature of 5000 F at the opposite face. The back surface temperature as a function of time was measured and it is obvious that the Ablatalite heats up at a much lower rate than any of the other materials. This is significant in this particular application, not only from the aspect of the char buildup, but because the nozzle itself could radiate to the surrounding structure if it reached excessive temperatures. Any large amount of thermal radiation from the nozzle to the surrounding skin and structure will lead to deterioration of these components.

Tests to simulate space conditions were made with samples of Ablatalite and compared to results from similar tests with other plastics. In these tests the samples were subjected to high vacuum conditions (3.8×10^{-5} mm of Hg).

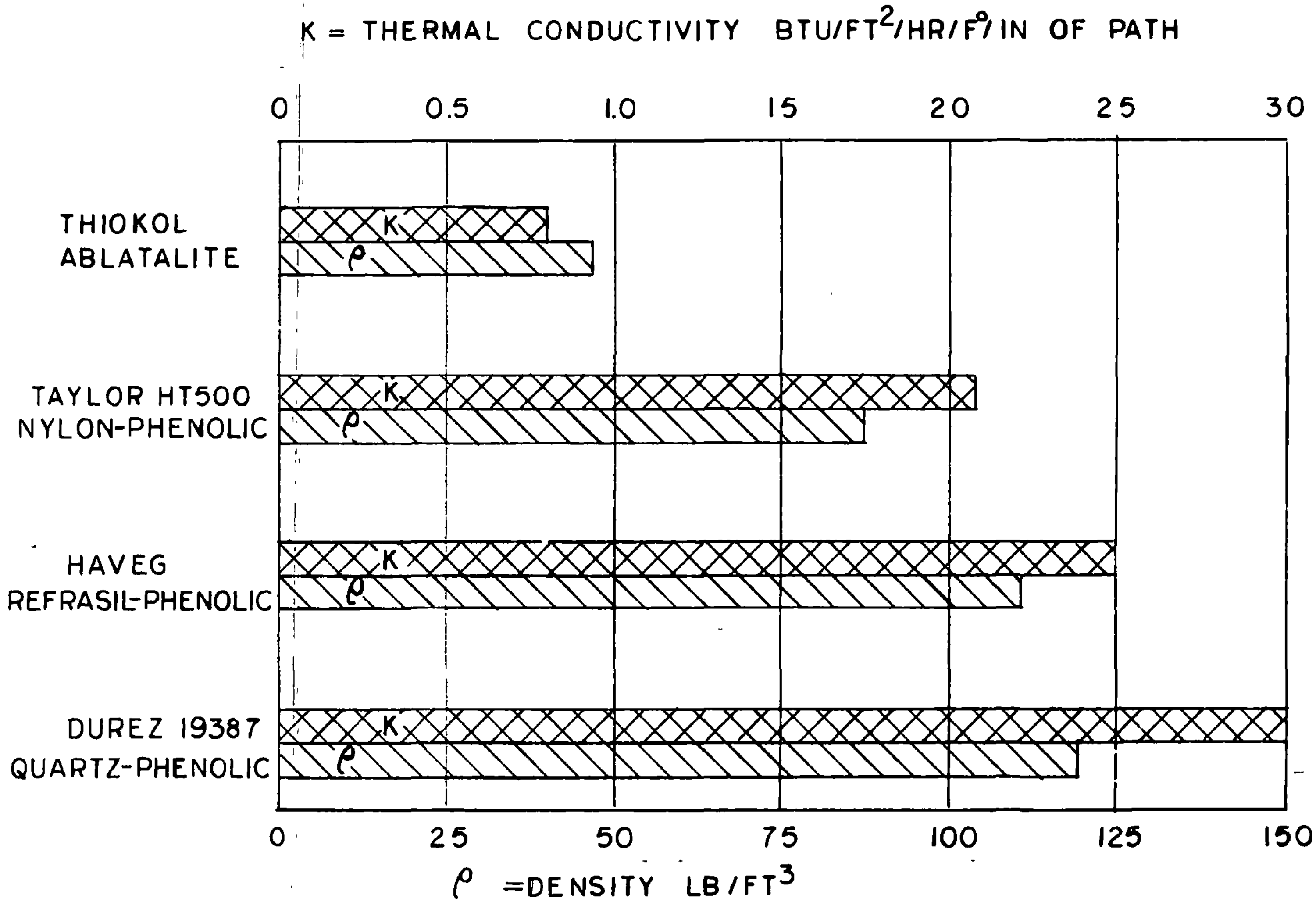
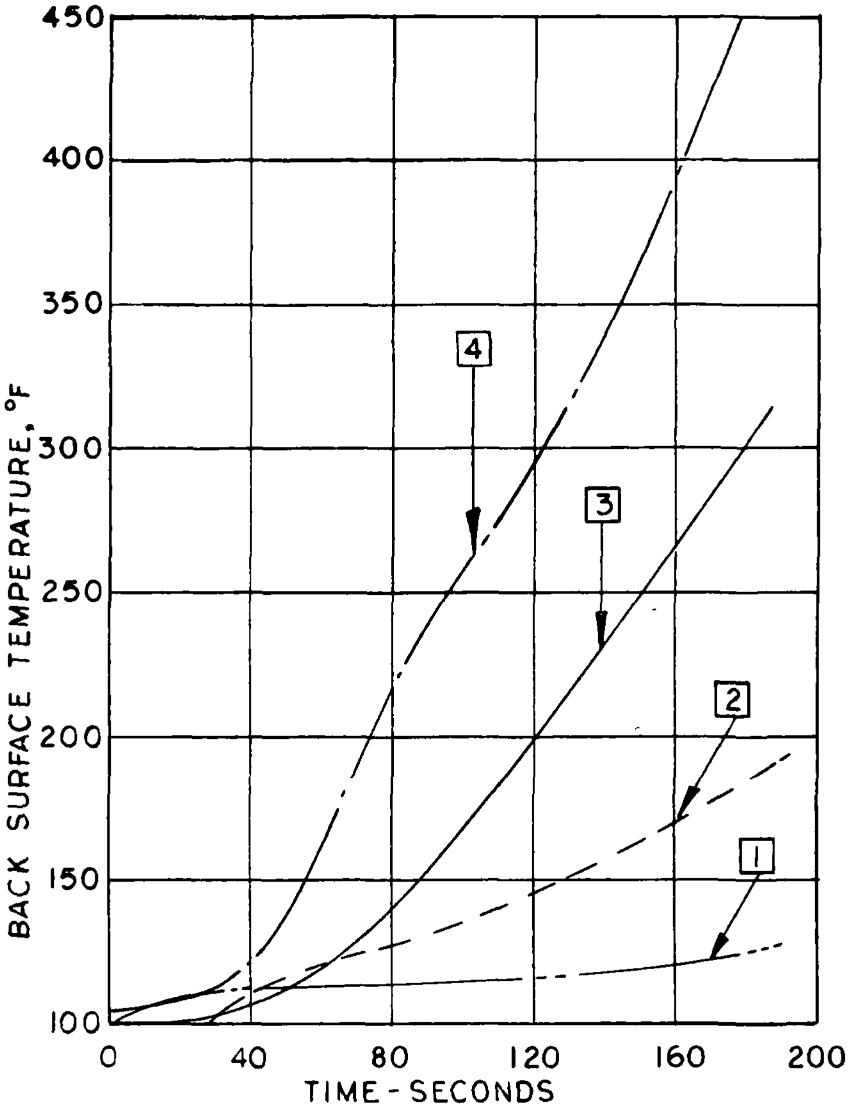


Figure 35. Thermal Conductivities of Ablatalite and other Materials



1 THIOKOL ABLATALITE, ρ = 43.7 LB/FT^3

2 TAYLOR HT500 (NYLON PHENOLIC) ρ = 87.5 LB/FT^3

3 HAVEG REFRASIL PHENOLIC ρ = 112 LB/FT^3

4 DUREZ 19387 (QUARTZ FILLED PHENOLIC) ρ = 118 LB/FT^3

BACK SURFACE TEMPERATURE

SAMPLE THICKNESS 3/8

HEAT SOURCE, OXY-ACETYLENE (5000°F)

Figure 36. Back Surface Temperatures of Ablatalite and other Materials

In comparison to the laminated plastics, the Ablatalite gassed-off at a lower rate. Tests were conducted with unheated samples and with samples heated on the surface to rocket operating conditions. (See Figures 37 and 38.) The surface of the Ablatalite after the hot tests was merely blackened, but did not crack.

A series of restart firings have been made on a small Ablatalite lined attitude-control size rocket. These tests were run at approximately stoichiometric mixture ratio with ethylene-air. Fifteen restarts were made. The run duration after initial checkout runs (5 to 24 sec) was 30 seconds each for 15 runs. The char layer was moderate and there was no apparent loss of thickness. The injector was a vortex type.

The design of the thrust chambers for the 1966 system also features vortex injection and a cooled ceramic throat insert. These chambers, of which there are two, are rated at 12,000 lb thrust each, at 200 psia chamber pressure. An expansion area ratio of 100:1 is used. In this installation, the requirement for gimbaling necessitates that the combustion chamber penetrate the present vehicle envelope when extended.

Therefore, the design of the vehicle structure in the vicinity of the propulsion unit must be coordinated with the engine design at an early date to achieve maximum efficiency of the overall unit. It is our recommendation that unnecessary structure in the vicinity of the powerplant nozzles be jettisoned soon after separation from the S-4 stage. This will enable the unimpaired radiation to space of the nozzle heat.

Further, the effect of jet flaring upon the structure will be eliminated as well as any possible contamination of the solar collector which might result from deposition of vaporized structural material upon the comparatively cool collector. Contamination of space in the vicinity of the vehicle due to high metallic ion concentration may also exist if the structure exposed to the jet is not jettisoned prior to firing. These factors are especially significant when one considers the fact that the combustion temperature of oxygen difluoride and monomethylhydrazine is quite high, 6100F.

A face shutoff vortex injector is again employed. In this design, however, the injector provides two-thrust level capability which enables operation of the chamber at either 6000 or 12,000 lb. The capability of the vortex injector to operate efficiently over a broad range is shown in Figure 39. A decrease in characteristic velocity of only 2% has been demonstrated

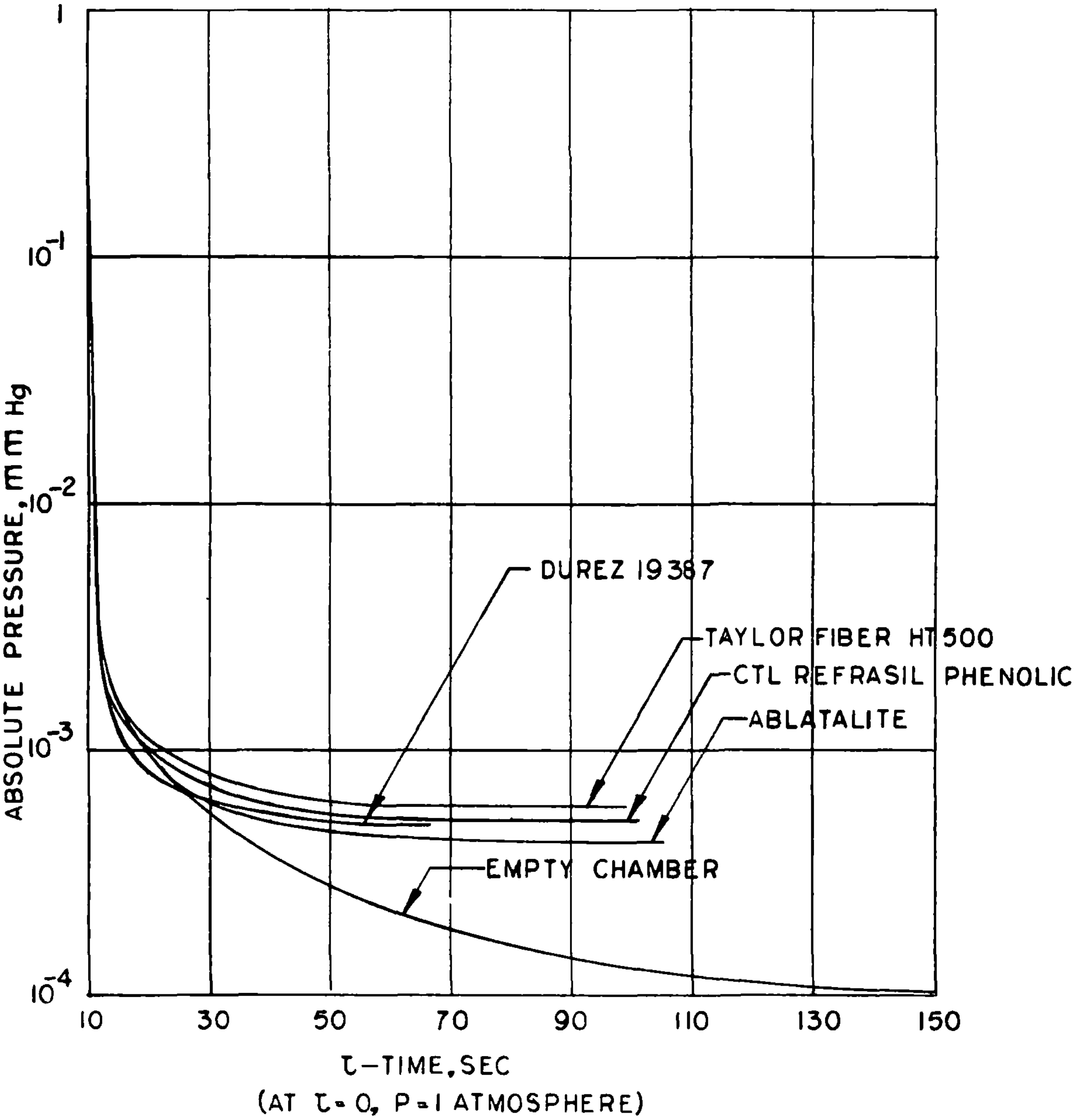


Figure 37. Rapid Decompression Tests, Unheated Samples

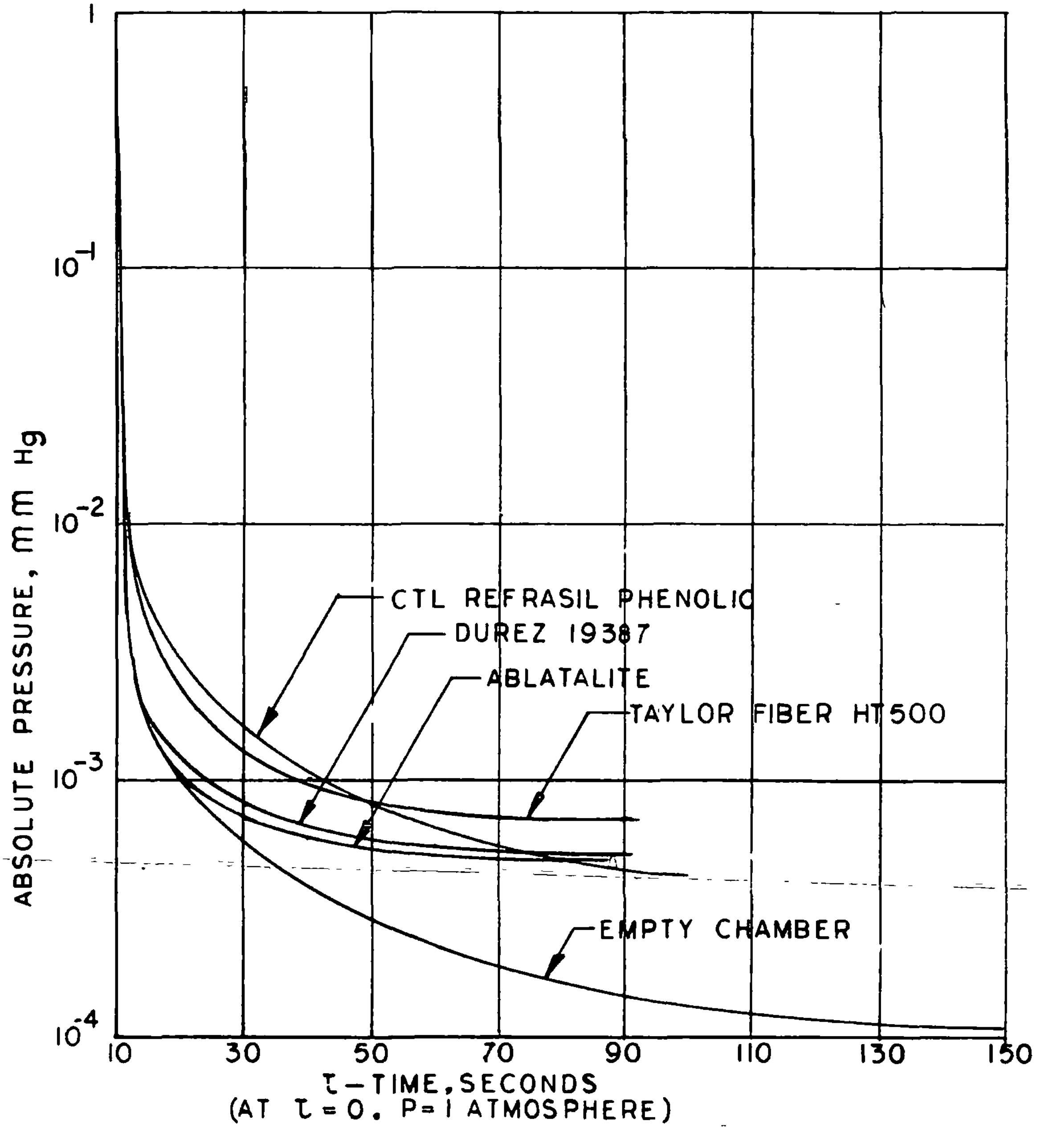


Figure 38. Rapid Decompression Tests, Heated Samples

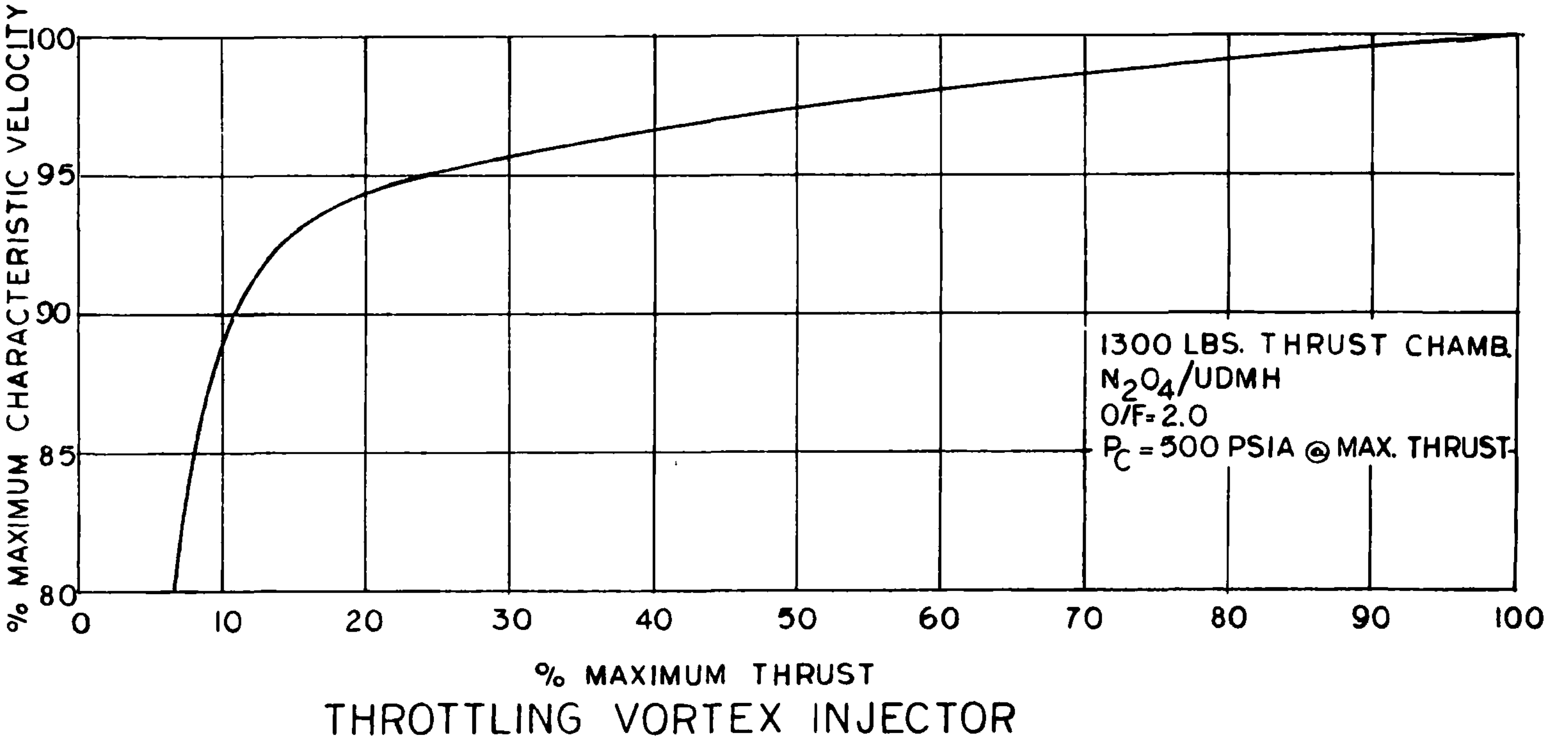


Figure 39. Throttling Vortex Injector

over the required 2/1 thrust range. The cylindrical portion of the chamber contains an ablative liner which in turn is protected by the introduction of a small quantity of helium as a film coolant. The injection of helium serves to locally dilute the combustion products and hence lower the temperature at the ablation surface. Helium has been selected as the film coolant since, because of its low molecular weight, its addition as a film coolant does not detract from performance. Actually, the performance is enhanced as can be seen from the table below. In addition, the availability of liquid helium from the pressurizing system makes it a logical choice.

EFFECT OF HELIUM ADDITION UPON PERFORMANCE (OF_2/MMH)

% Helium	Space Specific Impulse at O/F 2.5
0	425
2	428.5
5	431
10	429

The throat in this design is similar to that of the N_2O_4/MMH design in that it is a cooled ceramic. The high combustion temperature and the limited cooling ability of the fuel dictated that only the throat insert be regeneratively cooled. Dependent upon the severity of throat conditions, it may be necessary to supplement the regenerative cooling with a small amount of helium injection at the nozzle entrance.

The nozzle expansion section is radiation cooled. Molybdenum has been selected as the material since it is designed to operate at stabilized temperatures in excess of 2500F. The selection of this material is based upon our test experience with attitude control rockets and a recently completed design for a large molybdenum nozzle extension for the XLR99 engine for the X-15 aircraft. The experience gained in its fabrication will be directly applicable to this program.

Consideration of packaging and mission requirements for the 1966 Apollo led to the evolution of the unique advanced propulsion device concept shown in Figure 20. This basic propulsion device consists of eight conventional can-combustors located around the periphery of a conical-like, reverse flow supersonic expansion cone. Thrust is derived from the action that is basic to the well known plug nozzle. However, the basic plug nozzle concept using a full annular throat encounters design problems in fabricating and cooling a thin annular throat at large diameters. This problem is aggravated in high area ratio (100:1 in this case) applications since the circumference of the annulus increases with area ratio and the height must then decrease (since throat area is constant). The localized combustion gas injection approach to the high area ratio plug type nozzle becomes mandatory. In addition, the cost and time of development of individual combustors is obviously reduced, which represents appreciable savings in the development of relatively large propulsion devices.

The plug nozzle concept is generally characterized by some internal supersonic expansion in an annular passage from Mach 1.0 to some intermediate Mach number followed by an external expansion to the design Mach number which is developed on the plug surface. This latter external expansion is derived directly from flow direction change. In diameter limited applications, considerations of the final flow direction change available dictates the amount of internal supersonic expansion required. Consideration of the thrust requirements and available envelope for the Apollo vehicle revealed the feasibility in this case of employing full external expansion (no internal expansion).

Consideration was given to using a radial outflow nozzle of the same diameter as the reverse flow design. An estimation of weights for conical surfaces showed the radial outflow nozzle to be heavier than the reverse flow nozzle at the same performance level. This was due to an increase in surface area for the outflow nozzle over the reverse flow nozzle. Figure 40 shows net decrease of payload versus area ratio for the radial outflow nozzle as compared to the reverse flow nozzle (zero line) and both at the same equivalent thickness. The radial outflow nozzle offers certain packaging advantages which may permit decreases in supporting structural weight and thus provide a net advantage. This concept requires further analysis.

The design requirements for the propulsion system shown in Figure 20 were selected as follows:

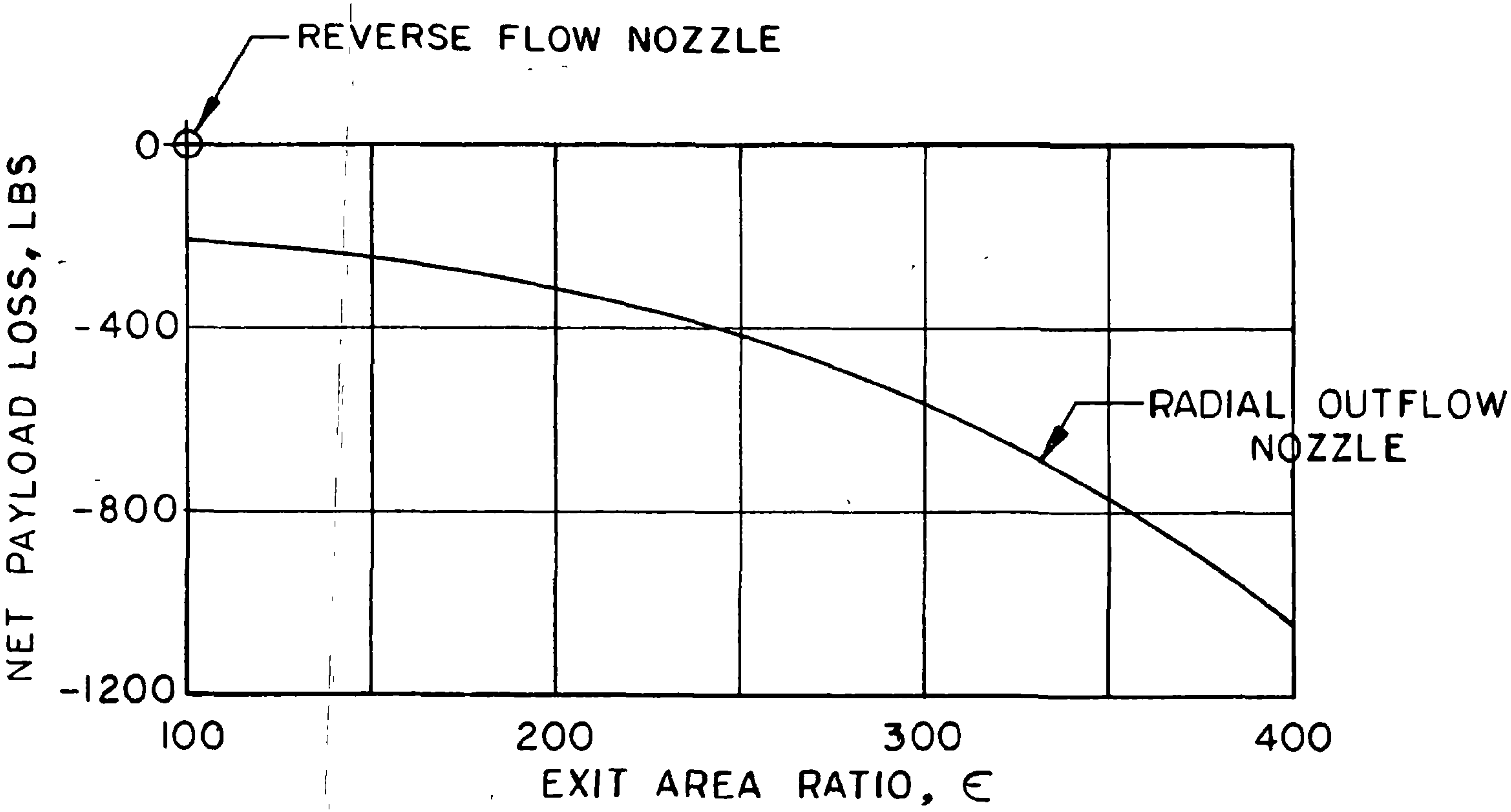


Figure 40. Nozzle Trade-Off Comparison

Thrust - 24,000 lb (for emergency abort)
6000 lb (for lunar orbit)

Chamber pressure - 200 psia

Expansion Ratio (E) - 100:1

Propellants - OF_2/MMH

O/F - 2.5

γ - 1.10

These requirements dictated a flow direction change of 114 degrees for full external expansion from Mach 1.0 to the design Mach number (E = 100:1). This angle is the basis for orientation of the individual combustors. The combustors are inclined inward and forward at 66 degrees to the vehicle axis. The final exit flow direction after turning is co-axial with this axis providing axial thrust. It has been experimentally demonstrated that both localized combustion gas injection in the plug type nozzle and full external expansion with turning angles greater than 90 degrees is feasible. However, available data are limited to nozzles with relatively narrow spacing between individual throats.

Published information also exists which demonstrates the feasibility of obtaining thrust vector orientation (relative to the physical nozzle axis) by throttling individual combustors located around the periphery of a plug nozzle. This approach was chosen since it was readily adaptable to the necessary individual combustor requirement. Table VIII shows the throttling requirements and other considerations which led to the choice of eight combustors for the Apollo application. System studies are required to determine the trade-off between the number of combustors, redundancy, control requirements, and system weights to optimize the design.

Table VIII shows the combustor thrust rating as a function of the number of combustors. The breakdown of multiples of four combustors was chosen on the basis that suitable throttling of four equally spaced (circumferentially) combustors could provide thrust vector orientation in any desired longitudinal plane. The table shows the operating chamber pressure required for a nominal thrust of 6000 lb with all chambers rated at 200 psia. Available data show that for an effective thrust vector angle of 5 degrees, a chamber pressure

TABLE VIII

SUMMARY OF EFFECT OF NUMBER OF COMBUSTORS ON SYSTEM REQUIREMENTS AND CHARACTERISTICS

Number of Combustors	4	8	8	16	16	16
Rated Chamber Pressure, psia	200	200	200	200	200	200
Combustor Thrust Rating	6000	3000	3000	1500	1500	1500
Number of Combustors Firing for Thrust Vector Orientation	4	4 (4 off)	8	4 (12 off)	8 (8 off)	16
Required Chamber Pressure for 6000 lb Nominal Thrust, psia	50	100	50	200	100	50
Maximum and Minimum Chamber Pressure Limits for Thrust Vector Orientation While Maintaining 6000 lb Nominal Thrust, psia	75 25 (12.5%) (Rated)	150 50 (25%) (Rated)	75 25 (12.5%) (Rated)	300 100 (50%) (Rated)	150 50 (25%) (Rated)	75 25 (12.5%) (Rated)
Redundant Systems Available for Nominal Thrust Requirements	None	One	None	Three	One	None

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Note: Thrust Requirements - 24,000 lb (Max Rating) Emergency Abort
6000 lb Nominal

variation (increase on one side and decrease on the opposite side) of about ± 50% would be required. This criterion was applied and a set of maximum and minimum chamber pressures required during nominal operation was developed as shown. Those systems shown in Table VIII that provided no system redundancy were immediately eliminated from further consideration from a reliability standpoint. Since the pressurization system would only allow a maximum chamber pressure of 200 psia, a 16-combustor design with 12 combustors off during nominal operation was impossible due to a 300 psia chamber pressure requirement for thrust vector orientation. Also, since either an 8- or 16-combustor design with both systems using one-half of the combustors during nominal operation had the same throttling requirements and provided the same redundancy characteristic, the simpler (less plumbing and valving) 8-combustor design was chosen.

The basic concept of the plug nozzle involves the employment of an isentropic plug contour. In this case, uniform and axial exhaust flow is developed. These isentropic plugs are quite long. The usual approach to this problem has been to discontinue the isentropic contour when it is at some suitable angle to the axis of the nozzle and substitute a simple conical surface of that half-angle. This results in a shorter and lighter plug nozzle. The resulting exhaust flow pattern distortion results in a nozzle performance loss which a limited amount of published data indicates is about 3% for a 40 degree half-angle cone. This approach was felt to be quite justified in the advanced Apollo vehicle design especially in view of the fact that an extremely long isentropic plug would be required for a 100:1 area ratio nozzle.

By employing individual throats, with wide spacing between throats, flow turning and expansion can not only occur in the axial plane but circumferentially also. Streamlines from two adjacent combustors may conceivably be directed at each other initially. However, they must turn and flow axially in a mixing zone formed by the two individual expanding jets. The effect on performance with wide spacing between throats is not known at this time. However, it is reasonable to assume that any losses incurred could be minimized by suitable special contouring approaches in the expansion cone regions near the combustors. Such contouring requirements are undefinable at present.

4. Space Ignition

Both altitude and temperature effects on hypergolic ignition have been demonstrated in the laboratory. The result of these effects is that the ignition delay increases with decreasing ambient pressure and propellant temperature.

In some cases, a pressure has been demonstrated below which hypergolic ignition does not occur. It is reasonable to assume that, for any hypergolic propellant system, a pressure-temperature boundary could be established beyond which ignition will not occur.

Thiokol-RMD has conducted tests in the laboratory to determine an ignition parameter for low pressure ignition of non-hypergolic propellants. The tests were made with propane-air and propane-O_2 propellants. Ignition was attained at pressures as low as 1 mm Hg abs.

The experiments have been conducted in a large bell jar in which pressures of a few microns Hg abs are attainable. Inside the bell jar is a sealed igniter with a volume approximately 1/300 of the jar. This is secured to a platform mounted on the base plate. In all runs to date, an approximately stoichiometric mixture of propane and oxygen has been loaded into the igniter. This loading has been achieved by the method of partial pressures. The total pressure in the igniter has been varied from one atmosphere to one quarter of an atmosphere. Upon ignition of the igniter mixture by a spark, the sealing diaphragm bursts and a flame jet enters the bell jar.

The bell jar contains the propellant combination to be studied at the desired test pressure. To date, approximately stoichiometric propane-air and propane-oxygen mixtures have been used.

Visual observations are made to determine whether the outer mixture ignites as a result of the flame jetting from the igniter orifice hole. This method has been adequate for most of these initial tests. However, under some operating conditions, i.e., at initial igniter pressures of one-half atmosphere rather than one-fourth atmosphere, the visual method is less certain. At the higher igniter pressure, the mass of fuel in the bell jar is only approximately twice that in the igniter for a typical run with propane-air as the test mixture. As a consequence, the intensity of the jetting flame tends to overshadow any flame due to combustion of the test mixture. However, the latter flame seems to linger for a brief instant after the former has expended itself. The brevity of this time differential for such runs renders results of the visual observations somewhat uncertain.

At initial igniter pressures of one quarter atmosphere, the visible jetting flame confines itself to a vertical path in the center of the bell jar unless ignition and combustion of the test mixture occurs. In this case, the entire volume of the bell jar is briefly lighted and the flame due to the test mixture lingers significantly longer than the jetting flame. Visual observations in such cases are quite unquestionable.

Critical pressures were obtained for propane-air and propane-oxygen mixtures for certain conditions in the test apparatus. Critical pressure is defined as that pressure below which ignition of the test mixture under the test conditions does not occur. Two significant facts have been obtained in the limited program to date. The first is that the diameter of the igniter flame jet has a marked effect on the critical pressure. Increasing the jet diameter decreases significantly the critical pressure. The second is that the critical pressure appears independent, or very nearly so, of the initial igniter pressure and therefore of the mass flow in the jet (see Figure 41).

From these tests, it can be concluded that the igniter hole diameter may be considered a dominant ignition parameter.

However, a literature search of the performance of actual rocket engine systems at very high simulated altitudes discloses the absence of any effects on hypergolic ignition delay as a result of decreased pressure or (within the areas studied) temperature. The conclusion reached by agencies who have conducted tests is that there is no discernible increase in ignition delay between sea level and high altitude conditions for rocket engine systems operating with conventional design chamber pressures. Figure 42 illustrates the range of test conditions covered. Figure 43 shows the earth's atmospheric pressure variation with altitude.

The apparent difference between test results obtained with laboratory type ignition delay measuring equipment and those obtained from rocket engines can be explained by the effect of the combustion chamber volume and throat in concentrating the propellants and increasing the local pressure within the combustion chamber. In the typical laboratory ignition delay measurement, the reactants are introduced into a relatively large evacuated container and diffuse away from the area of impingement or mixing. The density of the mixture of reactants remains low and as the concentration of reactants decreases, the reaction rate decreases. As a result, hypergolic systems take longer and longer to accumulate the heat required to initiate combustion.

In a thrust chamber, however, the typical hypergolic, storable, volatile oxidizer such as N_2O_4 or OF_2 rapidly vaporizes and pressurizes the combustion chamber, increasing the concentration of its vapors to the point where ignition occurs rapidly and smoothly when fuel is injected.

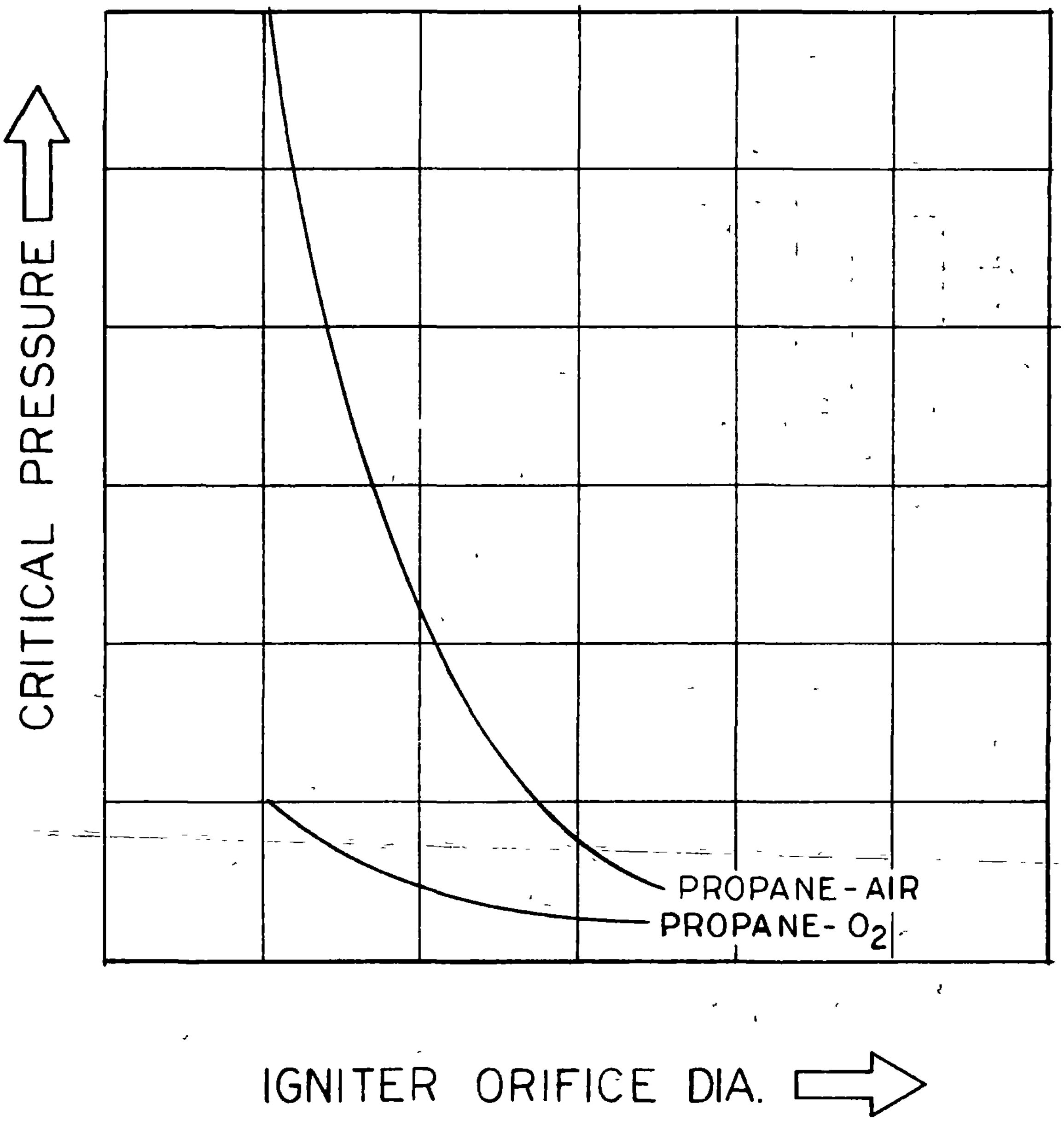

Figure 41. Low Pressure Ignition

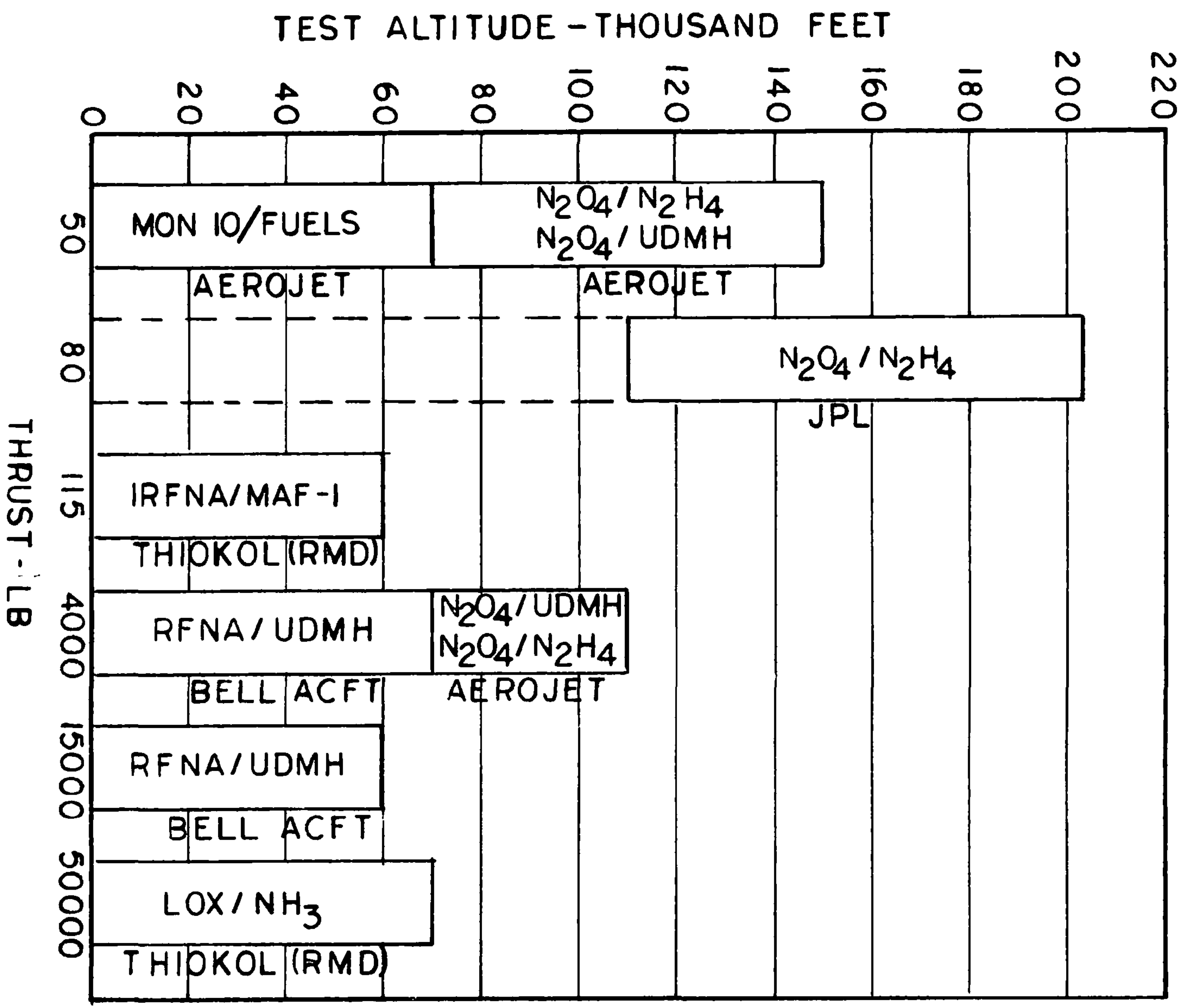

Figure 42. Range of Conditions Tested without Ignition Delay

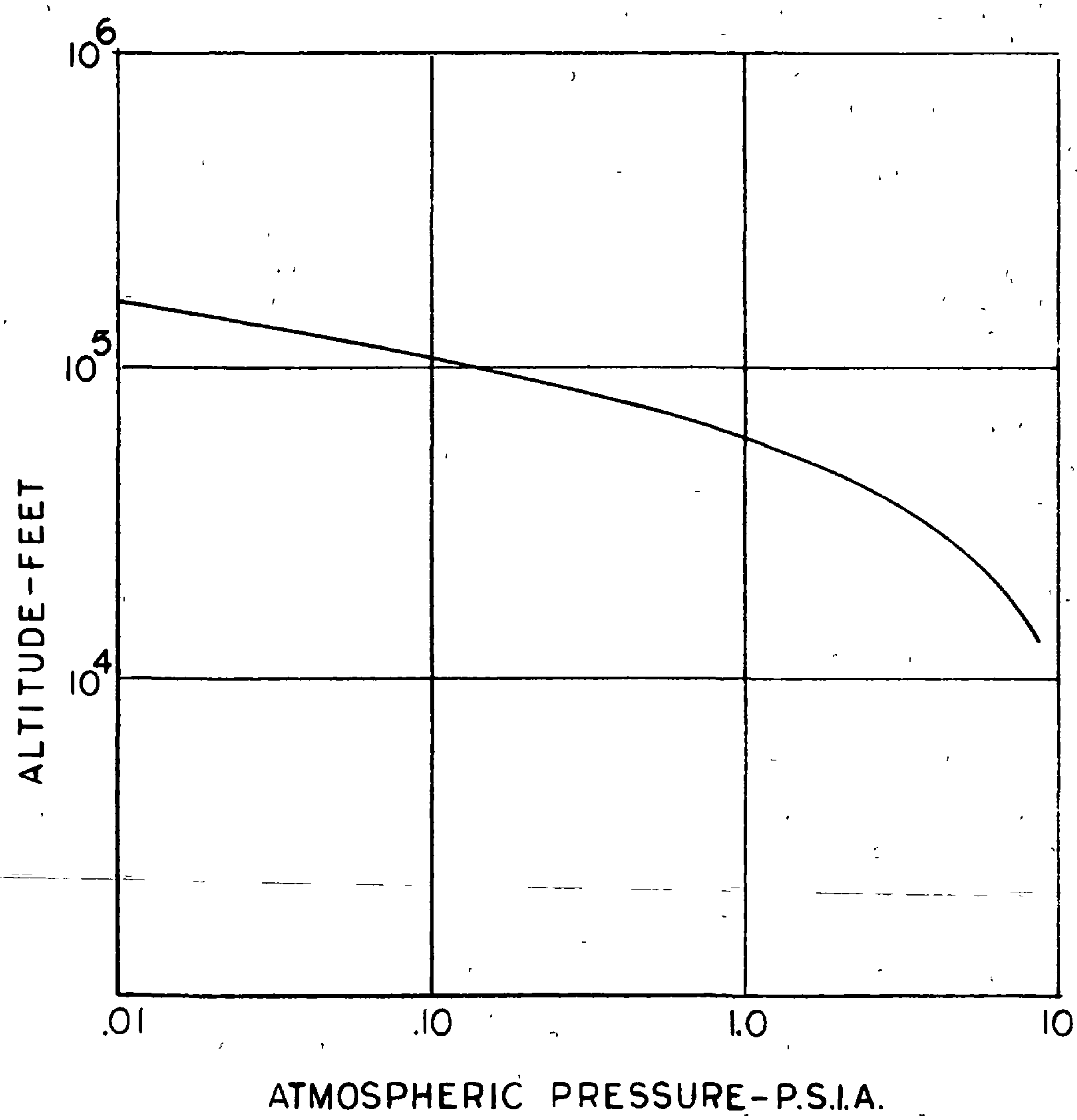

Figure 43. Atmospheric Pressure vs Altitude

The Reaction Motors Division of Thiokol Chemical Corporation has recently completed three programs investigating high altitude ignition. One involved testing a small 115 lb thrust drone engine using hypergolic propellants. Starting tests were completed at a simulated altitude of 60,000 ft. The second program involved ignition and operating tests with the XLR99 turborocket engine for the X-15 aircraft. These tests were made at various simulated altitudes up to 70,000 ft and were conducted at the AEDC facilities at Tullahoma, Tennessee. The third program was the testing of micro-pulse rocket engines and successful ignition was attained at altitudes up to 104,000 ft. The 104,000 ft is the maximum altitude attainable with the test equipment and not the limit of ignition for the hypergolic propellants used in the tests.

Aerojet has found (Aerojet-General Corporation Report No. 0106-01-9a) that for an ultra low pressure engine with a pressure feed system, a certain concentration (or pressure) of N_2O_4 is required in the thrust chamber to insure ignition. At pressures below this value (0.6 psia), ignition is marginal. At N_2O_4 chamber pressures just above this value, ignition is reliable and smooth; however, the ignition delays are very long when compared to sea level data (200 to 400 msec as compared to 10 msec). The large chamber volume and throat diameter (relative to the oxidizer flowrate) can reduce the concentration of reactants below that required for reliable ignition

Ignition may be accomplished at very high altitudes by pressurizing the thrust chamber to near sea level conditions with an inert gas bleed prior to the ignition transient. This system has been studied for flight use, and the amount of gas required for large engines is surprisingly low*. In a low pressure system, the hardware is correspondingly light in weight.

It is apparent then that low pressure and low temperature hypergolic ignition is dependent upon injection pattern and propellant flowrates. In the case of very low chamber pressures or flowrates, these conditions may very well determine whether or not there is ignition.

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* Symposium Proceedings, 1-2 December 1959, Simulated Altitude Testing of Rockets and Missile Components, Arnold Engineering Development Center, Air Research and Development Command, USAF: "Problems Requiring Tests of Liquid Propellant Rocket Engines at Simulated Altitudes", Lee F. Webster, Engine Test Facility. "Experimental and Theoretical Investigations into Ignition of Hypergolic Propellant Rockets at High Altitudes", Boorady and Rossman, Bell Aircraft Corporation. "Design Concepts and Simulated Altitude Testing of the X-15 Aircraft Rocket Engine", Gwozdz and Sjoberg, Reaction Motors Division, Thiokol Chemical Corporation.

F. DESCRIPTION OF COMPONENTS

1. Shutoff Valves

There are many different types of shutoff valves that can be used, such as solenoid, pneumatic, hydraulic, and squib. At the present time it is planned to use squib valves in the pressurizing system. Typical of this type is the helium bottle fill and pressurizing valve, which will be incorporated in one body to match a successful design that we have used in the past. This valve is currently designed to control the fill and release of nitrogen pressurization gas in a 6000 psi system. Procurement is covered by Thiokol-RMD Specification 20232. Figure 44 shows the installation and specification control drawing for this valve. For the application in question, consideration will be given to eliminating the gage shown on the figure. High vacuum and radiation tests will be conducted to insure that the O-ring seals and epoxy squib filler currently employed will not deteriorate in these environments. This valve has fired 28 times without malfunction during a current missile flight evaluation program.

2. Regulators

These components are usually tailored to fit the particular mission in question. The major considerations in design are the range of flow capacity required, allowable droop, and range of inlet to outlet pressure ratio. These major considerations can, at times, be handled by series and/or parallel arrangements. These components can be made versatile by providing for the ability to adjust downstream pressure. Also, if made in two stages, the second stage could be two units that would automatically compensate for temperature variations in a bipropellant rocket system, thus keeping O/F ratio within acceptable limits.

The type regulator planned for the re-entry roll control system is shown in Figure 45. This regulator is currently used as the top stage in a two stage regulation system reducing N_2 pressure from a nominal 6000 psi maximum to approximately 200 psi. Procurement is currently covered by Thiokol-RMD Specification 20211. High vacuum and radiation tests will be required to insure that O-ring seals will not deteriorate. The types of regulators under consideration for use in the main propulsion system are shown on Figure 46. These latter three units were all designed and developed at Thiokol-RMD.

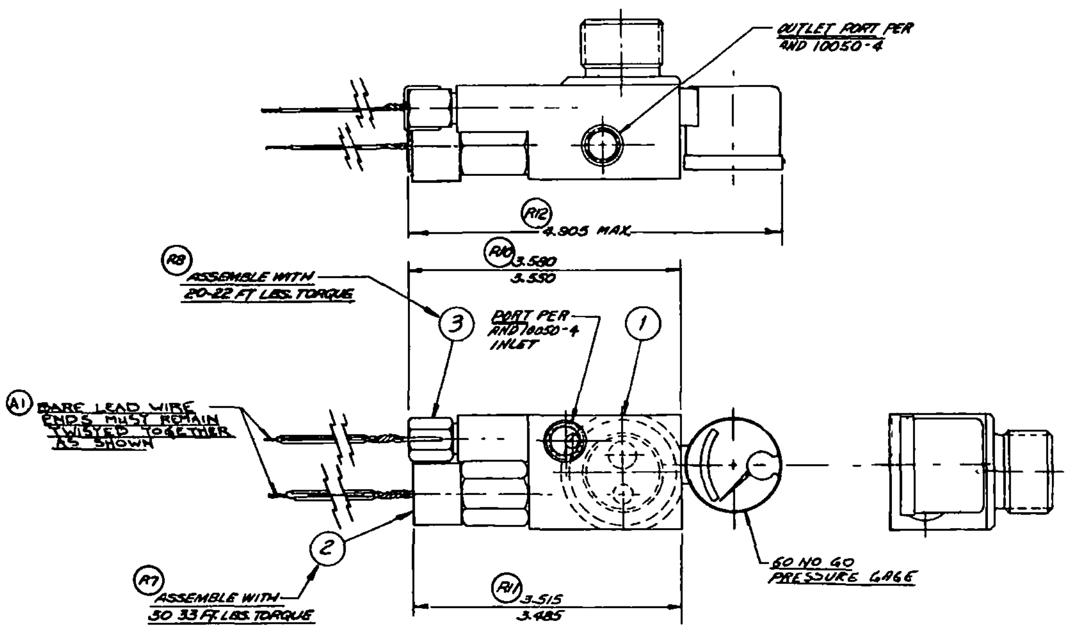


Figure 44. Squib Actuated Helium Fill and Release Valve Assembly (Dwg 304382)

	ITEM NO	RMI PART NO	MFRS PART NO.	DESCRIPTION	MATERIAL	NO. REQ'D
J18	1	306354-1	3001132 1	BODY	——	1
J19	2	306354-2	3001132 3	RETAINCE - SEAT	——	1
	3	306354-3	3001127-3	PLUG FITTING	——	1
J7	4	306354-4	3001132-4	PISTON	——	1
	5	306354 5	3001127-5	PIVOT - SPRING	——	1
	6	306394-6	3001127-6	NUT - LOCK	——	1
J8	7	306*54-7	3001132 2	SEAT	——	1
	8	306354-8	3001127-8	CAP - ADJUSTMENT	——	1
J9	9	306354-9	130-3	SPRING	——	1
J14	10	MS28782-7	MS28782-7	BACK-UP RING	——	1
J13	11	MS28775-014	MS28775 014	O RING	——	1
J11	12	MS28775 012	MS28775 012	"O" RING	——	1
J12	13	MS 28775 016	MS28775 016	O" RING	——	1
	14	AN995C25	AN995C25	LOCK WIRE	——	AS REQ'D
J10	15	306354-15	101-093	BALL	——	1
J15 G1	16					
J17	17	306354-17	3001132-5	LABEL	——	1

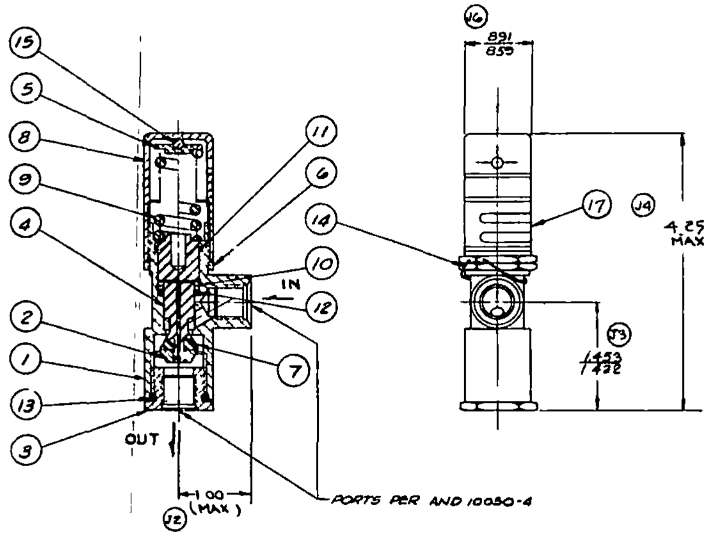
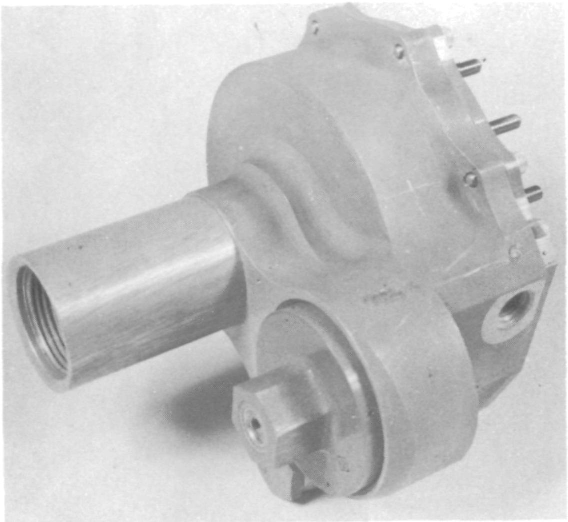
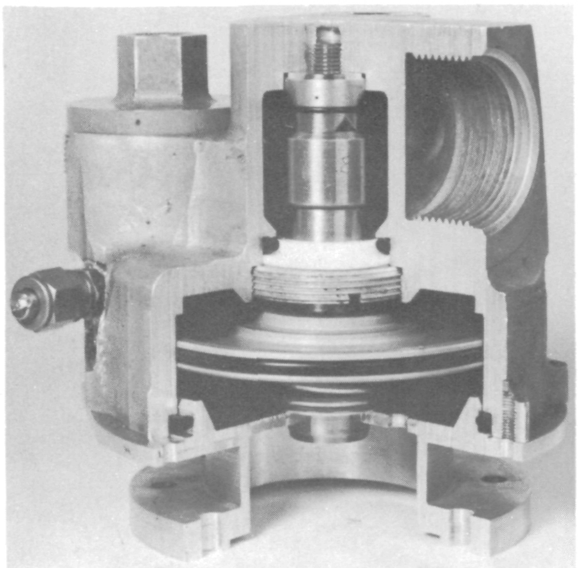


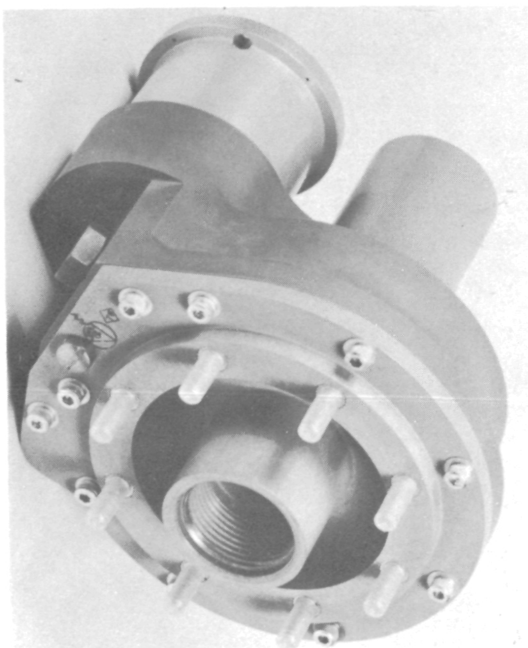
Figure 45. Helium Pressure Regulator (Dwg 306354)



(081-43)
Atlas Regulator.



(081-59)
Thor Regulator.



(081-87)

Figure 46. Types of Thiokol-RMD Regulators

3. Check Valves

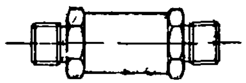
These components are primarily used to prevent back flow. In most cases they are included for reasons of safety and, since they are simple in construction, are extremely reliable. Where human safety is concerned, it has been found that quadruple check valves replacing a single check valve provides an extremely safe and reliable component. This scheme affords series and parallel redundancy. Units of this type have been made that are not appreciably larger than a single check valve. Figure 47 shows the installation and specification control drawing for a typical check valve. Procurement of these valves is covered by Thiokol-RMD Specification 20232.

4. Relief Valves

Relief valves in pressurizing systems are usually built into the pressure regulators, but can, of course, be made as separate units. They are strictly a safety feature to avoid overpressurization. If they are called upon to operate, the mission could be lost due to excessive loss of pressurant; therefore, immediate preventive steps are required. For example, a complete shutdown could be signalled, followed by an attempt to start again, or, if the mission is manned, repairs could possibly be made. In the case of the main propulsion system, the normal function of the fuel manifold relief valve is to operate after shutdown of any main thrust chamber to vent off cooling jacket boiloff. Figure 48 shows a typical fuel relief valve suitable for this application. Of particular concern in the operation of the relief valve (or any component containing sliding metal parts) which is vented to space is the possiblity of cold welding of metal parts so that the relief (or other function) is not performed. In this connection Thiokol-RMD will determine the practicality of using controlled vaporization of seal materials, or deliberately installed nonmetallic materials, to effect a continuous gassing off in the region of sliding parts, thereby eliminating concern over cold welding.

5. Burst Disc Valves

These components are used to insure a hermetic seal of propellant tankage. They are, of course, one shot items and consist of a metal disc which is ruptured by the fluid sealed at the appropriate time. These discs are extremely reliable and a wide range of burst pressures can be provided. In special cases the discs can be ruptured by a lancing type device that could be pressure or squib actuated.



ACTUAL SIZE

ITEM NO	RMI PART NO	MFR PART NO. 1	DESCRIPTION	MATERIAL	NO REQ'D
1	308694-1	10287A1	HOUSING	—	1
2	308694-2	154A1	SPRING GUIDE	—	1
C1 3	308694-3	155	SPRING	—	1
4	308694-4	5007-20	"O" RING	—	1
5	308694-5	153A1	POPPET	—	1
6	308694-6	1906	GASKET	—	1
C1 7	308694-7	10288A1	END	—	1

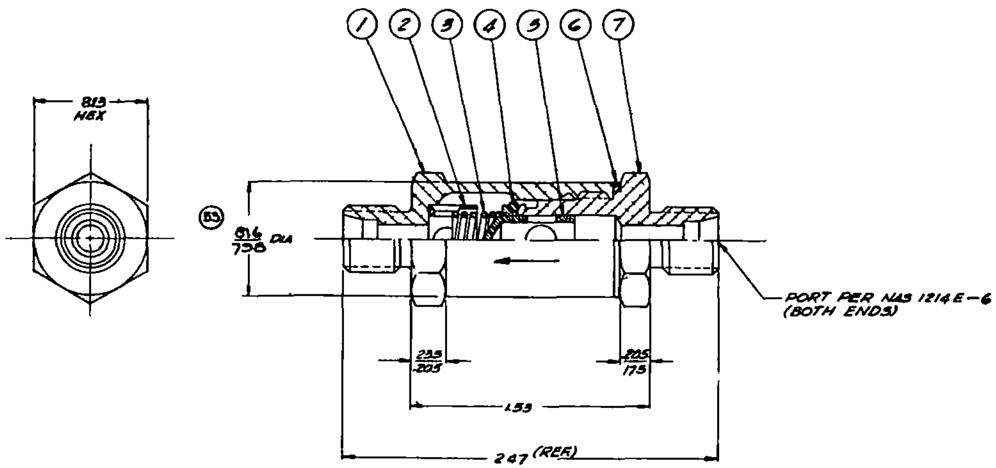
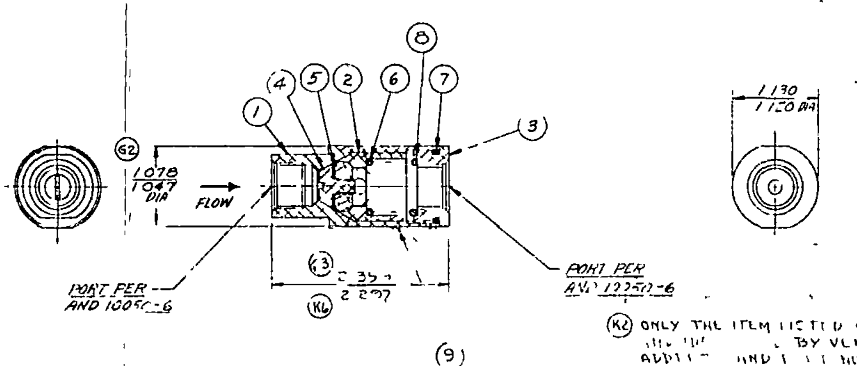


Figure 47. Oxidizer Tank Check Valve Assembly (Dwg 308694)

ITEM NO	PART NO	CODE	MFG PART NO		DESCRIPTION	NO REQ
1	304885-1	J	4001157-1	C	BODY	1
2	304885 2	J	4001157 2	B	PISTON	1
3	304885 3	J	4001157-3	N/C	PLUG	1
4	304885 4	J	4001157-4	N/C	SEAT RETAINER	1
5	304885 9	J	4001157 5	N/C	SEAT	1
6	304885 6	J	4001157 7	A	SPRING	1
7	304885-7	J	118-19	N/C	"O" RING	1
8	304885 8	J	139	N/C	SHIM	1
9	304885 10	K	4001158 1	N/C	NAME PLATE	1



ONLY THE ITEM LISTED ON THE DRAWING [illegible] BY VENDOR'S NAME, ADDRESS AND [illegible] NUMBER HAS BEEN TESTED AND FOUND SUITABLE BY RMD FOR USE IN A ROCKET ENGINE. A SUBSTITUTE [illegible] WITHOUT [illegible] AND [illegible] BY RMD

Figure 48. Fuel Tank Relief Valve (Dwg 304885)

6. Valve Sealing and Allowable Leakage

All propellant tanks are hermetically sealed. The helium bottles are also hermetically sealed prior to arm since the helium pressurizing squibs at the tank exits have a hermetic seal built into them and the bodies of these valves will be welded to the helium bottles. The helium fill squib valves incorporate a metal to metal drive fit for shutoff. A welded plug arrangement will be considered as a hermetic seal backup.

The leakage problem is thus restricted to those periods after an engine system is armed. For the main engine system and main attitude control systems, the period after the systems are armed is essentially 14 days. For the re-entry roll control system, the period is approximately 30 minutes. The most critical area for leakage centers on the propellant valves for the main attitude control thrust chambers. Here the armed period of 14 days is involved and there is a total of 24 propellant valves. A portion of the propellant reserve allowed for this system will be taken as the allowable leakage for all the valves. The probability that all valves will leak the maximum amount for 14 days is an unlikely event. The same maximum allowable leakage rate will be assigned to the propellant valves in the re-entry roll control system. The leakage rates for the main thrust chamber propellant valves have been established using the data from the valves used in the X-15 powerplant.

Table IX presents a listing of the propellant valves and the amount of maximum allowable leakage assigned to each. For the components in the helium pressurization system, allowable leakage rates consistent with the practices developed for the X-15 powerplant were established. An external leakage rate of ten cubic centimeters per minute of helium, at standard temperature and pressure, will be allowed for each component in the helium systems.

7. Tankage

The design of all the tankage on the powerplants being discussed has been predicated on design criteria compatible with the manned safety aspect of the program. The factors used are very close to those used for the design of the highly successful man-rated X-15 powerplant.

Two types of tanks were designed. One of these types was designed to contain propellants and the other to hold high pressure helium. They are

TABLE IX

MAX ALLOWABLE LEAKAGE RATES

Valve	Designation	Figure Number	Max Allowable Leakage Rate cc/min of Working Fluid	Number of Valves
Main Attitude Control fuel propellant valves	V6, V8, V10, V12, V14, V16, V18, V20, V22, V24, V26, V28	4	0.012	12
Main Attitude Control oxidizer propellant valves	V5, V7, V9, V11, V13, V15, V17, V19, V21, V23, V25, V27	4	0.018	12
Re-entry Roll Control fuel propellant valves	V6, V8, V10, V12, V14, V16, V18, V20	2	0.012	8
Re-entry Roll Control oxidizer propellant valves	V7, V9, V11, V13, V15, V17, V19, V21	2	0.018	8
Main Propulsion fuel portion of propellant valve	V1 - V4	4	0.2	4
Main Propulsion oxidizer portion of propellant valve	V1 - V4	4	0.2	4

shown in Figures 49 and 50. The material selected was a titanium of the meta-stable beta type. This material is compatible with the N_2O_4 and MMH being used for the earlier system and is also compatible with the OF_2 and MMH for the later version.

The wall thicknesses of the tanks have been chosen on the basis of the strength levels of the material and not upon considerations of fabrication. These thicknesses may not seem compatible with present state of the art techniques, but it is believed that with the progress now being made in this area, the extremely thin walls will be completely feasible. As an example of this, companies such as the Budd Company are doing extensive work in the fabrication of large diameter thin-walled tanks. They have recently delivered three tanks to NASA-Goddard Division. These tanks are spherical in shape, 36 inches in diameter, and are fabricated from 0.026-0.022 inch cold worked stainless steel. They have also fabricated a five-inch diameter tank of the same material with a thickness of 0.0055-0.0045 inches.

Careful consideration has been given to the method of mounting the thin-walled, large diameter propellant tanks. These tanks are held to the supporting structure by means of a skirt which is welded to the periphery of the shell. The tank must be thickened at the juncture of the skirt and the tank wall so that the loads may be spread evenly about the circumference.

Due to the high diameter-to-thickness ratio, the diameter of the tanks will grow under the influence of pressurization. As an example, the main oxidizer tank is 48.5 inches in diameter and is 0.035 inch thick. The calculated radial growth of this tank is 0.146 inches. Thus another reason for thickening the tank at the juncture of its wall with the supporting skirt is to prevent excessive radial loads on the skirt. The thickened skirt section is slowly tapered to the necessary vessel thickness to minimize the effects of the discontinuity stresses.

The typical tank shown in Figure 49 also reflects the necessary design requirements for a positive expulsion bladder. There is a wide neck at one end which is necessary for the insertion of the bladder. Bladder requirements also necessitate that there be no sharp edges on the inside of the tank. As a result all of the welds are of the butt type and all radii are extremely generous. There is a boss at the opposite end of the necked flange to support the breather tube and to provide an inlet for the pressurizing gas. A small bleeder tube is placed in through the breather tube so that the ullage can be controlled to bring the filled capacity to 100% if this is necessary.

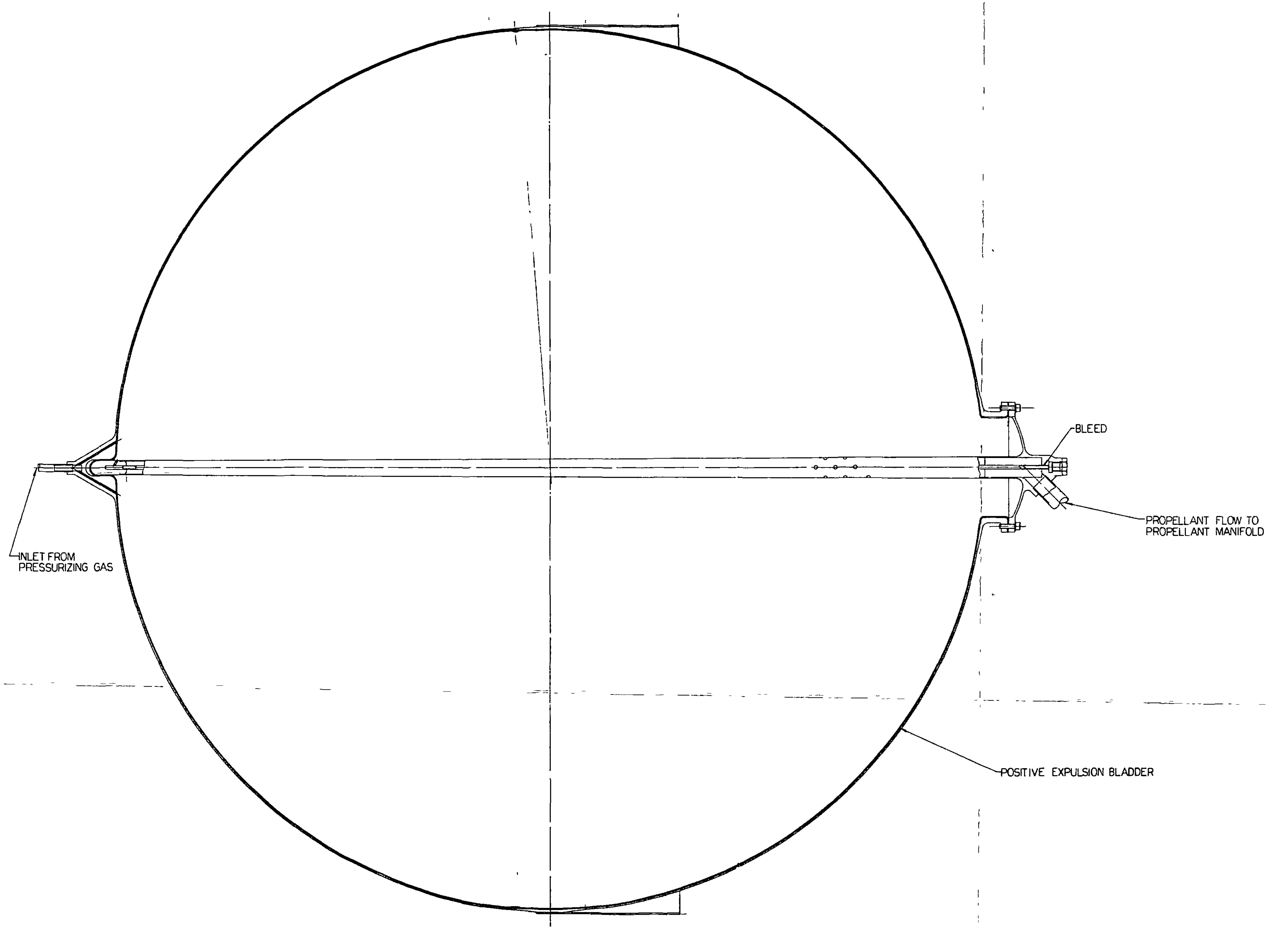


Figure 49. Propellant Tankage (Dwg 52095)

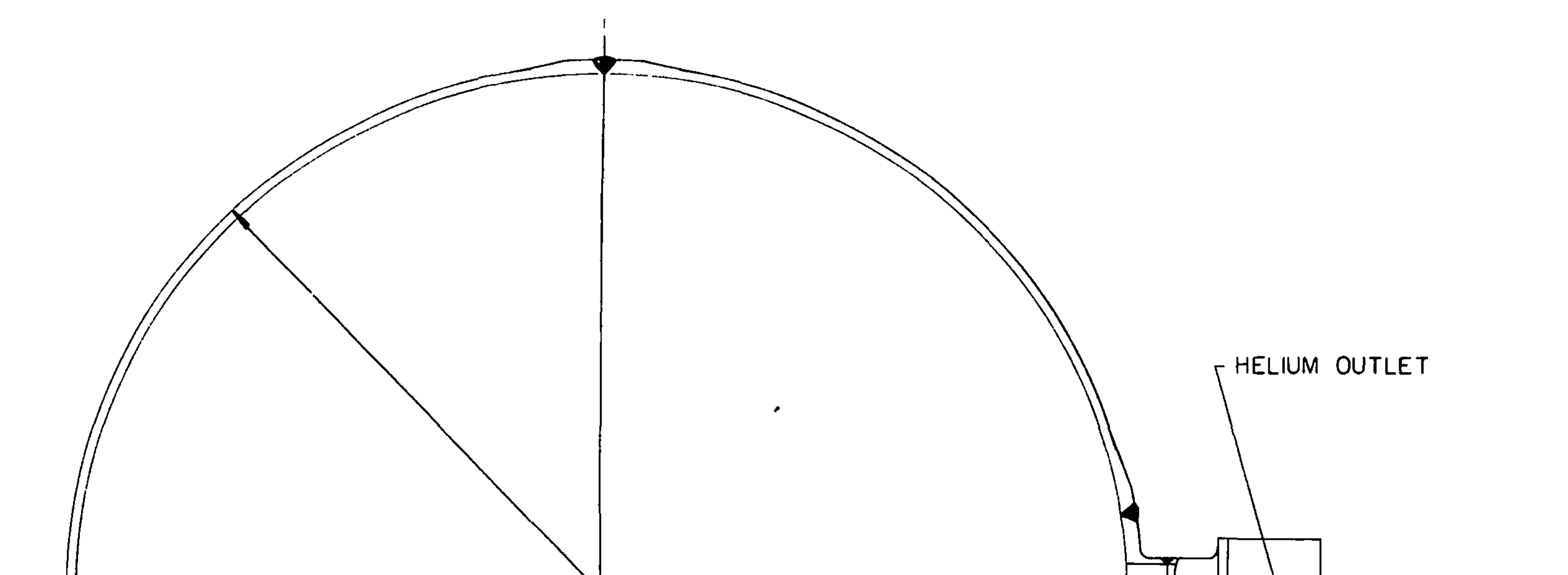


Figure 50. Helium Tankage (Dwg 52208)

The paramount criterion of the Apollo program will be the safe return of the flight crews to earth. To achieve this criterion, reliability must play a major roll in the preliminary design. However, it is impossible to achieve "absolute" reliability; therefore, to eliminate or reduce the probability of conditions which may represent hazards to the flight crews, redundancy becomes a useful tool.

Component and subsystem redundancy must be utilized discreetly. Detailed system studies and trade-off analyses are required to efficiently utilize redundancy. In the case of tankage for the propellants and pressurization supplies, the types of failure which could result in the loss of propellants or helium must be analyzed.

Tankage failures could be caused by excessive pressure levels and weld failures from excessive environmental conditions brought about by other vehicle-imposed malfunctions. However, in the case of excessive pressure levels, failure modes can be eliminated by proper design techniques of the pressure feed system. This has been accomplished in all Apollo feed systems through the utilization of redundancy and the study of malfunction conditions. Therefore, excessive pressure levels should not be encountered in the propellant or helium tankage under single malfunction conditions. In addition, the tankage design incorporates ultimate safety factors which preclude failure of tankage if pressure levels 50% above design pressure are actually experienced. Therefore, tankage failure due to excessive pressure levels is extremely remote. The case of weld failures or defective material can be eliminated by adequate quality control procedures, inspection, and acceptance test procedures. Again this type of tankage failure is practically nonexistent.

If redundant or compartmented tankage was incorporated into the system the probability of weld failures or defective materials would increase thereby increasing the probability of expending leakage propellants. In addition, redundant or compartmented tankage requires additional valving and associated piping which in turn increases the probability of leakage or unavailability of supply. To offset this additional complexity, redundancy of valving, etc. may be required, which in turn increases probability of leakage and burdens the system with additional weight. This weight may be more effectively utilized for shielding purposes against meteorites.

Thiokol-RMD believes that shielding represents a more feasible method of preventing propellant loss from meteorite penetration than redundancy or compartmented tankage. The probability of meteorite penetration can be made extremely low by proper shielding. An analysis will be performed to determine the trade-off between loss of propellants and/or helium as a function of meteorite penetration and the reliability of redundant or compartmented tankage. Table X indicates the order of magnitude of propellant and helium loss due to meteorite penetration of tankage. These data also show the remote probability of such a failure occurring.

In the Apollo program the helium supply tankage is made of titanium which is 0.140 inch thick. The propellant tanks are 0.040 inch thick. Such thicknesses eliminate the possibility of penetration of meteorites which have a reasonable probability of occurring. For example, the titanium helium sphere could be pierced by a meteorite particle having a diameter of 0.110 inch. However, if one looks at the probability of this occurring in a 24 hour period he would find that the probability of occurrence is one in two million. This probability value is based on a 9.58 foot diameter sphere, which approximates the surface area being utilized for the helium spheres in the Apollo system. The propellant tankage which is 0.040 inch thick will allow meteorite particles exceeding 0.035 inch to penetrate the tank. The probability of this occurring is one in fifty thousand. Therefore, the probability of meteorite penetration is extremely remote. One must also consider that the tankage is within the vehicle shell. Any particles which penetrate the outer shell will lose considerable kinetic energy and, therefore, have a lower probability of penetrating the tankage within the vehicle. Thus, the probabilities of encountering meteorite penetration are even less than indicated. Table X indicates particle size, the probability of occurrence, loss rate, and the time to expend the propellants or helium if a penetration did occur. However, the particle size of 0.0315 inch and 0.00362 inch will not pierce either the propellant tank or the helium tank. Therefore, the associated propellant expenditure times are fictitious. The first particle size in the table (0.362 inch diameter) will pierce either one of the tanks. However, the probability of occurrence (one in a hundred million) is so remote that it would be senseless to protect for this size particle penetration.

8. Lines

The design considerations presented above in regard to redundancy will also apply to lines and fittings. In the Apollo system, Thiokol-RMD will unitize components wherever possible thereby eliminating as many fittings

TABLE X

LOSS OF HELIUM AND PROPELLANTS DUE TO METEORITE PENETRATION

Propellant	Particle Size Diameter (Inches)	Probability of Occurence in 24 Hours (9.85 ft Sphere)	Loss Rate (lb/sec)	Time to Expend Full Tank (Hours)
Fuel	0.362	1.2×10^{-8}	6.03	0.1295
	0.0315	2.0×10^{-5}	0.0457	17.05
	0.00362	1.2×10^{-2}	0.000603	1295
Oxidizer	0.362	1.2×10^{-8}	7.78	0.211
	0.0315	2.0×10^{-5}	0.0592	27.8
	0.00362	1.2×10^{-2}	0.000778	2110
Helium	0.362	1.2×10^{-8}	Variable	0.00728
	0.0315	2.0×10^{-5}	Variable	0.963
	0.00362	1.2×10^{-2}	Variable	7.28

and lines as feasible. A typical unitized body will be used in the pressurization feed system. The parallel regulators, sensing valves, check valves, and relief valve would be integrated into one body with drilled passages. In this manner lines and fittings are minimized thereby reducing the probability of leakage.

Another method which has been successfully utilized on particular systems is the elimination of all fittings by welding. This, of course, limits interchangeability and field servicing but may be advantageous on an overall basis.

G. SYSTEM DESIGN ASPECTS

Several areas which bear on the overall system design were investigated separately and the results are reported here. These areas are:

- Meteoroid impact
- Space environmental effects
- Temperature control of propulsion module
- Multiple engine trade-off
- Information display

1. Meteoroid Impact

The hazards of, and the protection required for, meteoroid impact upon the propulsion system of the Apollo Vehicle were investigated. Both the determination of meteoroid distribution and the mechanics of meteoroid penetration are in early states of the art. However, sufficient information is available to design meteoroid shielding to insure that no penetrations of critical components of the Apollo propulsion system would occur during the entire mission with a probability of 0.9999.

Once a space vehicle has passed through the earth's protective envelope of air, radiation and meteoroid protection must provide sufficient attenuation so that unacceptable levels of structural, component, and personnel damage do not occur.

This discussion concerns only the meteoroid protection required for the propulsion system of the Apollo Vehicle.

The two basic problems involved are: (1) the determination of the distribution of meteoroids with regard to the type, size, weight, and velocity, and (2) the determination of the construction necessary to insure no penetration of critical structures (e.g., propellant and pressurizing tanks) within a chosen level of probability.

Both of these areas are in a relatively early state of the art and future changes of order of magnitude in the solutions are not inconceivable. However, progress has been so rapid in the past couple of years that it may be anticipated that by the time that the Apollo vehicle design is finalized, sufficient data will be available to accurately assess and, if necessary, to modify the design chosen on the basis of present day knowledge.

By choosing a level of probability of no penetration of critical structure of 0.9999 we risk the possibility that the true level of probability may be only 0.999 (or perhaps 0.9999) due to the uncertainties of the current state of the art.

This level of 0.999 is still high enough to warrant designs based on current information since there is only a possibility that the true reliability will be as low as 0.999 and modifications may still be made to the design, if necessary, to raise the reliability back up to 0.9999.

a. Meteoroid Distribution

The three terms, meteor, meteorite, and meteoroid, which are often confused, are defined below to ensure proper interpretation of the discussion.

Meteoroid: A solid particle, in a solar orbit (and relatively small compared to asteroids or comets)

Meteor: A meteoroid, passing through the earth's atmosphere and heated to incandescence so that it is visible

Meteorite: The intact portion of a meteoroid which has landed on the earth's surface

Variations of these definitions occur in the literature so that in one case the term meteor might be used to include meteoroids while in another case the term might be restricted to the sensible phenomena associated with a meteor's passage. However, for the purposes of this discussion, the terms will be used as defined.

b. Composition of Meteoroids

The first natural assumption was that the composition of meteoroids was the same as that of meteorites. Meteorites can be divided into three general classes of composition as follows (Ref 1):

Composition	Approximate Specific Gravity
Iron-nickle	7.8
Stone-iron	4.7 to 6.2
Stone	3.5

However, current research indicates that the vast majority of meteoroids consist of the ices of gases (comet debris) with specific gravities on the order of 0.05 (Ref 2) which is vastly lower than those tabulated above. It is obvious that such meteoroids would not survive the heating of atmospheric passage and thus are not represented on earth.

c. Distribution of Meteoroids

It is believed that meteoroids are concentrated in the plane of the ecliptic (the plane of the earth's orbit) and the flux data given below are for that region. Since the Apollo mission will be within that area, the data should be applicable.

Figure 51 shows the latest best guess for meteor flux (Ref 2). There are many earlier assumed distributions and undoubtedly there will be later, more accurate distributions. The one shown is based on photographic techniques from ground-based stations and by various measuring devices carried in rockets and satellites.

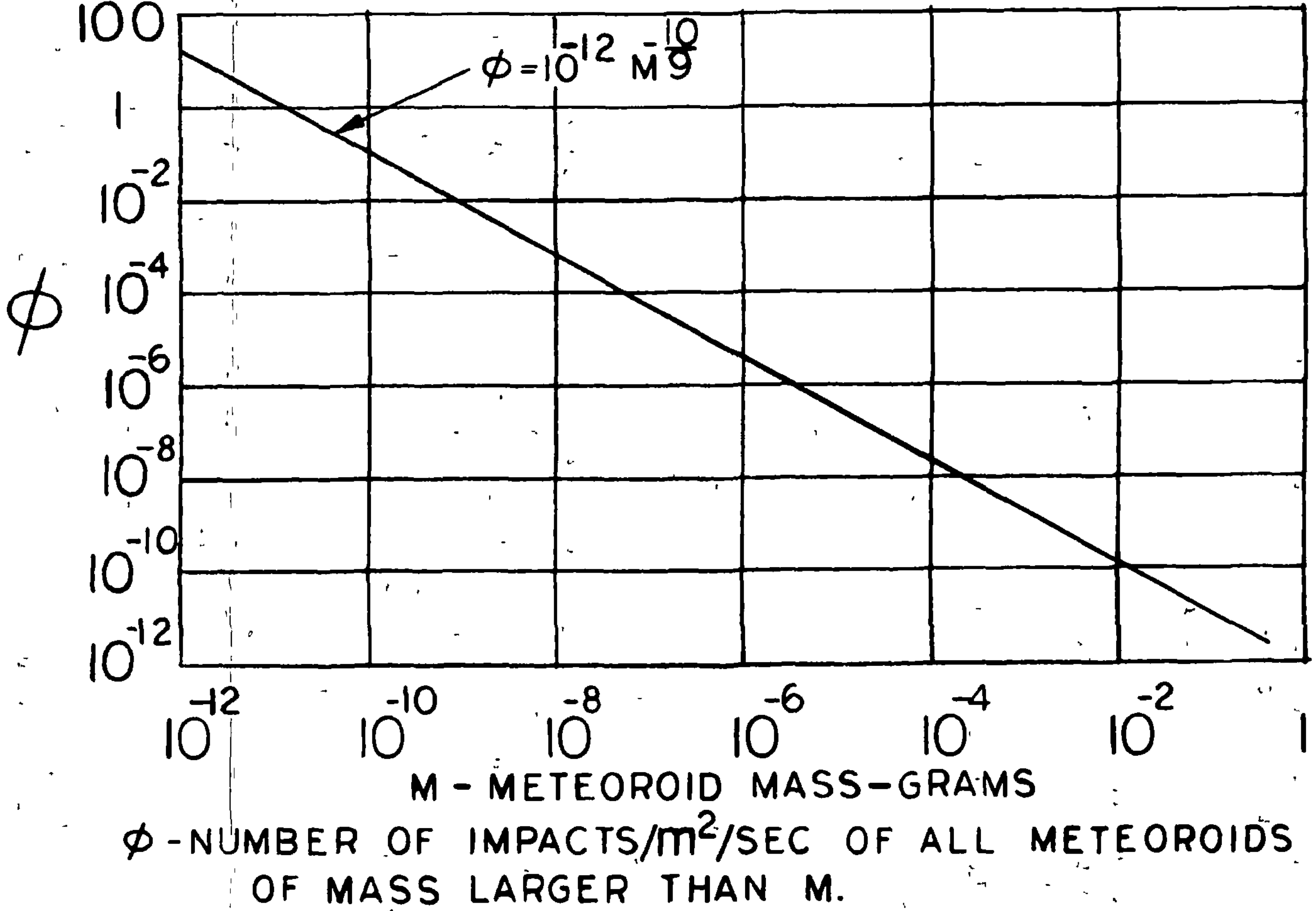


Figure 51. Meteoroid Flux in the Earth's Vicinity

The predominant feature of Figure 51 is that the meteoroid flux varies inversely with mass. The observations on which Figure 51 is based were all made in the vicinity of the earth where some shielding might be expected due to the earth itself. Accordingly, the flux density should be approximately double away from the earth.

d. Probability of Impact

The main propellant tanks of the Apollo vehicle propulsion system consist of two oxidizer tanks, 44.5 inches in diameter, and two fuel tanks, 48.5 inches in diameter.

The total surface area of these tanks is given by:

$$\text{Area} = A = 2\pi(44.5)^2 + 2\pi(48.5)^2 = 27,220 \text{ square inches}$$

$$A = \frac{27,220}{1,550} = 17.55 \text{ square meters}$$

However, this figure should be reduced because of the shielding of portions of each tank by adjacent tanks. On the other hand, lines, valves, engines, and secondary tanks have not been included.

The net effective area will be assumed as 15 square meters, or A = 15 square meters. The time for the flight is 14 days or t = 14(24)(3600) = 1.243×10^6 seconds. The desired probability of no puncture is 0.9999 which means that 0.0001 punctures are allowable per flight or N = 0.0001. To ensure this, a meteoroid of mass m which is just sufficient to cause puncture must not impact more than x times per second per square meter.

It is obvious that

$$x = \frac{N}{A_t} = \frac{0.0001}{15(1.243 \times 10^6)} = 0.535 \times 10^{-11}$$

From Figure 51, m = 0.22 grams for the earth's vicinity.

However, for the Apollo mission, $\emptyset$ should be doubled since the earth's shielding effect would not apply. Therefore, equivalent $x = 1/2\ (0.535 \times 10^{-11}) = 0.2675 \times 10^{-11}$.

From Figure 51, m = 0.41 grams.

e. Meteoroid Velocity (Ref 2)

All available data agree that the velocity range of meteors is from 11 to 72 km/sec. These two limits have physical significance in that the lower limit is the escape velocity from the earth while the upper limit is the maximum velocity a particle may have and still be a member of the solar system. To date, no meteoroid, whose motion has been accurately determined, has come from outside the solar system.

For a meteoroid in the 0.41 gram range, the average velocity is 28 km/sec. Thus, protection must be provided against a 0.41 gram meteoroid whose velocity is 28 km/sec.

f. Meteoroid Penetration Theory and Calculations

The currently accepted best guess for the mechanics of meteoroid penetration of shielding is a theory proposed by R. L. Bjork of the Rand Corporation (Ref 2). He neglects the strengths of the materials involved because the kinetic energies of the particles are far above those values. Instead he considers that the materials act as fluids. His results agree well with the highest velocity tests made to date. It should be noted that those tests were made at velocities substantially below the lowest meteoroid velocity of 11 km/sec.

His results are as follows:

For aluminum impacting on aluminum, $p = 1.09\ (mv)^{1/3}$

For iron impacting on iron, $p = 0.606\ (mv)^{1/3}$

where p = depth of penetration in a semi-infinite solid, cm

m = meteoroid mass, grams

and v = impact velocity, km/sec

For titanium shielding, the equation would be approximately

$$p = 0.8(mv)^{1/3}$$

(by interpolating between the equations for aluminum and iron)

for $m = 0.41$ gm

$v = 28$ km/sec

$$p = 0.8 \left[(0.41)(28)\right]^{1/3} = 0.8 \left[11.5\right]^{1/3}$$

$$= 0.8\ (2.26) = 1.8 \text{ cm}$$

The required thickness to prevent penetration is given by $t = 1.5\ p$

For our case $t = 1.5(1.8) = 2.7$ cm

converting to inches: $t = \frac{2.7}{2.54} = 1.06$ inches

This thickness of 1.06 inches represents the required thickness of a set of tanks which are otherwise unprotected against meteoroid impact. However, in the case of the Apollo, an outer skin, of sandwich construction protects the tanks. This allows a substantial reduction in wall thickness as shown below.

It has been estimated that a single meteoroid shield reduces the chance of puncture by a factor of from 10 to 100 (Ref 3). If this is accepted then a double shield (the sandwich construction outer wall) should reduce the chance of puncture by a factor of from 100 to 10,000. If 1000 is chosen as a mean value between these limits, then the value N becomes:

$$N = 0.0001\ (1000) = 0.1$$

The equivalent x becomes:

$$x = 1/2\ N/A_t = 1/2\ \frac{(0.1)}{(15)(1.243 \times 10^6)} = 0.2675 \times 10^{-8}$$

From Figure 51, m = 0.00077 gm

$$p = 0.8 \left[(0.00077)(28)\right]^{1/3} = 0.8 \left[0.02155\right]^{1/3}$$

$$p = 0.8 \;\; 0.2785 = 0.2225 \text{ cm}$$

$$t = 1.5p = 0.334 \text{ cm}$$

in inches: $t = \dfrac{0.334}{2.54} = 0.132$ inches

This is greater than the present tank thickness. Instead of increasing the tank thickness to the above amount, it may require less weight to double the shielding of the sandwich material by inserting two additional walls. The four walls would be separated by three layers of honeycomb material with the overall thickness being 1/2 inch .. Design trade-offs of this type will be the subjects of early, coordinated studies with the General Electric Company MSVD.

This construction should decrease the probability of puncture again by a factor of 1000 so that

$$x = 0.2675 \times 10^{-8} \times 10^{3} = 0.2675 \times 10^{-5}$$

From Figure 51, $m = 1.54 \times 10^{-6}$ gm

$$p = 0.8 \left[(1.54 \times 10^{-6})\ (28)\right]^{1/3} = 0.8 \left[43.1 \times 10^{-6}\right]^{1/3}$$

$$p = 0.8\ (3.5 \times 10^{-2}) = 0.028 \text{ cm}$$

$$t = 1.5\ (0.028) = 0.042 \text{ cm}$$

in inches. $t = \dfrac{0.042}{2.54} = 0.0165$ inches

which is less than the tank wall thickness.

g. Conclusions

It is feasible to design meteoroid shielding for the Apollo propulsion system to insure no penetrations of critical components during the entire mission with a probability of 0.9999. The actual probability may vary from this because of the early state of the art, but modifications may be made, if necessary, by the time the vehicle design is finalized. The shielding does involve a weight penalty but it is a small fraction of the weight of the entire propulsion system.

2. Space Environmental Effects

a. Radiation Effects

A study was made to investigate the effect of space radiation upon the propulsion system hardware and upon the propellants. The following conclusions reached may be modified as more experimental data become available but it is believed that they are reasonably accurate:

(1) The structural materials selected for this design will not suffer appreciable damage due to space radiation.

(2) It is felt that the propellants will not deteriorate significantly during the mission involved.

(3) The effect of radiation fields on the corrosivity and shock sensitivity of the propellants will have to be investigated in detail.

(4) Radiation can affect plastics. Each plastic used will have to be investigated in detail. Plastics such as Teflon which are known to be easily affected by radiation will be avoided in favor of more resistant materials.

(5) The effectiveness of the basic structure as a radiation shield should be investigated.

In general, the effects of radiation depend only on dose and are independent of dose rate. Theoretically, very high dose rates should produce different effects, but this has not yet been demonstrated experimentally, and it is unlikely that spaceships will encounter such high fluxes. Moreover, the amount of damage is proportional to the energy absorbed regardless of the type of radiation.

Among the changes which are generally produced in metals by radiation are increase in electrical resistance, in thermal resistance, in hardness, and in tensile strength. It appears that after exposure to radiations metals and alloys have higher yield strengths, lower percentage elongations, and somewhat higher ductile-brittle transition temperatures.

On the whole, the effects are less marked on hardened metal than in the annealed condition. The effects of radiation are generally smaller at elevated temperatures (Ref 5).

Elastomeric materials are particularly suitable for sealing metal surfaces to prevent passage of liquids or gases. They are employed in the form of gaskets and sealants. Their use is due primarily to good "spring back" and poor permeability to gases and liquids. These properties need to be retained after irradiation. Fluoro-rubber elastomers exhibit the highest known heat stability combined with excellent retention of their properties at elevated temperatures. Unfortunately, they are the poorest performers in a radiation environment. They lose their desirable physical properties at comparatively low radiation dosage and also liberate corrosive products that attack the metal surfaces touching them. They should be avoided where high radiation fluxes are present.

Among the materials used where the requirement is good resiliency, tear resistance, low coefficient of friction, temperature stability, and compatibility with the system fluid are Teflon, Kel-F, and Viton A. Teflon, although superior in non-nuclear applications, is one of the poorest materials for radiation resistance. Kel-F, followed by Viton A, offers the best replacement for Teflon (Ref 7). Table XI gives the properties for some fluorocarbon elastomers after irradiation.

Carbides subjected to high radiation are found to change in dimensions and properties. Graphite also changes in dimensions and in physical properties after radiation exposure. In general, there is an improvement in the mechanical properties of graphite after exposure (Ref 6).

The ablative materials and inserts used will be selected so as to insure a minimum of change as a result of exposure to radiation. These materials must be tested under realistic conditions.

Table XII gives a general idea of the tolerances of certain materials to integrated doses of radiation. The wide ranges represented here (from 10^2 roentgens for photo film to 10^9 roentgens for polystyrene insulator) is an indication that radiation does not affect each property to the same degree. The values given represent thresholds to damage or incipient loss of the critical physical property.

TABLE XI

(Ref 6)

EFFECT OF IRRADIATION ON PROPERTIES OF FLUOROCARBON ELASTOMERS

Material	Dose roentgens	Initial Properties and Change After Irradiation					
		Hardness		Elongation		Tensile Strength	
		Shore A	Change %	%	Change %	Ps.	Change %
Viton A-7		88		250		2270	
	1.040×10^8		12.5		-94		6.8
Viton A-8		79		180		2285	
	1.040×10^8		22.8		-88.9		-1.0
Viton A-9		78		165		1810	
	1.040×10^8		24.4		-84.7		46.3
Viton A-10		77		140		1765	
	1.040×10^8		23.4		-85.7		12.4
Viton A-11		79		125		2095	
	1.040×10^8		24.1		-80.3		27.9
Kel-F 5500		62		550		1810	
	1.040×10^6	--	0	---	9.1	----	44.2
	5.25×10^6	--	0	---	-8.2	----	19.6
	2.27×10^7	--	3.2	---	-41.8	----	-28.8
	5.73×10^7	--	16.1	---	-73.6	----	-24.9
	10.4×10^7	--	25.8	---	-80.0	----	-13.9

TABLE XII

(Refs 4 and 6)

TOLERANCE LEVEL OF SPACE VEHICLE PAYLOAD COMPONENTS

Material	Integrated Tolerance (Roentgens)
Photo Film	10^2
Man	4×10^2
Teflon	2×10^4
Optical Glass	10^5
I. R. Devices	10^6
Semi-Conductors	10^6
Lucite	10^6
Rubber	10^7
Typical Lubricant	10^7
Vacuum Tubes	10^7
Polystyrene Insulator	10^9

Of the propellants being considered for application in the Apollo vehicle, none has been examined in any detail.

While MMH itself has not been investigated, experiments have been performed at the Missiles and Space Division of Lockheed Aircraft Corp. using a 50/50 mixture of hydrazine and unsymmetrical dimethylhydrazine. They monitored the effect of Co^{60} radiation by measuring the rate of pressure rise in closed systems. At total integrated dosages of 10^6 rads (1.2×10^6 roentgens) they find approximately 3.5×10^{-5} moles of gas produced per mole of fuel irradiated.

While detailed irradiation experiments of the other propellants have not yet been reported, it is reasonable to expect that the degree of decomposition will be of an equivalent order of magnitude. Such small concentrations will have no noticeable effect on the energy value of the propellants; however, it is possible that certain deleterious properties of the propellants such as corrossitivity and shock sensitivity may be substantially enhanced by the radiation fields. For example, when Freon-11 (CCl_3F) is irradiated, the ions which result are not compatible with stainless steel. Such effects may be anticipated with other fluorine containing propellants and must be investigated (Ref 4).

For a one year orbit of a satellite at altitudes between 500 and 3000 miles (the inner Van Allen belt) approximately 10^6 roentgens will be experienced (Ref 4). This is beyond the threshold of damage for certain materials but it is practically negligible in regard to propellant damage. By ratioing the Apollo mission duration of two weeks to the foregoing radiation value of 10^6 roentgens, it is found that the vehicle will experience 3.8×10^4 roentgens during its mission. This is a conservative figure since the peak radiation level which will be experienced will occur within the Van Allen belts (not considering solar flares). Occasionally solar flares increase the flux density by about 100 times the average, dropping to about 10 times the average after six hours. Approximately 2 to 6 solar flares occur per year.

The effects of radiation environments on materials appear to be concentrated in the unshielded surfaces. This results in highly localized dose rates, which may cause surface degradation. The type and manner of this degradation is in many cases uncertain and there is only limited information available on radiation damage in a vacuum environment (Ref 8).

b. High Vacuum Effects

The effects of high vacuum on materials were studied. The following conclusions were reached which may be modified as a result of continuing study and experimental work.

(1) The metals chosen for this application will not suffer damage due to sublimation during the Apollo mission.

(2) The cross-linked plastics chosen for this application will be satisfactory in high vacuum for a mission of the Apollo type.

(3) Gas permeation through the metals chosen will not be a problem for a mission of the Apollo type.

(4) Self-welding of metals will be avoided by proper selection of materials and by the use of dry film lubricants.

(5) All lubricated surfaces will be sealed against the high vacuum or dry film lubricants will be used.

There are two important effects on solid materials which occur in a high vacuum. First, sublimation and evaporation are enhanced by the absence of an atmosphere in that molecules leaving the surface of a material do not make collisions that return them to the surface. Therefore, if a spacecraft is high enough, any molecule that leaves the surface can be assumed not to return.

The second effect is the partial or complete removal of the surface film of gas which covers all material in the sea level atmosphere. As a result of prolonged exposure to the vacuum of space, the character of the gas layer on spacecraft parts will change from the sea level condition.

It is expected that neither of the above effects will result in significant changes to the metals for the Apollo mission.

The effects of high vacuum on plastics are complex because of the variety of ingredients in the plastics. A general rule that will be followed will be the avoidance of plasticized plastics and the use of cross-linked plastics. All plastics used will be subjected to detailed testing under high vacuum conditions.

There are mechanical, chemical, and even electrostatic mechanisms involved in ordinary function and wear. The multiplicity of phenomena is one of the reasons for the conflicting data available.

In general, an oxide or other barrier must be present to prevent seizure and surface damage. Under normal conditions, a film of lubricating fluid assists in creating a barrier.

Conditions of high vacuum prevent the use of ordinary fluid application. It is true, however, that for short missions under vacuum conditions of 10^{-7} or 10^{-8} mm of Hg certain special low vapor pressure greases or fluids may be suitable for short periods of time.

Certain solid film lubricants appear promising for missions of the Apollo type. Intensive investigation of many types of lubricants would be conducted to insure optimum selection.

3. Temperature Control of Propulsion Module

The temperature environment experienced by the final stage propulsion and attitude control system are (1) aerodynamic heating during boost, (2) direct solar heating or radiation from the solar cells and collector during the cislunar flight, (3) alternate heating and cooling during the 3 to 7 days of lunar orbit, (4) intermittent heating during firing of the thrust chamber, and (5) possible heating from the adjacent control compartment. At the present stage in the design of the vehicle sufficient information is available to estimate the effects of items (1), (2), and (4). Engine components must be in readiness for start-up at any time between the end of booster firing and re-entry. Propellant loss due to boil-off (of cryogenics) or freezing (of storables) is to be eliminated or minimized by the use of insulation and skin surface finishes.

a. Aerodynamic Heating

During the first stage booster firing (0 to 207,000 feet) heating of the vehicle skin is mainly aerodynamic, combined with radiation from the earth and sun, and cooling by radiation from the skin. The heat input to a thin slab under aerodynamic and solar heating is:

$$\ddot{q} = h_a (T_r - T_w) - \epsilon \tau T_w^4 + \alpha_s I_s \quad (1)$$

where h_a = aerodynamic heat transfer coefficient

T_r = recovery temperature

T_w = instantaneous wall temperature

ϵ = emissivity of skin

τ = Stefan-Boltzmann constant

α_s = effective absorptivity of skin to solar radiation

I_s = intensity of solar radiation at any time.

The heat input represented by the term $\alpha_s I_s$ is dependent upon atmospheric conditions, vehicle attitude, and time of launch. For a point on the skin receiving direct and reflected radiation from the earth, $\alpha_s I_s$ may be replaced by a term $\alpha_e I_e$, in which α_e is the absorptivity to earth radiation due to the earth's direct radiation and its reflective radiation, or albedo.

By assuming quasi-steady heat input during a time interval Δt, the change in wall temperature may be computed numerically by

$$\Delta T_w = \bar{q}'' \frac{\Delta t}{\rho_m C_{pm} X} \tag{2}$$

in which $\bar{q}'' = 1/2\ (q''_1 - q''_2)$ = average heat flux in time interval Δt and $\rho_m C_{pm} X$ = thermal capacity of the wall.

To estimate skin heating for possible heat input to the propulsion compartment, equations (1) and (2) were numerically integrated for a 0.010 inch aluminum skin perfectly insulated on the cool side and having an emissivity of 0.5, neglecting radiation input. The maximum skin temperature of 620F is reached near the end of the first stage booster firing, based on a typical Saturn escape trajectory. Skin temperature is plotted on Figure 52. Convective heat transfer is almost negligible at the end of 127 seconds. The maximum skin temperature is not considered excessive and the total input to the skin represents only a few degrees of temperature rise in the interior metal parts and propellants. Because of its location downstream of the stagnation point, temperature control of the tankage compartment during boost should be no problem.

b. Temperature Control During Cislunar Flight and Moon Orbit

As mentioned above, the radiation input from the earth to a body near the earth depends on the body shape, attitude, cloud cover, etc. Data concerning the variation of earth radiation with altitude have been gathered during the development of the earth satellites. The amount of earth reflection has been taken as between 30% and 38% of the solar output (555 Btu/ft^2 hr), at satellite altitudes for estimating the skin temperatures of earth satellites. Above about 50,000 feet, direct solar radiation has reached its maximum value. Figure 53 shows the direct solar radiation as a function of altitude. Above 200,000 feet, convective heat transfer is almost negligible. The vehicle will start to approach a solar equilibrium temperature which is dependent upon the solar absorptivity and the emissivity of the vehicle skin, and the emitting and absorbing

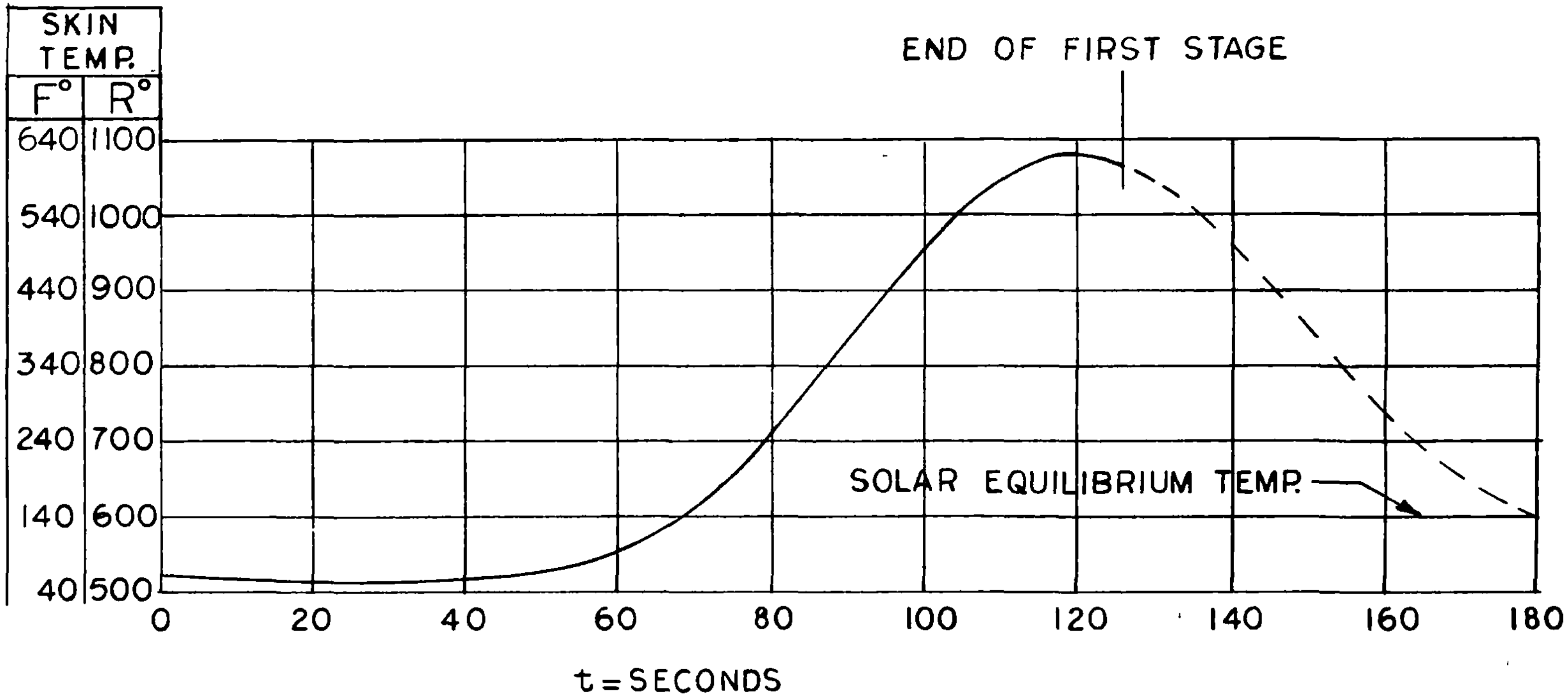


Figure 52. Aerodynamic Heating of Bare Wall (0.010 in. thick) at Sta 255.6

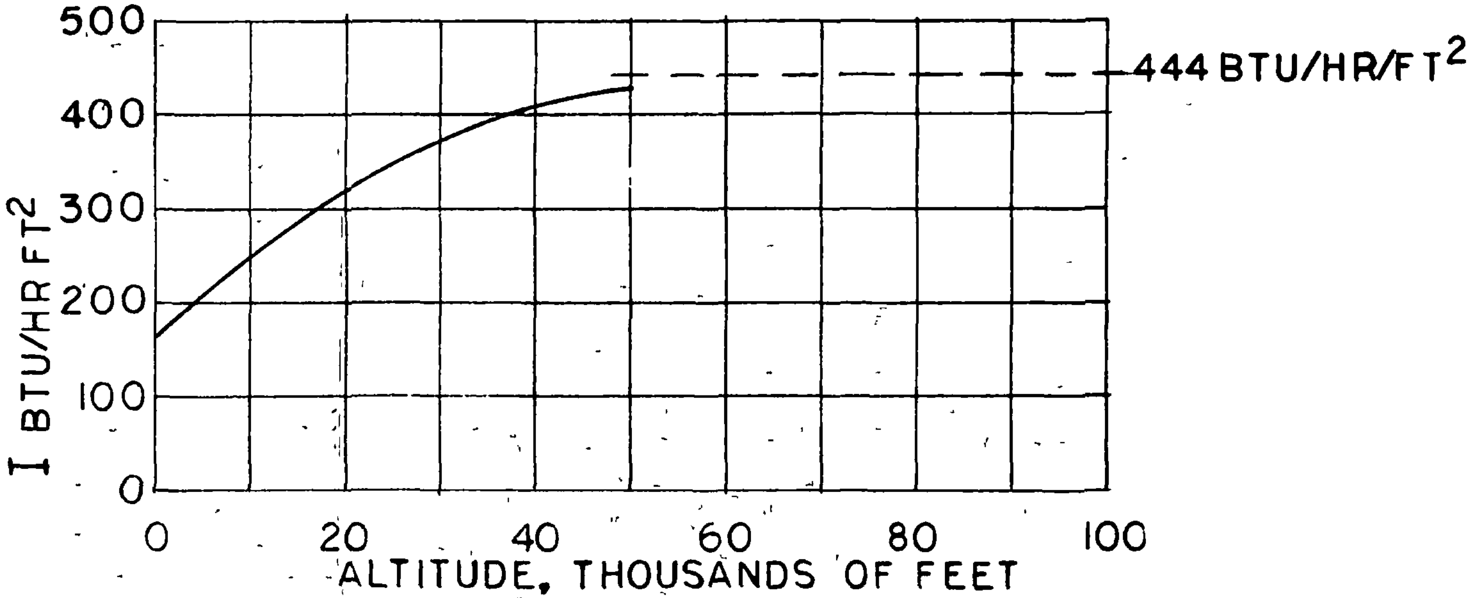


Figure 53. Solar Input in the Vicinity of the Earth

areas. The equilibrium temperature for an inert, perfectly conducting body under solar radiation is:

$$T_{eq} = \left[\frac{1}{\sigma} \frac{\alpha}{\epsilon} \frac{A_\alpha}{A_\epsilon} I \right]^{1/4}$$

A_α = absorbing area

A_ϵ = emitting area

α = absorptivity of skin to solar radiation

ϵ = emissivity of skin at the equilibrium temperature

I = 444 Btu/ft^2 hr in the vicinity of the earth.

A wide range of surface solar absorptivity to emissivity is available, all the way from α/ϵ = 0.15 for some highly reflective paints to 10 for ideally polished metals. Figure 54 shows the estimated solar equilibrium temperature of the tank compartment as a function of this ratio for two assumed attitudes of the vehicle with respect to the sun. Any desired value of effective α/ϵ from 0.15 to 10 can be obtained by using a surface alternately striped with painted and polished surfaces. The allowable compartment temperature variation for reliable utilization of propellants and operation of moving parts is much wider than has previously been obtained in the payload compartments of earth satellites. For example, an optimum compartment temperature range for the proposed 1963 propellant combination would be between 20F and 150F, which can be achieved with an α/ϵ ratio of between 1 and 3. The lower limit of 20F is fixed by the freezing point of N_2O_4 (12F). This limit can be lowered by adding NO to the N_2O_4. For example, 6% by weight of NO lowers the freezing point of the mixture to 0F; 10% by weight reduces the freezing point to -10F There is an ample margin in the design of the surface finish to allow for changes due to temperature and erosion effects. In the payload compartments of earth satellites, temperature variation has been kept down to a maximum difference of 36F For the 1966 propellants, compartment temperatures may vary between -63F and 150F, a range which is easily obtained under conditions of solar input equilibrium. Here the -63F lower limit is fixed by the freezing point of the MMH. The OF_2 will be in an insulated tank and will be independent of compartment temperatures. In practice the upper limit of 150F will probably be reduced by considerations other than propellant storage, e.g., O/F variations

If the compartment is shielded from direct solar radiation, either by a radiation shield or by the surface of a solar collector, the temperature of the shiled, the surface emissivities, and the geometric view factors between the

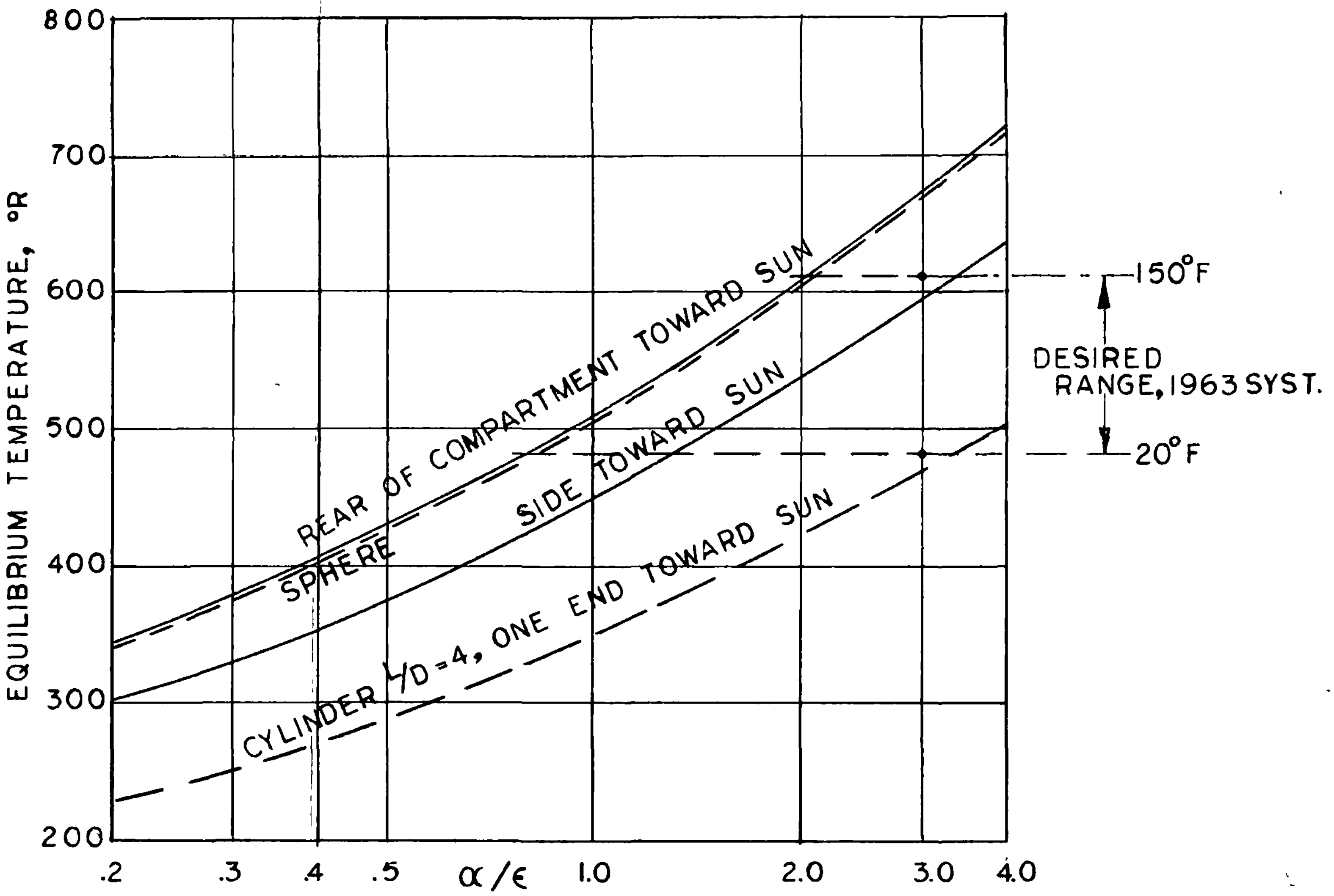


Figure 54. Solar Equilibrium Temperature vs Absorptivity to Emissivity Ratio of Surface

two surfaces will determine the compartment skin temperature at any time. In this case, the emissivity of the portions of the compartment skin receiving heat from the shield may be made as close to 1.0 as possible and the emissivity of the portions radiating into space as low as possible so that the desired range of compartment temperatures is achieved. Some of the surface characteristics available with present materials are given in the table below:

Surface	Emissivity (ϵ)	Solar Absorptivity (α)	α/ϵ
Polished Aluminum	0.04	0.204	5.1
Electrolytic Nickel	0.04	0.28	7.0
410 Stainless, Sandblasted	0.578	0.775	1.34
White Aluminum Oxide Paint	0.98	0.16	0.163
White Lead Carbonate Paint	0.89	0.12	0.135
Rokide A	---	---	0.3

The largest temperature fluctuations in the vehicle will occur during the orbiting of the moon. The surface of the moon is estimated to be about 200F on the side facing the sun and -250F on the dark side. While on the sunlit side of the moon, temperatures will be similar to those estimated for the cislunar flight. However, while passing the unlighted side of the moon, the vehicle will lose heat by radiation to space and to the cold surface of the moon. The 1963 system is required to make only one pass around the moon. Firing of the 6K rockets before entering the circum orbit will result in some heat being transferred through the flameshield into the propulsion compartment, which will help to keep the small lines and valves from reaching temperatures lower than 20F in the period of approximately one hour spent on the dark side of the moon. The metal parts also have heat sink capacity, which, combined with local insulation (less than 10 lb), can be used for this one hour period. Because of the large total heat capacity of the propellants, it is relatively easy to insulate them for a 14 day period for a range of environmental temperatures. This is illustrated by Figure 55, in which tank insulation weights required to prevent freezing or boiling of the fluids are shown for various propellants and compartment temperatures. For a given constant compartment temperature, the

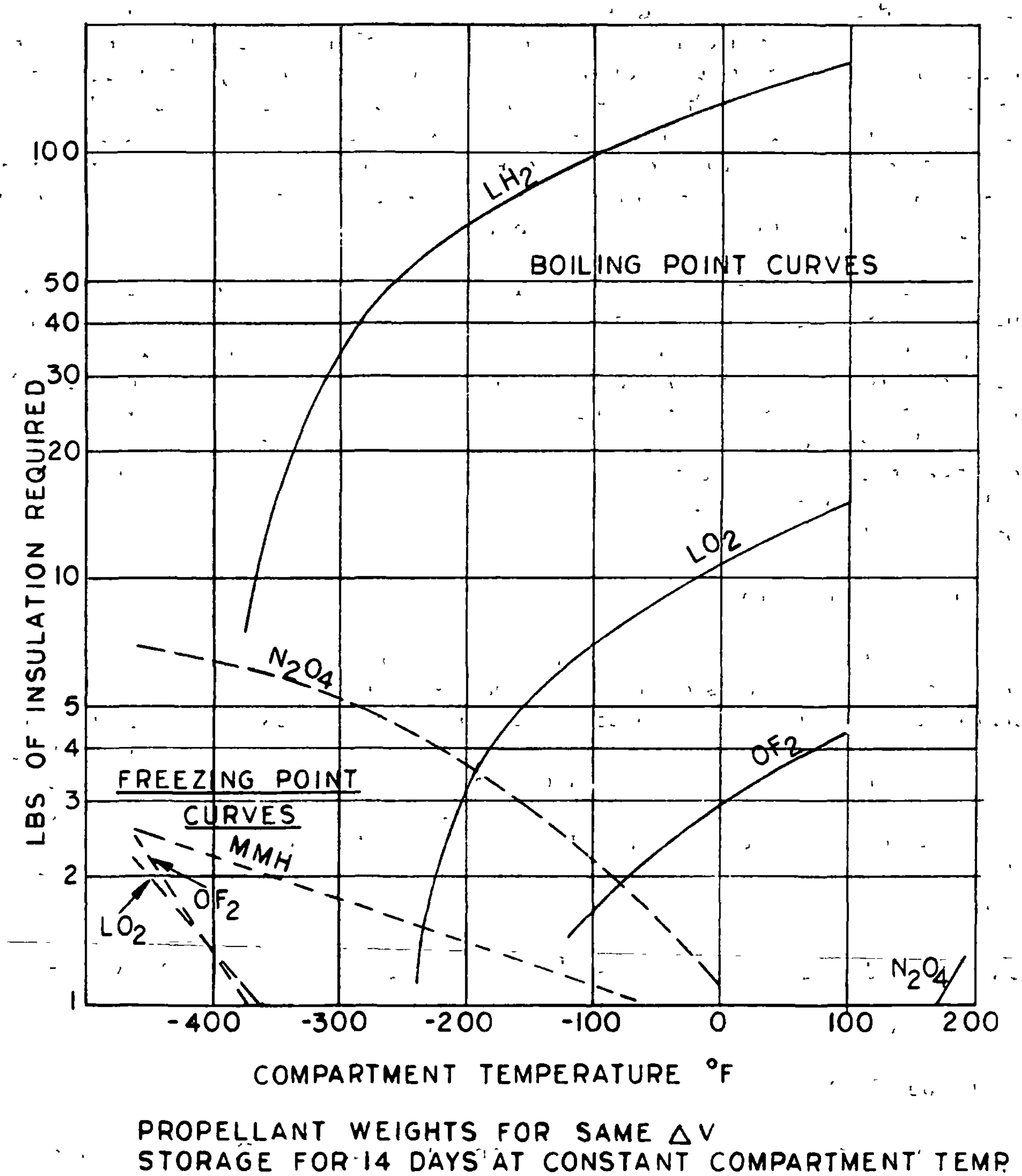


Figure 55. Comparison of Insulation Requirements for Spherical Tanks

intersection with the freezing point curve shows the weight of insulation required to prevent the particular propellant from freezing with the compartment maintained at that temperature for 14 days. Similarly, the intersection of a constant compartment temperature line and a boiling point curve for a particular propellant shows the weight of insulation required to prevent boiling for 14 days. In each case the propellant weight was that required for a 7500 ft/sec ΔV Apollo mission. The propellants were assumed to be loaded at 80F (or their atmospheric boiling point, whichever was lower) and their temperature rise restricted to reach a temperature corresponding to a vapor pressure of 150 psia at the end of the 14 day period.

At this time, detailed information needed to define the compartment temperatures during the 3 to 7 days of lunar orbit has not been available. The temperature variation in the compartment is dependent upon the amount of radiation received from the sun or solar collector, amount of heat radiated into space, and whether or not the compartment skin is used for heat rejection from other compartments. As mentioned above, protection of the propellant tankage is not so much a problem as keeping propellant carrying lines and valves in readiness for firing. The minimum desired temperature for such parts is -63F.

c. Tank Insulation

As shown in Figure 55, the insulation requirements for the two proposed propellant combinations compared to that required for liquid hydrogen are nominal over a wide range of compartment temperatures. Although insulations of higher conductivity could be used with storable propellants without a large weight penalty, comparison is made with cryogenics on the basis of superinsulations. These insulations are actually radiation shielding, composed of layers of highly reflective foil with Fiberglas mat or paper between layers. The conductivities (at vacuum conditions) are given below, along with weight data.

Type of Insulation	k, Btu/hr ft^2 ^{o}F/ft	density, lb-ft^3
Linde SI-4	2.5×10^{-5}	4.7
Linde SI-12	8×10^{-5}	3

The above conductivities are at conditions of 0.1 micron Hg abs and cold surface temperatures of 100R to 200R, and hot side temperatures of about 520R. Thermal conductivity increases, at pressures exceeding 0.1 micron of mercury, and also with hot-side temperature. The maximum allowable operating temperature of these insulations is quite high, about 1000F. When designing insulating jackets for the cryogenics, (LH_2 and LO_2), the effect of conduction losses through tank supporting structure or spacers in double-wall tanks is much more significant than in the insulation of the storables. This is because of the very low total heat transfer which is allowable in storing cryogenics. In the comparison made in Figure 55, an effective insulation conductivity has been used to account for conduction through supports.

d. Heating from Thrust Chambers

Jet boundaries under space conditions have been estimated for the mid-course propulsion chamber. On Figure 56, the boundary at 200 psia (maximum) chamber pressure is shown impinging on the compartment skin with the thrust chamber in its extreme deflected position. At the other extreme (thrust chamber axis parallel to vehicle axis), the jet will not impinge on the compartment wall. Thermal protection of the vehicle skin can be in the form of a low conductivity ablating material such as is used in the nozzle, or by using a high temperature material, such as molybdenum, and radiation-cooling. The former approach is probably more suited to the design as shown because of the location of the small attitude control rockets. However, radiation cooling whenever possible is attractive from the weight standpoint.

Base heating rates, under conditions in which heat transfer is by radiation only, are normally in the order of 5 or 19 Btu/ft^2 sec. Heating rates of this order of magnitude can result in temperature changes of uninsulated components in the compartment of several hundred degrees Fahrenheit over the maximum firing times of 118 seconds. A more detailed study of the base design and temperature conditions over the compartment skin is required to determine how much insulation is required to avoid overheating of components near the flameshield.

The base heating is locally more severe in the region between the nozzles when all four nozzles are firing simultaneously. The interaction of the four jets can create a region of backflow and high base pressure in which the local convective heat transfer rate can be several times the rate with only three jets operating. The degree of base heating depends upon how effective the surrounding structure is as a shroud. Experiments

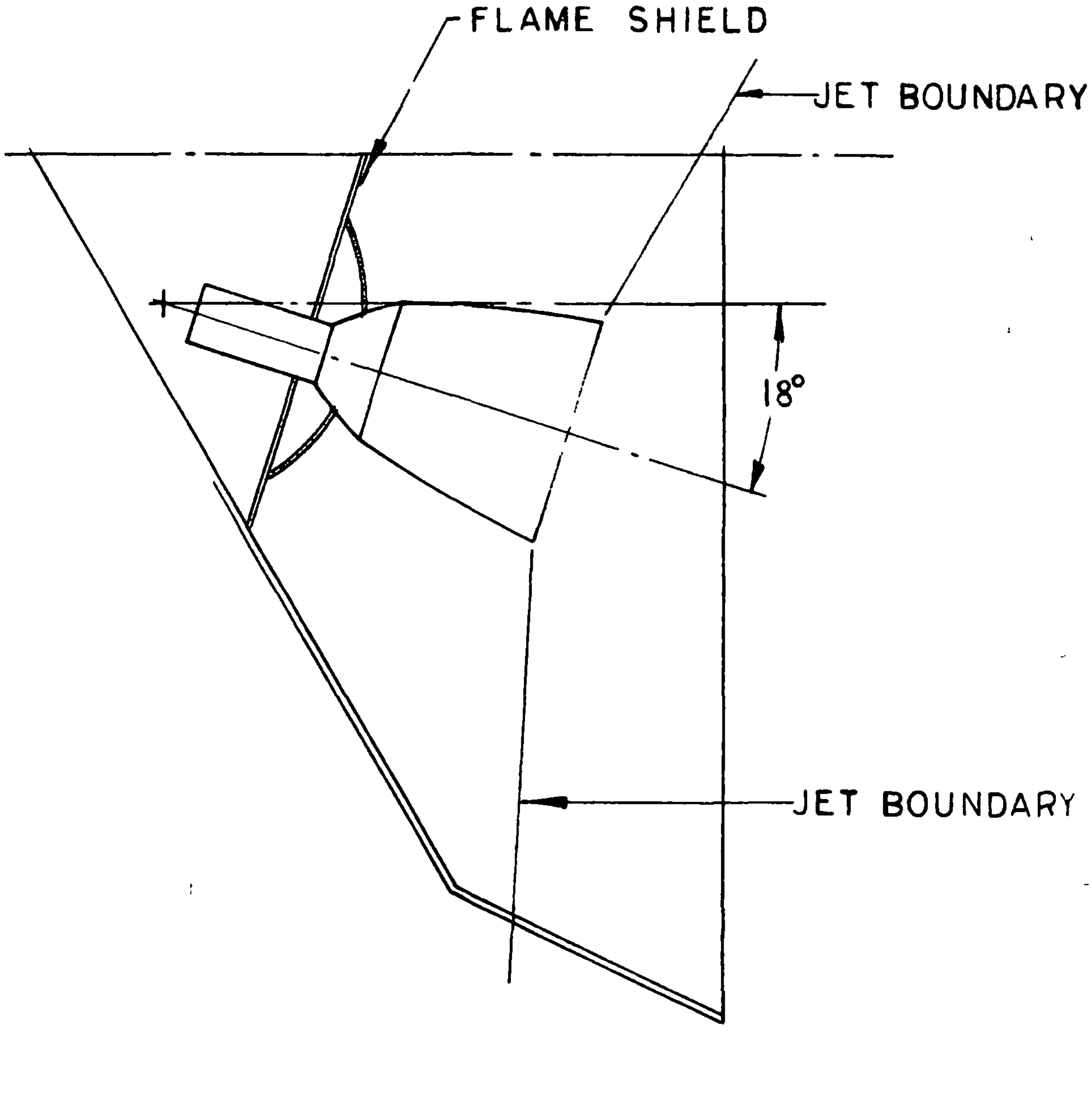


Figure 56. Jet Flaring in Space at 200 psi P_c

on four-nozzle configurations* have shown that this local peak in heat flux can be alleviated with a properly designed shroud. No adequate analysis exists for base heating combined with jet interaction. For the Apollo vehicle the design of an effective shroud will require an experimental program conducted at simulated high altitudes.

The location of the attitude control nozzles in the 1963 system presents a double problem. Protection of the skin, the chambers, and the propellant lines from the impinging jets of the 6K chambers is required. Temperature control of the propellant lines and the valves during the nonfiring periods (which can be as long as 7 days) is difficult because of the thermal isolation of these parts from the tankage compartment. Relocating the small attitude control nozzles closer to the tankage would alleviate these problems. If this is not possible, a means of intermittently heating the valves and lines will be required to eliminate the possibility of icing during the moon orbit.

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* AEDC TN 60-63, "An Evaluation of the Base Heating and Performance of a Missile Model Having Four Rocket Exhausts and a Single Base Shroud", LeRoy Barnes, J. R. Parker (Confidential), Arnold Engineering Development Center, April 1960.

4 Multiple Engine Trade-off

In a complex system the reliability of components, subsystems, and systems plays a major roll in the evolution of the overall system. If reliability is low, the probability of multiple malfunctions becomes significant, which of course increases the risks encountered by the flight crews. Therefore, to achieve safety, high reliability is an absolute necessity in complex systems. Figure 57 demonstrates the importance of reliability and its influence on multiple malfunctions. In this figure, the probability of the number of malfunctions as a function of component reliability and complexity is presented. When the probability of multiple malfunctions is significant, then redundancy becomes a useful tool in their reduction.

In the Apollo program, where safety and reliability are paramount, the case of engine redundancy can represent an important element in the achievement of both safety and reliability. Specifically, the 1963 main propulsion system incorporates four engines. This affords redundancy in performing orbital change requirements, but results in weight and overall performance penalties If two engines were utilized with step thrust capabilities, it would be possible to perform both orbital and abort requirements. The use of two engines would result in a weight saving and performance gain. However, the safety and reliability aspects of four engines versus two engines must be considered in detail to determine what penalty in safety one may expect to pay for the use of two engines.

There are two functions required from the main midcourse propulsion system, orbital change and abort. Each function must be considered separately for the case of four engines and two engines.

a. Orbital Change

To accomplish an orbital change with four engines, any one of the four engines is required to function, therefore the probability of performing an orbital change can be expressed as follows:

$$R_o = 1-(1-R_1)^4$$

where R_1 is the reliability of a single engine.

If one assumes a reliability of 90% for each engine for illustrative purposes only, we find that the probability of performing the orbital change function

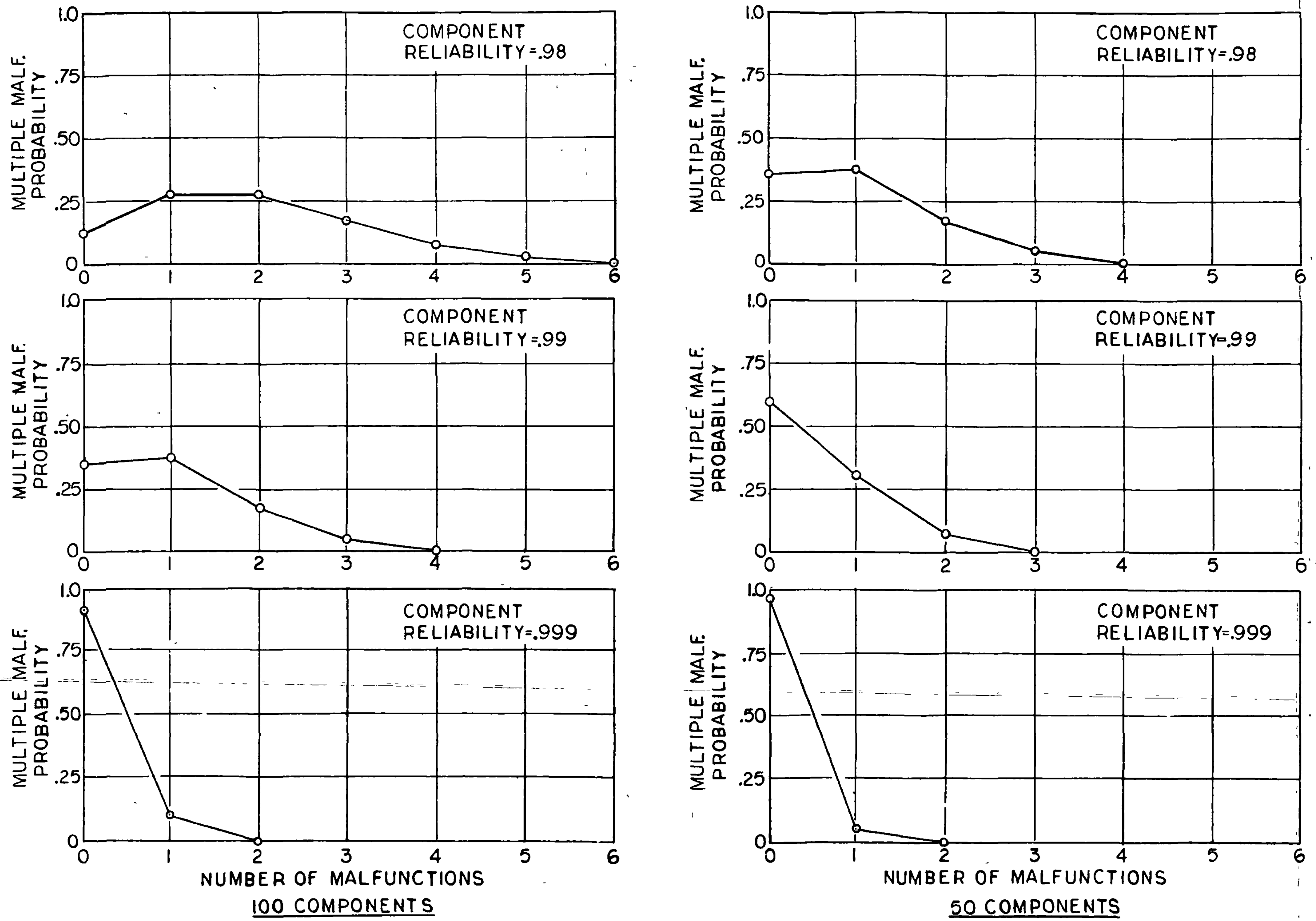

Figure 57. Probability of Multiple Malfunctions

is 99 99%. It can be seen that a quadruple malfunction is required before the orbital change function is destroyed.

With two engines, each engine is designed for step-thrust operation, that is, the engine is capable of 6000 or 12,000 lb thrust. This added requirement increases the engine complexity thereby reducing reliability. For illustrative purposes only, assume the engine reliability for the two engine configuration to be 89%. Therefore, the probability of performing the orbital change is:

$$R_o = 1-(1-R_1)^2$$

or 98.79%. In comparison, the four engine configuration offers greater safety and reliability in performing the orbit change function.

b. Emergency Escape (Abort)

With four engines, the case of the abort function will depend directly upon the thrust level required or the number of engines required to function to achieve a safe abort. The mathematical expression for the four engine configuration can be given as follows:

$$R_1^4 + 4R_1^3q + 6R_1^2q^2 + 4R_1q^3 + q^4 = 1$$

where

R_1^4 = the probability of no malfunction or reliability of firing all four engines.

$4R_1^3q$ = probability of a single malfunction

$6R_1^2q^2$ = probability of a double malfunction

$4R_1q^3$ = probability of a triple malfunction

q^4 = probability of a quadruple malfunction

If one assumes a reliability of 90% for each engine then the following table applies:

Number of Malfunctions	Term	Thrust Level (lb)	Probability of Occurrence (%)	Probability of Obtaining Specified Thrust or More (Cumulative Probability %)
0	R_1^4	24,000	65.61	65.61
1	$4R_1^3q$	18,000	29.16	94.77
2	$6R_1^2q^2$	12,000	4.86	99.63
3	$4R_1q^3$	6,000	0.36	99.99
4	q^4	0	0.0001	100.00

The mathematical expression for the two engine configuration can be given as follows:

$$R_1^2 + 2R_1q + q^2 = 1$$

where

R_1^2 = the probability of no malfunctions or reliability of firing two engines.

$2R_1q$ = probability of a single malfunction

q^2 = probability of a double malfunction

If one assumes a reliability of 89% for each engine, the following table applies:

Number of Malfunctions	Term	Thrust Level (lb)	Probability of Occurrence (%)	Probability of Obtaining Specified Thrust or More (Cumulative Probability %)
0	R_1^2	24,000	79.21	79.21
1	$2R_1^2q$	12,000	19.58	98.79
2	q^2	0	1.21	100.00

From the above charts and Figure 58 it can be seen that the level of thrust required to attain a successful abort will dictate which configuration would be more applicable to the abort function. If 24,000 lb of thrust were absolutely required, then the two engine configuration represents a better approach, however, once malfunctions are considered, the four engine system represents greater safety and reliability than that achieved with the two engine configuration. From an overall safety and reliability aspect it is more realistic to choose the four engine configuration. Safety cannot be compromised to achieve performance and weight gains. However, with increasing basic engine reliability (e.g. 95%) the differences between the two and four engine systems become relatively insignificant, permitting performance and weight consideration to play a larger role in determining the final approach. Since we have predicted an engine reliability of greater than 95% (Section II), the latter considerations may well prevail.

5. Information Display

Presentation of information to the pilot must satisfy three requirements. First, the pilot needs information which allows him to make normal decisions in which his judgment is a necessary part. Second, he needs information to maintain his psychological assurance of proper performance. Last, he needs information to evaluate an emergency situation and make a fast, accurate decision.

The optimum system in manned space flight is a well-integrated combination of man and machine which uses each to maximum capability to insure manned safety and complete the mission successfully.

Therefore, the above requirements must be defined by analyzing first the man-machine functions to determine which can be performed most effectively by the man and which by the machine. This is done by listing the controls which will be required to fulfill the mission of the system, evaluating each to decide on the most logical and expeditious manner of actuation, assigning certain controls to the pilot, and then establishing the data necessary to enable him to make the proper decision.

The decision of assigning controls to the man or to the machine depends on whether the task requires human intelligence, and whether it is within the scope of pilot capabilities of acquisition, response, actuation time, resolution, accuracy, and mental availability. If the pilot is physically incapable of performing a task, or if he is not available, then the function must be delegated

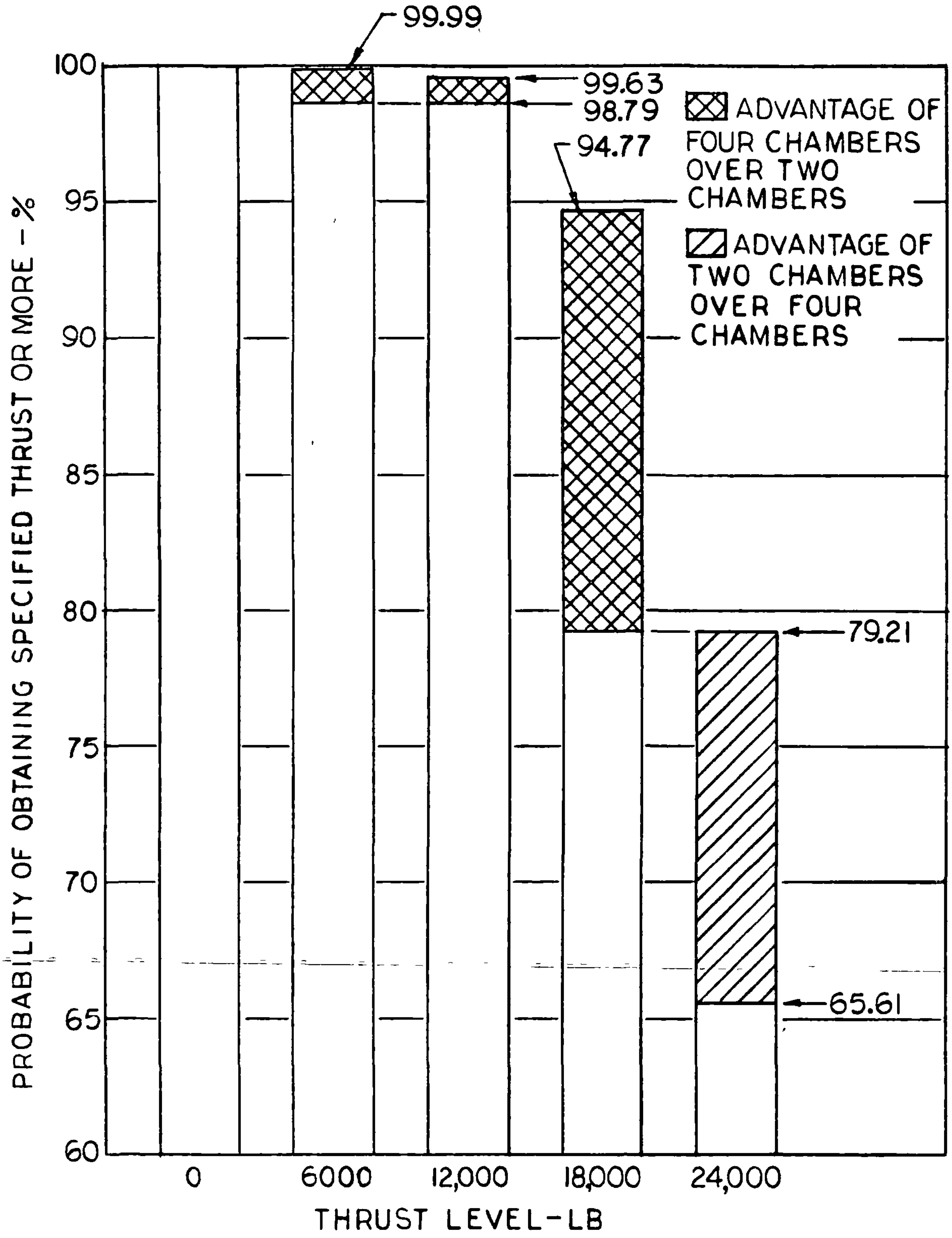

Figure 58. Probability of Obtaining Specified Thrust or More

to the machine, or the pilot must be dualized, that is, a copilot must be utilized. Similarly, if automatic equipment is unable to handle a function, or if it is performing another function when needed, then the task must be assigned to the pilot or duplicate equipment must be provided.

When the pilot controlled functions have been established, the parameters to be measured and monitored must be selected. This is accomplished by synthesizing the system equation and examining each variable to determine its amplitude and range. Then the variable must be investigated to decide whether direct measurement or computation from other variables will provide the simplest, most reliable operation. Next, all possible methods for measuring the desired variables must be investigated and judged. Thus, the data for pilot presentation are established.

In addition, the system must be analyzed to find the effect of a single malfunction of each component in the system (one at a time), determine if and what corrective action is required, and present information to the pilot to enable him to make the necessary correction.

Several modes of information presentation must be considered: binary (on-off indicator), analogue (meter or gauge), digital (counter), and display panel (cathode ray tube). The selection depends on the type of information required: whether it is quantitative or qualitative, what accuracy, amplitudes, rate of change, response, etc. are demanded. When the selections are made they must be "human engineered" into a display which satisfies the three original requirements with maximum simplicity and a negligible chance of error.

Figure 59 illustrates the engine instrumentation system which embodies the above philosophy. Four types of visual display indications are presented as inputs to the pilot (decision maker). Two pilot outputs are possible: changing the stored program to continue automatic control at a new level, or manual operation of those controls necessary either to restore normal system functioning or to abort the flight safely.

The mission of the system under discussion is to complete a round trip, lunar orbiting, manned space flight To accomplish this mission, there are several functions which must be performed by the man-machine combination. The mission has been divided into phases and the functions common to each phase grouped, so that they could be analyzed in a logical, sequential pattern.

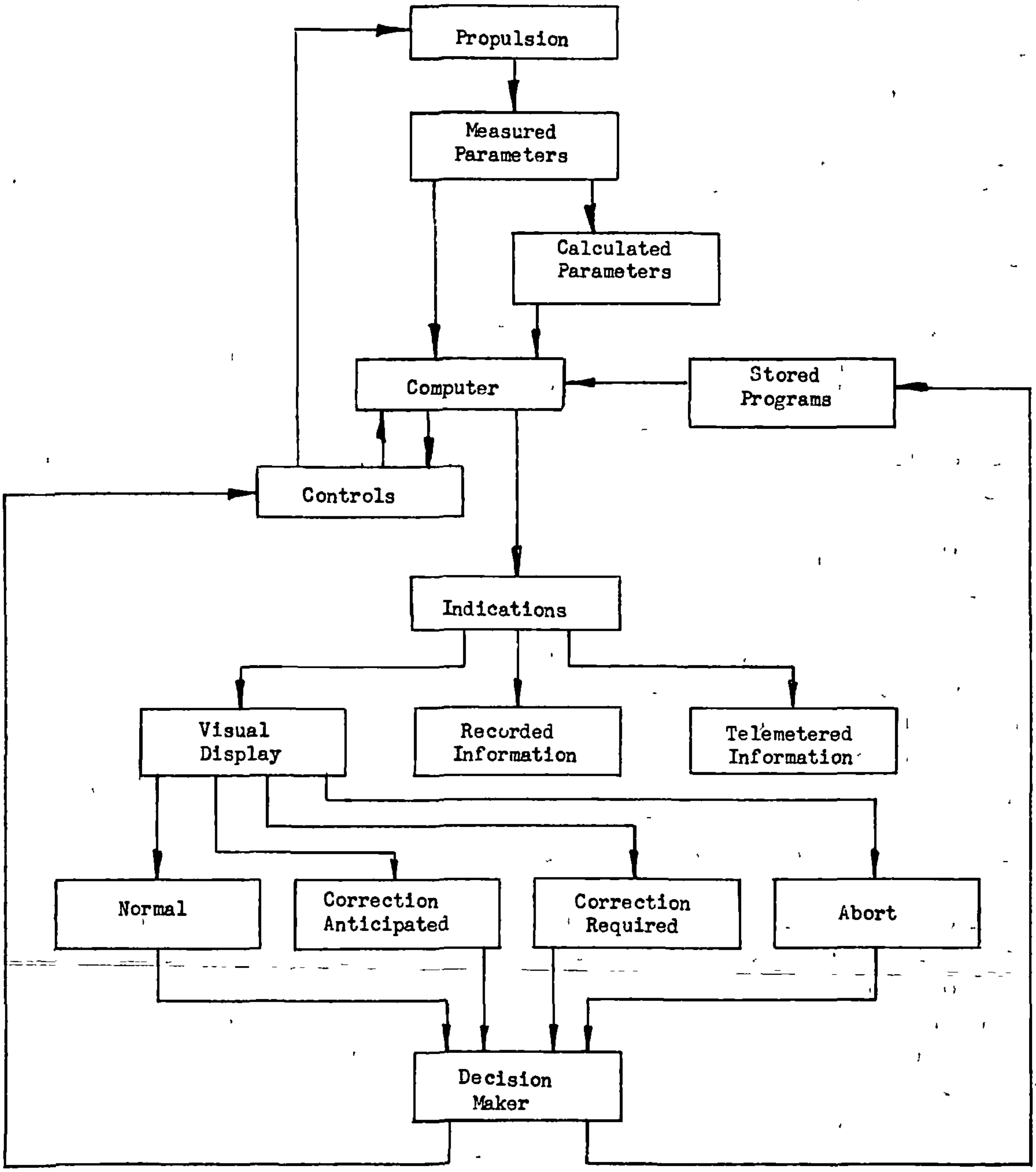

Figure 59. Block Diagram-Rocket Engine Instrumentation

Each function imposes certain requirements which must be communicated to the pilot. This is done with a display. The display must present the information needed by the pilot to evaluate the system operation and to take corrective action if necessary. It is achieved by analyzing each requirement on the basis of the display methods available--binary, analogue, digital, and panel--and selecting the method into which the requirement logically falls, or which is judged to be the simplest, most effective presentation.

The enclosed chart (Table XIII) lists the phases of operation, the functions common to each, the requirements for each function, and the method of display selected. The following discussion amplifies the material presented on the chart.

During the prelaunch phase, the only information required by the passengers is a warning of the impending launch so that they will be prepared for the acceleration to which they will be subjected. The requirement is similar whether a normal launch is to occur or an abort firing is signaled. Voice communication will exist between passengers and ground control. A normal countdown will proceed, subject to interruption by the pilot if it appears to be justified. Should an abort firing be initiated, both a visual and an auditory signal will be flashed to each man so that each has the earliest possible warning of this unscheduled event.

The pilot's station will be the master display station. However, for critical operations such as prelaunch and launch, where immediate action is mandatory, triplicate displays must be provided to give adequate information to all. For example, if a manual abort is necessitated by a booster failure during the high acceleration launch phase, the pilot may be unavailable physically or mentally. Therefore, the other passengers are given the information and controls to perform this function.

In a normal launch phase, the pilot monitors the attitude control system by means of a bearing deviation indicator (BDI) type cathode ray tube (CRT) which is designed to present pitch, yaw, and roll simultaneously, referenced to an artificial horizon. Three counters supply identical information in digital form. Position and velocity are indicated on a plan position indicator (PPI) type CRT, which places the earth at the center of the screen with concentric circles as a distance scale, and contains two moving pips. One pip represents vehicle position and the other pip is the reference position. As long as the two pips coincide, the programmed position and velocity are being maintained. Counters provide more accurate numerical information on both

TABLE XIII

INFORMATION DISPLAY CHART

Phase	Function	Type of Information Presented to Pilot	Display
1 Prelaunch	a Prepare to launch	Warning	Digital (auditory)
	b Fire abort rockets (if required)	Warning	Binary (light plus horn)
2 Launch	a Fire booster	Warning	Binary (light)
	b Control attitude	Monitoring	Digital (counters) Panel (BDI scope)
	c Fire abort rockets (if required)	Warning	Binary (light plus horn)
	d Attain escape velocity	Monitoring	Analogue (gauge) Panel (PPI scope) Binary (light)
	e Jettison abort rockets	Monitoring	Binary (lights)
	f Shut down booster	Monitoring	Binary (light)
	g Fire separation rockets	Monitoring	Binary (lights)
3 Midcourse	a Control attitude	Attitude Monitoring	Digital (counters) Panel (BDI scope)
		Propellant Monitoring	Analogue (gauge)
	b Control position	Position Monitoring	Digital (counters) Panel (BDI scope plus range scope)
		Propellant Monitoring	Analogue (gauge)
		Engine Monitoring	Binary (lights)
	c Abort (if required)	Monitoring	Binary (light plus horn)
	d Advance to next phase	Monitoring	Binary (light)
4 Lunary orbit injection	a Control attitude	Same as 3	Same as 3
	b Control position	Position Monitoring	Digital (counters) Panel (PPI scope)
		Propellant Monitoring	Analogue (gauge)
		Engine Monitoring	Binary (lights)
	c Abort (if required)	Same as 3	Same as 3
	d Advance to next phase	Same as 3	Same as 3
5 Lunar orbit	a Control attitude	Same as 4	Same as 4
	b Control position	Same as 4	Same as 4
	c Abort (if required)	Same as 4	Same as 4
	d Advance to next phase	Same as 4	Same as 4
6 Lunar disorbit	a Control attitude	Same as 4	Same as 4
	b Control position	Same as 4	Same as 4
	c Abort (if required)	Same as 4	Same as 4
	d Advance to next phase	Same as 4	Same as 4
7 Midcourse return	a Control attitude	Same as 3	Same as 3
	b Control position	Same as 3	Same as 3
	c Abort (if required)	Same as 3	Same as 3
	d Advance to next phase	Same as 3	Same as 3
8 Re-entry	a Control attitude	Attitude Monitoring	Digital (counters) Panel (BDI scope)
		Propellant Monitoring	Analogue (gauge)
	b Control position	Position Monitoring	Digital (counters) Panel (PPI scope)
		Propellant Monitoring	Analogue (gauge)
	c Separate main propulsion	Monitoring	Binary (lights)
	d Separate nose	Monitoring	Binary (lights)
	e Control capsule roll	Monitoring	Digital (counters) Panel (BDI scope)
	f Operate decelerator	Monitoring	Binary (lights)

predicted and actual distance and velocity. Lights are included to indicate that booster cutoff velocity has been reached, that the booster has been shut down, and that the abort rocket jettisoning and separation rocket firing should be initiated and have been completed.

The same attitude control display is utilized during the midcourse phase and all the following phases. A dual pointer gauge furnishes data on attitude control propellant supply versus the amount required to complete the mission. A similar gauge indicates the status of propellants for the main propulsion engines.

The position control display for the midcourse and midcourse return phases consists of two CRT presentations. One scope indicates predicted and actual distances between earth and moon by means of two pips on a narrow rectangular raster, suitably calibrated, with earth and moon at the extremes The second scope is a BDI type which supplies bearing information relative to the predicted course line. Counters are again provided to give parallel information to that on each scope. A light signals the termination of the phase. Other lights indicate malfunction of one of the propulsion engines, and an auxiliary light panel locates the source of trouble so that remedial action may be taken if possible.

The same display information is utilized for the lunar orbit (injection, orbiting, and disorbiting) phases except that the rectangular position scope is replaced by a PPI scope as used during the launch phase. In this instance the moon is at the center of the scope, and concentric rings indicate orbital distances from it. Two pips show predicted and actual positions.

The final phase is re-entry. Attitude and position control displays are the same as for the lunar phases except that on the PPI scope the earth is at the center. Light indicators signal the time at which to separate the main propulsion engines and the completion of this operation. Another pair of lights furnishes similar information on nose separation, and a third pair indicates the operation of the decelerating mechanism. After separation the attitude control includes roll only. This is presented on the BDI scope used before separation.

The primary function of the pilot throughout the mission is that of monitoring. He acts as the brain behind the brain. The mechanical brain or computer receives the stored program outputs and the measured and calculated parameter outputs and translates these into signals which are usable by the control system in maintaining the intended system performance. The pilot, by utilizing the display presented to him, evaluates the performance of the computer, exercises his judgment to decide whether action on his part is required, and, if so, assumes control over those portions of the system which he deems necessary.

The abort signal is an analysis by the computer that the mission cannot be completed as intended and that it has been aborted or should be aborted by the pilot. However, with the exception of certain critical conditions where abort must be initiated suddenly (such as booster malfunction during the pre-launch phase), the pilot should be able to interpret his display so that he anticipates the abort signal. Thus, he may initiate the abort in advance of the computer or take corrective action which eliminates the need to abort.

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VI. DEVELOPMENT PROGRAM

A. PROGRAM SCHEDULE, 1963 SYSTEM

Phase I - Design and Development

Component design and development will run for 12 months. At the beginning of this phase, final designs of the thrust chambers, attitude and roll control chambers, and thrust chamber propellant valves will be prepared. These designs will take advantage of the N_2O_4/MMH firings of the vortex injectors and the IR and D bipropellant pulse rocket testing, as well as experience gained on the Surveyor vernier powerplant program and other related programs currently underway. Designs for the propellant tanks and pressurizing gas storage tanks will be prepared in accordance with the installation and capacity requirements established by the design parameters and by consideration of meteorite penetration resistance and thermal control. Propellant tanks expulsion bladders will be procured from the Joclin Company or another qualified bladder manufacturer. Pressurization system components will be ordered using existing specifications, or modified specifications, where appropriate. The basic program is shown in Figure 60.

1. Thrust Chamber

Individual thrust chamber testing will commence at the start of the fourth month of the program. Sea level firings of the 6K chambers without nozzle extension will be performed to confirm performance (C*), stability, heat transfer, and durability. The equivalent of 40 runs, averaging 100 seconds duration, will be accomplished. An additional 20 runs, averaging 100 seconds, will be made so that at least two thrust chambers achieve a total of over 500 seconds operating time to demonstrate adequate lift. An additional 40 runs will be accomplished with the exhaust nozzle extension added to check space performance, durability of extension under ablation cooling conditions, and durability of extension-chamber joint. High altitude exhaust for the nozzle will be provided. Multiple starts and their effect on the ablation liner will be checked. At least two thrust chambers will be started a minimum of 14 times under very high altitude conditions and will achieve a total of over 500 seconds operating time each to demonstrate adequate ignition and liner life.

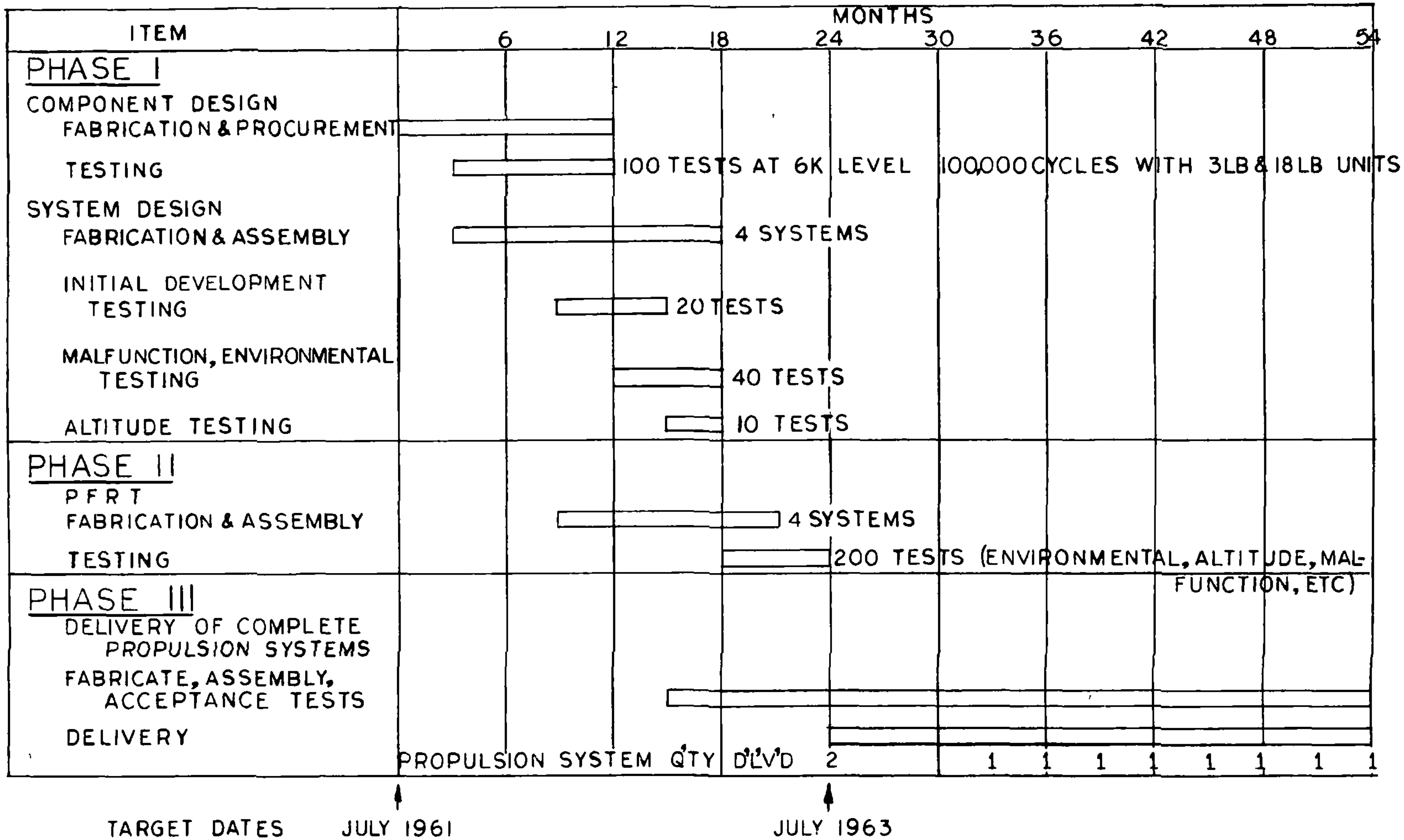


Figure 60. Program Schedule, Apollo Propulsion System, 1963

The 3 lb and 18 lb attitude control rockets will be tested for a total of 100,000 cycles averaging one second each. All testing will be done under high altitude simulation with flight type solenoid valves. Runs of up to 60 seconds duration on the 3 lb rockets will be made to establish durability. Testing in the clusters of three (both individually and with more than one unit firing) will be carried out to insure adequate radiation cooling with no cross-effects on adjacent chambers and valves.

2. Gas Storage Tanks

Gas storage tanks will be proof and burst tested to insure adequate strength and quality control. Loss of pressurization gas under high vacuum will be determined. Propellant tank and bladder assemblies will be checked for uniformity and completeness of expulsion at design pressure level and with a variety of attitudes. The effect of propellant sloshing and simulated chamber vibration on expulsion and bladder life will be investigated. The cycle life of the bladders expelling actual propellants will be established. The effect of simulated space radiation on the bladder expulsion capability, flexibility, and permeability will be investigated. The effect of the thermal environment on propellant storage will be established.

3. Components

Testing of the propellant solenoid valves, helium fill valve, regulators, relief valves, and other pneumatic and squib operated components will commence during the third month of the program. During initial testing of the propellant solenoid valves emphasis will be placed on:

a. Actuation reliability of the valves over the anticipated temperature range

b. Leaktightness of the valve when its environment and downstream port are exposed to very high vacuum and propellant pressure is maintained at the valve inlet.

c. Effect of space radiation on valve seals and other parts.

Testing and development of the valves will be continued as required to insure reliable leaktightness under the anticipated mission conditions. For this purpose leaktightness will be defined as leakage having a negligible effect upon ΔV during the 14 day trip time. Simulated missions will be made with the valves stored for 14 days in high vacuum, sealing propellants and periodically actuating according to a simulated flight plan.

The pressurization system components will be checked for operational characteristics with helium where the data are not available, e.g., leakage, pressure drop, and other effects. The effect of extended exposure to high vacuum and space radiation will be established. Simulated missions will be made.

4. System Design and Development

System design and development will begin during the fourth month and run for some 15 months. During this phase, system design and functional studies will be carried out in close liaison with the General Electric Company MSVD to interchange information and insure optimum configurations. This portion of the program is designed to insure that assembly of the entire system will not result in unsuspected system difficulties. Initial testing will be made of the complete system installations. A total of 20 tests is planned to begin in the tenth month. Some tests will be at each of the 20F and 80F temperature extremes. During all of these runs, high altitude exhaust conditions will be provided for the 6K chambers. During some of these runs an actual thrust-time mission profile will be simulated with the tanks and components exposed to a high vacuum environment. At least two of these 20 tests will be at the full 24K thrust value. The tests will be of progressively increasing complexity and will be designed to integrate all of the components into the system and to provide for changes in design as the need arises.

Prior to the PFRT phase, a series of complete system tests will be made to provide assurance that the flight configurations will function reliably and satisfy the model specification. The testing will feature environmental, malfunction, and altitude testing. A total of 50 tests is planned for this portion of the systems development test program. Forty tests will be conducted at Thiokol-RMD. These tests will be made first at the 20F and 80F temperature extremes. High altitude exhaust conditions will be used to insure full flow in the nozzle. Simulated thrust-time, high vacuum space environment for the tankage and components, and gimbaling programs will be followed. During some of these tests the full 24K thrust will be generated.

The remaining ten tests will be conducted in the Tullahoma high altitude test facilities. The three propulsion systems will be mounted as in the spacecraft and Tullahoma tunnel pressure set to simulate the highest possible altitude. Checkout runs will be made to investigate ignition, component leakage, and the problems, if any, resulting from base recirculation of exhaust gases. The 14 day mission will be simulated as closely as deemed practical at the Tullahoma facility. Runs will then be made simulating the actual mission as closely as possible. At least two tests will be made at the full 24K thrust.

All of the runs will be made under carefully controlled conditions. Servicing of the powerplants will be required between runs. Complete data will be taken and reviewed, and maintenance and test malfunction records will be kept. Every effort will be made to prepare the basic systems for the exacting jobs ahead - PFRT and actual flight missions.

Phase II - PFRT

The PFRT program will cover some 15 months, beginning during the ninth month and ending in the 24th month. The testing program will occupy the last six months of this period. The PFRT will be a carefully conducted testing program completed in accordance with a PFRT test program report to be approved beforehand. It is anticipated that approximately 200 tests will be needed to complete the entire program, which is aimed at establishing the reliability and safety aspects of the propulsion systems. The testing will include environmental, vibration, malfunction, and altitude testing. The largest single group of tests (between 80 and 100) will be used to demonstrate malfunction safety. Various malfunctions will be simulated and the reaction of the powerplant determined. These tests will measure the effectiveness of the malfunction studies. Penalty runs will be conducted when necessary and additional malfunction runs made as the need becomes evident.

Altitude tests will be run at the 6K and 24K level at the Tullahoma facilities. The longest practical mission simulation will be made, and thrust, gimbaling, and attitude control programs included. It is planned to conduct 10 tests at Tullahoma.

The remaining tests will be scheduled and conducted to demonstrate compliance with the model specification and other applicable specifications. They will include propulsion system tests and component tests. In all cases they will be conducted in accordance with written specifications covering requirements, servicing (if necessary), inspection, etc.

Phase III - Delivery

Powerplant deliveries will commence during the 25th month of the program. Delivery of ten complete propulsion systems will be made over a period of two and one half years in accordance with the schedule of Figure 60. Following a complete inspection of all parts (including pressure checks and flow checks, as required), the propulsion system will be assembled to simulate vehicle installation and then acceptance tested in a manner agreed upon with the General Electric Company. An acceptance test should include a short 10 second run with one 6K chamber, followed by a two hour shutdown period with the system propellant tanks pressurized, the complete system exposed to simulated space vacuum environment, and the main attitude control system cycling periodically. The test will then conclude with 10 seconds of operation at full 24K thrust. The gimbaling system shall be checked for operation and the re-entry roll control system cycled to insure proper functioning. This testing will be conducted with simulated high altitude exhaust to insure full flow in the nozzles.

Following completion of the acceptance test, the propulsion systems will be drained, serviced as required for flight operation, and shipped dry to a point to be determined. During the course of the development program, consideration will be given to the requirements of a prelaunch checkout for electrical and pneumatic equipment. A schedule of spare parts to be supplied will be established during this phase.

B. PROGRAM SCHEDULE, 1966 SYSTEM

Phase I - Applied Research, Design and Development

Prior to initiation of formal design and development, and concurrent with development of the 1963 system, applied research in performance, system requirements, and materials will be conducted. At the beginning of this phase, 100 lb thrust chambers will be designed based on the experience gained in current Thiokol-RMD IR and D programs utilizing vortex injectors and OF_2/MMH. Basic programs studying performance (C*), stability, and cooling will be conducted. Based on these data the program will be increased to 500 lb thrust and finally to 6K thrust levels, broadening in scope to encompass regenerative and ablative cooling of the chamber and throat insert, film cooling of the throat insert with liquid helium and fuel, radiation cooling of nozzle extensions, and other aspects of combustion research. Materials research will parallel this effort. Material compatibility, metallic and nonmetallic, and material choice as related to specific designs will be studied. The basic program is shown in Figure 61.

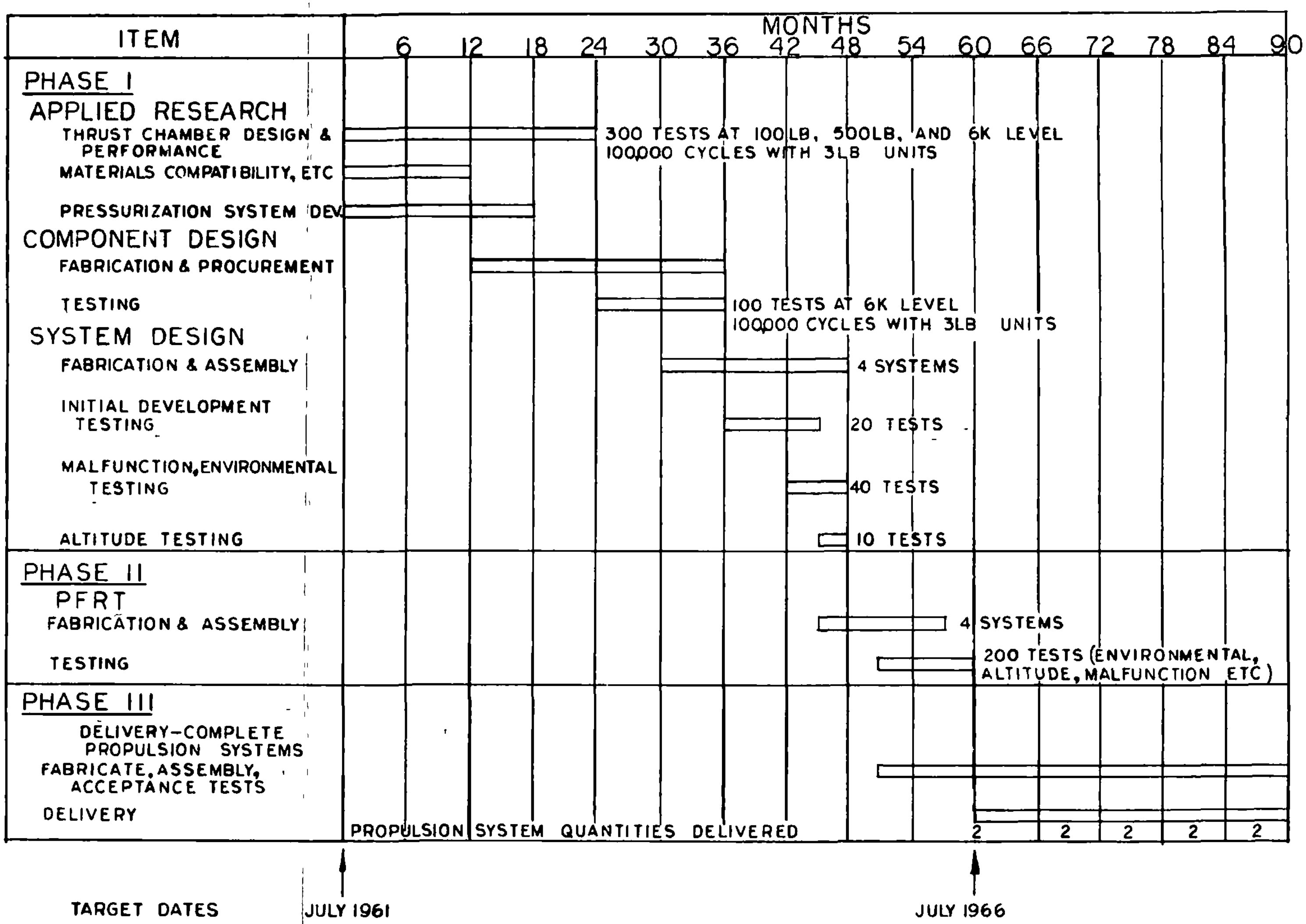


Figure 61. Program Schedule, Apollo Propulsion System, 1966

Pressurization system investigations will follow two paths, (1) a liquid helium system utilizing a heat exchanger, and (2) a hybrid chemical gas generation system producing an oxidizer-rich and a fuel-rich gas for pressurizing the respective main propellant tanks. Two checkpoints in this program are anticipated. At the end of six months the liquid helium program will be reviewed with a view towards incorporating it in the 1963 system in place of the stored gas system. At the end of 12 months the liquid helium and the hybrid systems will be compared with a view towards choosing between them for the 1966 system. As a part of the pressurization system studies, positive expulsion of the OF_2 will be investigated. Both nonmetallic and metallic bladders will be tested, as well as other devices.

Component design and development will run for 24 months. At the beginning of this phase final designs of the thrust chambers, attitude control chamber, and thrust chamber propellant valves will be prepared. These designs will be based on the OF_2/MMH basic data generated during the first year of the program. Designs for the pressurizing gas storage tanks will be prepared in accordance with the installation and capacity requirements established by the design parameters and by consideration of meteorite penetration resistance and thermal control. It is anticipated that the main propellant tanks and expulsion bladders developed for the 1963 system can be used here. Consideration will be given to thermal control relative to extended storage on space of OF_2. If other propellant acquisition means are to be employed, the tank designs will be modified accordingly. Pressurization system components will be ordered using existing specifications developed under the 1963 system program and based on the pressurization system found to be most promising during the research phase.

1. Thrust Chamber

Individual thrust chamber testing will commence at the start of the 24th month of the program. Sea level firings of the 12K chambers without nozzle extension will be performed to confirm performance (C*), stability, heat transfer, durability, and step-thrust capability. The equivalent of 40 runs, averaging 100 seconds duration, will be accomplished. An additional 20 runs, averaging 100 seconds, will be made so that at least two thrust chambers achieve a total of over 500 seconds operating time at the 6K level to demonstrate adequate life. At least two chambers will be operated for over 120 seconds continuous running at the 12K level. An additional 40 runs will be accomplished with the exhaust nozzle extension added to check space performance and durability of extension under radiation cooling conditions. High altitude exhaust for the nozzle will be provided. Multiple starts and their effect on the extension will be checked. At least two thrust chambers will be started a minimum of 14 times under very high altitude conditions and will achieve a total of over 500 seconds operating time at the 6K level each to demonstrate adequate ignition and chamber life.

The 3 lb attitude control rockets will be tested for a total of 100,000 cycles averaging one second each. All testing will be done under high altitude simulation with flight type solenoid valves. Runs of up to 60 seconds duration on the 3 lb rockets will be made to establish durability. Testing in the clusters of three (both individually and with more than one unit firing) will be carried out to insure adequate radiation cooling with no cross-effects on adjacent chambers and valves.

2. Storage Tanks

Gas and liquid helium storage tanks will be proof and burst tested to insure adequate strength and quality control. Loss of pressurization gas under high-vacuum will be determined. Liquid helium vaporization losses under simulated flight conditions will be established. Tank and bladder assemblies will be checked for uniformity and completeness of expulsion at design pressure level and with a variety of attitudes. The effect of propellant sloshing and simulated chamber vibration on expulsion and bladder life will be investigated. The cycle life of the bladders expelling actual propellants will be established. The effect of simulated space radiation on the bladder expulsion capability, flexibility, and permeability will be completely investigated.

3. Components

Testing of the propellant solenoid valves, helium fill valve, regulators, relief valves, and other pneumatic and squib operated components will commence during the 18th month of the program. During initial testing of the propellant solenoid valves emphasis will be placed on:

a. Actuation reliability of the valves over the anticipated temperature range

b. Leaktightness of the valve when its environment and downstream port are exposed to very high vacuum and propellant pressure is maintained at the valve inlet

c. Effect of space radiation on valve seals and other parts.

It is anticipated that a considerable amount of effort from the 1963 program will be applicable here. Testing and development of the valves will be continued as required to insure reliable leaktightness under the anticipated mission conditions. For this purpose leaktightness will be defined as leakage having a negligible effect upon ΔV during the 14 day trip time. Simulated missions will be made with the valves stored for 14 days in high vacuum, sealing propellants and periodically actuating according to a simulated flight plan.

The pressurization system components will be checked for operational characteristics with gaseous and liquid helium where the data are not available, e.g., leakage, pressure drop, and other effects. The effect of extended exposure to high vacuum and space radiation will be established. Simulated missions will be made.

4. System Design and Development

System design and development will begin during the 24th month and run for some 24 months. During this phase, system design and functional studies will be carried out in close liaison with the General Electric Company MSVD to interchange information and insure optimum configurations. This portion of the program is designed to insure that assembly of the entire system will not result in unsuspected system difficulties. Initial testing will be made of the complete system installations. A total of 20 tests is planned to begin in the 36th month. Some tests will be at each of the 20F and 80F temperature extremes. During all of these runs, high altitude exhaust conditions will be provided for the 12K chambers. During some of these runs an actual thrust-time mission profile will be simulated with the tanks and components exposed to a high vacuum environment. At least two of these 20 tests will be at the full 24K thrust value. The tests will be of progressively increasing complexity and will be designed to integrate all of the components into the system and to provide for changes in design as the need arises.

Prior to the PFRT phase a series of complete system tests will be made to provide assurance that the flight configurations will function reliably and satisfy the model specification. The testing will feature environmental, malfunction, and altitude testing. A total of 50 tests is planned for this phase. Forty tests will be conducted at Thiokol-RMD. These tests will be made first at the 20F and 80F temperature extremes. High altitude exhaust conditions will be used to insure full flow in the nozzle. Simulated thrust-time, high vacuum space environment for the tankage and components, and gimbaling programs will be followed. During some of these tests the full 24K thrust will be generated.

The remaining ten tests will be conducted in the Tullahoma high altitude test facilities. The three propulsion systems will be mounted as in the spacecraft and the Tullahoma tunnel pressure set to simulate the highest possible altitude. Checkout runs will be made to investigate ignition, component leakage, and the problems, if any, resulting from base recirculation of exhaust gases. The 14 day mission will be simulated as closely as deemed practical at the Tullahoma facility. Runs will then be made simulating the actual mission as closely as possible. At least two tests will be made at the full 24K thrust.

All of the runs will be made under carefully controlled conditions. Servicing of the powerplants will be required between runs. Complete data will be taken and reviewed and maintenance and test malfunction records will be kept. Every effort will be made to prepare the basic systems for the exacting jobs ahead - PFRT and actual flight missions.

Phase II - PFRT

The PFRT program will cover some 15 months, beginning during the 45th month and ending in the 60th month. The testing program will occupy the last nine months of this period. The PFRT will be a carefully conducted testing program completed in accordance with a PFRT test program report to be approved beforehand. It is anticipated that approximately 200 tests will be needed to complete the entire program, which is aimed at establishing the reliability and safety aspects of the propulsion systems. The testing will include environmental, vibration, malfunction, and altitude testing. The largest single group of tests (between 80 and 100) will be used to demonstrate malfunction safety. Various malfunctions will be simulated and the reaction of the powerplant determined. These tests will measure the effectiveness of the malfunction studies. Penalty runs will be conducted when necessary and additional malfunction runs made as the need becomes evident.

Altitude tests will be run at the 6K and 24K level at the Tullahoma facilities. The longest practical mission simulation will be made, and thrust, gimbaling, and attitude control programs included. It is planned to conduct 10 tests at Tullahoma.

The remaining tests will be scheduled and conducted to demonstrate compliance with the model specification and other applicable specifications. They will include propulsion system tests and component tests. In all cases they will be conducted in accordance with written specifications covering requirements, servicing (if necessary), inspection, etc.

Phase III - Delivery

Powerplant deliveries will commence during the 60th month of the program. Delivery of ten complete propulsion systems will be made over a period of two and one half years in accordance with the schedule of Figure 61. Following a complete inspection of all parts (including pressure checks and flow checks, as required), the propulsion system will be assembled to simulate vehicle installation and then acceptance tested in a manner agreed upon with the General Electric Company. An acceptance test should include a short 10 second run with one 6K chamber, followed by a two hour shutdown period with the system propellant tanks pressurized, the complete system exposed to simulated space vacuum environment, and the main attitude control system cycling periodically. The test should then conclude with 10 seconds of operation at full 24K thrust. The gimbaling system shall be checked for operation and the re-entry roll control system cycled to insure proper functioning. This testing will be conducted with simulated high altitude exhaust to insure full flow in the nozzles.

Following completion of the acceptance test, the propulsion systems will be drained, serviced as required for flight operation, and shipped dry to a point to be determined. During the course of the development program, consideration will be given to the requirements of a prelaunch checkout for electrical and pneumatic equipment. A schedule of spare parts to be supplied will be established during this phase.

C. TEST FACILITIES

Three areas in the proposed development program merit special consideration and discussion. They are: (1) high altitude exhaust simulation to insure full flow in the nozzle, (2) very high vacuum to permit simulation of exposure of components to space pressure, and (3) facilities for the test irradiation of fluids and components.

1. High Altitude Exhaust Simulation

During the course of the main attitude control and re-entry roll control development program it will be necessary to insure full flow in the nozzles. Further, it is required that the entire main attitude and re-entry roll control systems be subjected to as low a pressure as is practicable for extended periods of time to simulate the space mission environment. A system to provide this capacity is currently under development for the NASA Surveyor vernier powerplant system development program at Thiokol-RMD, and will be available for use on the Apollo program.

The altitude test apparatus will have the capacity to start and run at high altitudes. Altitude simulation for the applicable rocket will require a contoured rocket exhaust diffuser discharging into a jet ejector. Commonly, steam is employed to power the ejector; however, Thiokol-RMD has developed an "instant steam" system based on peroxide decomposition products reacted with gaseous hydrogen to provide steam at a much lower cost in installation and operation than an equivalent, conventional steam generating system

The Hyprox Evactor system proposed for the altitude testing phase of this program consists of an altitude chamber, rocket exhaust diffusers, exhaust cooler, first and second stage ejectors, a Hyprox combustor to supply ejector drive system, and controls. Figure 62 portrays a typical system.

The altitude chamber, approximately 8 ft in diameter and 20 ft long, houses the engine system and rocket exhaust diffusers. Each diffuser is essentially a duct consisting of gently converging entrance and exit cones separated by a short cylindrical throat section. The ducts are water cooled over their entire length.

The cooler section, immediately downstream of the diffusers, manifolds the engine exhaust gases, and reduces temperature from the expected value of approximately 5000F to 1200F. This reduces the otherwise severe service conditions of the first and second stage steam ejectors located immediately downstream of the cooler, and thereby permits the use of conventional steam ejector designs and materials for these stages. Also, the use of smaller units is permitted because of the increased density of gases being handled. The cooling operation is accomplished by means of water injection into the exhaust gas stream.

The first and second stage ejectors are coaxial with, and immediately downstream of, the exhaust cooler. Both function as conventional steam ejectors, the driving steam being generated in the Hyprox combustor and introduced axially at the entrance of each ejector through conventional supersonic steam nozzles.

The Hyprox combustor is the source of the driving steam for the first and second stage ejectors. The combustor consists of a single assembly with four major areas: a peroxide injection and decomposition chamber, a hydrogen gas preheating and injection section, a hydrogen combustion chamber, and a water injection section.

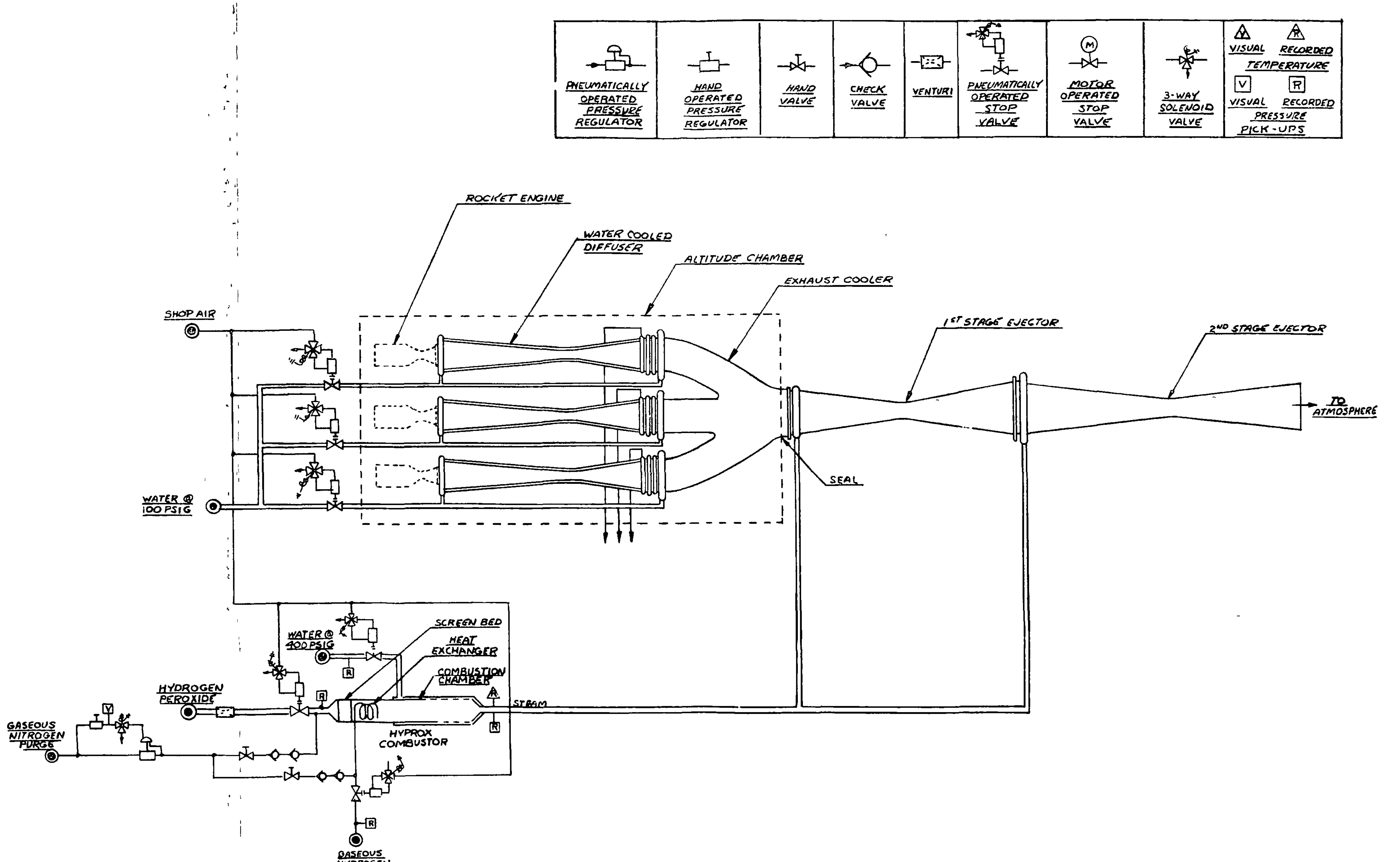


Figure 62. Typical Hyprox Evactor System

The introduction of gaseous hydrogen increases the steam generating quantity by burning the available oxygen in the decomposed peroxide, while the water injector cools the products of combustion and also increases the quantity of steam generated. The dry, saturated steam leaves the combustor and enters the drive gas inlets of the first and second stage ejectors through an uncooled pipeline.

The controls and instrumentation required to operate the Hyprox Evactor system are simple, consisting of commercially available, remotely operated valves and regulators. These control the fluids supplied to the Hyprox combustor and the cooling water supplied to the diffusers and exhaust cooler. Low response pressure monitoring instrumentation is required to monitor altitude chamber pressure levels.

The described system provides a high degree of operational flexibility. Specifically, it permits engine start at the desired altitude, and maintains the desired altitude during single or multiple engine operation, during restart, during thrust termination, and over a wide range of engine throttling.

Prior to start, the desired altitude is obtained by operation of the first and second stage ejection. This will establish a 100,000 ft altitude pressure equivalent in the altitude chamber. This altitude is maintained by these ejectors during engine start until the engine exhaust flow is sufficient to fill the rocket exhaust diffuser. At this point, the rocket exhaust diffuser and second state ejector handle the increased exhaust gas flow and maintain the desired altitude. During this period, the first stage ejector serves merely as a duct with its driving steam providing enough energy to offset duct losses which would be incurred if the steam were shut off.

The sequence of operation is reversed on shutdown. Thus, the desired altitude is always maintained throughout the system, providing for an immediate altitude restart.

Thiokol-RMD plans to conduct development testing of the on-board propulsion system at the 6K and 24K levels in a system which will provide space environment simulation for the entire system and multiple diffusers to maintain a low back pressure on the thrust chamber nozzles during operation. During the quality assurance and PFRT programs, a series of tests to finally demonstrate nozzle performance and cooling, high altitude ignition, and simulated space operation of all of the propulsion systems will be conducted in the Tullahoma high altitude test facilities.

2. Space Vacuum Simulation

The simulation of the very low pressure of space will be accomplished by use of special bell jars exhausted by diffusion pumps. Two jars approximately 1 cu ft and 5 cu ft are available for use in this program. They have a vacuum capability of 10^{-7} mm Hg, providing therefore a capacity and altitude simulation considered entirely adequate for testing components.

3. Radiation Testing

Thiokol-RMD plans to use the radiation test facilities of Brookhaven or Lawrence Radiation Labs for component and propellant irradiation. Consideration will be given to the exposure of propellant-containing containers to high energy proton bombardment in a proton synchrotron to simulate the effect of cosmic rays in the BEV range.

APPENDIX A

WEIGHT SUMMARY

APPENDIX A

WEIGHT SUMMARY

System		1963*		1966*		1963 Alternate **		1966 Alternate - * Reverse Flow
Main On-Board Propulsion								
Thrust Chambers	(4)	204.0 lb	(2)	390.0 lb	(4)	378.2 lb	(8)	222.2 lb
Fuel Tanks		98.0		56.0		78.0		83.2
Oxidizer Tanks		123.0		130.0		98.0		136.3
Helium Tanks		177.3		67.0		139.0		---
Valves and Controls, Misc.		109.3		85.6		104.9		362.6
Dry Weight		711.6 lb		728.6 lb		798.1 lb		804.3 lb
Helium Weight		37.0		43.0		29.0		---
Propellant Weight		8931.0 lb		7127.0 lb		7153.0 lb		7127.0 lb
Main Attitude Control								
Thrust Chambers (12)		12.0 lb		12.0 lb		12.0 lb		12.0 lb
Fuel Tank		4.6		3.6		3.6		3.6
Oxidizer Tank		7.3		6.6		5.8		6.6
Valves and Controls, Misc.		10.7		10.2		10.7		10.2
Dry Weight		34.6 lb		32.4 lb		32.1 lb		32.4 lb
Propellant Weight		390.0 lb		336.0 lb		220.0 lb		336.0 lb
Re-Entry Roll Control								
Thrust Chambers (8)		16.7 lb		16.7 lb		16.7 lb		16.7 lb
Fuel Tank		0.5		0.5		0.5		0.5
Oxidizer Tank		0.8		0.8		0.8		0.8
Valves and Controls, Misc.		23.4		23.4		23.4		23.4
Dry Weight		41.4 lb		41.4 lb		41.4 lb		41.4 lb
Propellant Weight		25.4 lb		25.4 lb		25.4 lb		25.4 lb
Total Loaded System Weight		10,171.0 lb		8333.8 lb		8299.0 lb		8366.5 lb

NOTES

* W_o = 15,715 lb; ΔV = 7500 ft/sec

** ΔV = 5600 ft/sec (circumlunar mission only)

APPENDIX B

MODEL SPECIFICATION NO. 15108
PROPULSION SYSTEM, LIQUID PROPELLANT

Manned Vehicle Main On-Board Propulsion
Attitude Control and
Re-Entry Roll Control Systems

MODEL SPECIFICATION FOR TD-281 PROPULSION SYSTEM

1. SCOPE

1.1 This specification covers the requirements for the General Electric Company Apollo Main On-board Propulsion and Control Rocket Systems.

1.2 Classification. The Model TD-281 Main On-board Propulsion and Control System consists of three subsystems, all designed to operate and be capable of restart under the conditions of cislunar and lunar space.

a. Subsystem A: Main On-board Propulsion.

The main on-board propulsion subsystem contains four (4) six thousand (6000) pound thrust chambers, with regeneratively cooled chamber and throat inserts and ablation cooled nozzle extensions. These chambers may be fired singly or in combination. The tankage, two spheres for each propellant, is pressurized with helium to 390 psia and utilizes positive expulsion to deliver N_2O_4 and MMH to the thrust chambers. The thrust chamber will attain an I_{sp} of 319 seconds operating at a P_{ch} of 200 psia with an O/F of 2:1 and an expansion ratio of $A_e/A_t = 40$. Each chamber can be independently actuated through a gimbal angle of $\pm 5^\circ$ in any direction. The engine operation and mode will be initiated and controlled by the pilot.

b. Subsystem B: Main Attitude Control.

The main attitude control subsystem contains 12 (twelve) 3 (three) pound thrust radiation cooled thrust chambers. These chambers will operate at a P_{ch} of 40 psia to obtain a minimum I_{sp} of 310 seconds with an O/F ratio of 2.4 and an expansion ratio of $A_e/A_t = 100$.

The positive expulsion tankage pressurized to 150 psia with helium from the same tanks used for the main on-board propulsion system will also provide adequate N_2O_4 and MMH to the 6000 pound thrust chambers to produce a 250 ft/sec velocity increment. The main attitude control subsystem will be pilot controlled.

c. Subsystem C: Re-Entry Roll Control.

The re-entry roll control subsystem contains eight (8) eighteen (18) pound thrust radiation cooled thrust chambers. These chambers will operate at a P_{ch} of 40 psia to obtain a minimum I_{sp} of 310 seconds with an O/F ratio of 2.4 and an expansion ratio $A_e/A_t = 100$. The positive expulsion tankage pressurized to 150 psia with helium will provide N_2O_4 and MMH to the thrust chambers. This subsystem is to be independently mounted on the re-entry vehicle and will be pilot controlled.

2. APPLICABLE DOCUMENTS

2.1 The following specifications form a part of this specification except as modified herein.

Specifications.

Military: to be supplied later

Reaction Motors Division: to be supplied later

2.2 Precedence of Specifications. The precedence of specifications shall be as follows in the case of conflict between requirements of different specifications.

a. Contract requirements
b. This specification
c. Other specifications

2.3 Definitions. For the purpose of this specification, Thiokol Chemical Corporation, Reaction Motors Division, shall be referred to as the contractor.

3. REQUIREMENTS

3.1 Mockup. For new types or models of propulsion systems, a full scale mockup shall be prepared for examination and approval as soon as the contractor has established the installation features of the engine. Any changes required by the using service shall be subject to negotiation.

3.2 Propulsion System Changes. The using service shall be notified of changes affecting the installation made after approval of the mockup. After the mockup has been approved, the installation drawings shall be forwarded to the using service for approval. The mockup shall be kept current with approved changes, at least through the first production contract, unless otherwise authorized.

3.3 Performance Characteristics. The ratings and curves shown are based on the terms and standard conditions defined herein.

3.3.1 Propulsion System Operating Regimes.

3.3.1.1 Pressure and Temperatures. The maximum ambient pressure at which the engines shall start, operate, and stop shall be 0.5 psia. The propulsion system shall be capable of satisfactory operation at the temperature extremes of 20°F and 80°F.

3.3.1.1.1 Static Exposure. The propulsion system shall be capable of satisfactory operation after static exposure for 8 hours at a minimum ambient temperature of 80°F when supplied with propellants at a minimum temperature of 80°F or 10°F below the respective boiling points of the propellants, whichever is lower; and after static exposure for 24 hours at a maximum ambient temperature of +20°F, followed by exposure for at least 48 hours at a maximum temperature of +20°F when supplied with propellants at +20°F or 10°F above their respective freezing points, whichever is higher.

3.3.1.2 Attitudes. The rocket engines shall perform satisfactorily in all attitudes under the loading conditions specified in paragraph 3.4.2.

3.3.2 Ratings. The performance ratings shall be as listed in Table B1. These data are based on the use of specified propellants and fluids under specified conditions (per paragraph 3.3.8) with an exhaust nozzle having an expansion ratio of 40:1 for the main on-board propulsion system and 100:1 for the main attitude and re-entry roll control systems.

3.3.3 Thrust Chambers. Performance parameters are given in Table B2.

TABLE B1

PERFORMANCE RATINGS SINGLE CHAMBER OPERATION

System	Nominal Design Thrust, lb	Minimum Instantaneous Specific Impulse, sec	Instantaneous Mixture Ratio	Nominal Chamber Pressure, psia	Duration per Cycle Minimum	Duration per Cycle Maximum	No of Starts
Main On-Board Propulsion	6,000	319	2 1	200	2 sec	104 sec	14
	24,000*	319	2 1	200	---	104 sec	1
Main Attitude Control	3	310	2 4	40	100 msec	67 sec	3000 (max)
Re-Entry Roll Control	18	310	2 4	40	100 msec	5 sec	500 (max)

* Four-chamber operation (emergency abort)

TABLE B2

PERFORMANCE PARAMETERS

Thrust Chamber	Thrust, lb	Chamber Pressure psia	Thrust Coefficient	Characteristic Velocity, ft/sec	O/F Ratio
Main On-Board Propulsion	6000	200	1 975	5200	2 1
Main Attitude Control	3	40	1 995	5000	2 4
Re-Entry Roll Control	18	40	1 995	5000	2 4

3.3.4 Starting.

3.3.4.1. Main On-Board Propulsion System Starting. Figures B1 and B2 present the fluid and electrical schematics of the main on-board propulsion system.

The following sequences are involved with starting the system:

a. Pre-Arm.

Prior to "arming" the system, the option exists to cycle open all the solenoid operated attitude control propellant valves (U5-U28). This can be done through the spacecraft controls and would perform the function of clearing the lines from the propellant tanks to the individual thrust chambers when vented to a vacuum atmosphere.

b. Arm.

The system is "armed" by closing the arm switch (S1). This allows current to fire open the dual squib operated pressurizing valve (U42). The squib portion of this valve is dualized to increase its reliability. Once the valve (U42) is opened, high pressure helium is allowed to flow from the storage bottles. After the helium leaves the storage bottles it branches into two parallel paths. Each path contains identical pressure regulators (RG1 and RG2). The regulators are sized so that each can handle the maximum flowrate capacity requirement of the engine system. Downstream of each regulator is a spring loaded helium pressure operated regulator malfunction valve (U43 and U44). The function of these valves is to shut off helium flow in the applicable branch in the event of a regulator failure in the wide open position to prevent full helium bottle pressure from pressurizing propellant tankage. A check valve (CK1 and CK2) is located downstream of each malfunction valve (U43 and U44) so that an abnormally high pressure in one branch will not feed back into the other branch. After leaving these check valves (CK1 and CK2) the pressurizing helium in the two branches is joined by a common line in which is located a relief valve (U45). This relief valve (U45) is sized to compensate for regulator lockup leakage and possible pressure pulses.

c. Control Gas.

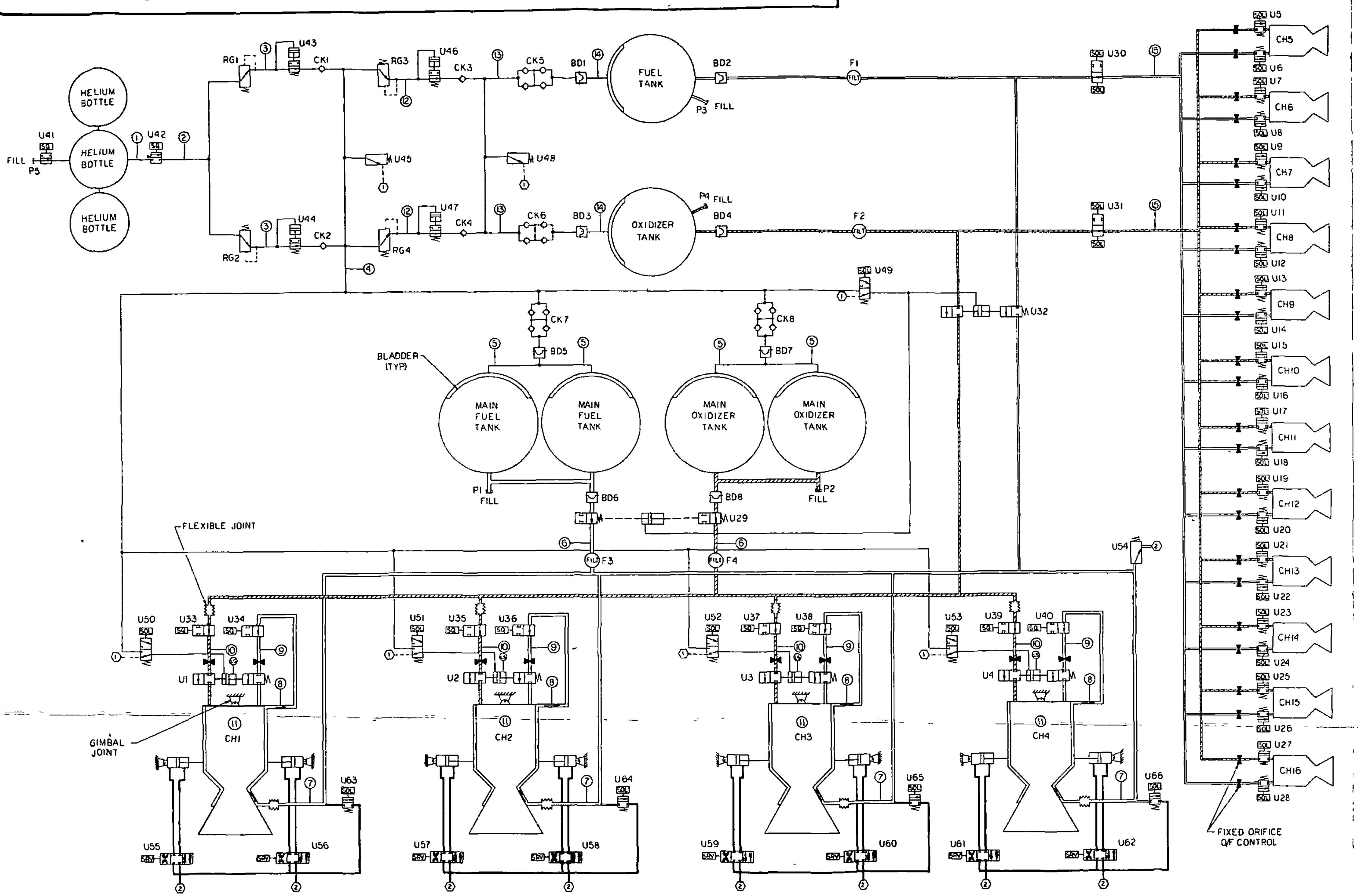
Pressurized helium gas is taken from the common line and is distributed to five (5) solenoid operated pilot valves (U49 - U53) to be utilized as control gas for operating pneumatic actuated valves.

d. Main Propulsion Tankage Pressurization.

Pressurized helium gas is taken from the common line and is sent to the main propulsion tankage system. In the main propulsion tankage system there are two tanks for each propellant. Each pair of tanks, for similar propellant, is manifolded together both at the inlet and exit portions. Each pair of main propellant tanks is fitted with burst disc valves (BD5 - BD8) at both the inlet and exit ports. These burst disc valves (BD5 - BD8) afford a hermetic seal of the propellant tankage until propellants are required. When the pressurizing gas reaches the inlet port burst disc valves (BD5 and BD7) this pressure will rupture the discs allowing the pressurizing gas to enter all main propellant tankage. Once the gas enters the tanks it pushes against the positive expulsion bladders in each tank and forces the propellant out of the exit ports of each tank. The fluid pressure of the propellants will build up rapidly and will rupture the tank exit burst disc valves (BD6 and BD8). Quadruple check valves (CK7 and CK8) are located at the entrance of both propellant tanks to preclude the possibility of any backflow or intermixing of propellants.

After the tank exit burst disc valves (BD6 and BD8) have been ruptured propellants are allowed to flow from the tanks to the main propellant manifolds. From the manifolds the propellants branch out to each individual main thrust chamber (CH1 - CH4) and the flow will be stopped at the individual main propellant valves (U1 - U4) and the entire main thrust system will be pressurized. In this process the regeneratively cooled portions of the thrust chambers (CH1 - CH4) will be filled with fuel as the main propellant valves (U1 - U4) are located downstream of the cooling jackets. Filters (F3 and F4) are located in the propellant lines leading to the propellant manifolds to remove any foreign particles that may exist that could possibly clog the thrust chamber injectors.

MODEL SPECIFICATION FOR TD-281 PROPULSION SYSTEM



ITEM	DESCRIPTION
CH1-CH4	MAIN THRUST CHAMBER, GIMBALED
CH5-CH16	ATTITUDE CONTROL THRUST CHAMBER
U1-U4	MAIN PROP VALVE, PNEU. OPERATED
U5-U28	ATTITUDE CONTL PROP VALVE SOLENOID OPERATED
U29	MAIN PROPELLANT TANK SHUTOFF VALVE PNEU OPERATED
U30 U31	ATTITUDE CONTROL SYS. SHUTOFF VALVE SOLENOID OPER.
U32	CROSS FEED PROPELLANT VALVE PNEU OPERATED
U33 U40	MAIN CHAMBER MALF SHUTOFF VALVE DUAL SQUIB OPER
U41	HELIUM FILL VALVE DUAL SQUIB OPERATED
U42	PRESSURIZING VALVE DUAL SQUIB OPERATED
U43 U44	MAIN HE REGULATOR MALF SHUTOFF VALVE PNEU OPER
U45	MAIN HELIUM PRESSURE RELIEF VALVE
U46 U47	ATTITUDE CONT'L HE REGULATOR MALF VALVE PNEU OPER
U48	ATTITUDE CONTL PRESSURE RELIEF VALVE
U49	MAIN PROP SHUTOFF & CROSSFEED VALVE PILOT VALVE, SOL OPER.
U50-U53	MAIN PROP VALVE PILOT VALVE SOLENOID OPERATED
U55-U62	MAIN THRUST CHAMBER GIMBAL VALVE SERVO OPERATED
CK1-CK4	CHECK VALVE SINGLE
CK5-CK8	CHECK VALVE QUAD.
BD1-BD4	ATTITUDE CONT'L PROPELLANT TANK BURST DISC
BD5-BD8	MAIN PROPELLANT TANK BURST DISC
F1 F2	ATTITUDE CONT'L PROPELLANT FILTER
F3, F4	MAIN PROPELLANT FILTER
P1 P2	MAIN PROPELLANT TANK FILL PLUG
P3 P4	ATTITUDE CONT L PROPELLANT TANK FILL PLUG
P5	HELIUM FILL PLUG
U54	MAIN FUEL MANIFOLD RELIEF VALVE
U63-U66	SERVO FEED VALVE, SOLENOID OPERATED
RG1-RG4	HELIUM PRESSURE REGULATOR

Figure B1. Main Propulsion and Main Attitude Control System Fluid Schematic, 1963 (Dwg 52214)

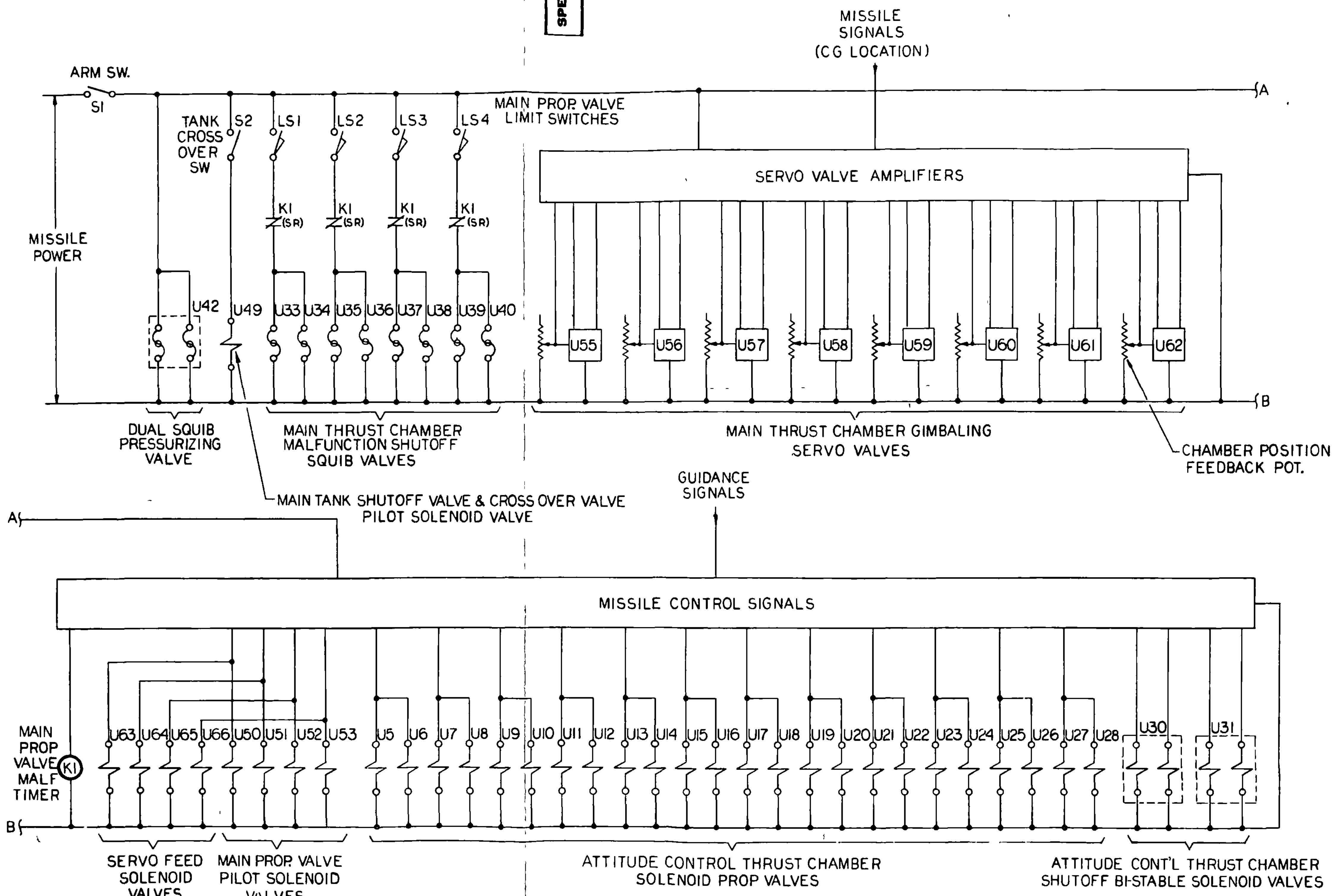


Figure B2. Main Propulsion and Main Attitude Control System Electrical Schematic, 1963 (Dwg 52215)

e. Fire.

Once the system has been armed any one or any combination of the four main thrust chambers (CH1 - CH4) can be fired upon command from the vehicle controls. When a particular thrust chamber is required to fire, the vehicle control system will energize the appropriate solenoid operated servo fluid valve (U63 - U66) and its corresponding solenoid operated main propellant valve pilot valve (U50 - U53). Operating the servo fluid valve (U63 - U66) will allow pressurized fuel to be available to the corresponding servo operated main thrust chamber gimbal valve (U55 - U62). The gimbal valves (U55 - U62) receiving information from the spacecraft controls, along with data from the chamber position feedback potentiometer, will provide pressurized fuel to the thrust chamber gimbal actuators in such a way as to position the thrust vector through the instantaneous center of gravity.

While this is occurring, the main propellant valve pilot valve (U50 - U53) has opened and allowed control gas (helium) to enter the actuating piston of the pneumatic operated main propellant valve (U1 - U4). The control gas will force the main propellant valve (U1 - U4) to the open position and both propellants will enter the combustion chamber (CH1 - CH4). Ignition will occur upon contact of the two hypergolic propellants.

3.3.4.2 Main Attitude Control System Starting. Figures B1 and B2 describing the main propulsion system also show the fluid and electrical schematics of the main attitude control system.

The two systems are linked through the helium storage bottles, pressure regulators, and relief valves. Consequently the pre-arm and arm phases are the same.

a. Attitude Control Tankage Pressurization.

The attitude control propellant tankage is pressurized to approximately one half the level of the pressure in the main propellant tankage. Another stage of pressure regulation is required. Pressurizing helium gas is taken from the common line and is directed into two parallel paths. Each path contains identical pressure regulators (RG3 and RG4). The regulators are sized so that each can handle the maximum flowrate capacity of the attitude control engine system and one main thrust chamber.

Thus in the event of a failure such that either of the regulators (RG3 or RG4) does not allow any flow when required, the other will take over the entire function. Downstream of each regulator is a spring loaded helium pressure operated regulator malfunction valve (U46 and U47).

b. Fire.

Once the system has been armed, any one or any combination of attitude control thrust chambers (CH5 - CH16) can be fired upon command from the vehicle controls. Both the oxidizer and fuel solenoid valves for a specific thrust chamber are actuated by a common signal. Ignition will occur upon contact of the two propellants due to their hypergolicity.

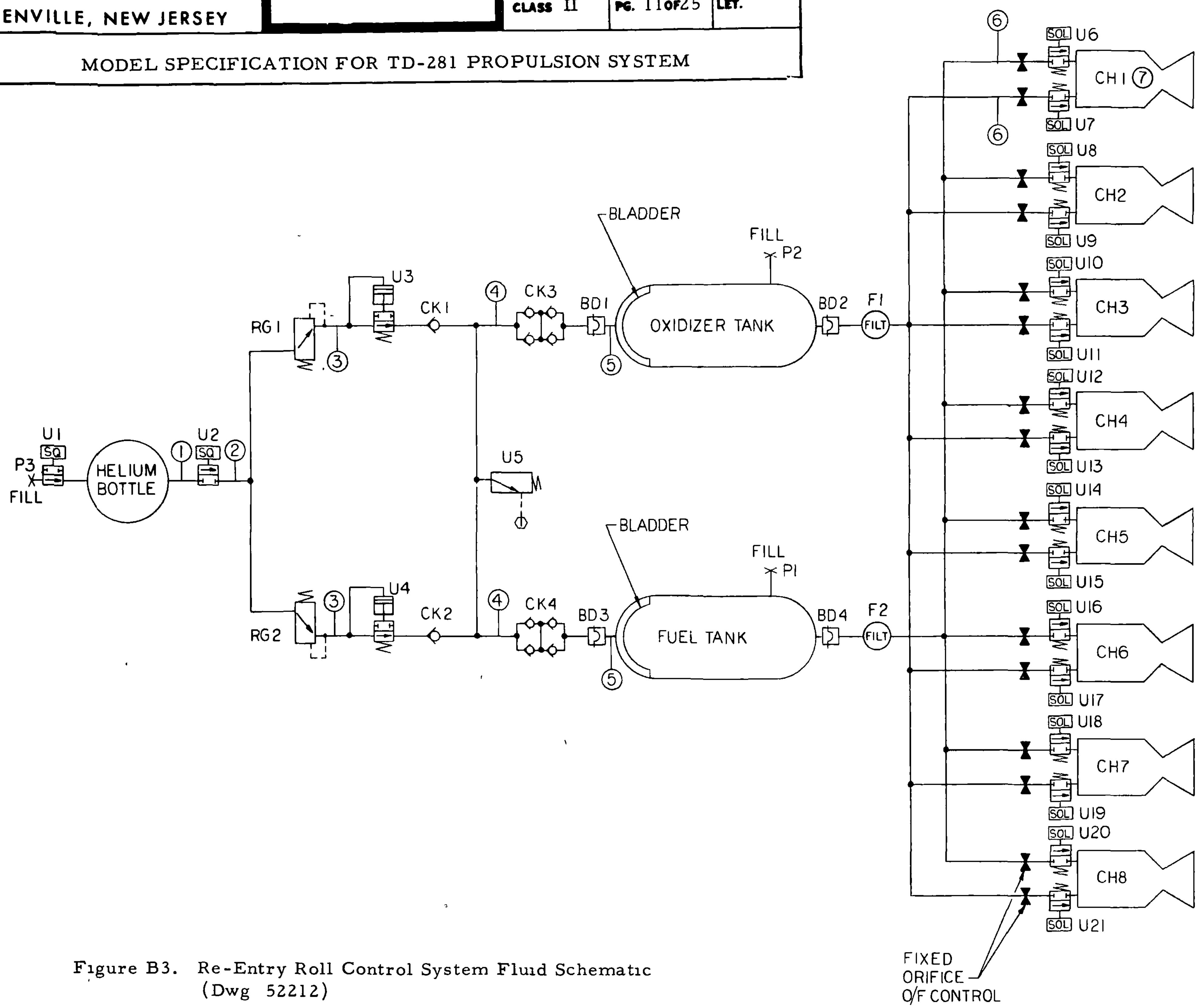
3.3.4.3 Re-Entry Roll Control System Starting. Figures B3 and B4 show the fluid and electrical schematics of the re-entry roll control system. The following sequences are involved with starting the system.

a. Pre-Arm.

Prior to "arming" the system, the option exists to cycle open all the solenoid operated propellant valves (U6 - U21). This can be accomplished through the spacecraft controls once the power switch (S1) has been closed and would perform the function of clearing the lines from the propellant tanks to the individual thrust chambers when vented to a vacuum atmosphere.

b. Arm.

The system is "armed" by depressing the momentary contact arm switch (S1). This allows current to fire open the dual squib operated pressurizing valve (U2). The squib position of this valve is dualized to increase its reliability. Once the valve (U2) is opened, high pressure helium is allowed to flow from the storage bottle. After the helium leaves the storage bottle, it branches into two parallel paths. Each path contains identical pressure regulators (RG1 and RG2). The regulators are sized so that each can handle the maximum flowrate capacity requirement of the engine system. Downstream of each regulator is a spring loaded helium pressure operated regulator malfunction valve (U3 and U4). The function of these valves is to



ITEM	DESCRIPTION
U1	DUAL SQUIB OPERATED, HELIUM FILL VALVE
U2	DUAL SQUIB OPERATED, PRESSURIZING VALVE
U3,U4	PRESSURE OPERATED, REGULATOR MALF VALVE
U5	HELIUM RELIEF VALVE
U6-U21	SOLENOID OPERATED, PROPELLANT VALVE
CH1-CH4	ROLL CONTROL THRUST CHAMBER
CH5-CH8	REDUNDANT ROLL CONTROL THRUST CHAMBER
RG1,RG2	HELIUM PRESSURE REGULATOR
CK1,CK2	CHECK VALVE
CK3,CK4	QUAD. CHECK VALVE
BD1-BD4	FUEL TANK & OX TANK BURST DISCS
P1-P3	FILL PLUGS
F1	OXIDIZER FILTER
F2	FUEL FILTER

(1) HELIUM OVERBOARD DRAIN

Figure B3. Re-Entry Roll Control System Fluid Schematic
(Dwg 52212)

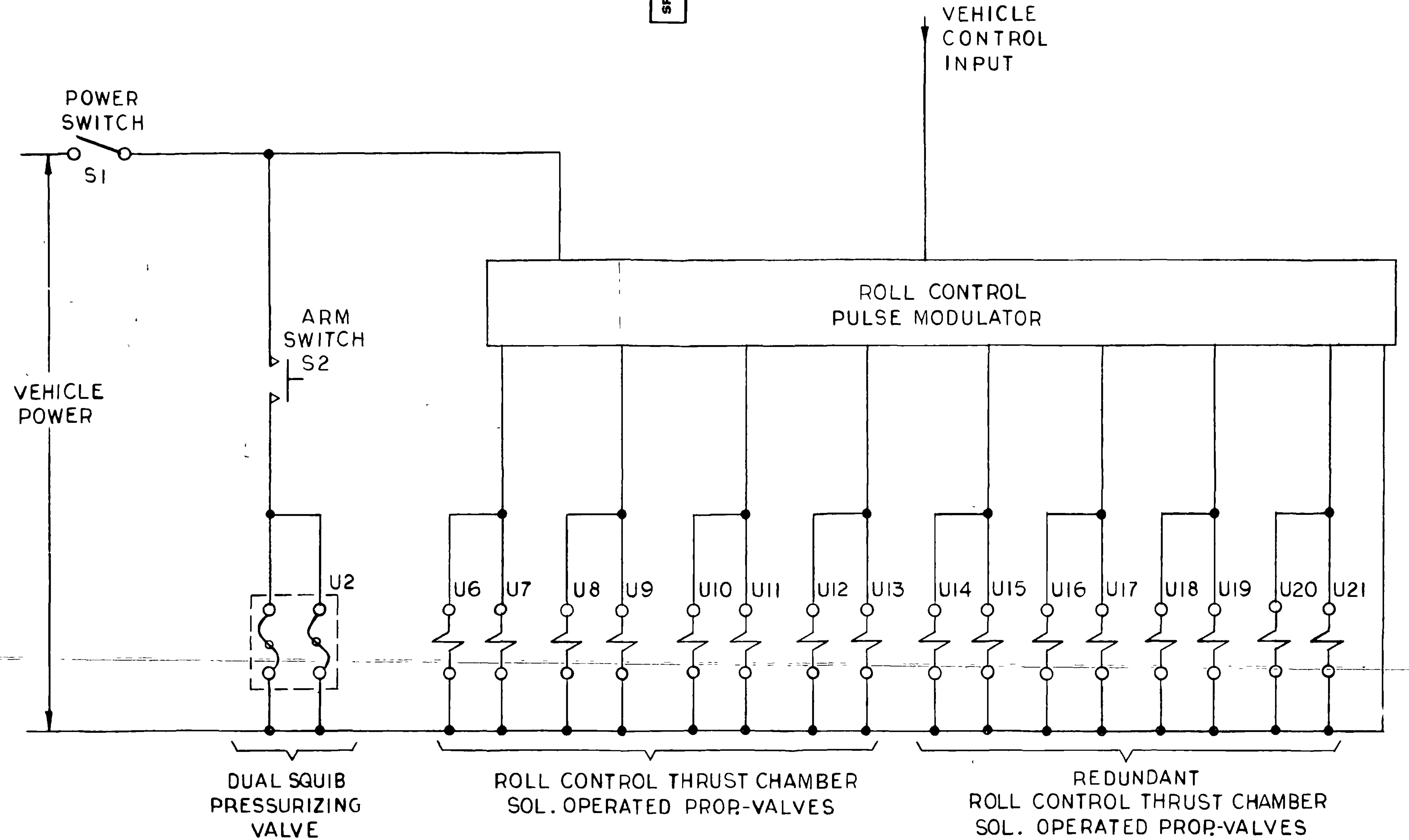


Figure B4. Re-Entry Roll Control System Electrical Schematic (Dwg 52213) stop helium flow through a regulator which fails in the open position and prevent excessive helium flow into the propellant tanks. A check valve (CK1 and CK2) is located downstream of each malfunction valve (U3 and U4) so that an abnormally high pressure in one branch will not feed back into the other branch. After leaving these check valves (CK1 and CK2) the pressurizing helium in the two parallel branches is joined by a common line in which is located a relief valve (U5). This relief valve (U5) is sized to compensate for regulator lockup leakage and possible pressure pulses. The common link permits flow from either pressure regulator to supply both the oxidizer and fuel tank in case one regulator malfunctions.

c. Fire.

Once the system has been armed, any one or any combination of thrust chambers can be fired upon command from the vehicle controls. Both the oxidizer and fuel solenoid valves for a specific thrust chamber are actuated by a common signal. Ignition occurs upon contact of the two propellants due to their hypergolicity.

3.3.5 Shutdown.

3.3.5.1 Main On-Board Propulsion System Shutdown. Once the required velocity correction has been obtained the vehicle controls will signal the thrust chamber (CH1 - CH4) in operation to shut down by discontinuing power to the appropriate main propellant valve pilot valve (U50 - U53) and servo fluid valve (U63 - U66). Discontinuing power to the main propellant valve pilot valve (U50 - U53) will allow it to return to its normal position, under a spring load, and the control gas will be shut off and the actuating piston of the main propellant valve (U1 - U4) will have its pressurizing gas vented to atmosphere by means of appropriate porting in the pilot valve (U50 - U53). Once the pressurizing gas is vented, the main propellant valve (U1 - U4) will close under the action of its spring loading. With the flow of propellants discontinued, the main thrust chamber (CH1 - CH4) will cease operation.

In the event that a main thrust chamber (CH1 - CH4) does not discontinue operation when required, erroneous velocity correction and loss of propellant will occur as well as presenting a hazardous condition. To eliminate this condition, the following safety circuitry has been incorporated.

Whenever a thrust chamber is fired a slow release timer is energized. Energizing this timer will cause its four (4) normally closed contacts (K1) to open. Each one of these normally open contacts is wired in an individual series circuit which includes a main propellant valve limit switch (LS1 - LS4) and two squib operated main chamber malfunction shutoff valves (U33 - U40) which are wired in parallel. One each of the four limit switches (LS1 - LS4) is located physically on each of the main propellant valve actuators (U1 - U4) so that when the valve (U1 - U4) is in the fully closed position the limit switch (LS1 - LS4) is held open. At any time that the valve (U1 - U4) is off its seat the limit switch (LS1 - LS4) is in its closed position. The squib operated shutoff valves (U33 - U40) are installed one each in the propellant lines leading to each main thrust chamber (CH1 - CH4). They are normally open, so that once they are actuated, the thrust chamber in question will no longer be operable. Three thrust chambers remain, ensuring that the system is still safe.

Thus when the vehicle controls signal an operating thrust chamber to shut down, power will be discontinued to the main propellant valve malfunction timer (K1). This will start the timer on its count (2 - 4 seconds). When this count has been completed, the timer contacts (K1) will return to their normally closed positions. If the particular main propellant valve (U1 - U4) in question has not returned to its fully closed position, then the corresponding limit switch (LS1 - LS4) will remain in the closed position and a circuit will be completed through the normally closed K1 contacts. This completed circuit will fire the appropriate pair of squib operated main chamber malfunction shutoff valves (U33 - U40) which will stop propellant flow to the main thrust chamber (CH1 - CH4) in question.

Located in the main fuel manifold is a relief valve (U54) that is sized to handle excess fuel vapor pressure resulting from residual main thrust chamber (CH1 - CH4) heat on shutdown.

An additional requirement imposed on the entire system is that provision shall be made for operating any of the main thrust chambers on propellants stored in the attitude control propellant tankage. This was accomplished by the inclusion of the pneumatically operated cross feed propellant valve (U32). When this feature is required, the solenoid operated main propellant shutoff and cross feed valve pilot valve (U49) is energized by the vehicle controls allowing control gas (helium) to enter the actuating piston of the pneumatically operated cross feed propellant valve (U32). This will force this valve to the open position and allow propellant tanks to the main thrust chamber propellant manifolds. At this point any of the main thrust chambers (CH1 - CH4) can be fired, as described previously. Concurrent with the vave action above, the pilot valve (U49) also allows control gas (helium) to enter the actuating piston of the pneumatically operated main propellant tank shutoff valve (U29). This will force this valve to the closed position thus precluding the possiblity of any type of flow in either direction during the cross feed operation of one of the main thrust chambers (CH1 - CH4). The system is returned to normal by deactuating the pilot valve (49) which shuts off the control gas (helium) and vents to atmosphere the pressurizing gases in the actuating cylinders of the two propellant valves (U29 and U32).

3.3.5.2 Main Attitude Control Shutdown. The chambers can be fired for long durations or they can be pulsed 10 - 15 cycles per second. Fixed O/F control orifices are shown in each propellant line. These orifices will be part of the propellant valves and will be sized during or just prior to acceptance tests.

In the event that any of the attitude control propellant valves (U5 - U28) stick in the open position, propellant will be lost through this open valve. The fact that propellant is being expended when no thrust chamber is required to be fired will be sensed by the vehicle's propellant management system. Once the malfunction has been detected, the vehicle controls will signal the appropriate solenoid operated, bistable, attitude control system shutoff valve (U30 - U31) to close. Thus no propellant will be lost through a malfunctioned open attitude control thrust chamber propellant valve (U5 - U28) during extended periods of shutdown. The action of the bistable (dual solenoid) valve (U30 and U31) is such that when the appropriate solenoid is energized, the normally open valve actuates closed and is held in the closed position by a detent. In this way, power to the actuating solenoid can be removed during the expected long periods of attitude control thrust chamber shutdown. When the attitude control system is required to operate, the vehicle controls will energize the other solenoid on the bistable valve which will remove the detent and allow the valve to return to its normally open position under the action of a spring.

3.3.5.3 Re-Entry and Roll Control Shutdown. The chambers can be fired for long durations or they can be pulsed 10 - 15 cycles per second. Fixed O/F control orifices are shown in each propellant line. These orifices will be part of the propellant valves and will be sized during or just prior to acceptance tests.

3.3.6 Malfunction. The propulsion system shall, under any single or critical multiple malfunction within the system or its associated propellant and environmental control systems, present no hazard to the spacecraft and flight crew and must remain operable or be capable of becoming operable within the limitations of on-board repair capability, redundancy, or the use of alternate propulsion devices. The term "operable" is defined to mean the ability to conclude the mission in a safe manner.

3.3.7 External Power.

3.3.7.1 Valves and Solenoids. The maximum power required (during emergency abort) is 780 watts.

3.3.7.2 Heating. A maximum of 2000 Btu/hour will be required for heating the propulsion system compartment during periods of lunar orbit.

3.3.8 Propellants and Fluids.

3.3.8.1 Propellants.

a. Oxidizer: N_2O_4 - Nitrogen Tetroxide

b. Fuel. MMH - Monomethylhydrazine

3.3.8.2 Pressurizing Gas: Helium Gas, Welding Grade. Welding grade helium shall be oil free and have a maximum dewpoint of -70°F. Helium purity shall be greater than 99.9 percent.

3.3.8.3 Lubricants: None required.

3.3.8.4 Other Fluids: None required.

3.3.8.5 Leakage of Rocket Engine Fluids: Table B3 presents a listing of the propellant valves and the amount of maximum allowable leakage assigned to each. This leakage produces a negligible velocity increment. Vented gas will not apply any torque or other force to the spacecraft.

3.3.9 Control.

3.3.9.1 Accuracy. The control shall be such that the rocket engine will operate within the limits specified in paragraph 3.3.2 entitled "Ratings."

3.3.9.1.1 Mixture Ratio. The mixture ratio shall be controlled within safe operating limits at set points and during thrust increase and decrease. Mixture ratio limits at set points specified in Table B1.

3.3.9.2 Thrust.

3.3.9.2.1 Change Rate. There shall be no rate limitations on selection of set performance point(s) regardless of means used to effect the changes.

3.3.9.2.2 Increase. Time intervals in the starting sequence, between starting and selected set performance point(s), and between selected set performance points shall be as follows:

		Starting and Shutdown Transients	
Thrust Chamber	Thrust, lb	Signal "On" to 90% Thrust	Signal "Off" to 5% Thrust
Main On-Board Propulsion	6000	250 msec	250 msec
Main Attitude Control	3	20-25 msec	25-30 msec
Re-Entry Roll Control	18	20-25 msec	25-30 msec

3.4 Environmental and Load Factors.

TABLE B3

MAXIMUM ALLOWABLE LEAKAGE RATE

Valve	Designation	Figure Number	Max Allowable Leakage Rate cc/Min of Working Fluid	Number of Valves
Main Attitude Control fuel propellant valves	V6, V8, V10, V12, V14, V16, V18, V20, V22, V24, V26, V28	B1	0.012	12
Main Attitude Control oxidizer propellant valves	V5, V7, V9, V11, V13, V15, V17, V19, V21, V23, V25, V27	B1	0.018	12
Re-Entry Roll Control fuel propellant valves	V6, V8, V10, V12, V14, V16, V18, V20	B3	0.012	8
Re-Entry Roll Control oxidizer propellant valves	V7, V9, V11, V13, V15, V17, V19, V21	B3	0.018	8
Main Propulsion fuel portion of propellant valve	V1 - V4	B1	0.2	4
Main Propulsion oxidizer portion of propellant valve	V1 - V4	B1	0.2	4

Note. An external leakage rate of ten (10) cubic centimeters per minute of helium at standard temperature and pressure will be allowed for each component in the helium system.

3.4.1 Environmental Conditions.

3.4.1.1 Temperature. The propulsion system dry hardware shall suffer no detrimental effects after exposure to a temperature range of -65°F to 160°F for a period of one year. The fully loaded propulsion system shall suffer no detrimental effects after exposure to a temperature range of 20°F to 80°F for a period of 30 days.

3.4.1.2 Humidity. The propulsion system shall suffer no detrimental effects from exposure to an atmosphere having 95 percent relative humidity throughout the temperature range of 20°F to 80°F for 24 hours.

3.4.1.3 Transportation. The propulsion system dry hardware installed on the spacecraft structural frame shall be capable of transportation in any attitude while contained in an appropriate shipping container (container to be furnished by The General Electric Company) by rail, truck, or air.

3.4.1.4 Vibration. The fully loaded propulsion system installed on the spacecraft structural frame shall not suffer any detrimental effects and shall operate satisfactorily after exposure to the following flight vibrations: to be specified.

3.4.1.5 Radiation. The propulsion system shall not suffer any detrimental effects from exposure to the following (during 14 days of spacecraft cislunar and lunar travel):

a. Cosmic Rays
b. Solar Flares
c. Trapped Radiation of the Van Allen Belts
   (1) Inner belt
   (2) Outer belt
d. Electromagnetic Radiation. solar ultraviolet, x-rays, infrared, and visible.

The intensity of these radiations shall be specified in future revisions of this specification.

3.4.2 Flight Loading Conditions. The rocket engine and its supports shall withstand, without permanent deformation or failure, the largest forces resulting from all critical combinations of load factors and rotational accelerations specified on or within the limits defined by paragraphs 3.4.2.1 and 3.4.2.2. For design purposes, the ultimate strength shall provide for a minimum of 1.5 times the forces resulting from the loading conditions.

3.4.2.1 Boost Phase. Plus 7 1/2 g longitudinal and ± 3 g side loading.

3.4.2.2 Mid-Course Corrections and Lunar Orbit and Disorbit. Zero to 5 g longitudinal.

3.4.2.3 Re-Entry Phase. Not applicable.

3.4.2.4 Space Abort Conditions. 5 g longitudinal.

3.5 Drawings and Data. The following Thiokol Chemical Corporation, Reaction Motors Division drawings and data form a part of this specification.

3.5.1 The following table lists the applicable drawings:

	Drawing Number		
	Main On-board Propulsion	Attitude Control	Re-Entry Roll Control
Installation	52210	52210	52210
Propellant System	52214	52214	52212
Electrical System	52215	52215	52213
Control System			

3.5.2 Not applicable.

3.5.3 System Weights. The weights for each system is tabulated below.

Main On-board Propulsion	
Thrust Chambers (4)	204.0 lb
Fuel Tanks	98.0
Oxidizer Tanks	123.0
Helium Tanks	177.3
Valves and Controls, Misc.	109.3
Dry Weight	711.6 lb
Helium Weight	37.0
Propellant Weight	8,931.0 lb
Main Attitude Control	
Thrust Chambers (12)	12.0 lb
Fuel Tank	4.6
Oxidizer Tank	7.3
Valves and Controls, Misc.	10.7
Dry Weight	34.6 lb
Propellant Weight	390.0 lb
Re-Entry Roll Control	
Thrust Chambers (8)	16.7 lb
Fuel Tank	0.5
Oxidizer Tank	0.8
Valves and Controls, Misc.	23.4
Dry Weight	41.4 lb
Propellant Weight	25.4 lb
Total Loaded System Weight	10,171.0 lb

3.5.4 Envelope Dimensions. The envelope dimensions of the propulsion system shall be those shown on Thiokol Chemical Corporation, Reaction Motors Division Drawing Number 52210.

3.6 Components and Systems.

3.6.1 Electrical System.

3.6.1.1 Electrical Power. All components using electrical power from the vehicle power system shall comply with the requirements of specification MIL-E-7894 except that an input voltage of 24 - 28 volts DC is required.

3.6.2 Ignition System. Ignition results from contact of the hypergolic propellants.

3.6.3 Lubrication System. Not applicable.

3.6.4 Tanks. Tankage data are tabulated in Table B4.

3.7 Fabrication.

3.7.1 Materials.

3.7.1.1 Quality. Materials used in the manufacture of military rocket engines shall be of high quality, suitable for the purpose and shall conform to applicable specifications in accordance with ANA Bulletin No. 343.

3.7.2 Processes.

3.7.2.1 Quality. The use of nongovernmental specifications shall not constitute waiver of Government inspection.

3.7.2.2 Workmanship. The workmanship and finish shall be of sufficiently high grade to insure satisfactory operation, reliability, and durability consistent with the service life and application of the rocket engine.

TABLE B4

TANKAGE DATA

System	No. of Tanks	Fluid	Capacity (each)	Pressure
Main On-board Propulsion	2	Oxidizer	31.75 ft^3	390 psia
	2	Fuel	26.8 ft^3	390 psia
	3	Helium	5.67 ft^3	3000 psia
	1	Oxidizer	3.04 ft^3	150 psia
Main Attitude Control	1	Fuel	2.11 ft^3	150 psia
	The same helium tank is used for both the main vehicle propulsion system and the main attitude control system			
Re-Entry Roll Control	1	Oxidizer	370 $in.^3$	150 psia
	1	Fuel	247 $in.^3$	150 psia
	1	Helium	42.4 $in.^3$	3000 psia

3.7.2.3 Protective Treatment. With the exception of working surfaces and drive pad faces, all parts shall be corrosion resistant or suitably protected.

3.7.3 Standards.

3.7.3.1 Parts. AN, JAN, or MIL Standard parts shall be used wherever they are suitable for the purpose, and shall be identified by their Standard part numbers. The use of nonstandard parts will be acceptable only when standard parts have been determined to be unsuitable.

3.7.3.2 Design. MS and AN Design Standards shall be used wherever applicable.

3.7.3.3 Threads. Conventional straight screw threads shall conform to the requirements of Specification MIL-S-7742. Tapered pipe threads may be employed only for permanently installed fittings or plugs.

3.7.4 Parts List. The parts list for the rocket engine which successfully completes the Qualification tests shall constitute the approved parts list for subsequent engines of the same model. Changes to the approved rocket engine parts list shall be governed by the requirements specified in paragraph 3.7.5 entitled "Changes in Design."

3.7.5 Changes in Design. Changes made in the design or materials of parts listed in an approved rocket engine parts list shall be approved in accordance with the provisions of ANA Bulletin No. 391.

3.7.5.1 Class I Changes. Class I changes affect contract requirements covering weight, performance, cost, interchangeability, or affect durability of either parts or the complete rocket engine.

3.7.5.2 Class II Changes. All other changes are classified as Class II changes.

3.7.5.3 Approval of Changes. Approval of changes does not relieve the contractor of full responsibility for the results of such changes on rocket engine characteristics.

3.8 Identification. Equipment, assemblies, and parts shall be suitably marked and identified.

3.8.1 Connections. The rocket engine shall be permanently marked to indicate all connections shown on the installation drawing for instrumentation, propellant, and other fluid connections. All fluid lines shall be marked in accordance with Drawing AND10375.

MODEL SPECIFICATION FOR TD-281 PROPULSION SYSTEM

4. QUALITY ASSURANCE PROVISIONS

4.1 Classification of Tests. The testing of the systems described herein shall be classified as follows:

a. Preliminary Flight Rating Tests.

The purpose of these tests is to demonstrate the suitability of the Model TD-281 Propulsion System for its intended function.

b. Acceptance Tests.

These tests provide the criteria for acceptance by the purchaser for those TD-281 Propulsion Systems which are manufactured for delivery and flight use.

4.2 Tests and Methods.

4.2.1 Preliminary Flight Rating Tests. A PFRT Specification shall be prepared for approval by The General Electric Company prior to commencing these tests.

4.2.2 Acceptance Tests. An Acceptance Test Specification shall be prepared for approval by The General Electric Company prior to commencing these tests.

APPENDIX C

SAMPLE MALFUNCTION ANALYSIS

APPENDIX C

SAMPLE MALFUNCTION ANALYSIS

Malfunction analyses are performed on rocket engine systems for all manned vehicles. When preparing such an analysis for a manned craft traveling in the earth's atmosphere, a basic criterion must be followed. This criterion is that under any single condition of malfunction the engine(s) will continue running in a safe manner or the engine(s) will shut down safely. For a vehicle traveling in space, this criterion must be altered radically. For a vehicle designed to travel in the earth's atmosphere, an engine that shuts down safely, and is not restartable, is not a problem insofar as human life is concerned. Of course, the mission, all or part, has been lost, but the pilot need only glide his craft back to earth for a safe landing. Now for a spacecraft, if an engine shuts down due to a malfunction, the pilot cannot glide back to earth for a landing. Thus one of the following alternatives must be provided:

(1) Once the engine shuts down, it must be restartable and must run safely.

(2) Redundancy must be provided on the thrust chamber level so that the failed chamber will be replaced by another.

(3) Redundancy and detection units must be provided on the component level to lock out or bypass a malfunctioned component.

(4) Repair in space must be possible and the necessary equipment provided.

It is felt that the final approach will not be any one of the above items, but a combination of any or all of them.

On the forms that follow, a sample of a single malfunction analysis is shown. Several components in the re-entry roll control system were selected at random to provide a feel for the type of detailed single malfunction analysis that would be performed on all the systems down to the last contact point.

Several ground rules had to be established concerning the single malfunction analysis. They are as follows:

(1) As this is only a sample preliminary malfunction analysis, only single conditions of malfunction were considered. A more detailed analysis, made at a later date, would be more comprehensive.

(2) Listed in the first column, under the title of component, is the reference print number that should be consulted to more clearly follow this analysis. Also listed in the column is the component designation and name.

(3) If an "H" appears in the last column to the right, then a protective measure does not exist and provision must be made for eliminating the hazard.

(4) It is assumed that once a malfunction occurs, it remains in the system. Thus a component that malfunctions intermittently is not considered.

(5) It was assumed that if roll control was not available when required, then this constituted a hazard. Actually, it may be that the relative time when a malfunction occurs for a particular flight plan will determine whether a failure is hazardous or not.

On the sheets that follow, item 6 results in an "H" (Hazard to personnel) in the last column. This indicates how a malfunction analysis picks up potentially bad conditions. At this point it is the responsibility of the proper people to take corrective action. A possible solution is indicated on the malfunction sheet.

SINGLE MALFUNCTION ANALYSIS: RE-ENTRY ATTITUDE CONTROL SYSTEM

*LEGEND
H - HAZARD TO PERSONNEL
N - NOT HAZARDOUS

REFERENCE NO. COMPONENT	TYPE OF FAILURE	RESULTING CONDITION	*L	EXISTING PROTECTIVE MEASURE REMARKS	*L
1. 52212 (U2) Dual Squib Operated Pressurizing Valve	1. Inoperative - Closed Position	Pressurizing Helium Gas will not be available to the propellant tanks. Roll control will not be available	H	The squib valve is an extremely reliable device, but in order to increase its reliability dual squibs will be used with two fuse wires in each.	N
	2. Actuates Prematurely	The entire system will be pressurized prematurely	N	The squibs will be shielded so that stray signals will not fire them Also the arm switch (52) 52213 will not supply current until the power switch (S1) 52213 is actuated. Still, if the malfunction does occur, propellants will not be lost as the propellant valves (U6-U21) 52212 will stop flow.	N
2 52212 (RG1) helium Pressure Regulator	1. Inoperative (Jammed Closed)	Pressurizing gas will not be available to expell propellants from the tankage. Decaying or no thrust will be available from the roll chambers	H	Parallel redundancy has been provided with regulator (RG2) 52212. This regulator has the capacity to handle the entire maximum flow rate demand.	N
	2. Inoperative - Open	Full helium bottle pressure will be allowed to enter the propellant tankage. This tankage is not designed to handle pressure of this magnitude.	H	A pressure operated regulator malfunction valve (U3) has been included in the system. This valve is spring loaded and is designed to remain open at the nominal regulated pressure level. At some increment above this nominal pressure the valve (U3) will close and prevent excessive pressure from reaching the propellant tankage. The parallel regulator system will now handle the flow requirements.	N
3. 52212 (RG2) Helium Pressure Regulator	1. Inoperative (Jammed Closed)	Pressurizing gas will not be avilable to expell propellants from the tankage. Decaying or no thrust will be available from the roll chambers.	H	See 2.1 Substitute (RG1 for (RG2)	N
	2 Inoperative - Open	Full helium bottle pressure will be allowed to enter the propellant tankage. This tankage is not designed to handle pressure of this magnitude.	H	See 2.2 Substitute (U4) for (U3)	N
4. 42212 (U3) Pressure Operated Regulator Malfunction Valve.	1. Inoperative - Jammed in the Normally Open Position	This condition will not affect the system under normal operation The only time this valve is required to operate is during a malfunction(wide open)of the regulator (RG1) 52212. Thus if both the regulator and the valve (U3) fail at the same time this constitutes a dual malfunction and is not covered in this analysis	N	The helium relief valve (U5) 52212 will not allow excessive pressure to reach the propellant tankage as it will vent.	N
	2. Inoperative - Jammed in the closed or Actuated Position	See 2.1	H	This valve is designed to actuate only during a malfunction (open)of the regulator (RG1) 52212. Thus if it sticks in the actuated position this constitutes a dual malfunction. however in the event the dual redundancy of the pressurizing system will handle the flow requirements	N

SINGLE MALFUNCTION ANALYSIS: RE-ENTRY ATTITUDE CONTROL SYSTEM

*LEGEND
H - HAZARD TO PERSONNEL
N - NOT HAZARDOUS

REFERENCE NO.

COMPONENT	TYPE OF FAILURE	RESULTING CONDITION	*L	EXISTING PROTECTIVE MEASURE REMARKS	*L
5. 52212 (U4) Pressure Operated Regulator Malfunction Valve	1 Inoperative - Jammed in the Normally Open Position	See 4.1 Substitute (RG2) & (U4) for (RG1) & (U3)	N	See 4.1	N
	2. Inoperative - Jammed in the Closed or Actuated Position	See 2.1	H	See 4.2 Substitute (RG2) for (RG1)	
6. 52212 (U6-U21) Solenoid Operated Propellant Valves.	1 Inoperative - Stuck in the Closed Position	If any one of these valves fails to open when required the particular chamber in question will not develop thrust. The improper amount of roll control thrust plus some vehicle translation will be produced.	H	Four (4) roll control thrust chambers are required for proper operation. Complete redundancy has been provided on the thrust chamber level. Thus if improper roll control is being obtained the four (4) other chambers can be placed in operation as provided to afford correction	N
	2. Inoperative - Stuck in the Open Position	If any one of these valves fails in the open position any time after the system is armed the associated propellant will be lost overboard and the total impulse requirement for roll control will not be met.	H	No protective measure exists for this malfunction The question arises as to whether the system will be armed for a period long enough so that the failed valve will lose enough propellants to cause a hazard. Of course the answer to this is not known at this time so the condition must be termed hazardous.	H
				Remarks A protective measure must be provided. A method would be to install normally closed solenoid valves in each propellant line just downstream of the filters (F1 & F2) 52212. When roll control is required, these additional valves will be cycled open.	N

APPENDIX D

SEQUENCE OF OPERATION, 1966 SYSTEM

APPENDIX D

SEQUENCE OF OPERATION, 1966 SYSTEM

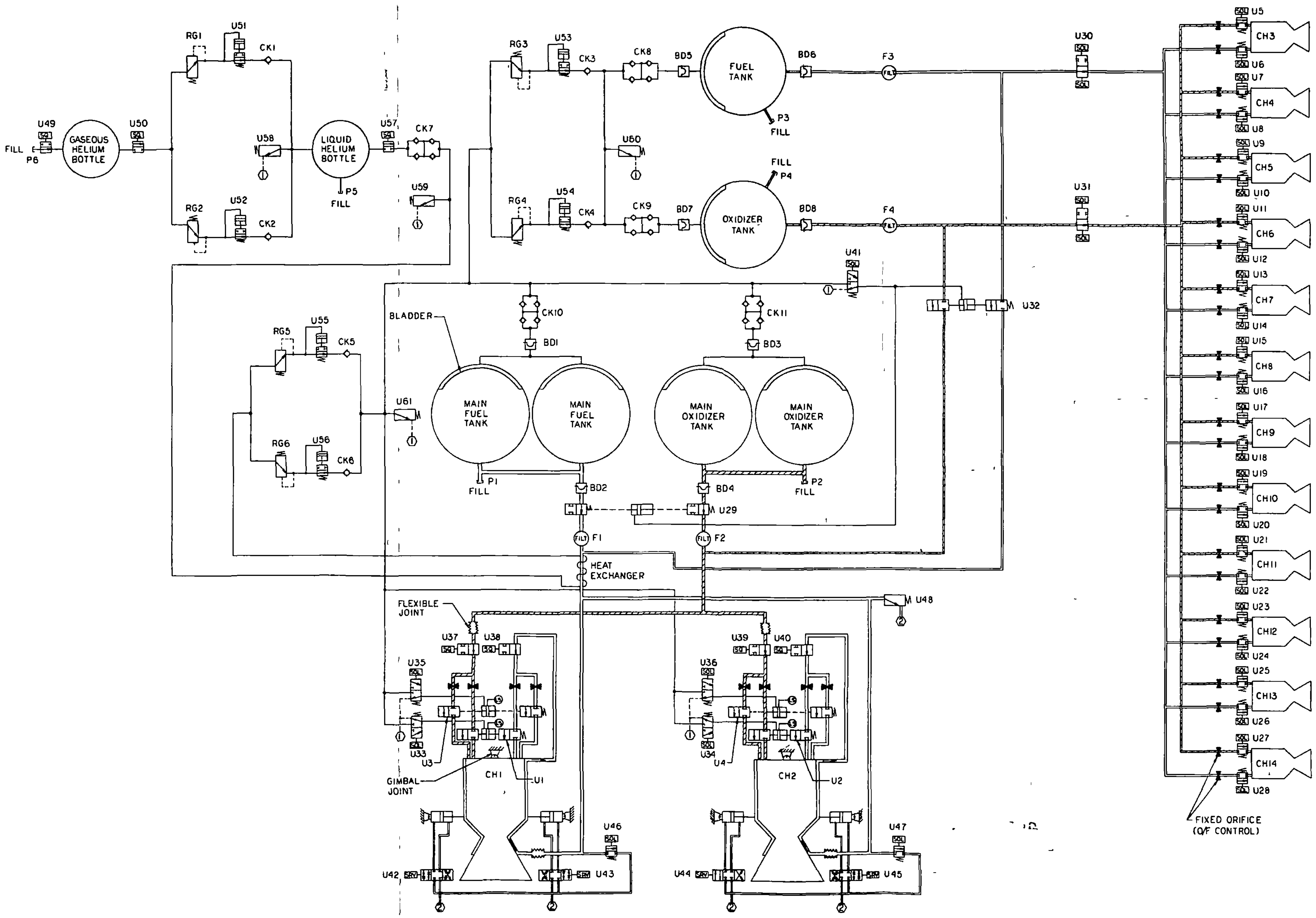
The following discussion is based on Figures D1 and D2, which present the fluid and electrical schematic drawings, respectively, of the 1966 main on-board propulsion system.

A. GROUND SERVICE

The propellant tanks and the helium bottles can be loaded just prior to launch or the fuel tank and the gaseous helium bottle, which are storable, can be loaded prior to launch operations when the spacecraft is in a subassembly stage. The gaseous helium bottle is filled through the dual squib operated helium fill valve (V49) When the helium bottle is filled to the correct pressure, the valve (V49) is fired closed and a fill plug (P6) is then installed in the fill port of the valve to afford greater confidence against leakage from the fill port. The liquid helium bottle is filled through its fill port and when the bottle has been filled with liquid helium the bottle is sealed by installing the fill plug (P5). The attitude control portion of the system contains one fuel tank and one oxidizer tank. The main propulsion portion of the system contains two fuel tanks, which are manifolded together, and two oxidizer tanks manifolded together. As the dual tankage for each propellant is connected, only one fill port is required for each. The oxidizer and fuel tanks are filled through their respective fill ports and once the proper amounts of propellants have been placed in the tanks, the tanks are sealed by installing the fill plugs (P1-P4)

B. PRE-ARM

Prior to arming the system the solenoid operated attitude control propellant valves (V5-V28) can be cycled open through the spacecraft controls to clear the propellant lines when vented to a vacuum atmosphere. This procedure also would check for proper valve operation with the aid of valve limit switches which could be incorporated if desired A defective component could then be electrically locked out of the system The twelve attitude control thrust chambers (CH3-CH14) are operated in pairs to produce a couple. As many as four of



ITEM	DESCRIPTION
CH1 CH2	MAIN THRUST CHAMBER GIMBALED
CH3-CH14	ATTITUDE CONTROL THRUST CHAMBER
U1 U2	1ST STAGE MAIN PROP VALVE PNEU OPERATED
U3 U4	2ND STAGE MAIN PROP VALVE, PNEU OPERATED
U5-U28	ATTITUDE CONTROL PROP VALVE, SOLENOID OPERATED
U29	MAIN PROPELLANT TANK SHUTOFF VALVE, PNEU OPERATED
U30,U31	ATTITUDE CONTROL SYSTEM SHUTOFF VALVE, SOL OPERATED
U32	CROSSFEED PROPELLANT VALVE SOLENOID OPERATED
U33,U34	1ST STAGE MAIN PROP VALVE PILOT VALVE SOL OPERATED
U35,U36	2ND STAGE MAIN PROP VALVE PILOT VALVE SOL OPERATED
U37-U40	MAIN CHAMBER MALF SHUTOFF VALVE SQUIB OPERATED
U41	MAIN PROP TANK SHUTOFF VALVE & CROSSOVER VALVE PILOT VALVE
U42-U45	MAIN CHAMBER GIMBAL VALVE, SERVO OPERATED
U46 U47	SERVO FEED VALVE SOLENOID OPERATED
U48	MAIN FUEL MANIFOLD RELIEF VALVE
U49	GASEOUS HELIUM FILL VALVE, DUAL SQUIB OPERATED
U50	PRESSURIZING VALVE DUAL SQUIB OPERATED
U51-U56	HELIUM REGULATOR MALF SHUTOFF VALVE PNEU OPER
U57	HELIUM SUPPLY VALVE DUAL SQUIB OPERATED
U58-U61	HELIUM PRESSURE RELIEF VALVE
RG1-RG6	PRESSURIZING GAS REGULATOR
CK1-CK6	CHECK VALVE, SINGLE
CK7-CK11	CHECK VALVE, QUAD
BD1-BD4	MAIN PROPELLANT TANK BURST DISC
BD5-BD8	ATTITUDE CONT'L. PROPELLANT TANK BURST DISC
F1-F4	PROPELLANT FILTER
P1-P6	FILL PLUGS

Figure D1. Main On-Board Propulsion System Schematic, 1966

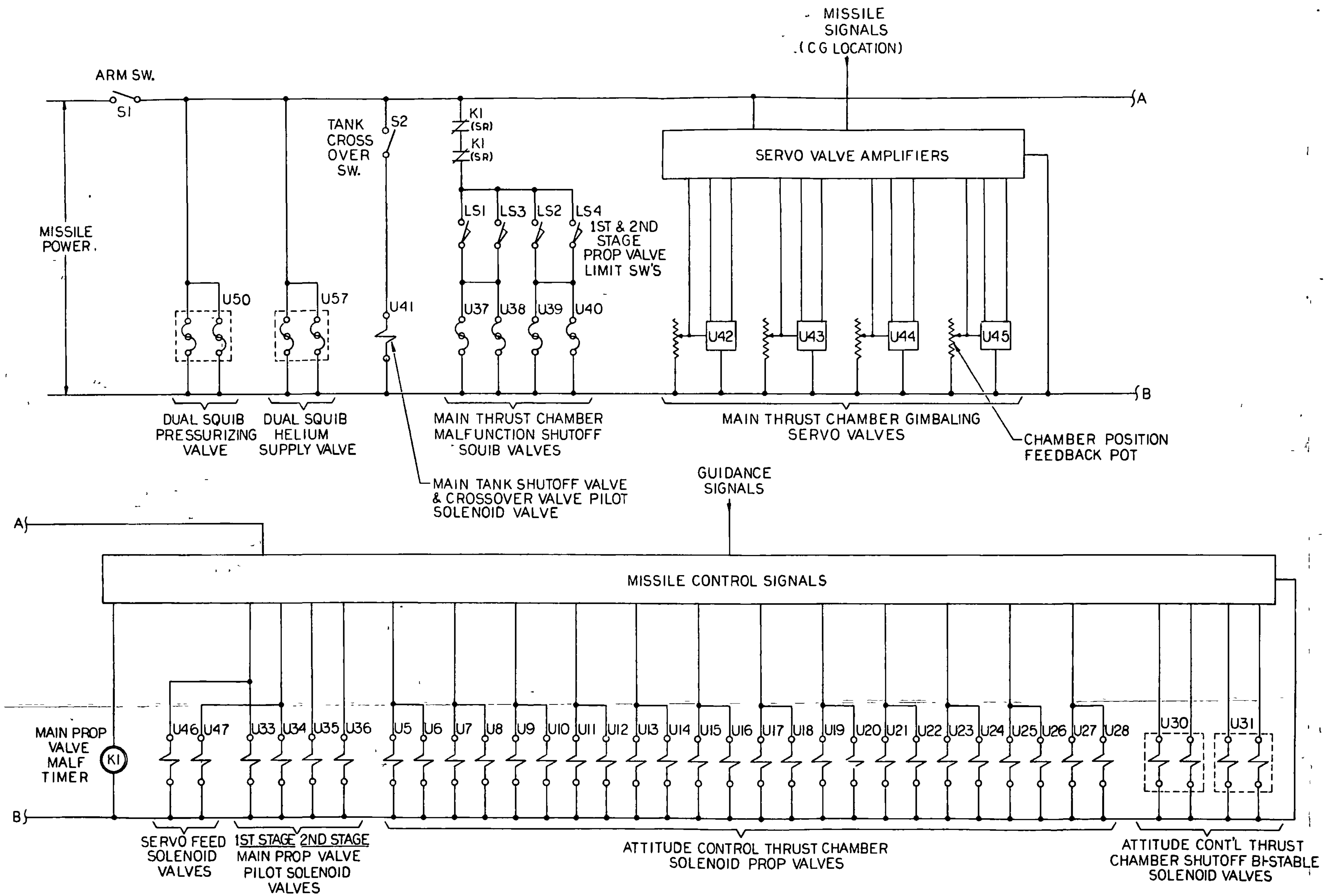


Figure D2. Main On-Board Propulsion System Electrical Schematic, 1966 these thrust chambers could become inoperative and attitude control would still be obtainable, but translation of the vehicle would be involved. In this respect, redundancy is provided for the attitude control system.

C ARM

The system is armed by closing the arm switch (S1) which energizes the electrical control switches and fires open the dual squib pressurizing valves (V50 and V57). The squib portion of this valve is dualized to increase its reliability. Upon opening of these valves, helium gas flows from the gaseous helium bottle to the liquid helium bottle. The helium gas pressure is regulated by parallel pressure regulators (RG1 and RG2) each of which is sized to handle the maximum flowrate requirement of the engine system. In the event of a regulator failure, the other regulator will maintain proper pressure. Downstream of each regulator is a spring loaded helium pressure operated regulator malfunction valve (V51 and V52). The function of these valves is to shut off helium flow in the applicable branch in event of a regulator failure which would result in overpressurization of the liquid helium tank. A check valve (CK1, CK2) is located downstream of each malfunction valve to prevent feedback of high pressure in one branch into the other branch. After leaving these check valves the two branches are joined by a common line in which is located a relief valve (V58) which is sized to compensate for regulator lockup leakage, possible pressure pulses, and expansion due to heat transfer into the liquid helium bottle.

Helium gas flow into the liquid helium bottle will result in heating of the liquid helium. As the liquid helium temperature passes through the critical temperature the liquid helium will be vaporized. The gas mass and temperature in the liquid helium bottle will continue to increase until the bottle pressure reaches the regulated helium pressure. Simultaneously, helium flows from the liquid helium bottle through quadruple check valve (CK7) to the fuel line heat exchanger where it is heated nearly to the fuel temperature. The check valve (CK7) prevents relatively hot helium gas flow from the heat exchanger back to the liquid helium bottle which would result in excessive helium consumption. A relief valve (V59) is provided between the check valve and the heat exchanger to relieve the helium pressure buildup between firings.

From the heat exchanger the helium flows to the main propulsion tanks pressure regulators

1. Main Propulsion Tankage Pressurization

The helium flows from the heat exchanger to the main propulsion tankage pressure regulators (RG 5 and RG6) These regulators and associated malfunction valves (V55, V56), check valves (CK5, CK6), and relief valve (V61) provide the same malfunction protection, pressure regulation and relief features as the regulator system described previously for the liquid helium pressurizing system.

Pressure regulated helium gas flows from the pressure regulating system to the main propulsion tankage system. In the main propulsion tankage system there are two tanks for each propellant. Each pair of tanks, for the same propellant, is manifolded together at both the inlet and exit lines. Each pair of main propellant tanks is fitted with burst disc valves (BD1-BD4) at both the inlet and exit ports. These burst disc valves afford a hermetic seal of the propellant tankage until propellants are required. When the pressurizing gas reaches the inlet port burst disc valves (BD1 and BD3), this pressure will rupture the discs allowing the pressurizing gas to enter all main propellant tankage. Once the gas enters the tanks it pushes against the positive expulsion bladders in each tank and forces the propellant out of the exit ports of each tank. The fluid pressure of the propellants will build up rapidly and will rupture the tank exit burst disc valves (BD2 and BD4). Quadruple check valves (CK10 and CK11) are located at the entrance of both propellant tanks to preclude the possibility of any propellant backflow and intermixing. After the tank exit burst disc valves (BD2 and BD4) have been ruptured, propellants are allowed to flow from the tanks to the main propellant manifolds. From the manifolds the propellants branch out to each individual main thrust chamber (CH1 and CH2) and the flow will be stopped at the individual main propellant valves (V1-V4) and the entire main thrust system will be pressurized. In this process the regeneratively cooled portions of the thrust chambers (CH1, CH2) will be filled with fuel as the main propellant valves (V1-V4) are located downstream of the cooling jackets. Filters (F1 and F2) are located in the propellant lines leading to the propellant manifolds to remove any foreign particles that may exist that could possibly clog the thrust chamber injectors.

2. Control Gas

Pressurized helium gas is extracted from the main propulsion tankage pressure regulating system discharge line and is distributed to five solenoid operated pilot valves (V33-V36, V41) to be utilized as control gas for operating pneumatic actuated valves (V1-V4, V29, V32).

3. Attitude Control Tankage Pressurization

The attitude control propellant tankage is pressurized to approximately one half of the level of the pressure in the main propellant tankage. Thus, another stage of pressure regulation is required. The helium flows from the main propulsion tankage pressure regulating system discharge line to the attitude control tankage pressure regulators (RG3, RG4). These regulators and associated malfunction valves (V53, V54), check valves (CK3, CK4) and relief valve (V60) provide the same malfunction protection, pressure regulation and relief features as the regulator system described previously for the liquid helium pressurizing system. From the common discharge line of the pressure regulating system, the pressurizing gas branches to each attitude control propellant tank. Each attitude control propellant tank is fitted with burst disc valves (BD5-BD8) at both the inlet and exit ports. These burst disc valves (BD5-BD8) afford a hermetic seal of the attitude control propellant tankage until propellants are required. When the pressurizing gas reaches the inlet port burst disc valves (BD5 and BD7) its pressure will rupture the discs allowing the pressurizing gas to enter both tanks. Once the gas enters the tanks it pushes against the positive expulsion bladders in each tank and forces the propellant out of the exit ports of each tank. The fluid pressure of the propellants will build up rapidly and will rupture the tank exit burst disc valves (BD6 and BD8). Quadruple check valves (CK8 and CK9) are located at the entrance of both attitude control propellant tanks to preclude the possibility of any propellant backflow and intermixing. After the tank exit burst disc valves (BD6 and BD8) have been ruptured, propellants are allowed to flow from the tanks to the propellant manifolds. From the manifolds the propellants branch out to each individual thrust chamber (CH3-CH14) and the flow will be stopped at the individual propellant valves (V5-V28) and the entire system will be pressurized. Filters (F3 and F4) are located in the propellant lines leading to the attitude control propellant manifolds to remove any foreign particles that may exist that could possibly clog the thrust chamber injectors.

D. FIRE

1. Attitude Control System

Once the system has been armed, any one or any combination of attitude control thrust chambers (CH3-CH14) can be fired upon command from the vehicle controls. Both propellant valves on the chamber required to fire, will open and the propellants will enter the combustion chamber. Ignition will occur upon contact of the two hypergolic propellants. The chambers can be fired for long durations or they can be pulsed 10-15 cycles per second. Fixed O/F control orifices are shown in each propellant line These orifices will be part of the propellant valves and will be sized during or just prior to acceptance tests.

In the event that any of the altitude control propellant valves (V5-V28) sticks in the open position, propellant will be lost through this open valve. The fact that propellant is being expended when no thrust chamber is required to be fired will be sensed by the vehicle's propellant management system. Once the malfunction has been detected, the vehicle controls will signal the appropriate solenoid operated, bistable, attitude control system shutoff valve (V30-V31) to close. Thus no propellant will be lost through a malfunctioned open attitude control thrust chamber propellant valve (V5-V28) during extended periods of shutdown. The action of the bistable (dual solenoid) valve (V30 and V31) is such that when the appropriate solenoid is energized, the normally open valve is actuated closed and held in the closed position by a detent. In this way power to the actuating solenoid can be removed during the expected long periods of attitude control thrust chamber shutdown. When the attitude control system is required to operate, the vehicle controls will energize the other solenoid on the bistable valve which will remove the detent and allow the valve to return to its normally open position under the action of a spring. The above procedure will repeat on any subsequent shutdown-start cycle.

2. Main Propulsion System

With the system armed, either or both of the first stages of the two main thrust chambers can be fired upon command from the vehicle controls. When a particular thrust chamber first stage is required to fire, the control system will energize the appropriate solenoid operated gimbal servo feed valve (V46, V47) and its corresponding solenoid operated first stage propellant valve pilot valve (V33, V34). Operating the servo feed valve (V46, V47) will allow pressurized fuel to be available to the corresponding servo operated main thrust chamber gimbal valve (V42-V45). The gimbal valves (V42-V45) receiving information from the spacecraft controls, along with data from the chamber position feedback potentiometer, will provide pressurized fuel to the thrust chamber gimbal actuators in such a way as to position the thrust vector through the instantaneous center of gravity. While this is occurring the first stage propellant valve pilot valve (V33, V34) has opened and allowed control gas (helium) to enter the actuating piston of the pneumatic operated first stage propellant valve (V1, V2). The control gas will force the propellant valve

(V1, V2) to the open position and both propellants will enter the combustion chamber (CH1, CH2). Ignition will occur upon contact of the two hypergolic propellants. With the thrust chamber first stage in operation, the thrust chamber second stage can be fired upon command from the vehicle controls. When a particular thrust chamber second stage is required to fire, the control system will energize the appropriate solenoid operated second stage propellant valve pilot valve (V35, V36). The second stage propellant valve pilot valve opens and allows control gas to enter the actuating piston of the pneumatic actuated second stage propellant valve (V3, V4). The control gas will force the second stage propellant valve open and both propellants will enter the thrust chamber and ignite. Fixed O/F control orifices are shown in each propellant line. These orifices will be part of the propellant valves and will be sized during or just prior to acceptance tests. Once the required vehicle velocity correction has been obtained the vehicle controls will signal the thrust chamber (CH1, CH2) in operation to shut down by discontinuing power to the appropriate propellant valve pilot valve (V33, V36) and servo fluid valve (V46, V47). Discontinuing power to the propellant valve pilot valve (V33-V36) will allow it to return to its normal position, under a spring load, and the control gas will be shut off and the actuating piston of the propellant valve (V1-V4) will have its pressurizing gas vented to atmosphere by means of appropriate porting in the pilot valve (V33-V36). Once the pressurizing gas is vented, the propellant valve (V1-V4) will close under the action of its spring loading. With the flow of propellants discontinued, the main thrust chamber (CH1, CH2) will cease operation.

In the event that a main thrust chamber stage does not discontinue to operate when required, erroneous velocity correction and loss of propellant will occur as well as a hazardous condition. To eliminate this condition, the following safety circuitry has been incorporated.

Whenever a thrust chamber is fired, a slow release timer is energized. Energizing this timer will cause its two normally closed contacts (K1) to open. Propellant valve limit switches (LS1-LS4) are located physically on each of the propellant valve actuators so that when the valve is in the fully closed position, the limit switch is held open. The limit switches are wired to the K1 contacts so that if the K1 timer is not energized and a propellant valve is not fully closed, squib operated shutoff valves (V37-V40) will be fired, and close off the propellant flow to the applicable thrust chamber. This thrust chamber will thereafter be inoperable.

Thus when the vehicle controls signal an operating thrust chamber to shut down, power will be discontinued to the main propellant valve malfunction timer (K1). This will start the timer on its count (2-4 seconds). When this count has been completed, the timer contacts (K1) will return to their normally closed positions. If the particular propellant valve (V1-V4) in question has not returned to its fully closed position, then the corresponding limit switch (LS1-LS4) will remain in the closed position and a circuit will be completed through the normally closed K1 contacts. This completed circuit will fire the appropriate pair of squib operated main chamber malfunction shutoff valves (V37-V40) which will stop propellant flow to the main thrust chamber (CH1, CH2) in question.

Located in the main fuel manifold is a relief valve (V48) that is sized to handle excess fuel vapor pressure resulting from the residual main thrust chamber (CH1-CH2) heat on shutdown.

An additional requirement imposed on the entire system was that provision shall be made for operating the main thrust chambers on propellants stored in the attitude control propellant tankage. This was accomplished by the inclusion of the pneumatically operated cross-feed propellant valve (V32). When this feature is required, the solenoid operated main propellant shutoff and cross-feed valve pilot valve (V41) is energized by the vehicle controls allowing control gas (helium) to enter the actuating piston of the pneumatic operated cross-feed propellant valve (V32). This will force this valve to the open position and allow propellant flow to the main thrust chamber propellant manifolds. At this time either or both of the main thrust chambers (CH1, CH2) can be fired, as described previously. Concurrent with the valve action above, the pilot valve (V41) also allows control gas (helium) to enter the actuating piston of the pneumatically operated main propellant tank shutoff valve (V29). This will force this valve to the closed position thus precluding the possibility of any type of flow in either direction during the cross-feed operation of one of the main thrust chambers (CH1, CH2). The system is returned to normal by deactuating the pilot valve (V41) which shuts off the control gas (helium) and vents to atmosphere the pressurizing gases in the actuating cylinders of the two propellant valves (V29 and V32).

APPENDIX E

RELIABILITY ORGANIZATION AND METHODS

APPENDIX E

RELIABILITY ORGANIZATION AND METHODS

A. RELIABILITY ORGANIZATION

The Thiokol Chemical Corporation, Reaction Motors Division reliability engineering section is functionally aligned to apply the specialized talents and techniques of the reliability concept in an efficient manner. A functional organization chart is given in Table E1 which shows the four basic areas of reliability control i.e., vendor, design engineering, manufacturing, and development testing, and gives a broad definition of the functional elements of control. The organization is "in line" with the Chief Engineer who is directly responsible for the conduct of the program.

Activities are coordinated with the requirements of the program by the reliability engineering section which insures that adequate emphasis is placed on reliability requirements in the detail design, the product specifications, and the development test specifications. In addition, the group assures that the testing program does define design capabilities, that corrective action, if needed, is prompt and that reliability growth rate is adequate to meet the reliability goals relative to development schedule.

1. Reliability Experience at Thiokol-RMD

Thiokol-RMD has been active in statistical reliability analysis, studies, programs, and demonstration for over six years. Several of the programs and demonstrations are given below:

a. XLR99 Rocket Engine Program

The XLR99 rocket engine system is dramatic evidence of the effectiveness of the procedures and philosophies outlined above. The XLR99 engine system fulfills the extremely complex requirements of manned rated safety together with high reliability and at the same time provides wide range throttling, unlimited restartability, and an idle capability which are all under the direct control of the human pilot. This engine is presently operational in the X-15 aerospacecraft.

TABLE E1

RELIABILITY ENGINEERING SECTION FUNCTIONAL CHART

Reliability Engineering Section

Reliability Control Vendor	Reliability Control Design Engineering	Reliability Control Development Testing	Reliability Control Manufacturing
Vendor Reliability Review	Design and Specification Review	Reliability Analysis and Evaluation	Quality and Inspection Standards Review
Vendor Quality Review	Engineering Procedure Review	Reliability Test Planning	Process Control Review
Associate Contractor Liaison	Environmental Activities Analysis	Reliability Records and Reports	Quality Test Specifications Review
	Reliability Analysis and Production	Field Test Liaison	Acceptance Specification Review
	Indoctrination and Training		Statistical Quality Control Standards Review
	Reliability Control Procedures		Material Review
			Reliability and Quality Audit

Although reliability was not a contractual requirement, reliability disciplines were applied throughout the program with the result that the engine system demonstrated 98.5 actual experienced reliability in a special demonstration following PFRT (after a 3-year development program) and has currently demonstrated 100% actual experienced mission reliability. In all tests to date, engine safety has been 100%. This record is shown below in the following table.

	No. of Runs	Actual Experienced Reliability	A Posteriori Reliability at 90% Confidence Level
Post PFRT Demonstration	68	98.5%	*94.4%
Field Operation**	300+	97.0%	*95.0%
Mission	5	100%	*74.0%
Combined Overall	373	97.8%	95.5%

* Statistical interpretation: From the results experienced in the limited numbers of tests involved, there is a 90% probability that the true reliability is greater than the reliability shown.

** Counts all engine runs made in the field (EAFB). Includes training runs, checkout runs, and flight operations with various crews and pilots.

b. Titan Nose Cone

In this program Thiokol-RMD developed the Attitude Control System (ACS) for the nose cone. Reliability was a primary contractual obligation. The requirements were 99.9% at a 95% confidence level.

In this program, special statistical techniques were developed to demonstrate system, subsystem, and component reliabilities. Sample size, test programs, and testing techniques were developed to attain maximum information at a minimum cost consistent with the stringent reliability requirements.

Special test-to-failure techniques were utilized at the component level, while subsystem and system levels were reliability tested utilizing the basic theory and principles of sequential analysis. The reliability test program and techniques were approved and adopted by the procuring activity.

c. Corvus Program

In this program a complete rocket engine system was developed for the Corvus missile system. Thiokol-RMD was one of the first companies to perform under contractual obligation a "Reliability Monitoring Program," and developed methods of reporting system and subsystem reliability, failure mode analysis, and reliability predicting techniques. All testing in development, PFRT, and acceptance was continually monitored for reliability purposes. Maximum utilization of test data was obtained to determine system reliability. Reliability progress was reported to the procuring activity in monthly reliability analysis reports. With this method of extracting maximum data from every test accomplished, design deficiencies were discovered in the program and corrective action followed. Reliability could be determine without entering a costly reliability demonstration program.

B. RELIABILITY METHODS

1. Definitions

To avoid confusion in discussions of reliability it is useful to distinguish between general types of reliability numbers.

Reliability Goals: These values result from the apportionment of the customer reliability goal for the system into reliability goals for components and subsystems. These numbers represent requirements that must be met by components and subsystems if the system goal is to be achieved. They do not represent predictions or estimates of the reliability that can be, or have been, achieved by the equipment.

Reliability Estimates: These values, empirically derived from a specific set of test results, represent the best estimates of currently achieved system reliability.

True Reliability: The true reliability is that value of success/failure probability that would be observed if an infinite number of samples were tested. It cannot be measured but is a statistical parameter of importance in discussing reliability.

Calculated Reliability: System reliability may be calculated by using an appropriate mathematical combinator of the reliabilities of the lower level parts that make up the system. When system reliability is determined this way, the resulting number is a calculated reliability. Reliability predictions and estimates can both be calculated values in this sense.

2. Reliability Concepts

It is important that the concept of inherent reliability and the degradation effects of manufacturing, handling and use be understood.

Inherent Reliability: This is the reliability which has been designed into a component, subsystem, or system. It is strictly a paper reliability value based upon malfunction analysis and failure mode analysis of the individual parts of the component. It is the reliability given to the product by the design and is sometimes referred to as design reliability prediction.

Manufactured Reliability. This definition of reliability implies the reliability of the manufactured product, i.e., the product cannot be fabricated exactly to the original design since there are inherent variances in any manufacturing process which tend to alter the inherent reliability of the product. These variances coupled with the basic material variances result in the reliability of the manufactured product.

Use Reliability: This definition of reliability is influenced by the field use a product may receive during its life cycle. Such factors as capability of operating and maintenance personnel; operating and maintenance procedures; auxiliary and support equipment; effects of shipping, storage, and handling all affect the reliability of the product. The resulting reliability as a function of field use is designated as "Use Reliability".

Operational Reliability: This definition is the basic reliability value influencing the customer. It is the actual reliability received from the product, taking into account the inherent reliability, manufactured reliability, and use reliability. Operational reliability can be defined as follows:

$$R_O = (R_I)(R_M)(R_U)$$

R_O = Operational Reliability

R_I = Inherent Reliability

R_M = Manufactured Reliability

R_U = Use Reliability

It is clear from the foregoing that the first step in achieving reliability is adequate design for the environment anticipated during the life cycle of the system.

3. Designing for Reliability

Experience has shown that highly complex equipment can be reliable only through the implementation of sound engineering practices coupled with a thorough, integrated test and product program.

The Thiokol-RMD approach to reliability is based on the premise that reliability is a parameter which can be quantitatively specified, estimated, assessed, and measured at predesignated points of a system's life cycle. It is also based upon the premise that reliability can be controlled throughout the phases of design, development, manufacture, and use. While reliability is stated in terms of probability and measured by statistical methods, it is basically achieved and improved in a product by good engineering practices and technical ability founded on a thorough understanding of the sciences involved. Thus the inherent reliability of a system is determined by the validity of the basic design. The degree of reliability actually realized also reflects the care, skill, and control exercised in the systems manufacture and the degree to which it is properly employed and maintained.

The inherent or design reliability of a product is established and determined by its basic design. The design engineer and reliability engineer are responsible for achieving reliability in their equipment. Therefore, maximum reliability effort should be applied in the design phase of any product. During this design phase the reliability engineer offers specialized assistance to the designer to fulfill the responsibility of reliable products.

It must be remembered that manufacture and use cannot improve reliability, only degrade the inherent reliability. For this reason the inherent reliability given to a product by the designer must be extremely high. Remember also that reliability cannot be tested into a product, it must be designed with it. Testing is essential, however, to determine the level of reliability and to insure the quality of the parts contributing to it. Therefore, the reliability problem reduces to a maximum effort applied to a product during the design phase. In keeping with this philosophy the following summary of procedures should be followed in an effort to achieve high inherent reliability.

a. Prior to Design

(1) Establish a complete system schematic indicating the components required to perform the overall system function.

(2) Establish the overall reliability requirements of the system at a specified confidence level.

(3) Reduce the system schematic to a functional block diagram in which each block represents a system component.

(4) In each component block specify the component type with operating parameters such as pressure levels, flowrates, temperature limits, time, etc. Be complete and detailed during this phase as the component block serves as a useful tool to the designer and specification engineer; it spells out the component requirements to fulfill system requirements.

(5) In each component block specify a predicted reliability based upon past performance of the component type under similar environments. Reduce all components to series operation striving to maintain independence of the components. Then utilizing the product rule calculate the system reliability and compare to specified reliability requirements. This procedure is fairly easy to apply and will yield the following information:

- A measure of design adequacy relative to reliability

- Estimation of the degree of redundancy required to achieve a specific reliability.

- Indication of area in which design changes would be most beneficial or necessary.

- Basis for comparing two or more designs.

(6) If the calculated reliability is below the specified reliability, determine the amount of redundancy required by adding parallel components. The parallel components can be either of the following:

- Concurrent Elements: In concurrent elements all parallel elements are operating throughout the entire time period and the reliability of each remains constant.

  The reliability of n such items is expressed as follows:

  $$R = 1 - (1 - R_1)(1 - R_2) - - - - - - - - - - - - - - (1 - R_n)$$

  If all n elements are identical then

  $$R = 1 - (1 - R_1)^n$$

  When $n = 2$

  $$R = 2R_1 - R_1^2$$

- Time-Sequenced elements: In time-sequenced elements the second does not begin to operate until the first has failed, the third does not operate until the second has failed, and so on. It is assumed that the component failure probability is essentially zero when the component is not operating. For n elements which have the same failure rate λ, the overall reliability is:

  $$R = e^{-\lambda t} \sum_{i=0}^{i=n-1} \frac{(\lambda t)^i}{i!}$$

  t is the entire operating period for the group. If only two such components have the common reliability R_1, the formula reduces to the following approximation:

  $$R = R_1 (1 - \ln R_1)$$

In the foregoing discussion of parallel components, the reliability of malfunction detection and switching devices has been assumed to be unity. Modifications must be made if this assumption does not hold.

(7) Once the calculated reliability approaches the specified value, the components which comprise the system can enter the design phase.

(8) System analysis continues into the Malfunction Analysis Phase. In this phase each component of the system in turn is assumed to fail in the actuated and deactuated position. The influence on system operation is analyzed. Any conditions which result in potential hazards to the vehicle or human on board must be eliminated. Other malfunction conditions which prevent desired system operation are tabulated.

(9) Present to each designer the method in which his component is integrated into the overall system. Explain to him the influence on system performance if his equipment fails.

(10) Write a complete detailed specification for each component including packaging, handling, transportation, and storage requirements. Utilize the component block diagram as a guide, section (4) above.

(11) Minimize the number of components through proper design techniques; utilize systems and combine functions wherever possible. Strive for reduction of individual component complexity, for simplicity of design.

b. During Design

(1) Establish realistic design and performance specifications for the production design; coordinate closely with Production Engineering on their capabilities of meeting the designers requirements.

(2) Design items which can be made; redesign and modify to achieve reliability within capabilities of production techniques.

(3) Utilizing the results of the malfunction analysis performed in section a (8) above, list the types of malfunctions which prevent desired system operation for each component. Then design the individual component to eliminate each type failure.

(4) Perform "Reliability Design Reviews" as the design of each component progresses.

Analyze every possible mode of failure within each component, i.e., clearance, jamming, leakage, etc.

Determine what modes of failure will cause the entire component to fail and the consequent influence upon system operation.

Through redesign eliminate as many failure modes as possible.

(5) Specify all acceptance and approval criteria for each component.

(6) Perform a reliability prediction analysis on each component. In this analysis all possible causes of component failure are tabulated. The probability of occurrence of each failure is estimated by several qualified people based on past experience. Then the probability of component failure if the individual failure occurs is estimated. Then by utilizing the following formula an estimated component reliability can be achieved.

$$R_c = \prod_{i=1}^{i=n} \left(1 - P\left\{f_i\right\} \; P\left\{F/f_i\right\}\right)$$

R_c = Component Overall Reliability

$\prod\limits_{i=1}^{i=n}$ = Product of n Terms

$f_i = i^{th}$ = Cause of Failure of the Component Type

$P\left\{f_i\right\}$ = Probability of Occurrence of f_i

$P\left\{F/f_i\right\}$ = Probability of Component Failure if f_i Occurs.

The above prediction phase serves the following:

- Makes known the possible causes of component failure; makes designer and others aware of the types of failure which are possible.

- Creates an atmosphere to eliminate failure modes.
- Gives an indication of how design phase is progressing compared to initial predicted reliability and specified reliability.
- Points to components requiring additional design and redesign effort.

c. Completion of Design

(1) Upon completion of each component design, perform a complete design review. The design review is accomplished by Reliability Engineering in which the complete design is analyzed from the detailed specification to insure all requirements are consistent with design. Failure mode analysis is again accomplished. Consultation between design engineering and reliability engineering eliminates as many modes of failure as feasible through proper design techniques.

(2) Production Engineering reviews all drawings to insure that the design is within their capabilities.

4. Reliability Control in Manufacturing

The validity of reliability evaluations is dependent on an assumption that the future quantities of components and systems consistently represent the design. This assumption requires that reproducible manufacturing methods and equipment are utilized together with effective process controls.

The control function which assures that such controls are implemented is Quality Engineering.

The primary objective of Quality Engineering is to assure that the inherent reliability of the design is maintained during the manufacture, handling, storage, and use of the system. Included in the function are process and acceptance inspection control, laboratory test control, and statistical quality control activities.

The functions are accomplished by establishing and enforcing procedures covering:

- Statistical Control of processes
- Correction or removal of all parts, materials, etc. which are outside usable limits
- Quality evaluations and audit.

These activities are monitored by the reliability engineering group which is responsible for the control specifications utilized.

APPENDIX F

OXYGEN DIFLUORIDE (OF_2) BULK TEMPERATURE AS A FUNCTION OF MISSION TIME FOR A HOT GAS PRESSURIZATION SYSTEM

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OXYGEN DIFLUORIDE (OF_2) BULK TEMPERATURE AS A FUNCTION OF MISSION TIME FOR A HOT GAS PRESSURIZATION SYSTEM

The temperature of the OF_2 stored in the two OF_2 tanks was calculated as a function of mission time. The tanks were assumed to be perfectly insulated so that there was minimum heat loss to the environment. The heat input from the hot pressurizing gases was then transferred to the OF_2 remaining in the tank. The final temperature was the equilibrium temperature between the pressurizing gases (assumed injected at 700F) and the liquid OF_2 This was done for the thrust chamber operating profile shown in Table F1 The heat capacity of the bladder separating the gases from the OF_2 was negelected. The initial OF_2 temperature was assumed to be that of its saturation temperature at 1 atm (230R). Figure F1 shows that the final OF_2 temperature is below its saturation temperature at 400 psi (500R), assuming a 10% oxidizer reserve in the tanks.

The above calculation model assumed that enough time between firings was available so that equilibrium was reached between the gas and the liquid OF_2. In the transient heating period, because of the effect of zero gravity, the heat transferred from the gas to the liquid would be by conduction. Temperature gradients then would exist in the liquid. When the interface temperature of the liquid (gas-bladder-liquid) reaches its saturation temperature, local vapor formation occurs. This vapor boundary will act as an insulation layer, and the time to reach equilibrium conditions would be increased. Radiation to space from the outside of the tank surface will then lower the pressurizing gas temperature, and the final temperature of the OF_2 liquid will be less than predicted here. The presence of the layer of OF_2 vapor formed by boiling at the surface will tend to move the flexible bladder outward and form a larger vapor pocket.

TABLE F1

THRUST CHAMBER OPERATING PROFILE -
APOLLO VEHICLE
1966 SYSTEM

Elapsed Time to Start of Firing, hours	Duration at 6K Thrust, seconds
Zero (projection into escape orbit)	
0.5	4
21.5	4
42.5	4
63.5	4
83.5	4
84	118
86	118
251	86
252	86
252.5	2
273.5	2
294.5	2
315.5	2
335.5	2
336	Re-entry

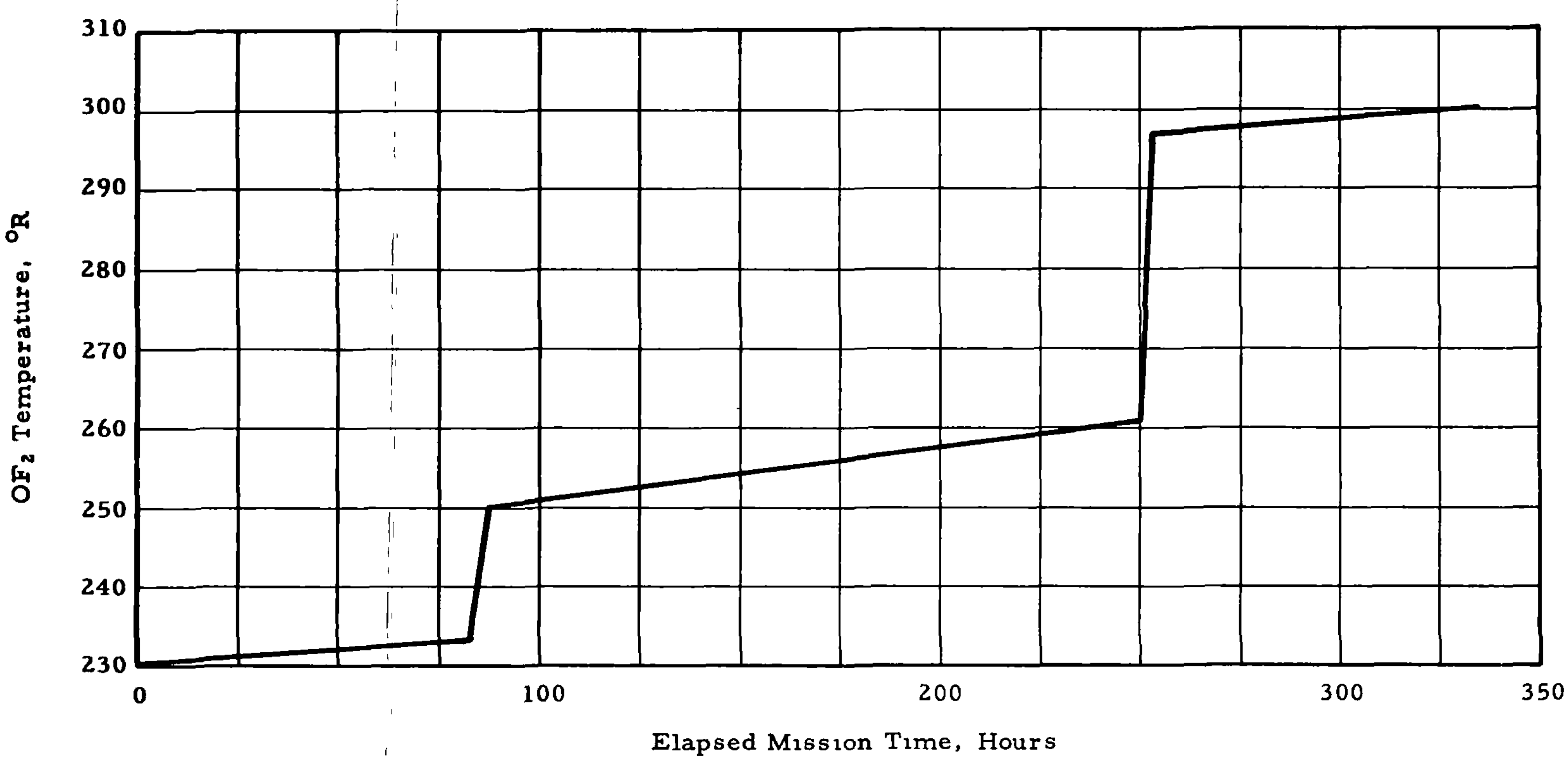
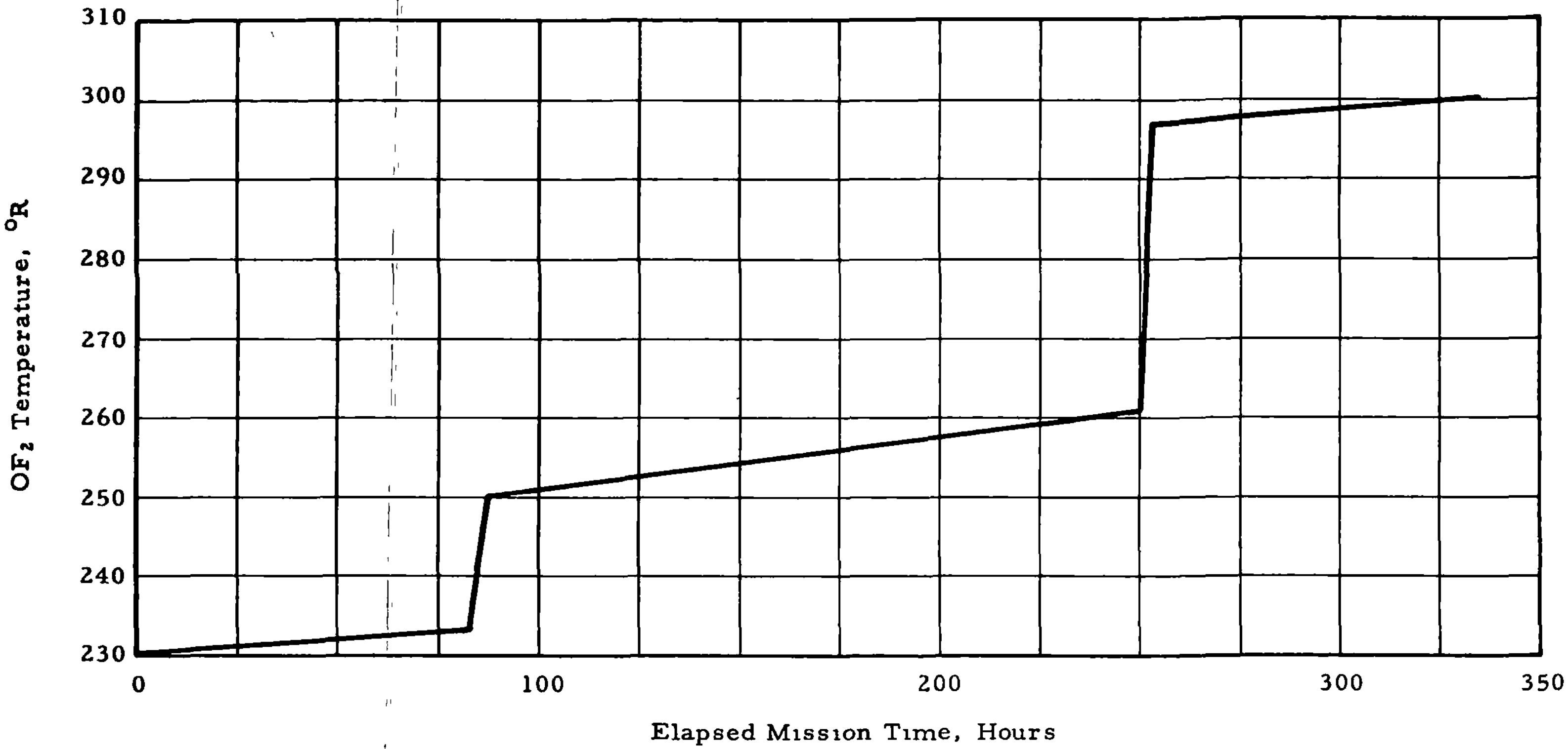


Figure F1. OF_2 Bulk Temperature as a Function of Operation Profile



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