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CR-133677

MISSILE AND SPACE VEHICLE DEPARTMENT

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME IV. ON-BOARD PROPULSION

Book 3 — Appendix P-B

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NASA Contract NAS 5-302

PROJECT APOLLO

A Feasibility Study of an Advanced
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FINAL REPORT

VOLUME IV. ON-BOARD PROPULSION
Book 3 — Appendix P-B

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10 refs

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
3198 Chestnut Street, Philadelphia 4, Penna.

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FINAL REPORT

APOLLO PROPULSION

SYSTEMS

REPORT NO. 7110-945003

REVISION A

1 MAY 1961

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APPENDICES

I. INTRODUCTION

This report presents the results of the propulsion system studies conducted by the Bell Aerosystems Company as part of the Apollo Advanced Manned Spacecraft and System Feasibility Study of GE/MSVD. After careful analysis of the requirements for the on-board propulsion as defined by the Spacecraft and System Study and with particular emphasis placed on the guide lines specified for propulsion by the NASA in the original RFP-302, namely: A. Multipurpose, B. Multiple firings, C. Reliability and D. Redundancy, the preliminary design of the on-board propulsion system presented herein was prepared.

The on-board propulsion consists of the main propulsion system for the lunar orbit mission and two auxiliary systems, one for ullage/attitude control in the propulsion module and the second for attitude control in the re-entry vehicle.

II. SUMMARY

A main propulsion system, a mission attitude control system and a re-entry vehicle attitude control system are defined based in part on requirements set forth by GE/MSVD. Those requirements for the 14 day, manned, lunar Apollo mission include an upper thrust capability of 24,000 pounds for mission abort and a maximum of 15 restarts of the main engines in space for course correction and to achieve lunar orbit and disorbit. The main propulsion and mission attitude control systems are mounted and supported in one complete spacecraft section adhering to an envelope definition of essentially the frustrum of a right circular cone with base and top drameters of 18 and 10 feet. The total impulse capabilities of the main propulsion system and mission attitude control systems are approximately 3.0×10^6 and 74,000 pound-seconds, respectively.

The propellant combinations selected for the main propulsion system, liquid fluorine and liquid hydrogen, and both attitude control systems, mixed oxides of nitrogen and un-symmetrical dimethyle hydrazine, are based on a critical analysis of mission performance requirements and the necessity for achieving the highest reliability. System design philosophy emphasizes the utilization of the highest performance propellant combination F_2-H_2 in an overall system utilizing only highly reliable state-of-the-art concepts; the majority of which have been demonstrated by the 100% flight success of the Bell Aerosystems Discoverer/ Agena Engines. Feasibility firings of an F_2-H_2 engine utilizing modified Agena engine components, have been conducted at Bell Aerosystems Company. The MON/UDMH combination provides the desired high performance for the requirements of vehicle attitude control. A MON/UDMH secondary propulsion system for upper stage vehicles is in the final stages of development at Bell Aerosystems Company.

The main propulsion system is defined to operate one or both pump fed 12,000 pound thrust chambers for the lunar orbit and disorbit firings expending approximately 93% of the total usable weight of propellants. Midcourse corrections are accomplished from a separate Helium pressure feed system and represent approximately 73% of the total number of firings for the maximum mission. Pressure fed firings are made using the two main engines step thrusted to 4000 pounds. There is a negligible loss of reliability between the pump/pressure fed system and an equivalent all-helium pressure fed system. The pump/pressure fed system is shown to be approximately 735 pounds lighter than the equivalent pressure fed system.

Detailed technical discussions, schematics, performance, preliminary design sketches, and estimated weight breakdowns of all components and subsystems are presented. Systems operating sequences, safety shut down controls and monitoring instrumentation are reviewed. Sections of the report are devoted to reliability analysis, background and experience, effects of cosmic radiation, and micrometeorites. A program plan and the facilities available and additional facilities required for its implementation are presented.

III. REQUIREMENTS

The final detailed requirements for the Apollo mission propulsion and attitude control systems were transmitted to BAC from G.E. MSVD. Two time periods were to be considered, mid-1963 and mid-1966. The later version of the propulsion systems should reflect possible weight savings and performance increases available with the additional 3 years of research and development. The 1963 and 1966 versions must be capable of operation with the Bell glide vehicle and the G.E. MSVD semi-ballistic vehicle. The maximum propulsion system envelope is presented in Figure III-1. Additional requirements are as follows:

A. Vehicle Weight

Vehicle weight at escape 15,715 (1963)
14,715 (1966)

B. Attitude Control

	Mission System	Re-entry Vehicle System
Total impulse, lb. sec.	60,000	7,000
Maximum Number of Starts	3,000	500
Maximum Single Impulse, lb. sec.	200	100
Unit Thrust, lbs.	3	18
Number of Units	12	4
Location of Units, Lever Arm, ft.	9	9

C. Mid-Course Correction (outbound)

Outbound ΔV, feet/second	250
Minimum g's	.25
Maximum g's	1.5
Number of starts (maximum)	5

D. Entering Lunar Orbit

Required ΔV, feet/second	3,500
Minimum g's	.25
Maximum g's	1.5
Number of starts	2*

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

[redacted]

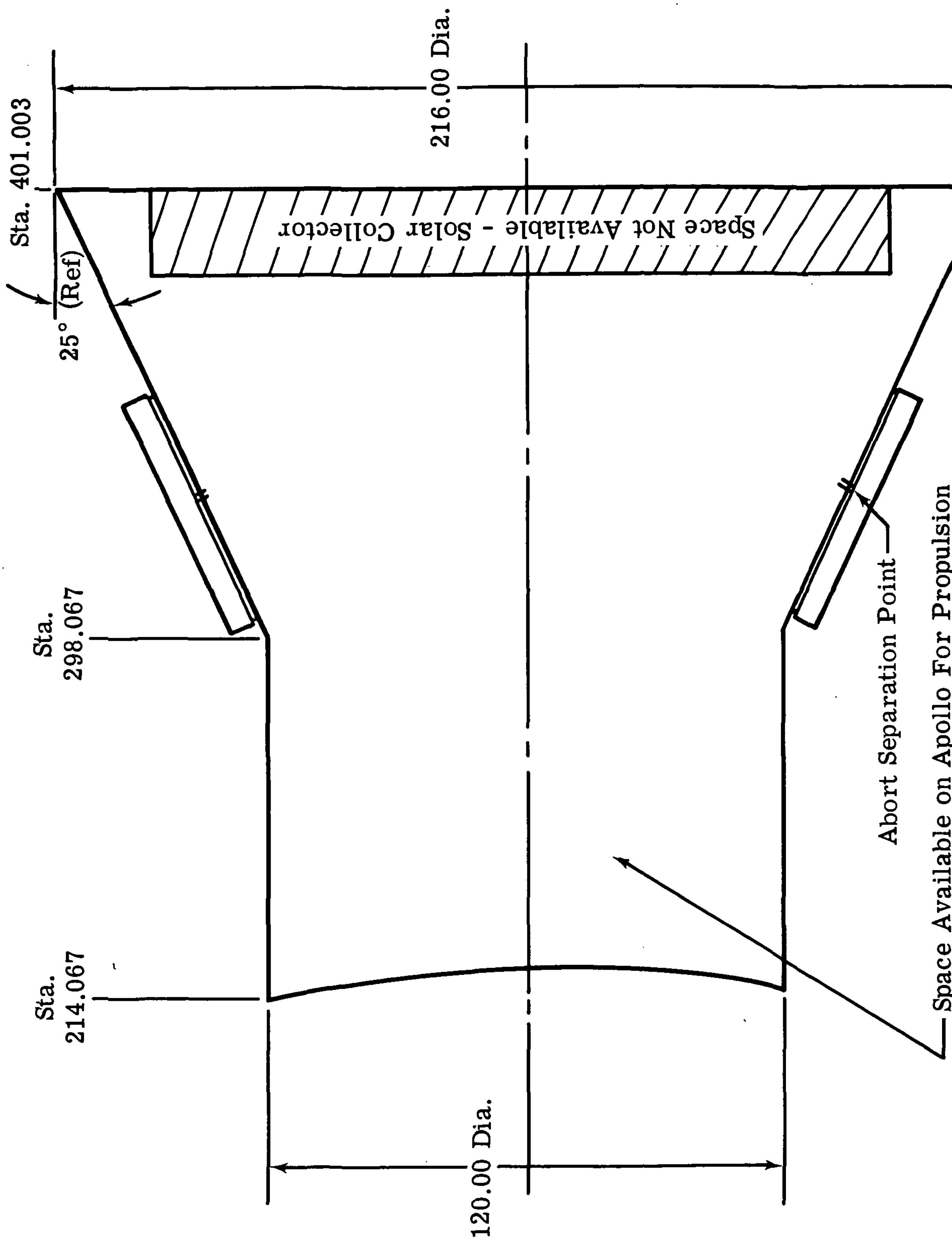


Figure III-1. Propulsion Module Envelope

[redacted]

E. Leaving Lunar Orbit

Required ΔV, feet/second	3,500
Minimum g's	.33
Maximum g's	2 (approximately)
Number of starts (maximum)	2

F. Mid-Course Correction (Inbound)

Inbound ΔV, feet/second	250
Minimum g's	.5 (approximately)
Maximum g's	3 (approximately)
Number of starts (maximum)	5

G. General

Mission Duration	14 days
Type of Design	1963 System
	1966 System

H. 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 Reserves	10%
Residual Propellant (Outage Boil-Off Allowance)	5%
a. Assuming non-vented pressurized tanks	0%
b. Assuming a pumped system	As required

* 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 sufficiently such that it may be better to risk a tank puncture.

Specific items to be covered in the final report include the following:

A. Drawings

Powerplant schematics
Powerplant installed in vehicle
Details of important or unique components
Method of Tankage Compartmenting
Pertinent Dimensions

B. Technical Data

Weights - breakdown by major components
Operating parameters such as chamber pressure, thrust, expansion ratio, specific impulse, etc.
Reliability level expected
Trade-off studies made in selecting chamber pressure, expansion ratio, etc.
Data on alternate systems studied, such as different propellants, etc.
Discussion of critical items such as pressurization techniques, pumps if used, flow control, two phase flow operation on starting and running.
Applicable experience with propellants, components, etc.
Heating analysis (heat input to propellant tanks)
Provisions for system redundancy
For liquid system: Predictability of start and shutdown transients, i.e., deviations from nominal
For solid system: Curves of thrust vs. time at sea level and selected altitudes.
Cockpit display parameters

C. Development Plan

The development plan shall consist of a program starting in July 1961 and terminating in July 1963 for the early version and July 1966 in the advanced version.

BAC shall supply details of a development program up through PFRT for a manned system for use in 1963 and 1966.

IV. DESIGN PHILOSOPHY

Initial payload requirements for the Apollo spacecraft, in conjunction with Saturn booster system capabilities scheduled for the mid-1960's, dictate a rigorous approach in order to achieve maximum performance for the spacecraft on-board propulsion system. In addition, the large costs attendant to the launching of the Saturn make mandatory the early and, necessarily, premanned flight achievement of a high level of reliability. It is economically unacceptable to program a large number of flight tests of the full, or even selected portions thereof, booster system to checkout advanced propulsion concepts in a space environment.

Initial consideration of the seemingly contradictory requirements for both super performance and high reliability fortunately yields several workable solutions to the on-board propulsion problem. In general, these approaches can be categorized as: (1) maximizing hardware and system simplicity with high performance propellants, (2) combining state-of-the-art hardware technology with the highest performance propellants and (3) utilizing moderately high performance propellants with high densities in light-weight, state-of-the-art systems. The first approach yields high performance (ΔV as a function of spacecraft gross weight, W_g) primarily through the use of high energy cryogenic propellants, and high inherent reliability through the implementation of simple, but, in comparison with the state-of-the-art, extremely sophisticated design precepts. The second concept may have a higher or lower system performance depending upon the choice of propellants and the method of supplying them to the reaction chamber but will have a definite edge in reliability. This "edge" will be absolute or transient depending on the relative complexity of the systems being compared. The third approach may combine features of both the initial concepts but can maintain a competitive stature only through achievement of a much higher $\lambda \left(\frac{\text{Propulsion System Propellant Weight}}{\text{Propulsion System Loaded Weight}}\right)$ to counterbalance a specific impulse decrement.

Numerical comparisons of these basic approaches are supplied in subsequent sections. It is the purpose of the present analysis to discuss their significant features and, through qualitative screening, to demonstrate the process through which the selected configuration was established.

Major system parameters or characteristics considered in arriving at the final configuration were:

Propellants

(a) Specific Impulse
(b) Bulk Density
(c) Physical Properties
(d) Chemical Properties
(e) Experience With

Reliability

(a) Design Complexity
(b) Effect of Past Experience on Reliability Growth Integral
(c) Redundancy

Propellant Feed

(a) Pump Feed
(b) Pressure Feed
(c) Combination (in parallel) of Pump and Pressure Feed

Tank Design

(a) Vehicle Geometry Restrictions
(b) Surface to Volume Ratio
(c) Positive Expulsion

Pressurization Methods

(a) Inert Gas Pressurization
(b) Evaporated Propellants or Inert/Compatible Liquids
(c) Vapor Pressure

Reaction Devices

(a) Regeneratively Cooled Thrust Chambers
(b) Regeneratively Cooled/Radiation Cooled Thrust Chambers
(c) Ablation Cooled/Radiation Cooled Thrust Chambers
(d) Heat Sink Thrust Chambers
(e) Combinations of above

As either direct or independent functions of the above, the achievement of a high propellant mass fraction and satisfactory space storability were prime objectives in the consideration of all the possible propulsion system characteristics.

Choice of propellants is limited to combinations yielding desired vehicle performance and for which an arbitrarily established minimum of development has been performed. The following table lists possible high performance propellant combinations on which the necessary advanced development work has been accomplished and which are worthy of serious consideration. A spacecraft gross weight of 15,000 pounds and a propellant fraction of 0.4 are assumed.

TABLE IV-1

Propellant Combination	Vacuum Isp, Seconds Actual	Volume Required $Ft.^3$	Required Volume Compared to N_2O_4/N_2H_4 Blend	Vehicle Velocity Increment Ft/Sec
N_2O_4/N_2H_4 Blend	315	79	1.00	5200
N_2H_4/B_5H_9	350	117	1.48	5800
F_2/N_2H_4 Blend	392	75	0.95	6500
O_2/H_2	428	280	3.55	7100
F_2/H_2	446	165	2.08	7400

Note: Adjustment has not been made for the implied variation in λ which in this chart penalizes the high density propellant combinations.

Of the five combinations presented in this comparison, four are operational or in an advanced state of development. The fifth, N_2H_4/B_5H_9, is considered here because of its relatively high performance potential and its suitability to extended low pressure storage at earth ambient bulk temperature conditions. Bell Aerosystems has had developmental or operational experience with all except O_2/H_2. The latter's properties as a high performance rocket propellant combination are, however, well known and a valid comparison can be easily performed.

The significance of this chart is both in the direct specific impulse and indirect comparison of tank weights required to contain a given weight of propellant. This latter criteria is a key systems design parameter and is usually expressed in the form:

$$\frac{\text{Propellant Weight}}{\text{Propellant Tankage Plus Propellant Weight}}$$

which is a more restrictive definition of the parameter $\bar{\lambda}$. Since the weight of ideal spherical tanks is proportional to the first power of propellant volume, weight of the O_2/H_2 system tanks would be twice those of F_2/H_2 which in turn would be twice those of F_2/N_2H_4. In a low mass fraction system like the Apollo spacecraft, the effect of low propulsion system $\bar{\lambda}$ on overall system performance is not so pronounced as it is in high mass fraction systems. The high density of the fluorine/hydrazine combination does not, therefore, compensate for its lower specific impulse. On the other hand, the fluorine/hydrogen propellant combination has the highest specific impulse of the group and a better bulk density than any combination except the N_2O_4/hydrazine blend and mixed cryogenic/storable, F_2/hydrazine blend. The result is a large net overall performance superiority compared to all other combinations.

Fluorine/hydrogen also combines with this positive performance increment the desired property of vacuum hypergolicity without additives. It does, of course, necessitate bulk temperature control to maintain vapor pressure within system operational limits. Again, however, the lean integrated mixture ratio of this combination, in conjunction with its high performance, minimizes the surface area of the deep cryogenic tanks (H_2).

Both fluorine and hydrogen exhibit excellent compatibility with most spacecraft structural materials (e.g. stainless steel and aluminum). Certain of the highly fluorinated polymers (e.g. Teflon, Kel-F) are also satisfactory for sealing purposes. Experience with fluorine at Bell Aerosystems covers a span of nearly 5 years during which time an extensive operational and design capability has been evolved. This technology is considered equally as advanced as any other industry experience with high energy dual cryogenics. The underlying reason for the equality is the adaptation during the 5-year H_2/F_2 development period of operational Agena space engine hardware to fluorine oxidizer. As a consequence, the reliability growth integral associated with the Agena engine, is to a very significant extent, directly applicable to a pump fed Apollo engine. The Agena engine has not only proven to be the most reliable high performance space engine in existence (100% flight performance in 23 Agena launches to date) but has had the most extensive formal ground test program (more than 500 firings) ever completed on a space engine. It also possesses inherent mixture ratio and thrust control. The marriage of fluorine to the Agena engine is a logical one and promises a combination superior in terms of both design confidence and performance to any other approach considered.

To add a measure of redundancy to a system comprised of pump fed engines and thrust chambers, tank feed and pressurization system parameters can be established to permit completion of a complete (or any portion thereof) Apollo mission operating in either the pump or pressure fed modes. In accomplishing this, the increase in system weight is negligible but the growth possibilities most significant. Increase in thrust beyond the 24,000 pounds desired for super-orbital mission abort can be accomplished easily and without compromise to acquired system reliability. Conversely, system total impulse can be increased without serious weight increase because of the low pressure tank designs possible with a pump fed system. Using low pressure tanks ($\angle$ 100 psia) the alternative to a pump fed system is excessively heavy thrust chambers required to achieve 24K or more.

In the system selected for detail study, both pump and pressure feed systems are utilized to maximize the respective capabilities and minimize the shortcomings. High thrust is provided in the pump fed mode for a relatively few starts and low thrust in the pressure fed mode for the majority of the starts which are required for mid-course guidance and/or trajectory adjustments.

Another factor in the design consideration was that of vehicle geometry. Because of the propulsion module space restrictions imposed by the Bell Aerosystems lenticular vehicle concept, it became necessary to utilize multiple tanks to achieve acceptable packaging. While not yielding the most advantageous design from the standpoint of functioning weight, this approach none the less eliminates the need for approximately 10 feet of cylindrical vehicle structure. It also provides for a high degree of redundancy in the event of failure of single tank components from internal or external (meteors or explosion) sources.

Pressurization of the propellant tanks can be accomplished with cold, high pressure helium with little weight penalty as compared to other systems. Hot gas is unacceptable due to the necessity of maintaining cryogenic temperatures in the propellant tanks for a 14-day period. More sophisticated systems, such as vaporized propellant, offer little weight advantage in a nominally pump fluorine/hydrogen propulsion system because of the low tankage volumes (high propellant densities) and low operating pressures (pump fed system).

Bell Aerosystems has recently conducted extensive studies on other programs of advanced tank pressurization techniques and has concluded that many do have merit for applications to high mass fraction systems where a few pounds of weight are important. Of particular interest in these investigations was the "booster PVP" concept which supplements normal vapor pressure with a selective use of an inert gas "overpressure" to maintain mixture ratio. Most desirable application of this concept is where propellant temperature can be absolutely controlled so that overpressure can be used without intermittent tank venting being required. Without the inert gas overpressure, propellant utilization becomes a problem solvable only with mixed-phase mass flowmeters of a suitably high accuracy.

Thrust chamber designs considered for the Apollo on-board propulsion were of both the regeneratively cooled and the uncooled types. The Bell Agena engine combines both features in implementing a regeneratively cooled combustion chamber and convergent nozzle along with a radiation cooled extension. This design approach offers a thrust to weight superior to all other methods for high chamber pressure operation; the reason being the large weight required for heat sink or ablative purposes in the alternative non-cooled designs. Reliability of the cooled design is an attractive feature in that durability of the thrust chamber is many times that required for a given mission in terms of both structural and dimensional integrity. This feature is desirable in that considerable test time can be accumulated on a thrust chamber prior to flight to insure that the high "infant mortality" period characteristic of some rocket hardware has been passed. Performance repeatability is excellent and an equilibrium operating mode places no restriction on restarts or cycle duration; a shortcoming of most heat sink designs. Lastly, the commonly ascribed shortcoming of regeneratively cooled space engines, freezing of the coolant, is absent in the case of liquid hydrogen.

The mission attitude control system approach will be discussed in detail in subsequent sections. The high performance system selected for the Apollo mission utilizes hardware and design concepts from the Bell Aerosystems Agena secondary propulsion system which is now in production. This Model 8101, utilizing MON/UDMH propellants, is the highest performing system of its total impulse class (i.e., 40,000 pound seconds) in existence. A high degree of reliability can be expected of this basic system by 1965 with the programmed launches of Agena during this 4-year period.

The subsequent sections deal in considerable detail with all design and performance characteristics of the selected system. Numerical computations are used whenever deemed desirable to back-up directly applicable empirical data and developmental experience.

V. COMPARISON OF SYSTEMS STUDIED, SUMMARY

A. General

The general requirement for the propulsion module is to provide the required usable propellants and engine thrust in the most efficient integrated manner. That is, with the best engine performance, least possible dead weight, and high reliability. An optimization of the module involves a trade-off of reliability, tank volume and weight, system enclosing structural weight, pressurization system weight and the number, performance and weight of thrust chambers. The large number of possible variations precluded a detailed study of each. Estimations of the effects of the major parameters were made which resulted in a system selection and detailed definition thereof. The alternate systems were then compared exactly to the final system on a reliability basis. The results of the reliability analyses and estimates of weight difference between the final selection and the other systems and configurations considered are summarized herein.

B. Reliability

The main propulsion arrangement defined as a result of the study is described as a pump/pressure fed system. Of the maximum of 15 firings for the mission, 11 are made pressure fed and 4 are performed on the pump fed side of the system. The system and its components are described in detail in subsequent sections of this report. Portions of that system were used to define all pump-fed and all helium pressurized pressure fed propulsion systems in Section XI, Reliability Analysis. The analyses show that the pressure-fed propulsion system has the highest predicted reliability of the three systems defined. The estimated missions between failures of the pump-pressure fed and the pump-fed propulsion systems are 1% and 14% lower, respectively, than the pressure-fed system. The reliability analyses presented in Section XI confirm the preliminary estimates that there would be only a slight loss of reliability by defining a pump-pressure-fed rather than an all pressure-fed propulsion system. The all pump-fed version was eliminated from further consideration because of its relatively low reliability.

C. Mixture Ratio

The liquid fluorine-liquid hydrogen propellant combination, while providing a high specific impulse, does possess a low bulk density relative to storable liquid propellant and F_2 plus other fuels. At the range of mixture ratios of interest, the liquid hydrogen fuel, while representing a low percentage of propellant weight, does, because of its low density, represent the major portion of required tank volume. Since reducing the tank volume will benefit the overall payload capability of the propulsion system by reducing tank weight, weight of gas required to pressurize the tank volume and the weight of the module enclosing structure, it is desirable to operate the engine at the highest oxidizer to fuel ratio possible, provided that any gains resulting from reduced system weights are not offset by an overriding loss in engine specific impulse. This discussion is based on the pump-fed side of the system which expends approximately 93% of the total weight of propellants.

The maximum value of the oxidizer to fuel ratio is set at approximately 13:1 by the thrust chamber cooling calculations presented in a later section. The overall effect of increasing the mixture ratio from 9 to 13 to 1 is to reduce the propulsion module weight. The increase of specific impulse at 10:1, relative to 13:1, represents 22 less pounds of propellant weight. The tankage weight and helium pressurization system weight is (based on spherical propellant tanks) increased approximately 10 pounds when proceeding from 13:1 to 10:1. The module envelope definition presented in the requirements section of this report results in a considerable difference in skin structure weight when the ratio is reduced below 13:1. At 13:1, a tankage configuration can be obtained which utilizes a minimum of skin structure. Below 13:1 an entirely different tank configuration must be employed. The increased weight of structure to continue to use spherical tanks is estimated at 150 pounds minimum. The smaller envelope could be maintained only by resorting to toroidal tanks with a weight penalty of approximately 200 pounds. Because the 13:1 mixture ratio presents a just fit arrangement of propellant tanks in a minimum weight of enclosing envelope, a mixture ratio of 13:1 was selected for the system.

D. Thrust Level

The reliability analyses for the pump/pressure-fed and the all pressure-fed systems indicate that there is little reliability increase by providing 4 rather than 2 thrust chambers. The pump/pressure-fed system proposed has two 12,000 pound thrust chambers and the pressure-fed system analyzed has four 6,000 pound units to provide the 24K upper thrust requirement. The envelope requirements limit the thrust chamber lengths to approximately 60 inches which dictates 2 pump-fed (P_c = 300 psia) chambers with an expansion ratio of 45:1. The pressure-fed chambers (P_c = 50 psia) have a length of 60 inches at an expansion ratio of approximately 25:1. The higher expansion ratio and chamber pressure of the pump/pressure-fed system provides a specific impulse approximately 20 seconds higher than the pressure-fed version. Twenty seconds of I_{sp} represents 220 pounds of usable propellant for the mission. The two, 12/4K, pump, pressure-fed chambers also allow the minimum envelope of propulsion module as described in the next section. An additional consideration is that four 6K chamber assemblies with valving, mounting and gimballing are heavier than two 12K chambers with turbine pumps, gas generator, valves gimballing and mounting. The weight difference is estimated at approximately 100 pounds.

E. Module Tankage Arrangement

Table I shows five cases of module tankage arrangements based on an overall mixture ratio of approximately 12:1. The 12:1 ratio results from the pressure-fed thrust chamber operation at 10:1 (7% of propellant weight is used in pressure-fed operation) and propellant utilized at a 1.4 O/F ratio in the gas generator of the turbine pump assembly and propellant lost during bleed. Tankage configuration numbers 4 and 5 can be considered only on the basis of the two 12/4K pump pressure-fed chambers. Four 6K pressure-fed chambers admit only cases 1, 2 and 3 because of their greater overall envelope.

F. Weight Summary

The total estimated weight difference between the pump/pressure fed and the pressure-fed H_2-F_2 systems is summarized as follows:

Item		
1	Additional Weight of Propellants	200 lbs
2	Additional Weight of Tankage for Item 1	10 lbs
3	Additional Weight of Propellant Tanks for 100 psi Working Pressure	55 lbs
4	Additional Weight of Thrust Chambers	100 lbs
5	Additional Weight of Structure	150 lbs
6	Additional Weight of Helium Pressurization System	200 lbs
		735 lbs

A propulsion system utilizing F_2-N_2H_4 propellants was investigated briefly. The difference in I_{sp} between pump/pressure-fed F_2-N_2H_4 and F_2-H_2 systems is approximately 50 seconds or 550 pounds of total loadable propellants; the F_2-N_2H_4 system having the lower I_{sp}. Because the minimum envelope of structure is approached by the F_2-H_2 system, the lower total volume of the F_2-N_2H_4 tankage cannot be utilized to reduce the weight of the envelope structure. The reduction in propellant tanks and helium system weight for the F_2-N_2H_4 system is estimated at 100 pounds which does not offset the increased weight of propellant tanked.

[redacted]

TABLE V-1

MAIN PROPULSION SYSTEM TANKAGE COMPARISON

ITEMS	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
			1+2			4+5		3+6+7					9+10+ 11+12		8+13+ 14		15+16		17+18
CONFIGURATIONS	F_2 Tank	F_2 Tank Insul.	Total F_2 Tank	H_2 Tank	H_2 Tank Insul.	Total H_2 Tank	Power* Plant Stru.	Total Stru.	Motor*	Motor Mount	Propel. System	Press* Syst.	Total Propul. System	Total Propel.	Total Propul. System	Skirt @1.20 lb per ft^2	Total	Attit. Cont. Pkg.	Total Syst.
Case I 2 Spheres	84	21	105	116	75	191	250	546	440	-	80	110	630	6769	7945	986	8931	507	9438
Case II			105			421	275	801	440	-	95	110	645	6769	8215	769	8984	507	9491
1-F_2 Sphere	84	21	105																
8-H_2 Spheres				121	300	421													
Case III			105			382	200	687	440	-	75	110	625	6769	8081	757	8838	507	9345
1-F_2 Sphere	84	21	105																
1-H_2 Torus				188	194	382													
Case IV			102			241	300	643	440	-	125	110	675	6769	8087	721	8808	507	9315
2-F_2 L.P. Spheres	51	33	84																
2-H_2 L.P. Spheres				65	123	188													
2-F_2 H.P. Spheres	7	11	18																
2-H_2 H.P. Spheres				31	22	53													
Case V			135			241	300	676	440	-	125	110	675	6769	8120	563	8683	507	9190
4-F_2 L.P. Spheres	64	53	117																
2-H_2 L.P. Spheres				65	123	188													
2-F_2 H.P. Spheres	7	11	18																
2-H_2 H.P. Spheres				31	22	53													

* - Early estimates
H.P. - High pressure (midcourse correction)
L.P. - High pressure (lunar orbit)

[redacted]

VI. MAIN PROPULSION SYSTEM

A. Propulsion System Description

1. Schematic and General Arrangement

The schematic of the main propulsion system is presented in Figure VI-1. The general features of the system will be discussed in this section. Operating pressures, controls, sequence of operation, preliminary design details, etc., will be covered in subsequent sections of this report.

The two thrust chambers are gas generator-turbine pump fed from "low pressure" spherical tanks. The chambers can also be operated pressure fed from "high pressure tanks". The thrust level of each chamber operating pump fed is 12,000 pounds. Each chamber fires at 3983 pounds thrust when operated pressure fed.

A helium tank is buried in one of the low pressure H_2 tanks to provide positive suction head requirements of the pumps through a solenoid start valve and pressure regulator. The helium is also fed through a second solenoid valve and regulator circuit to the high pressure H_2 tanks. A relief valve is installed on the downstream side of each pressure regulator to protect against overpressurization of the tanks in the event of malfunction of the pressure regulator or excessive pressure buildup in the propellant tanks due to increased vapor and helium pressure from heat transfer to the tank. The solenoid start valves are shut off between engine firings, reducing the possibility of He leakage from the He storage to the propellant tanks. The fluorine propellant high- and low-pressure tankage helium supply circuit is similar to that of the hydrogen tanks.

The two turbine pump assemblies are powered by gas generators which utilize F_2 and H_2 from the high pressure tanks for starting and propellants bled from the pump casing to bootstrap to full power operation. Check valves in the gas generator feed lines isolate the starting and pump discharge flows during the transient to full power. The flow control of the gas generator is accomplished by cavitating venturis. The thrust chamber propellant valves for the pump fed operation provide bleed flow to cool the propellant pumps prior to starting the gas generator. Temperature sensing elements are provided to indicate completion of bleed. After thrust chamber shutdown, the bleed ports vent the propellants trapped between the pump inlet valves and the propellant valves. Propellant acquisition for the pump fed operation is obtained by ullage rockets incorporated in the mission attitude control system augmented by the thrust developed by the gas generator exhaust duct.

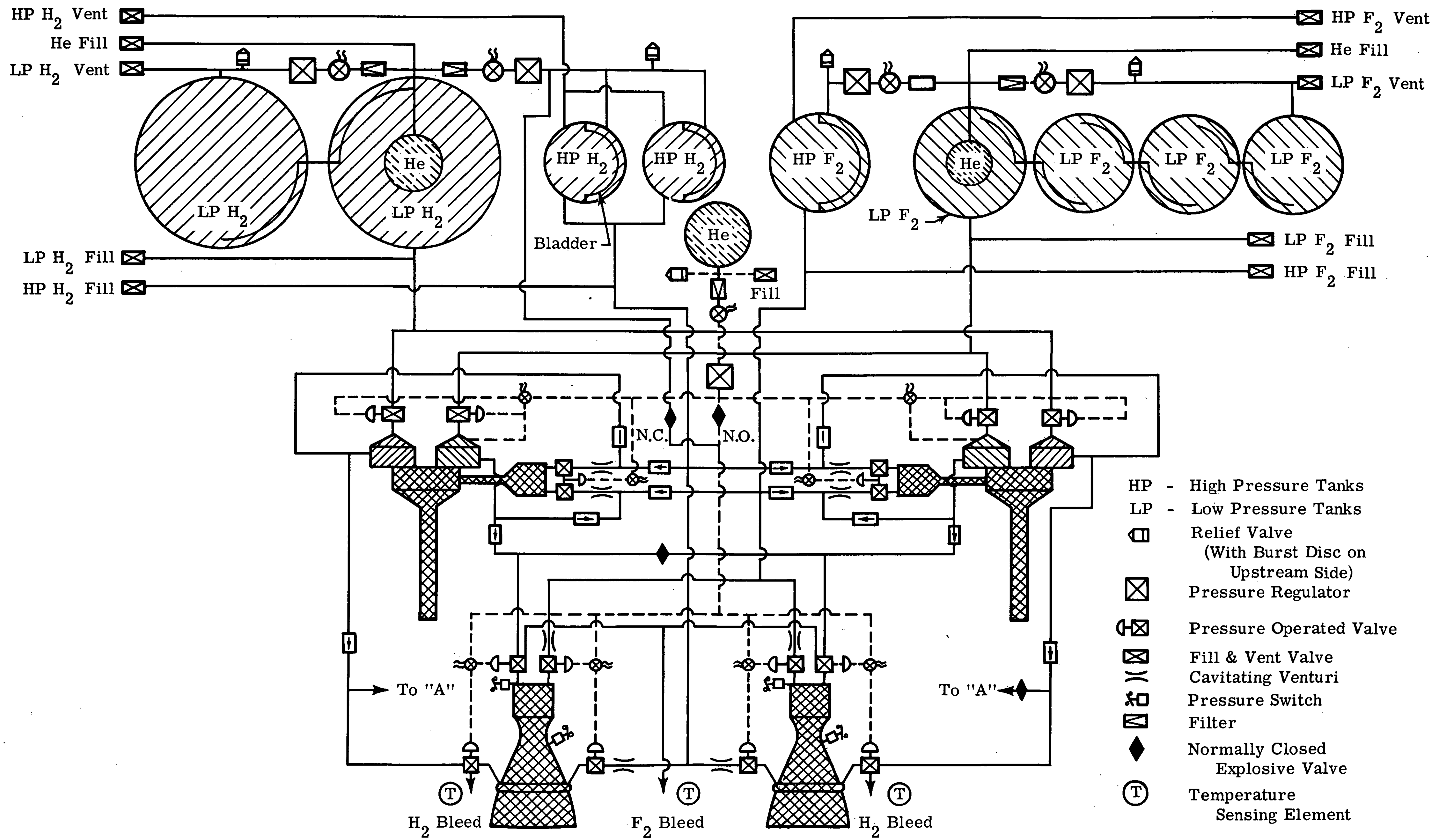


Figure VI-1. F_2-H_2 Main Propulsion System Schematic

The high pressure H_2 and F_2 tanks incorporate bladders for positive expulsion. Propellant flow for pressure-fed thrust chamber operation is accomplished by cavitating venturis. Each thrust chamber has two sets of propellant valves, one for pump fed operation; the other for pressure fed operation. The thrust chamber propellant valves and the F_2 and H_2 pump inlet isolation valves are pressure operated from a third helium tank, start valve and regulator circuit. That helium supply is also fed to a seal in the fluorine pumps.

Additional redundance is provided by:

(1) Normally closed explosive valves in the pump discharge lines enabling pump fed firing of one chamber with either turbine pump assembly.

(2) Cross-connection of the H_2 tankage helium supply and the helium circuit for valve operation and F_2 pump seals. In the event of failure of the valve/seal He supply tanks, start valve or regulator, helium can be obtained from the H_2 tankage circuit for the valve/seal operation. The cross-connection line would be routed through the warm sections of the propulsion module skin structure to increase its temperature, reducing the weight of helium bled from the H_2 helium tankage circuit.

(3) The ullage rockets also provide a back-up for the bladders of the high pressure H_2 and F_2 tanks.

(4) In the event that gas generator starting propellant is not available from one or both high pressure tanks, the gas generator will start from the pressure available in the low pressure tanks. The normal operation of gas generator starting from the high pressure tanks was provided for more rapid pump fed firing start transients. The capability of making pump fed firings independent of the pressure fed subsystem plus the 10% propellant reserves in the low pressure tanks makes the pump fed side of the system completely redundant to the pressure fed side.

(5) The pressure fed operation of the thrust chambers can be affected if there is a failure of the helium supply to the high pressure tanks part-way through the mission. The three to one step thrust operation of the thrust chambers is conservative. Four to one step thrust operation can be accomplished. If helium supply to the tanks is lost when the propellants are partially expended, pressure fed operation of the thrust chambers could be continued with the helium already in the tanks. For example, if the helium partial pressure in the H_2 tank is 200 psia and the tank is 40% full, the discharge of the last increment of H_2 would be based on 120 psia helium pressure. The flow rate at 120 psia would be approximately equal to the flow at 200 psia divided by $1/\sqrt{1.667}$

or 77.5% of rated flow. The thrust chamber for the case considered would be operating at approximately a four to one step when the last increment of fuel is expelled.

(6) Additional emergency modes of thrust chamber operation are discussed in section VI F.

The general arrangement of the propulsion module is depicted in Figure VI-2.

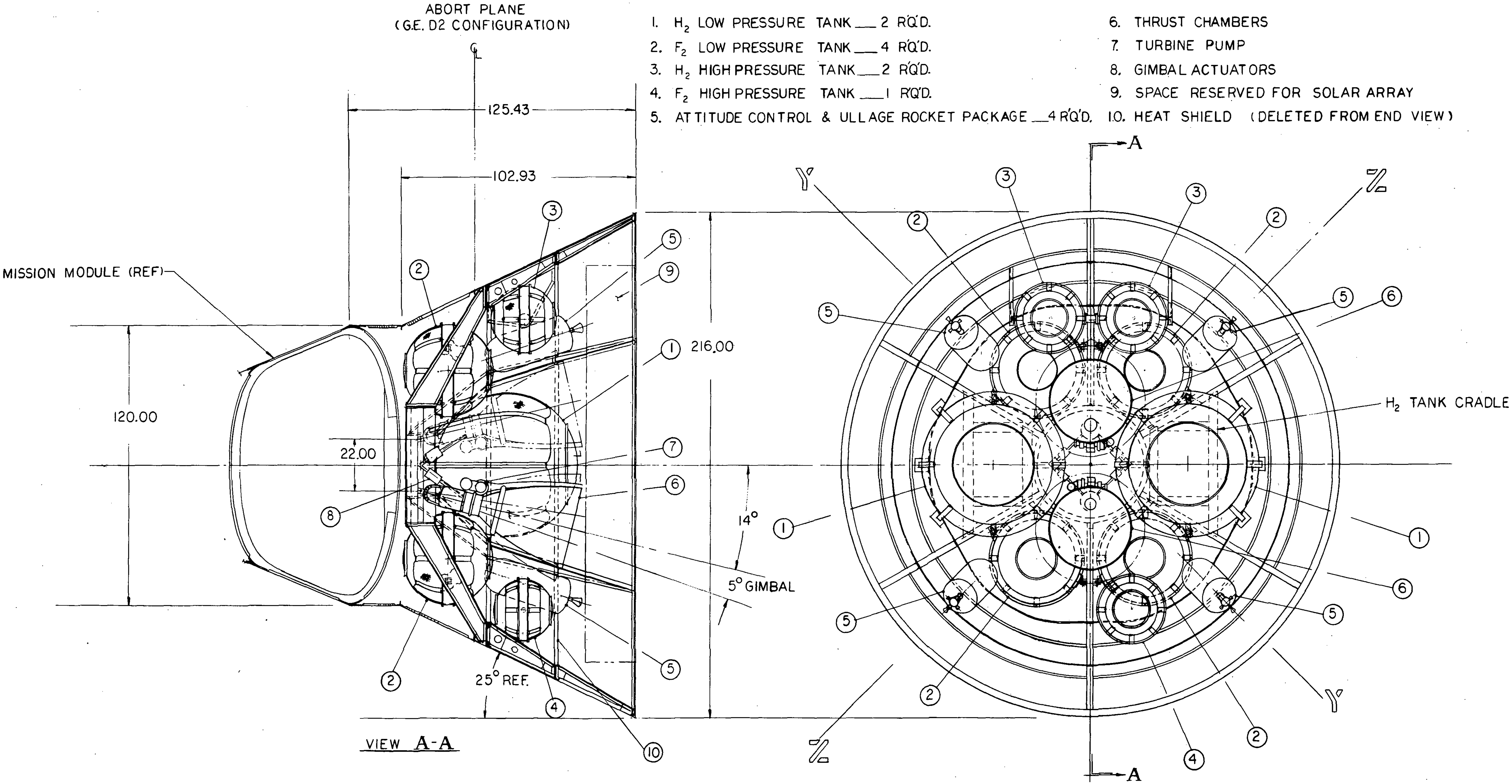


Figure VI-2. Propulsion Module Installation

2. System Operation Sequence

The maximum mission is defined as 5 midcourse corrections earth to moon, 3 firings to achieve lunar orbit, 2 firings to disorbit and 5 midcourse corrections, moon to earth. The pump/pressure-fed system defined herein would be operated pressure-fed for the 10 midcourse corrections and the third lunar orbit injection firing. The initial 2 firings to achieve lunar orbit and the 2 firings to disorbit the moon would be made on the pump-fed side of the system.

The normal operating sequence for a pressure-fed firing consists of the following:

a. An electrical signal actuates the normally closed solenoid valves upstream of the pressure regulators of the high pressure tank helium pressurization circuits. The same signal opens the solenoid valve of the valve actuation helium pressure circuit.

b. An electrical signal opens the helium pilot valve supplying pressure to the thrust chamber propellant valves.

c. The hydrogen propellant valve opens first. Hydrogen is forced out of the high pressure tanks by the tank bladders and fills the thrust chamber coolant jacket with hydrogen.

c. Hydrogen reaches the injector and flows through the injector into the chamber.

e. The fluorine valve opens after the hydrogen reaches the chamber because of a mechanical delay in the fluorine valve.

f. Fluorine is forced from the tank by the bladder and flows to the injector and into the chamber and ignites hypergolically with the hydrogen.

g. Thrust is terminated by de-energizing the propellant valves' helium pilot valve. The fluorine valve closes before the hydrogen valve.

h. The pressure-fed side of the system is secured by closing the 3 helium pressurizing circuit valves.

The normal operation sequence of the pump-fed side of the system consists of the following steps:

a. An electrical signal actuates the normally closed solenoid valves in the helium pressurization circuits of the low pressure tanks and in the valve/seal actuation helium supply circuit.

b. The same electrical signal actuates the solenoid pilot valve supplying actuation pressure to the pump shutoff valves of the one of the turbine-pump assemblies. The solenoid valve also supplied helium pressure to a seal inside the fluorine pump.

c. The pump inlet F_2 and H_2 isolation valves open. F_2 and H_2 bleed through the pumps and overboard through the bleed ports of the pump side thrust chamber propellant valves.

d. An electrical signal fires two 50 pound thrust ullage rockets in the mission attitude control/ullage rocket subsystem.

e. Power is applied to a relay in the circuit of the solenoid valve supplying helium to the gas generator propellant valves.

f. When the proper valve housing temperatures are obtained by the overboard bleed of propellants, the relay circuit of the pilot valve of the gas generator propellant valve closes actuating the pilot valve.

g. The gas generator propellant valve opens admitting F_2 and H_2 from the high pressure tanks to the combustion chamber of the gas generator. Ignition occurs hypergolically.

h. The gas generator exhausts through the turbine, accelerating the propellant pumps.

i. The exhaust of the gas generator provides ullage thrust. The 50 pound ullage thrust chambers are shutdown.

j. The thrust chamber H_2 and F_2 propellant valves are actuated to the run position by a solenoid pilot valve. The H_2 valve actuation preceeds the F_2 valve due to a mechanical delay in the F_2 valve.

k. H_2 fills the coolant passages of the thrust chamber and injector and enters the chamber. F_2 enters the chamber and ignition is obtained.

l. When the pump discharge pressure exceeds the H_2 and F_2 feed pressure from the high pressure tanks, check valves in lines from the pump casings open, allowing flow to the gas generator from the pumps. The check valves close in the lines between the pump fed gas generator and the high pressure tanks.

m. Pump discharge flow to the gas generator bootstraps the chamber to the 12,000 pound thrust level.

n. Thrust termination is initiated by de-energizing the gas generator propellant valve.

o. The H_2 and F_2 isolation valves at the pump inlet are closed.

p. The propellant valves at the thrust chamber are returned to the bleed position with the F_2 valve preceding the H_2 valve.

q. The thrust chamber propellant valves bleed the H_2 and F_2 from the pumps and lines overboard.

r. The solenoid valves in the helium pressurization circuits are de-energized.

The third and fourth pump-fed firings (lunar disorbit) are accomplished without additional low pressure propellant tank He pressurization. The partial pressure of the helium fed into the tanks during lunar orbit injection remains above the pump NPSH requirements during the third and fourth firings. The helium pressure time history in the low pressure tanks is discussed in detail in the next section of this report.

Automatic shutdown of the thrust chambers due to malfunction is discussed in Section VI.G. "System Controls". System monitoring instrumentation is also discussed in that section.

3. <u>Performance and Weight Summary</u>

The performance summary of the Apollo main propulsion system is presented in Table VI-1A and the weight summary of the 1963 and 1966 versions of the propulsion system is presented in Table VI-1B.

TABLE VI-1A

APOLLO MAIN PROPULSION PERFORMANCE SUMMARY

(1) Requirements

Vehicle Gross Weight	14,715 Lb
ΔV Total	7,500 Ft/Sec
ΔV Midcourse Corrections	500 Ft/Sec
ΔV Lunar Orbit Exit and Entry	7,000 Ft/Sec

(2) Engine Performance

a. Pump Fed Engine

Propellants	Liquid Fluorine/Liquid Hydrogen
Engine Vacuum Thrust, Lbs	12,083
Engine Mixture Ratio, O/F	11.92 ± 1½%
Engine I_{sp}, Nominal, Sec	446.2
Engine I_{sp}, Minimum Observed Guarantee, Sec	443.6
Thrust Chamber Vacuum Thrust, Lbs	12,000
Chamber Pressure, Psia	300
Mixture Ratio, O/F, Thrust Chamber	13
Area Ratio, A_e/A_t	45
I_{sp}, Nominal Thrust Chamber, Sec	448.2
I_{sp}, Minimum Observed Guarantee Thrust Chamber, Sec	446.2
Fuel Pump Discharge Pressure, Psia	465
Oxidizer Pump Discharge Pressure, Psia	400
Turbine Fuel Consumption, Lb/Sec	0.33
Turbine Exhaust Thrust, Lb	83
Exhaust Gas I_{sp}, Sec	250

b. Pressure Fed Engine

Propellants	Liquid Fluorine/Liquid Hydrogen
Vacuum Thrust, Lbs	3,983
Mixture Ratio, O/F	10
Chamber Pressure, Psia	100
Area Ratio, A_e/A_t	45
I_{sp}, Nominal, Sec	448
I_{sp}, Minimum Observed Guarantee, Sec	445.8

TABLE VI-1B

APOLLO MAIN PROPULSION WEIGHT SUMMARY

	1963 Version	1966 Version
Thrust Chamber Assemblies Including valves, engine mount, turbine pump assembly, gimbal actuators, etc.	555.1	515.1
Propellant System Including low pressure and high pressure tanks, insulation lines, valves, module structure, tank supports and interconnecting lines	899.0	744.0
Pressurization System Including helium tanks, tank supports, lines and valves	126.9	86.9
Instrumentation Pick-Ups	25.0	25.0
Loadable Propellant and Helium	6784.9	6739.9
Mission Ullage/Attitude Control System	507.0	417.0
Propulsion Module Skin Weight	563.0	563.0
TOTALS	9460.9	9090.9

Re-Entry Vehicle

Attitude Control System	70.97	70.97

4. Structure and Base Shield

a. Support Structure for Engine and Propellant Tanks

For the purposes of designing the support structure to withstand engine thrust loads and propellant tank acceleration loads the following design criteria and loads were applied. Acceleration loads are 5.18 g in the longitudinal direction combined with 1 g in the side direction. Both factors are for limit loads which are applied during boost phase of flight. In the design of the engine support a hard start condition loading of 22,500 pounds (limit) was assumed and applied for the thrust value of each engine.

The main support structure, which attaches the tanks and engines to the shell, consists of a large circular frame, stringers, and a smaller circular frame. Each frame is a built-up I-beam made of sheet and extruded beryllium. For stringers, bent-up sheet beryllium is employed. The larger frame is needed to introduce engine thrust loads into the outer shell structure, while the stringers and second frame are needed to introduce bending moments from the loaded inner engine support into the outer shell. In addition, the stringers are used to help support the two high pressure hydrogen tanks and one high pressure fluorine tank.

The engine support structure, which ties to the outer shell large frame, consists of built-up I-beams and channels made of sheet and extruded beryllium. Just behind the mission module an inner circular frame, made of an extruded channel section, is used. Within this circle two I-beams, which intersect, support the two 12,000 pound thrust engines and the four actuators for gimballing. These four members carry the engine and actuator loads to six spokes which extend from the inner circular frame to the large outer shell circular frame. The spokes are built-up I-beams made of sheet and extruded beryllium. These spokes also will support the four low pressure fluorine tanks and the two low pressure hydrogen tanks.

b. Tank Support Structure

Hydrogen and fluorine tank support structure will consist of straps, end coverings, and girdles or cradles which will surround the tank and extend to the primary structure leading from the engines or to the shell of the space vehicle skirt. Material for this structure is sheet beryllium. The purpose of the end coverings is to carry the longitudinal loading of tank and propellant to the straps and also to help seat the tank on the girdle or cradle through the straps. The straps then carry tank and propellant loads to the girdle or cradle. Tension is applied through the straps to seat both the end covering and the girdle or cradle to the tank. In the case of the fluorine tanks a channel surrounds the center of the tank. Added to this girdle are trunnions which enable the entire structure tank, straps, and girdle to be tied to primary structure of the engines. In the case of the large hydrogen tanks a cradle, at an angle to the tank, is used to seat the tank and tie into the engine support structure. For both the girdle and cradle bent-up sheet beryllium is again used. An additional factor taken into account in designing all of these structural items is that the tank insulation may withstand only 10 psi in compression.

c. Base Shield

The propulsion module will require an enclosing structure or base shield in a plane below the propellant tanks. The G.E. overall spacecraft configuration has a solar collector in sight of the aft end of the propulsion module making a base shield necessary to maintain a low radiation temperature environment about the propellant tanks. Undoubtedly, the base shield is also needed to prevent the circulation of main thrust chamber exhaust gases into the propulsion module. The under-expanded supersonic jet exhausting from a nozzle and from a turbine exhaust at infinite altitude will continue to expand and may flow perpendicular to the nozzle axis. In addition, two spreading exhaust jets from simultaneously firing main engines could cause a reverse flow of high temperature, low pressure gases into the space between the chambers. The affect of the conical module structure on the exhaust pattern and the possible requirement of flame shield structure in the module base shield will have to be determined experimentally.

Theoretical and experimental studies of the pattern of exhaust jets at high altitude have been conducted and reported in the literature (Reference 1). Investigations of the complex phenomena associated with the recirculation of hot gases from clustered rocket engines are being carried out at the USAF Arnold Engineering Development Center (References 2 and 3).

B. PROPELLANT TANKAGE

1. Tank Sizes and Materials

a. Total Usable Propellants

Figure VI-1, the propulsion system schematic, indicates the general approach used to store propellants for the midcourse and lunar orbit entry and exit rocket engine requirements. The number and size of tanks in general have been dictated by the vehicle envelope limitations, vehicle arrangement weight optimization, and the environmental factors associated with the propulsion module as discussed in other sections of this report. Spherical tanks are used throughout to minimize both tank hardware weights and the tank thermal insulation weights.

The midcourse correction pressure fed propulsion system has been sized to provide vehicle velocity corrections of 250 ft/sec during both the earth to moon and moon to earth mission phases. The lunar orbit propulsion system is designed to provide vehicle velocity increments of 3500 ft/sec for both the lunar orbit entry and exit phases of the mission. In addition, a 10% propellant margin is included in both systems as dictated by an established ground rule for the vehicle. Propellant requirements for each phase - midcourse out, lunar orbit entry, lunar orbit exit and midcourse out - were established by a step-by-step calculation for each firing using the equation $\Delta V = I_{sp}\, g \ln w_1/w_2$ for each phases where:

ΔV = required velocity increment, ft/sec

I_{sp} = rocket engine specific impulse, lb-sec/lb

g = earth gravitational constant, 32.2 ft/sec^2

w_1 = vehicle initial weight, lbs

w_2 = vehicle weight at end of firing, lbs

The vehicle initial weight was taken as 14,715 pounds prior to the first velocity correction and reduced by the propellant consumed for each subsequent velocity increment.

In computing the required propellant weights by the above method, nominal Isp (performance) was used. Normally, the minimum guaranteed performance value would be employed in these calculations to account for three sigma variations. The 10% propellant margin which is carried by the system will effectively cover three sigma variations in performance. Table VI-1 presents the rocket engines' performance and operating parameters, Table VI-2 is compiled to give the pertinent data on the propellant storage and feed systems.

b. High Pressure (Midcourse) Storage System

Tank pressures for this feed system were established from the following requirements:

	Fuel	Oxidizer
Chamber Pressure	100 psia	100 psia
Injector ΔP	30	7
Coolant Jacket ΔP	21	-
Propellant Valve ΔP	25	25
Venturi and Line ΔP	24	18
Required Tank Pressure	200	150

TABLE VI-2

APOLLO PROPELLANT FEED SYSTEM SUMMARY

I. High Pressure (Midcourse) System

A. Hydrogen

Usable Hydrogen	33.2 lbs	
Residual (1%)	0.3	
Mismatch (3%)	1.0	
Residual Vapor	-	
Pump Fed Eng. Start 0.35 lb/start	1.4	
	35.9	
10% Contingency	3.6	
TOTAL LOADED	39.5	39.5 lbs

Hydrogen Tanks -	Two 27.6 in. ID spherical tanks 6061-T6 Al. Al. 0.040, 200 psi W.P. with Mylar bladder assembly - Total	30.0
	SI-4 insulation 0.82 in. thick (5 BTU/hr/tank)	15.6

B. Fluorine

Usable Fluorine	331.8	
Residual (1%)	3.3	
Mismatch (in H_2 tank)	-	
Residual Vapor	-	
Pump Fed Eng. Start 0.50 lb/start	2.0	
	337.1	
10% Contingency	33.7	
F_2 LOADED	370.8	370.8

Fluorine Tank -	One 24.5 in. ID spherical tank 6061-T6 Al. Al. 0.032 in. thick, 150 psi W.P. with Al. Al. Bladder Assembly	7.0
	SI-4 insulation 0.56 in. thick (5 BTU/hr)	4.3

TABLE VI-2
(Continued)

II. Low Pressure (Lunar Orbit Entry and Exit) System

A. Hydrogen

Usable Hydrogen	432.0	
Residual (0.5%)	2.2	
Engine Precool (7 lb/Eng. Start)	28.0	
Lost on Shutdown (1 lb Eng. Run)	4.0	
Mismatch (1%)	4.4	
Residual Vapor	12.1	
	482.7	
10% Contingency	48.3	
H_2 LOADED	531.0	531.0

Hydrogen Tanks -	Two 60.0 in. ID spherical tanks 6061-T6 Al. Al. 0.032 in. thick 73 psi W.P.	78.0
	SI-4 insulation 1.27 in. thick (15 BTU/hr/tank)	113.2

B. Fluorine

Usable Fluorine	5158.0	
Residual (0.5%)	25.8	
Precool (10 lb/Eng. Start)	40.0	
Lost on Shutdown (4 lb/Eng. Run)	16.0	
Mismatch (in H_2 System)	-	
Residual Vapor	16.6	
	5256.4	
10% contingency	525.6	
F_2 LOADED	5782.0	5782.0

Fluorine Tank -	Four 37.6 in. ID spherical tanks 6061-T6 Al. Al. 0.032 in. thick 57 psi W.P.	60.0
	SI-4 insulation 0.64 in. thick (10 BTU/hr/tank)	44.0

TABLE VI-2
(Continued)

III. Helium Storage System		
A. For Hydrogen System		
Usable Helium	49.7	
Residual at 225 psi and -400°F	5.8	
TOTAL LOADED AT 3000 PSI AND -423°F	55.5	55.5
Helium Tank - One 24.5 in. ID spherical tank 6 Al-4V Titanium 0.135 in. thick		47.0
B. For Fluorine System		
Usable Helium	4.13	
Residual at 175 psi and -300°F	.51	
TOTAL LOADED AT 3000 PSI AND -306°F	4.64	4.6
Helium Tank - One 14.1 in. ID spherical tank PH15-7 MO 0.127 in. thick		24.0
IV. Total Propellant and He Loaded		6783.4 lbs.

Residual propellants (unusable) were calculated to be 1% of the usable and consist of propellants trapped in feed lines and in the tank diffuser tubes, percentages consistent with experimental data obtained with similar tanks. Propellant mismatch is accounted for by tanking an excess of one propellant to assume adequate propellant for the impulse requirements. Minimum system weight is achieved by providing an excess of the lighter (total loaded weight) propellant. The material selected for both the hydrogen and fluorine high pressure tanks is 6061-T6 aluminum alloy. This material was selected since no advantage could be realized from the employment of higher strength alloys due to minimum gage fabrication and handling feasibility. In addition, this material is easily formed and fabricated, possesses excellent low temperature properties and is compatible with both propellants. Linde SI-4 vacuum type insulation is provided on the tanks to maintain the heat leak to both fuel and oxidizer systems below 5 BTU/hr/tank. This heat leak is based on the predicted average environmental temperature of -30 to -20°F in the propulsion module.

c. Low Pressure (Lunar Orbit Entry and Exit) System

Storage conditions and operating procedures for this system are dictated by the requirements of the rocket engine propellant pumps. The suction pressure requirements of the pumps are discussed in the rocket engine description section. The pump suction pressure requirements are essentially a constant increment above the vapor pressure of the fluid at the pump inlet flanges. Since the pump suction pressures, and hence tank pressures, are closely keyed to the fluid vapor pressure, the heat leak to the propellant storage system becomes an important consideration.

An analysis of the pump pressure requirements and system heat leak led to definition of operation of the system by phase in the following sequence: During the three- to four-day period from vehicle launch to retrofiring into a lunar orbit, the propellant tank vents are closed and heat input is absorbed as sensible heat in the liquid. Sufficient insulation is provided to maintain the fluorine system heat leak to 10 BTU/hr/tank (40 BTU/hr total) and 15 BTU/hr/tank (30 BTU/hr total) to the hydrogen tank. These heat fluxes represent a temperature rise of 3°R in the hydrogen system and 2°R in the fluorine system during a four-day period. Five per cent ullage volume in the fuel system and 3% in the oxidizer system are respectively provided to allow for the propellant density increase due to the temperature changes.

For the engine firings required to enter lunar orbit, the tanks are pressurized to 35 psia using helium stored within the respective propellant systems. Thirty-five psi was selected to provide the necessary pump suppression pressure for the initial firing and also to assure sufficient tank pressure for subsequent firings without additional helium. Helium is added at approximately the temperature of the liquid propellant since the addition of heat to the gas would result in only a temporary lower gas consumption. It would also represent an additional heat load to the tankage system. Figures VI-3 and VI-4 present time histories of the low pressure propellant tank pressures and temperatures throughout the various mission phases.

During the six-day lunar orbit, heat input to the propellant tankage system is absorbed by the propellant. Figures VI-3 and VI-4 indicate the magnitude of this heat transfer on both propellant systems' temperatures and pressures. For the final engine firings used to escape from lunar orbit, the gaseous helium and hydrogen within the propellant tanks is allowed to expand with sufficient net positive suction pressure being available for engine operation without recourse to the use of additional helium.

It should be noted that for lunar orbit periods of less than six days, the helium requirements would be less than that provided. Additionally, as discussed in a following section, the location of the vehicle solar collector and cabin cooling radiator represent heat loads to the propulsion module which result in penalties to the propulsion system in the form of insulation weight and weight for higher operating tank pressures.

Tank material for both the fluorine and hydrogen tanks is 6061-T6 for the reasons presented in the high pressure propellant tank discussion. Linde SI-4 insulation is provided as discussed above. The four fluorine and two hydrogen tanks are connected in a series flow configuration for the following reasons: (1) to minimize trapped (residual) propellants; (2) to minimize the number of plumbing connections to the tanks which are a major source of heat leakage; and (3) to eliminate the need for a flow balance system. With this method, propellant liquid residuals are reduced to that trapped in the propellant feed lines and in one tank in each system, representing 0.5% of the loaded weight for each propellant.

d. Mismatch

Propellant mixture ratio in this system is accomplished by employing the unique control feature which is inherent in the rocket engine. This engine control system has been used with singular success on RASCAL, HUSTLER, and AGENA series engines over the past eight years. The heart of the flow control system is the turbine pump drive which incorporates mechanically linked propellant pumps having volutes and cavitating venturis in the discharge legs. The cavitating venturis maintain a constant volumetric flow at constant speed and inlet conditions, independent of the pressure downstream of the diffuser. The gas generator is supplied propellants from

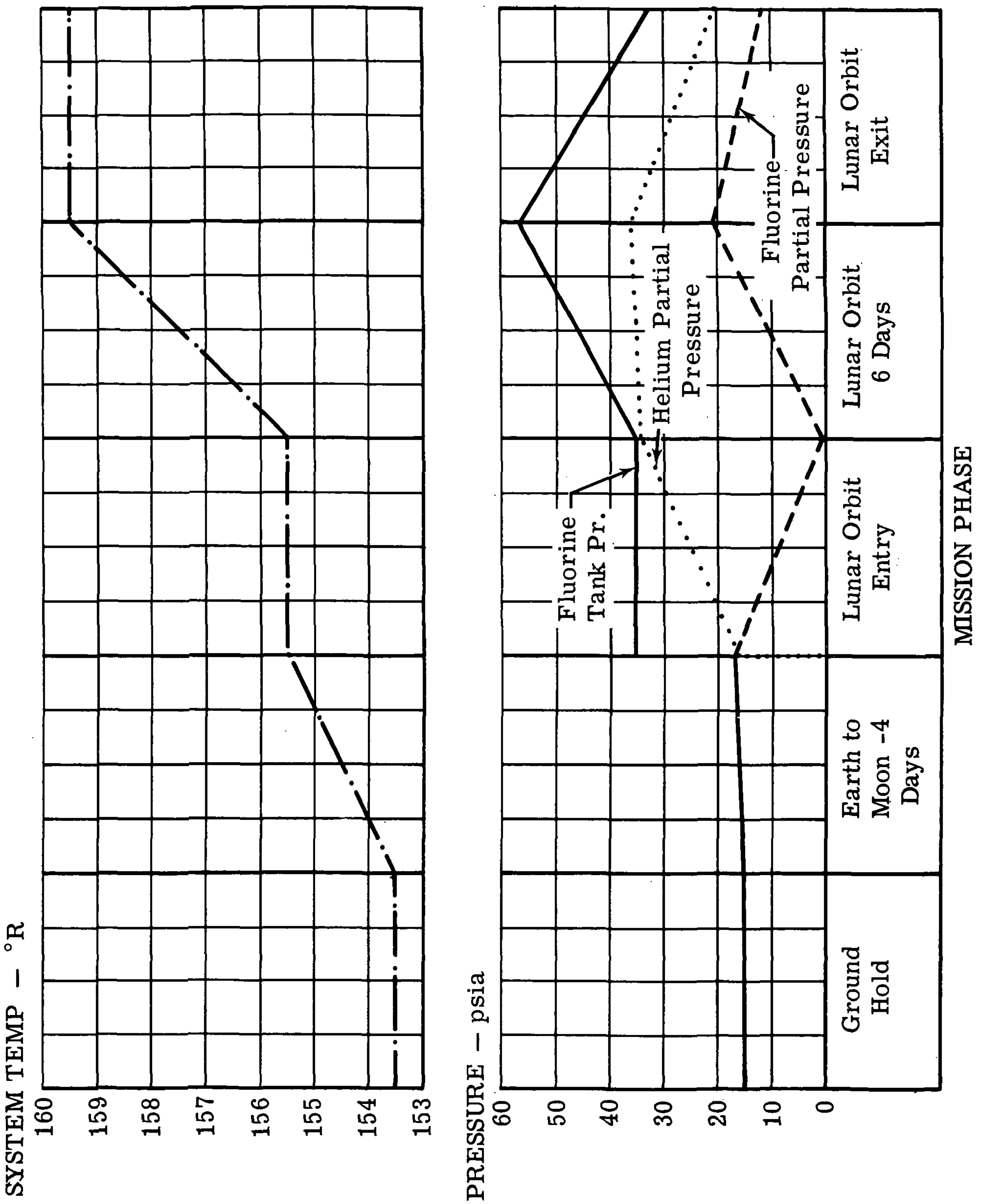


Figure VI-3. Low Pressure Fluorine Tank Time History

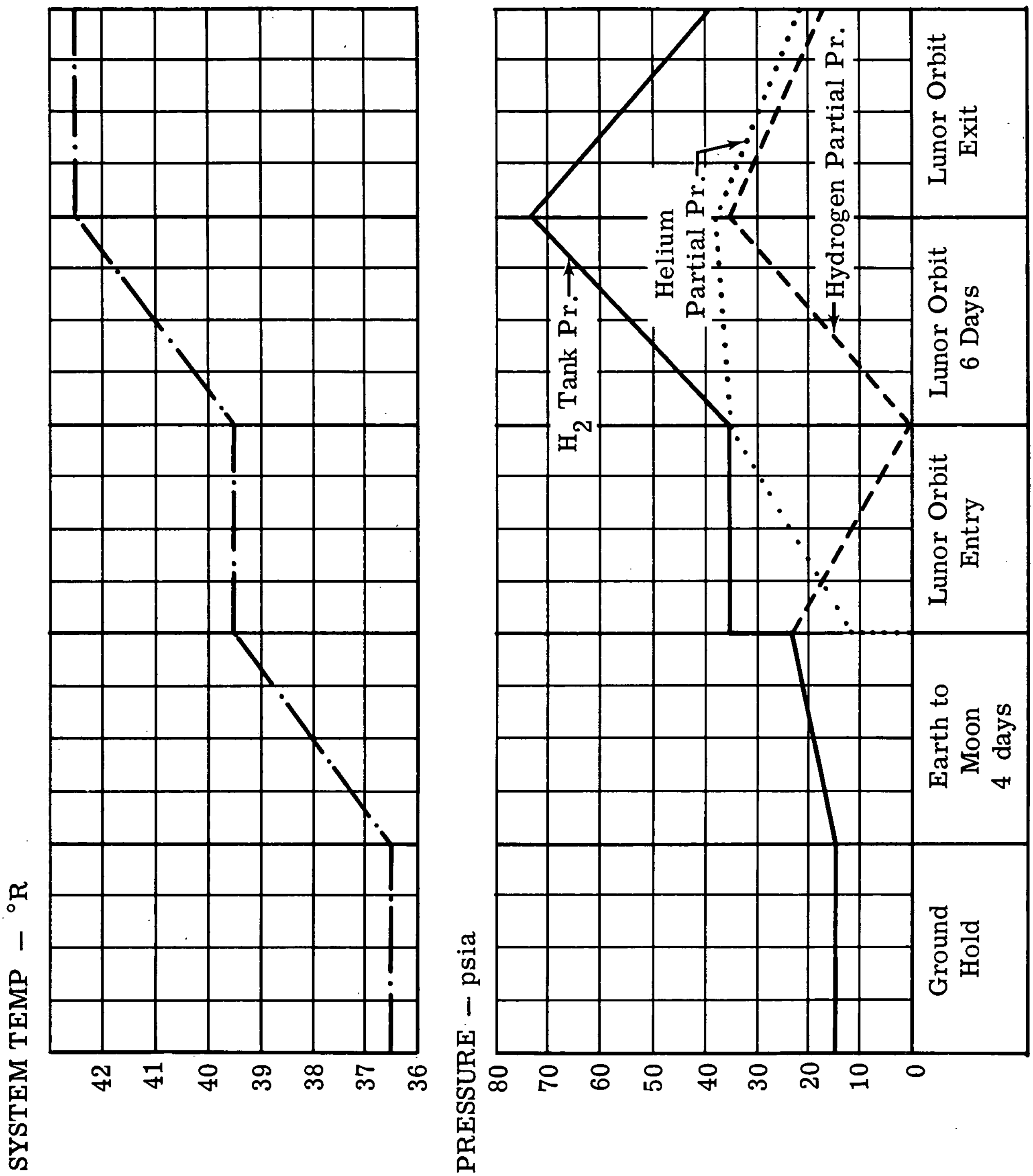


Figure VI-4. Low Pressure Hydrogen Tank Time History tangential pump ports through cavitating venturis, making the gas generator flow also insensitive to downstream hydraulic resistance. The turbine pump drive, therefore, operates as a stable open loop control system and at constant pump inlet conditions supplies constant power to the turbine from the gas generator resulting in constant mixture ratio and thrust.

The mixture ratio tolerances which are achieved with this control system are guaranteed to be ±1-1/2 per cent from engine to engine and ±1 per cent from run to run on a given engine. Since the volumetric mixture ratio control inherent in the engine can be predicted to within ±1 per cent, no auxiliary mixture ratio control system need be employed in the propellant feed system. This is the method of operation which is currently being successfully utilized on space stages which employ Bell Aerosystems engines.

Using the rocket engine as the control device, propellant utilization will be accomplished by loading 1 per cent excess hydrogen, intending to always run to oxidizer exhaustion. If the engine runs at the set (mean) mixture ratio, 1 per cent hydrogen will remain at oxidizer exhaustion. If the engine operates at the 1 per cent oxidizer rich limit, 2 per cent of the fuel will remain. Operation at the 1 per cent fuel rich limit results in simultaneous propellant exhaustion. Table VI-2 indicates the 1 per cent fuel addition provided for propellant utilization (mismatch).

e. Helium Tanks

Helium for both the high- and low-pressure hydrogen and fluorine systems is stored in spherical tanks within the respective propellant tanks. As discussed above, it is neither desirable nor necessary to add heat to the pressurization gases since, during the long-term mission with frequent engine firings, equilibrium temperature conditions will be achieved within the propellant tanks.

As shown in Table VI-2, helium for the hydrogen system is stored in a spherical, annealed titanium tank located within the last emptied low pressure hydrogen tank. This material has the highest strength-density ratio of the available materials for high pressure gas storage bottles. The alloy, 6AL-4V, in the annealed condition is satisfactory for use at -423°F because of its toughness and weldability.

Helium pressurization gas for the high- and low-pressure fluorine propellant tanks is stored in the last emptied, low-pressure fluorine tank. Since titanium alloys are not compatible with fluorine, a stainless steel material, PH15-7MO, which is compatible with the propellant, has been substituted for this application.

f. Bleed Estimates

The initial start, and restarts, of the pump-fed side of the system require peopellant bleed to cool the F_2 and H_2 pumps to a temperature such that the boiling rate and subsequent vapor formation do not "vapor-lock" the pumps. For given pump designs, the cool-down transient is primarily influenced by two parameters, the initial temperature of the pumps and the mechanism of heat transfer occurring. The state-of-the-art knowledge concerning boiling of liquid fluorine and hydrogen, as well as the effect of gravity on the heat transfer rate, is very limited and the analysis required to determine the initial temperature of the pump components is complex. Also, it is almost impossible to obtain a single heat transfer correlation equation to describe the effect of the various parameters over the entire range of ΔT since the heat transfer mechanism differs radically in the various regimes of boiling. In general, the regimes for free and forced convection, nucleate boiling and film boiling are correlated in the literature, however, regime transition, i.e. nucleate to film, and the temperature difference can only be determined experimentally for each fluid and no data exists or is readily available for fluorine and hydrogen for the pump configurations advanced in a later section.

The preceding discussion is presented to justify a simplified estimate of the propellant bleed requirements. A more detailed analysis could not be undertaken within the scope of the propulsion system study work. The estimate is based on the total heat capacity of the pumps over the range of cool-down, the heat of evaporation of the propellants and an assumed heat transfer efficiency. A relatively low initial pump temperature is based on the location of the pumps relative to the propellant tanks and the use of shielding about the pumps to reduce their view factor of the relatively warm propulsion module structure. The bleed estimates were obtained as follows:

(1)	Weight of Hydrogen Pump, Approx.	9 pounds
	Initial Pump Temperature, T_1	-200°F
	Final Pump Temperature, T_2	-420°F
	Mean Specific Heat of Pump (Aluminum Alloy), T_1 to T_2	0.13 BTU/lb/°R
	Heat Required to Cool Pump $Wcp\,\Delta T = 9\ (0.13)\ 220 =$	260 BTU
	Heat of Vaporization of Hydrogen	190 BTU/pound
	Weight of Hydrogen = 260/190	1.37 pounds

	Heat Transfer Efficiency (Assumed)	20%
	Total Pounds of Hydrogen Bleed/ Start 5 x 1.37 =	7 pounds
(2)	Weight of Fluorine Pump. Approx.	8 pounds
	Initial Pump Temperature, T_1	-200°F
	Final Pump Temperature, T_2	-305°F
	Mean Specific Heat of Pump (Aluminum Alloy) T_1 to T_2	0.17 BTU/lb/°R
	Heat Required to Cool Pump	143 BTU
	Heat of Vaporization of Fluorine	71.5 BTU/lb
	Weight of Fluorine = 143/71.5 =	2.0 pounds
	Heat Transfer Efficiency (Assumed)	20%
	Total Weight of Fluorine Bleed/ Start	10 pounds

2. <u>Propellant Acquisition</u>

The normal mode of propellant acquisition (positive expulsion) for pressure-fed firings consists of the operation of diaphragms in the high pressure propellant tanks. Pump-fed firings are accomplished with the aid of ullage rockets incorporated in the mission attitude control system. The ullage rockets will also provide a backup for the high pressure tank diaphragms. In addition, pressure-fed firings could be made in an emergency without propellant acquisition. Fluorine-hydrogen thrust chamber tests which have been conducted at Bell Aerosystems Company normally start with both propellants in the gaseous phase. The period of operation with gaseous F_2 resulted from fluorine evaporation in feed lines and in the injector of the thrust chamber. The hydrogen evaporates completely in the coolant jacket just prior to engine starting and is subsequently heated to the superheated gas region by the combustion process. If propellant acquisition were not available for an Apollo pressure-fed firing emergency operation, the initial flow of propellants into the chamber would consist of two-phase mixtures which, based on test experience, would provide satisfactory hypergolic ignition. The thrust produced would then provide propellant settling. The uncontrolled loss of helium gas that can occur during two-phase propellant starting dictated the normal modes of diaphragm and ullage rocket for propellant acquisition. The small size of the high pressure tanks influenced the selection of positive expulsion devices for those tanks because of the additional weight of propellants and tankage for complete ullage rocket propellant acquisition. The small tank size reduces the development time and cost for the expulsion devices. The remainder of this section will present a general discussion of positive expulsion devices and test experience to substantiate the current selection of diaphragms for the high pressure tanks.

In designing and developing a positive expulsion device, there are certain general characteristics which must be utilized as design criteria. An important consideration in the design of the tank and expulsion device system are the restraints placed on the design by geometry restrictions. As an example, a reversing diaphragm works extremely well in a spherical tank whereas use of such a device in a cylindrical tank of moderate to high L/D ratio is generally undesirable. In the design of an overall vehicle, careful attention must be attached to the price which is paid for a specific tank geometry.

Some of the more pertinent design requirements which must be considered when designing a positive expulsion device are listed below:

(1) Reliability

(2) Low Weight

(3) High Expulsion Efficiency

(4) Compatibility with the Propellants

(5) Ease of Loading

(6) Ease of Manufacture

(7) Low Cost

(8) Early Availability

(9) Ability to Undergo Repeated Expulsions

(10) Ease of Replacement

(11) Low Permeability of the Propellants

(12) Ability to Withstand Acceleration and Vibration

(13) Ability to Function Reliably Within the Range of Operating Temperatures

Several types of positive expulsion devices were analyzed for the Apollo application. Included in these concepts are piston tanks, collapsing bellows, reversible diaphragms and expanding or collapsing bladders. The piston type tank is generally limited to systems requiring a low total impulse where the tank diameter is small and heavy walls impose no weight problems. In larger tanks (over 12 inches), piston jamming due to the tank walls deflecting, renders the piston concept impractical. The collapsing bellows expulsion method was eliminated from consideration since this device requires a small cylindrical tank with a large length to diameter ratio. In addition, this type of positive expulsion device is very inefficient with regard to propellant volume utilization. A reversing plastic or metal diaphragm is suited to the high pressure propellant tanks because the tanks are spherical. The plastic diaphragm is capable of repeated expulsions whereas the metal diaphragm expulsion repeatability is uncertain. Only actual testing will reveal its limitations.

Expanding or collapsing bladders are also well suited to this application. Bell Aerosystems Company has used this type of bladder successfully in many applications. Beech Aircraft Corporation of Boulder, Colorado, used a gas expanding bladder in a liquid hydrogen positive expulsion research program. The expanding or collapsing bladders, however, start expulsion or complete expulsion with considerable folding. Bell Aerosystems Company's bladder experience has shown that certain types of folding can result in bladder failure at low temperatures. Since this is a manned vehicle application, the most reliable device has been selected; a reversing diaphragm.

Another but equally important design consideration in the selection of a positive expulsion device is the influence of the material properties on the design; for example, permeability, corrosion, strength, flexure characteristics, tear resistance and many others. In addition, state-of-the-art fabrication methods and their influence on material properties is another important factor in the selection of a specific material. Desirable properties of metal expulsion devices are resistance to work hardening and high elongation. Resistance to work hardening is important when plastic deformation occurs during forming, installation or repeated expulsion. Examples of materials that have excellent low work hardening properties are pure aluminum, gold and tin. In contrast, 300 series stainless steel, although having a large span between the yield and ultimate strengths, work hardens very rapidly. The property of uniform elongation in a metal is closely allied with its work hardening properties; however, it can also be affected by local variations in heat treatment and non-uniformity in alloying. Desirable materials from this standpoint are therefore pure non-heat treatable metals.

Desirable properties of plastic materials include low temperature flexibility, adequate tensile strength at both high and low temperatures, resistance to three-corner fold failures, low porosity, ability to act as a gasket seal and resistance to cold flow under high compression loads.

A soft metal diaphragm is utilized for the purpose of providing positive expulsion of liquid fluorine in the high pressure tanks of the pressure-fed portion of the system. Since the tank is spherical in shape, a diaphragm of 1100-0 aluminum alloy is ideally suited for this purpose. The tank will be of the flanged and bolted design and the diaphragm will be sealed at the equator by a tongue-and-groove joint. The diaphragm will be drawn from sheet stock and will be annealed after forming. At ambient temperature 1100-0 aluminum has a low modulus of elasticity and an extremely good elongation. Figure VI-5 is a comparison of strength and ductility of various welded aluminum alloys. When subjected to cryogenic temperatures, the strength and ductility are substantially increased. This condition is graphically illustrated in Figure VI-6.

An aluminum alloy bladder has been designed and successfully tested at Bell as part of a research program. This effort proved conclusively that the concept of an aluminum expulsion bladder is practical. In fact, the best test results were obtained when expelling liquid nitrogen. As a further justification for the use of this concept, it can be noted that the Naval Ordnance Test Station at Inyokern, California has flight tested a liquid propellant target rocket that utilizes 1100-0 aluminum bladders. Figure VI-7 shows an aluminum alloy test bladder.

A Mylar diaphragm is used for providing positive expulsion in the liquid hydrogen tank, of the pressure-fed portion of the system. The H_2 tank design is the same as that previously described for the F_2 high pressure tank. The proposed diaphragm is fabricated from sheet Mylar plastic film of approximately 4 mils in thickness. The flexibility characteristics of this material at liquid hydrogen temperatures is remarkably good. In addition to its excellent flexural characteristics, Mylar has a very high tensile strength. The diaphragm is fabricated from orange peel type segments that are joined together with a bonding agent. The bonded joints also have excellent flexural characteristics at liquid hydrogen temperature. The flange of the diaphragm will be built-up of several layers of Mylar bonded together to form a thick "gasket-like" sealing surface. The entire surface area of the diaphragm will be impregnated with vacuum deposited aluminum. This coating will help to reduce the permeability factor.

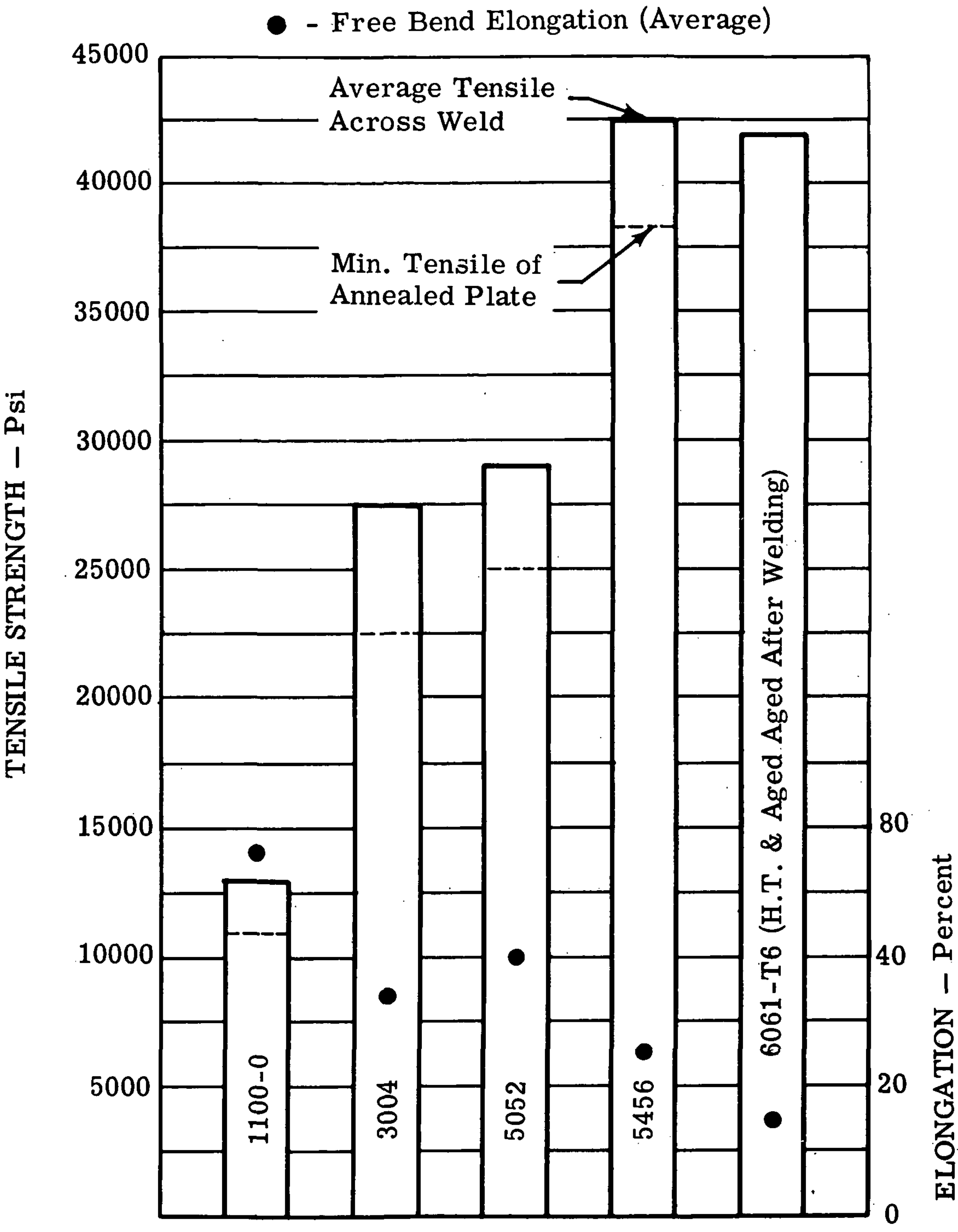


Figure VI-5. Comparison of Strength and Ductility in Welded Aluminum Alloys

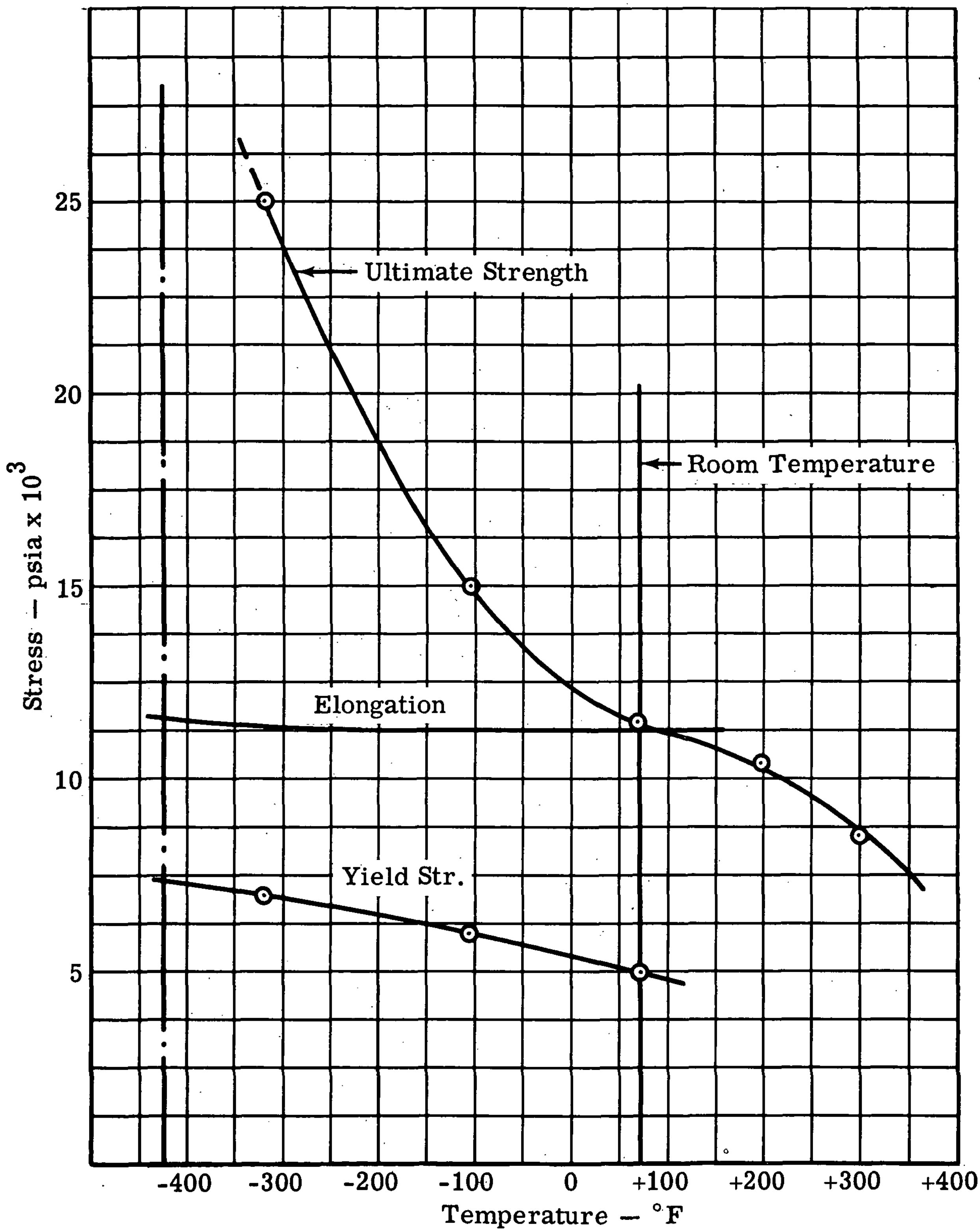
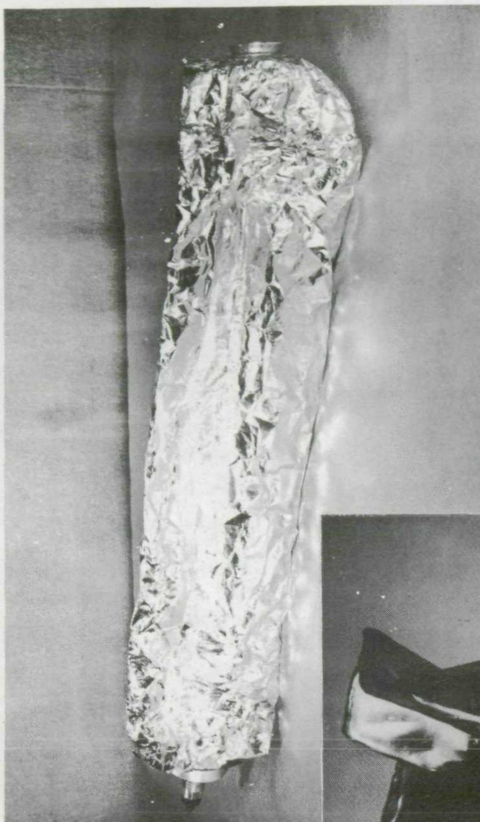
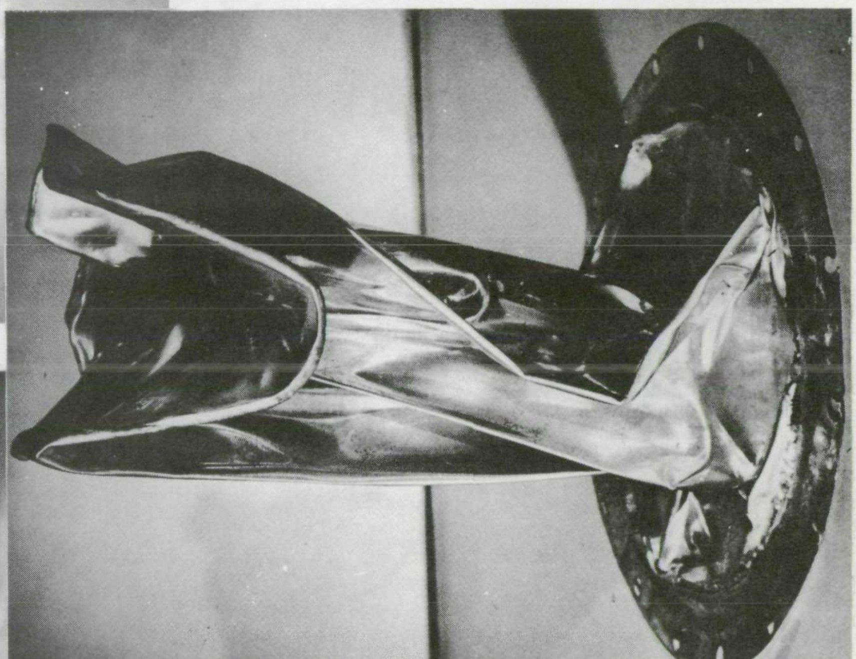


Figure VI-6. Low Temperature Properties of 1100-0(2SO) Aluminum



Aluminum Foil Bladder -- Collapsed

Aluminum Foil Bladder -- Expanded



Aluminum Sheet Bladder -- Collapsed

Figure VI-7. Aluminum Foil Test Bladder

Extensive testing has been carried out by Beech Aircraft Corporation of Boulder, Colorado, in the area of developing a liquid hydrogen bladder. As part of a research program, a 24-inch diameter gas expanded Mylar bladder was developed and tested. To observe the operation of the bladder during expulsion with liquid hydrogen, a 24-inch glass Dewar was used as a tank. The final test configuration resulted in a bladder that underwent 75 repeated expulsions without a failure of any kind. The results of this test program are outlined in Beech Aircraft Engineering Report Number 9501 by C. W. Spieth and is entitled "Development of Positive Expulsion Techniques for Cryogenic Fluids".

3. Insulation Analysis

A preliminary investigation of the insulation requirements and the heat input to the fluorine and hydrogen tanks was made in order to make a weight estimate and to set forth a workable pressurization and expulsion scheme. The analysis has been done for both the actual mission and for the ground hold. The mission should be separated into two parts, 1) Earth-Moon and Moon-Earth transit and 2) Moon orbit, however because of the rather preliminary nature of this study, one environment for the mission was assumed to prevail throughout. The insulation system has been developed for the mission, independent of the ground hold. The latter can be accomplished with appropriate ground support equipment.

a. Mission Environment

The vehicle is oriented such that one side is facing the sun. There is a solar collector on this side that reaches an equilibrium temperature of 220° in the center, 90° on the periphery. The 220° core is behind the base of the propulsion module. There is a cabin radiator that covers 175 square feed of the periphery of the propulsion module that operates at 70° to 80°F. The remainder of the periphery of the propulsion module does not have any significant heat input and will assume a low temperature, about -250°F, due to radiation to space. Appropriate view factors and temperature levels of these various components were determined and it was found that the propulsion module environment would be approximately -20°F.

b. Insulation Requirements for Mission

The above source temperature was used to determine the insulation required to keep the heat flux below certain predetermined values so that the tanks could be sealed for the entire mission. These values are listed below.

Tank	No. of Tanks	Allowable Heat Flux BTU/hr Tank	Tank Radius Inches
Low Pressure H_2	2	15	29.5
Low Pressure F_2	4	10	18.65
High Press. H_2	2	5	13.6
High Press. F_2	1	5	12.2

The insulation thicknesses were determined in the following manner:

The equivalent thickness of insulation for a flat surface may be written as

$$\tau_e = \frac{R\,\tau}{R + \tau} \tag{1}$$

$$q/A = \frac{K\ \Delta T}{\tau_e} \tag{2}$$

$$q = \frac{K(4\pi R^2)}{144\ \tau_e}\quad T \tag{3}$$

Combining equations (1) and (3):

$$\tau = \frac{R}{\dfrac{36q}{\pi R\ K\,\Delta T} - 1} \tag{4}$$

where:

A = area of sphere		Ft^2
τ = insulation thickness		Inches
τ_e = equivalent thickness of insulation for a flat surface		Inches
R = radius of tank		Inches
q = heat flux		BTU/Tank

K = conductivity of insulation BTU in/ft^{2}hr °F

ΔT = temperature gradient across insulation °F

Further, it was assumed that 25% of the allowable heat input to the tank flows through the lines and supports. Linde SI-4 was assumed as the insulation for the tanks. Due to the attachments required to hold the insulation on the tank during periods of high inertial loads, both the conductivity and density will be greater than for the normal uncompressed condition. For this application, the conductivity is 0.00045 BTU in./ft^2 hr °F for the temperature range from -423° to -20°F and 0.00054 for the range from -306° to -20°F and the density is 7.5 lb/ft^3. The following insulation thicknesses were obtained:

Tank	Insulation Thickness	Approximate Insulation Weight per Tank
Low Pressure H_2	1.27 inch	63.7
Low Pressure F_2	.64 inch	12.5
High Pressure H_2	.82 inch	8.7
High Pressure F_2	.56 inch	4.8

Thus the insulation around the tanks weighs approximately 210 pounds. The lines must be insulated also to restrict the heat leak. A weight of 7.5 pounds has been allowed for insulation of the lines.

c. Ground Hold

The ground hold for cryogenic propellants is not simple especially when hydrogen is being considered; however it does not impose any insurmountable problems. The boiling point of hydrogen is so low that it will liquify and solidify air if it is allowed to come in contact with the hydrogen tank. For the Apollo propulsion module there appear to be two feasible ground hold procedures: 1) insulate the tanks as stated above for the mission, seal up the compartment reasonably well and provide a helium environment; 2) draw a vacuum on all of the insulation around the tanks by enclosing each of the tanks in a sealed Mylar bag, pulling a partial vacuum and allowing them to cryo-pump to a good vacuum. This latter method would present accessibility difficulties and it may be difficult to prevent leaks. However, the savings in the amount of boiloff would be significant as can be seen in the following table.

Tank	Boil-off per Tank - Pounds/hr	
	He Atm.	Evacuated
H_2 Low Pressure	96	3.2
F_2 Low Pressure	184	6.1
H_2 High Pressure	32	1.1
F_2 High Pressure	91	3.0

The ground hold operation will be keyed to the ground support equipment that is needed for the SATURN booster.

4. Propellant Sloshing and Control

A primary problem associated with the use of liquid fuel is the possibility of dynamic coupling between the sloshing of propellants in tanks and rigid and/or flexible degrees of freedom of the spacecraft with the control system to produce a system instability. As fuel is consumed, the frequencies of the fundamental rigid and flexible degrees of freedom of the spacecraft are altered through changes in weight and inertia distribution. Moreover, fuel consumption results in changes in the slosh frequencies of the propellant. Such frequency changes may act to produce a coupling with the frequency of the control system resulting in an instability of the overall system. The principal method available for relieving adverse coupling effects between fuel sloshing and control system response is by means of a baffle system arrangement which will alter the damping-frequency characteristics of the sloshing propellant by breaking up the flow. An instability resulting from adverse coupling of the spacecraft with the control system would require modification of the "stiffness" characteristics of one or the other.

With regard to the configurations presently being considered, analytical methods are available for ascertaining sloshing frequencies for the multiple sphere configuration. In addition, analytical methods for establishing the effect of certain baffling systems are available and will be used for the tank design. Reference 4 presents <u>non-dimensionalized</u> results which may be used to establish slosh frequencies for the vehicle tankage configuration. Such results are compared in Reference 4 with analytical predictions based on the method presented in Reference 5. Correlation is extremely good. Lastly, Reference 6 presents experimental data to establish the damping effectiveness of fuel sloshing baffles applied to ring baffles in cylindrical tanks.

Using these data, provision will be made for slosh control in the propellant tank. Verification testing of slosh motion and frequencies will be performed during the development period.

C. 12,000/4,000 POUND THRUST CHAMBERS

1. Predicted Performance:

Considerable effort has been expended to determine the actual performance attainable with fluorine oxidizer propellant combinations. Due to the high combustion temperatures, a substantial amount of energy is absorbed in dissociation of the products of combustion. To approach the theoretically attainable performance levels, the energy lost in dissociation must be released by re-association as the exhaust gases expanded in the divergent nozzle. A comprehensive investigation of the problem was completed at the Bell Aerosystems Company test facility as part of a feasibility contract with the Air Force WADC Propulsion Laboratory. A series of tests utilizing fluorine and hydrogen, was conducted with 14.85:1 and 15.0:1 area ratio nozzles at chamber pressures of 400 psia and 180 psia respectively. Reference 7 is the final report on this work. With the use of this data and the NASA effort described in Reference 8 the minimum combustion and nozzle efficiencies were established. Both agencies, Bell Aerosystems and NASA, produced similar results: The combustion efficiency decreased with increasing mixture ratio. These combustion efficiencies are:

Mixture Ratio	Combustion Efficiency (Min.)
10	97.0%
11	96.9%
12	96.7%
13	96.5%
15	95.5%

The nozzle efficiency, applicable to contoured nozzles, was estimated based on three-dimensional analytical studies verified by altitude testing of various large area ratio and length nozzles at the Arnold Engineering Development Center.

High altitude upper stage applications require large area ratio nozzles. However, the performance increase with each area ratio increase must be weighed against the inert mass increase of the propulsion system. Also, for any given nozzle length, care must be taken to assure that the optimum area ratio is not exceeded. For the same area ratio, gains in three-dimensional losses may be reduced by extending the length. Such trade-off requires a lengthy study such as performed during the Bell Aerosystems Agena Engine Program.

The results of analysis and tests have enabled Bell Aerosystems Company to design optimum thrust chamber configurations for the proposed application. Nozzle efficiencies of 97% of shifting equilibrium theoretical values, repeatedly demonstrated with fluorine and hydrogen, can be readily maintained with large area ratio nozzles of minimum length and weight.

The specific impulse at various mixture ratios is shown in Figure VI-8. The nozzle efficiency, 97% of equilibrium theoretical, is for a divergent nozzle 80% the length of a 15 degree semi-divergent conical nozzle.

2. Thrust Chamber Heat Transfer

High performance levels achieved with high chamber pressures and minimum physical size must be compatible with cooling requirements. The maximum design chamber pressure is dictated primarily by the limiting heat flux, gas side wall temperature and the minimum tube size available in tapered sizes. The 0.010 inch thick tube wall with a 0.075 inch I.D. will result from a maximum heat flux of 10.6 BTU/in.^{2}sec and a chamber pressure of 300 psia, see Figure VI-9. Larger I.D. tubes might be employed if nickel were used as the tube material since its thermal conductivity is three times higher than stainless steel. Certain metallurgical drawbacks, however, primarily grain growth, preclude its use. The tube wall thickness must be held to a minimum but must still be formable and structurally adequate. For the proposed design, the size was selected by evaluating the coolant film coefficient accounting for variation in transport properties of the hydrogen at the large bulk to coolant side wall temperature ratios.

The hydrogen offers a tremendous potential as a heat sink because its high specific heat and low bulk temperature provides a large temperature rise potential.

The preliminary vehicle layout with the propulsion system has the thrust chamber nested among the propellant and pressurant tanks. This arrangement precludes the use of a radiation cooled divergent nozzle extension and the entire nozzle is regeneratively cooled. The dry weight of radiation cooled and regeneratively cooled nozzles is essentially the same and with the low density hydrogen coolant, no significant weight increase results as the tubes are filled. Other propellants, pure liquids, have limited heat sink and the use of the radiation cooled extension is required to meet cooling envelope requirements. Hydrogen, a gas, suffers only from increased free stream temperature and decreased density.

Several methods for computing the gas film coefficients have been reviewed for basic heat transfer applications. A modification of Bartz'method as offered in Reference 9 was found to fit test results of chemical rockets. This was found to be especially true in those cases where the combustion gases have attained homogeneous composition entering the convergent nozzle and heat rejected to the nozzle only was measured. Bartz's relationship, as stated, is

$$hg = \frac{.026}{D_T^{2}} \left[\frac{\mu^{.2}\, C_P}{P_R^{\ .6}}\right]_o \left[\frac{P_c\, g}{c^*}\right]^{.8} \left[\frac{D_T}{R_c}\right]^{.1} \left[\frac{A_T}{A}\right]^{.9} \sigma$$

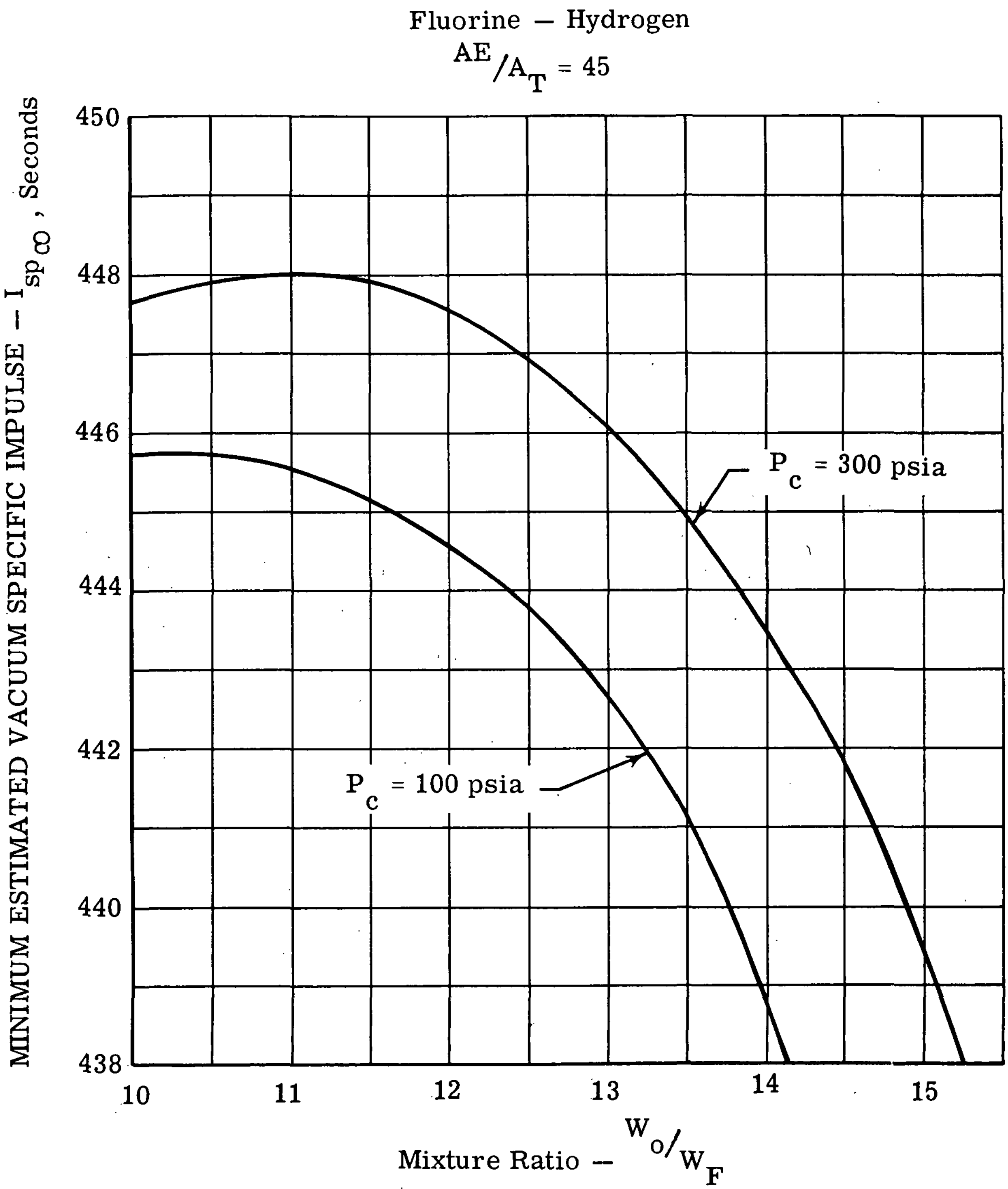


Figure VI-8. Thrust Chamber Estimated Vacuum Specific Impulse Versus Mixture Ratio

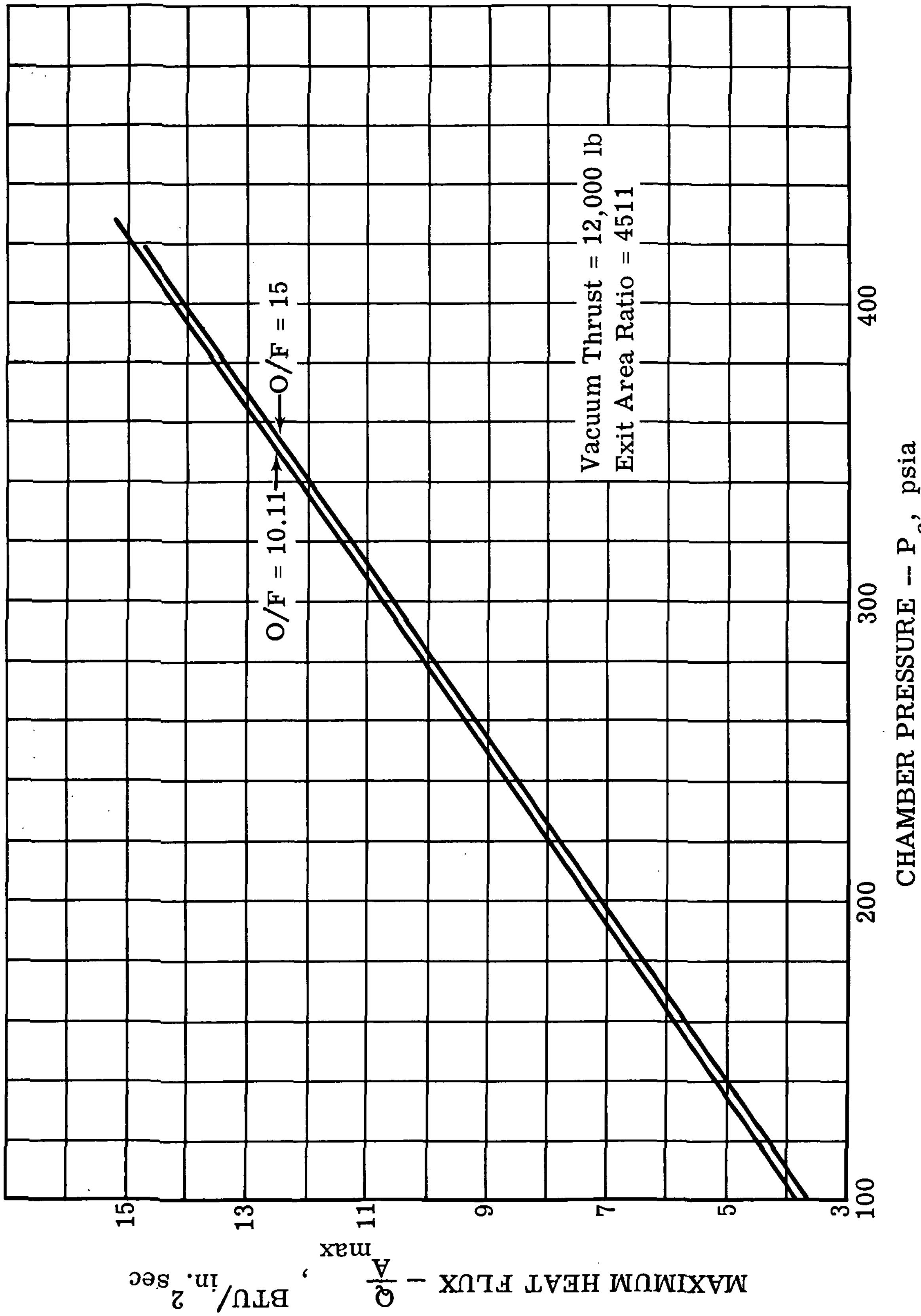


Figure VI-9. Maximum Heat Flux Versus Chamber Pressure — Fluorine-Hydrogen where P_c = chamber pressure, psia

c^* = characteristic exhaust velocity, FPS

D_T = throat diameter, inches

R_C = throat contour radius, inches

hg = gas film coefficient

μ = gas viscosity, lb/in. sec

C_P = specific heat at constant pressure, BTU/lb. °F

P_R = Prandtl number $C_P \mu / K$

σ = dimensionless factor accounting for variation of gas density and viscosity values across the gas boundary layer

Subscript o = properties at stagnation conditions

By rearrangement a simplified modification is:

$$hg = .026\ C_P \frac{W}{A} \left[\frac{W}{A} \frac{D}{\mu}\right]^{-.2} \left(P_R^{\ -.6}\right) \sigma$$

The gas properties are taken at stagnation conditions. The maximum heat flux is shown in Figure VI-9. The computed values are found by

$$\frac{Q}{A_{max}} = hg_{max} \left(T_C - T_{WG}\right)$$

where $\frac{Q}{A_{max}}$ = heat flux at throat, BTU/in.^{2}sec

hg_{max} = gas film coefficient at the throat, BTU/in.^{2}sec °F

T_c = actual gas temperature (stagnating conditions), °R

T_{WG} = gas side wall temperature, °R

Computation of gas film coefficients at stations other than the throat are described below.

The gas film coefficients for divergent nozzles at area ratios greater than 4 were computed as

$$hg_x = 1.515\ hg_{max} \left[\frac{A_T}{A_x}\right]^{0.9}$$

with hg_x = gas film coefficient at area A_x

hg_{max} = gas film coefficient at the throat, A_T

For divergent nozzle area ratios less than 4

$$hg_x = hg_{max} \left[\frac{A_T}{A_x}\right]^{1.2}$$

For convergent nozzles

$$hg_x = hg_{max} \left[\frac{A_T}{A_x}\right]^{0.8}$$

The above three adjustments for area ratio are based on extensive experimental heat transfer work conducted at Bell on chemical rockets with large expansion area ratios.

The main difficulty in correlating the heat rejection data is the fuel rich outer boundary of the thrust chamber. With the injectors designed for film cooling of the barrel or straight portion of the thrust chamber, the theoretical rejected heat, based upon a homogeneous gas mixture throughout the nozzle, will indicate higher rejected heats.

For the Bell Aerosystems Model 8031 fluorine-hydrogen thrust chamber, the total rejected heat was

computed 4666 BTU/sec

test 3580 BTU/sec

percent test/computed = 76.8%

The divergent nozzle absorbed 43% of the total rejected heat.

3. Thrust Chamber Design Details

The internal surface of the thrust chamber must be selected so that a minimum size is used that will still produce adequate combustion efficiency. The design approach is to evaluate experimentally the effect of chamber volume on combustion efficiency; a critical point becomes fairly evident where a reduction in chamber volume is reflected in reduction in combustion efficiency.

The contraction ratio, chamber area to throat area, is again a result of design experience of the injector where the drilling patterns dictate maximum mass flow (or flow density).

By designing to a minimum internal surface area, the exposed surface area will produce a minimum rejected heat. This will aid in preventing the hydrogen coolant from attaining exceptionally high velocities in the coolant passages. Very high frictional losses will occur as a result of high velocities at the end of the coolant jacketing. In the normal jacket operating regime, the specific volume of the hydrogen increases 20-40 times.

For the design study here a contraction ratio of 2.5:1 and a characteristic length (L*) of 30 in. were used.

Figure VI-10. shows the proposed stainless "tapered tube" thrust chamber configuration, which is based on the use of coolant passages made from stainless steel tubing with a pressure tight external jacket of interlocking foil wrapping.

The use of the tapered tubing provides a coolant jacket of minimum weight and permits the use of circular cross-sections at the throat which facilitate the design of a minimum weight cooling envelope.

The chamber and nozzle portion of the thrust chamber incorporates an external wrapping of interlocking foil made of 0.0125 inch thick AM 350. This jacket provides a second pressure tight wall which supplements that formed by the brazed joints between adjacent tubes.

The use of this interlocking foil was first demonstrated at Bell Aerosystems Company on a 12,000 pound thrust F_2-H_2thrust chamber. This design used the wrapping as both the outer wall of the coolant jacket and as the hoop load carrying external structure. Problems encountered in providing adequate attachment of the spacers from the inside shell to the wrapping led to the selection of tubes for the coolant passages of the unit proposed.

Two types of tubes will be used to form the coolant jacket of the thrust chamber. Both will be made from type 347 stainless steel and have a .010 inch wall thickness. The first type, called the primary, will be a double tapered tube having a 0.150 inch diameter at its forward or chamber end, a 0.095 inch diameter at the throat and 0.310 inch diameter at a station equivalent to the location of the coolant inlet manifold on the divergent nozzle.

The second tube type, or secondary tube, provides additional surface area in the divergent nozzle as the area ratio increases. This tube of constant circumference equivalent to a .310 inch diameter will be flattened toward the forward end. Its length will be equivalent to the untapered portion of the primary tube.

Each primary and secondary tube will be fully inspected upon receipt from the vendor and will undergo certain bending and squeezing operations prior to assembly into a thrust chamber.

The secondary tubes will be bent to a curved shape and will have their forward ends bent up to form the inlet to the coolant passages. The tube will also be formed to an approximate thickness of 1/2 its diameter at the forward end tapering to full diameter at its aft end.

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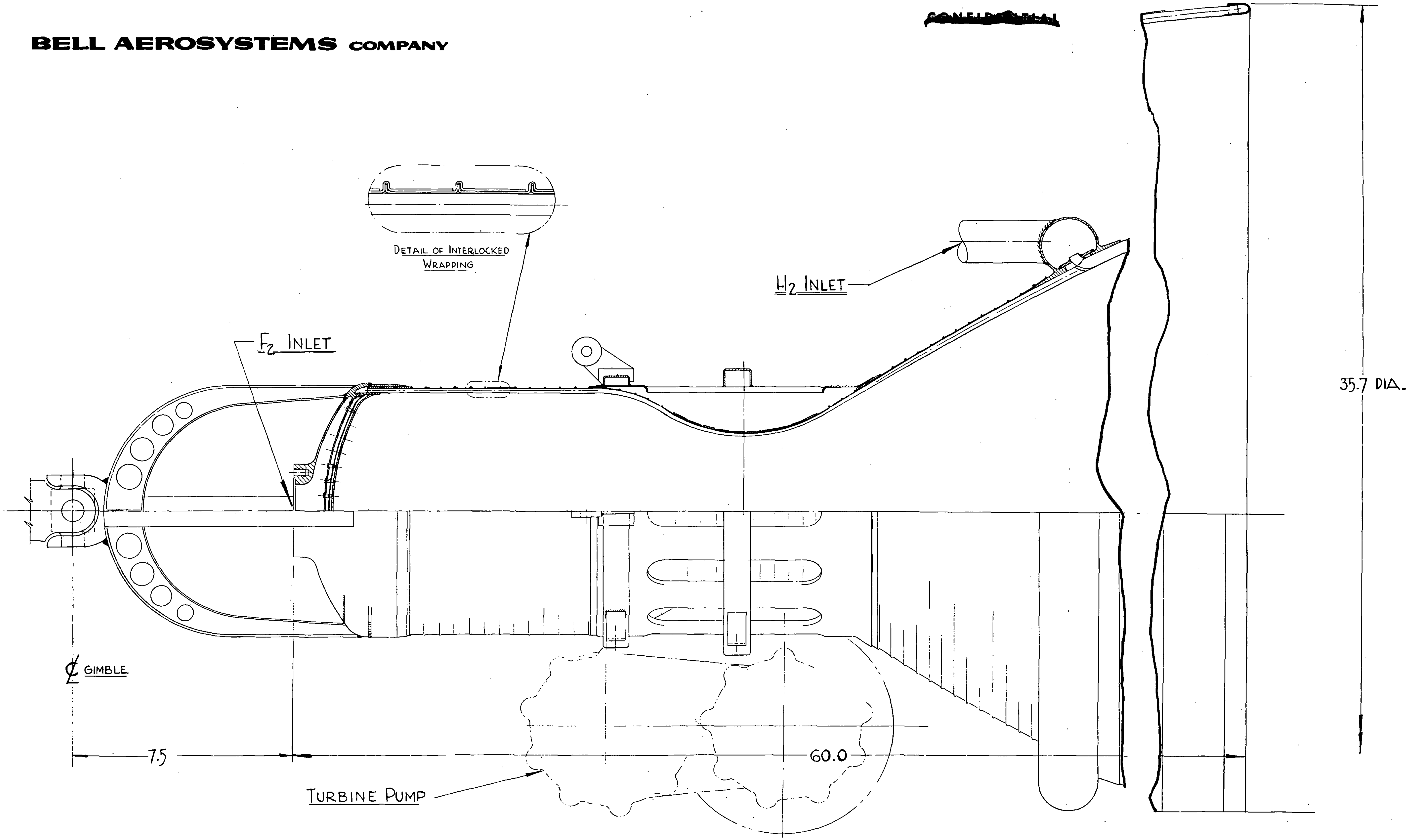


Figure VI-10. 12K Thrust Chamber

This forming will also result in a curvature matching the divergent nozzle contour.

A tube bundle will be fabricated using a mandrel which duplicates the inside contour of the thrust chamber. It will consist of 240 each of the primary and secondary tubes arranged alternately.

Following the initial brazing operation, the .0125 inch thick interlocking AM 350 foil wrapping will be applied from the injector end to the inlet manifold of the combustion chamber. In these areas where attachments are to be made to the foil wrapping such as at the injector end and at the throat, a filler strip is wound between the interlocks to provide additional area and mass for brazing and welding.

4. Injector Design

The paramount requirements in the design of high performance injectors are the proper distribution of the fluids in the manifolds and the choice of an orifice pattern in the injector face through which the propellant is injected into the combustion chamber in a manner which insures proper propellant conditioning (atomization, vaporization and mixing) which, in turn, promotes even and rapid combustion.

The amount of propellant conditioning required is dependent upon the properties or the propellants used. Generally speaking, maximum conditioning is required for non-cryogenic and non-hypergolic propellants. The degree of conditioning is somewhat less critical with non-cryogenic but hypergolic propellants. While with cryogenic propellants, with their high volatility, require the least extensive mechanical breakup and atomization in order to promote rapid vaporization. Fluorine, a cryogenic, when used as a propellant, requires little more than to be injected into the thrust chamber in reasonable sized streams in order to promote and produce rapid vaporization.

The injection of hydrogen, in a regeneratively cooled thrust chamber application, produces a stream of gas at the injector face. Because of this characteristic, unique in a rocket fuel application, the normally critical nature of the injector design requirements are minimized. Moreover, when employed as a fuel, it is readily utilized in the combustion process, and because of its low molecular weight results in high specific impulse values.

Intensive investigative programs seeking the optimum injector configuration for use with fluorine and hydrogen included tests of designs which evaluated impinging showerheads, random showerheads and triplet configurations over a wide range of injection density. Little or no change in efficiency was noted from one orifice drilling configuration to another. For this reason the simplest showerhead configuration in terms of distribution of propellants and reasonable heat rejection was decided upon in order to take maximum advantage of its manufacturing potentialities.

The injector proposed for this application is shown in Figure VI-10. Structurally the design of this unit is based on an elliptical back plate concept. This approach lends itself to the solution of the problem of proper manifolding and distribution of the propellants. The distribution plates for both the fuel and oxidizer are utilized as secondary structures with the oxidizer injection tubes acting as posts to tie these plates into an integral unit. The oxidizer is distributed under the elliptical back plate and over the fuel cross section of the injector. The fuel is distributed through a manifold across the injector then through a baffle to the injector orifices. This arrangement forces the fuel across the injector face both distributing the fuel and accomplishing injector face cooling. Fuel film cooling may be accomplished at the periphery of the injector face and during the initial testing program the quantity and arrangement of this fuel film cooling will be determined.

The face plate, baffle and injection tubes will be made from type 300 series stainless steel while the back plate will be made from AM 350 alloy steel.

The injector is brazed into the thrust chamber; however, the final joint which continues the thrust chamber coolant jacket into the injector is a welded joint. After the completion of the thrust chamber to injector assembly, the entire unit will be temperature conditioned to elevate the properties of the AM 350 alloy to a SCT condition.

Varying thrust by varying the flow of propellants to the injector is not a new concept. A considerable volume of testing to evaluate this method of controlling thrust has been conducted on numerous propellant combinations. Because of the critical injection requirements on conventional propellants, success in this area has been limited. However, in the case of the fluorine-hydrogen propellant combination, non-critical injection latitudes are greater, so that thrust variances in the order of three-to-one have been successfully and reliably performed with no loss in injector efficiency (Reference 8). The injection phenomenon which occurs as a result of a shift in flow rate in the case of fluorine-hydrogen is unique both because of the cryogenic properties of the oxidizer and the fact that the fuel is injected in the gaseous state. Here, the significant change in pressure loss through the injection orifices occurs on the liquid propellant side, while the gaseous propellant, because of shifts in density and velocity which accompany changes in flow rate, shows little change in pressure loss through the injection orifices.

TABLE VI-3

THRUST CHAMBER CHARACTERISTICS

	Rated Conditions	Step Thrust Conditions
Vacuum Thrust, lb	12,000	3,983
Thrust Chamber Pressure, psia	300	100
Mixture Ratio, W_o/W_f	13.0	10.0
Vacuum Specific Impulse, lb sec/lb	446.2*	445.8*
Propellant Flow Rate, lb/sec	26.90	8.92
Regeneratively Cooled Divergent Area Ratio	45	45
Throat Area, sq. in.	21.545	21.545
Nozzle Exit Diameter, in.	35.1	35.1
Oxidizer Feed Pressure, psia	400	111
Fuel Feed Pressure, psia	465	145
Cooling Fluid	Fuel	Fuel

* Minimum observed guarantee vacuum specific impulse; does not allow for 0.7% instrumentation error.

4. Gimbal Control System

a. Introduction

Two methods which may be utilized for positioning the engines about a two axis gimbal for controlling the thrust vector include: (1) conventional hydraulic powered servo-actuators, and (2) electro-mechanical actuators. In general, these systems function by utilizing two linear motion actuators for each thrust chamber installed at right angles to provide the desired gimbal pattern.

The hydraulic powered system incorporates piston type actuators with integral servo valves which meter the flow in response to guidance commands. The electro-mechanical type consists of an electrical motor driving a linear screw-jack through a reduction gear train into a pair of clutches which control the direction of actuator travel.

The advantages and disadvantages of these systems including specific recommendations for this application are outlined below.

b. Hydraulic Powered System

The conventional hydraulic system operating over a wide temperature and pressure range has demonstrated satisfactory performance in numerous applications of this nature. These systems are highly reliable and are universally used for both continuous and intermittent service where large inertia loads must be accelerated rapidly and positioned with a high degree of accuracy. However, the ability of these systems to function in a satisfactory manner after long periods of inactivity is questionable. Failure of the system could occur in numerous ways including loss of fluid due to seal deterioration and freezing of valve spools from lack of use. The low temperature conditions of the 14 day Apollo mission coupled with long periods of inactivity would result in sluggish response of the gimballing system and the consequent probability of vehicle instability.

c. Electro-Mechanical System

The electro-mechanical servo actuators are relatively small in size and provide high power amplification and rapid response with the added advantage of low weight.

The unit consists of a continuously rotating motor (either AC or DC) which drives two contra-rotating clutch housings. Both clutch pinions mesh with the output shaft, thus providing rotation in either direction. When one clutch is energized the output shaft rotates in a corresponding direction. Energizing the other clutch reverses the output. Rapid reversal is possible because the main elements of inertia are in the motor and clutch housings which always have a single direction of rotation.

Power is transmitted by means of dry magnetic powder. Energizing the clutch causes the magnetic particles to form linked chains across the gap completing the flux path set up in the magnetic core. The torque transmitted from the continuously rotating housing to the output shaft is proportional to the coil current over a wide range of input currents. Although the output of the dry-powder magnetic clutch servo is non-linear, this characteristic is avoided by operating the clutches in pairs with a bias current in each clutch. This results in a net torque that is linear and proportional to the differential current. For each value of differential input current, the maximum torque is applied until the limiting velocity is reached. At this point, the accelerating torque levels off in a manner similar to that of a hydraulic servo.

In actual operation, the torque generated by a constant speed motor is converted into linear motion by a recirculating ball screw which applies force to the thrust chamber. The relative difference between the actuator body and the actuator shaft is measured by a synchro. This position is compared with the desired actuator position and the resulting error amplified. This amplified signal controls the output torque from the clutches and the axial force generated tends to drive the error to zero. A rate generator is incorporated in the actuator to provide the necessary system damping. The gimbal system schematic is shown on Figure VI-11 and the electrical system block diagram on Figure VI-12.

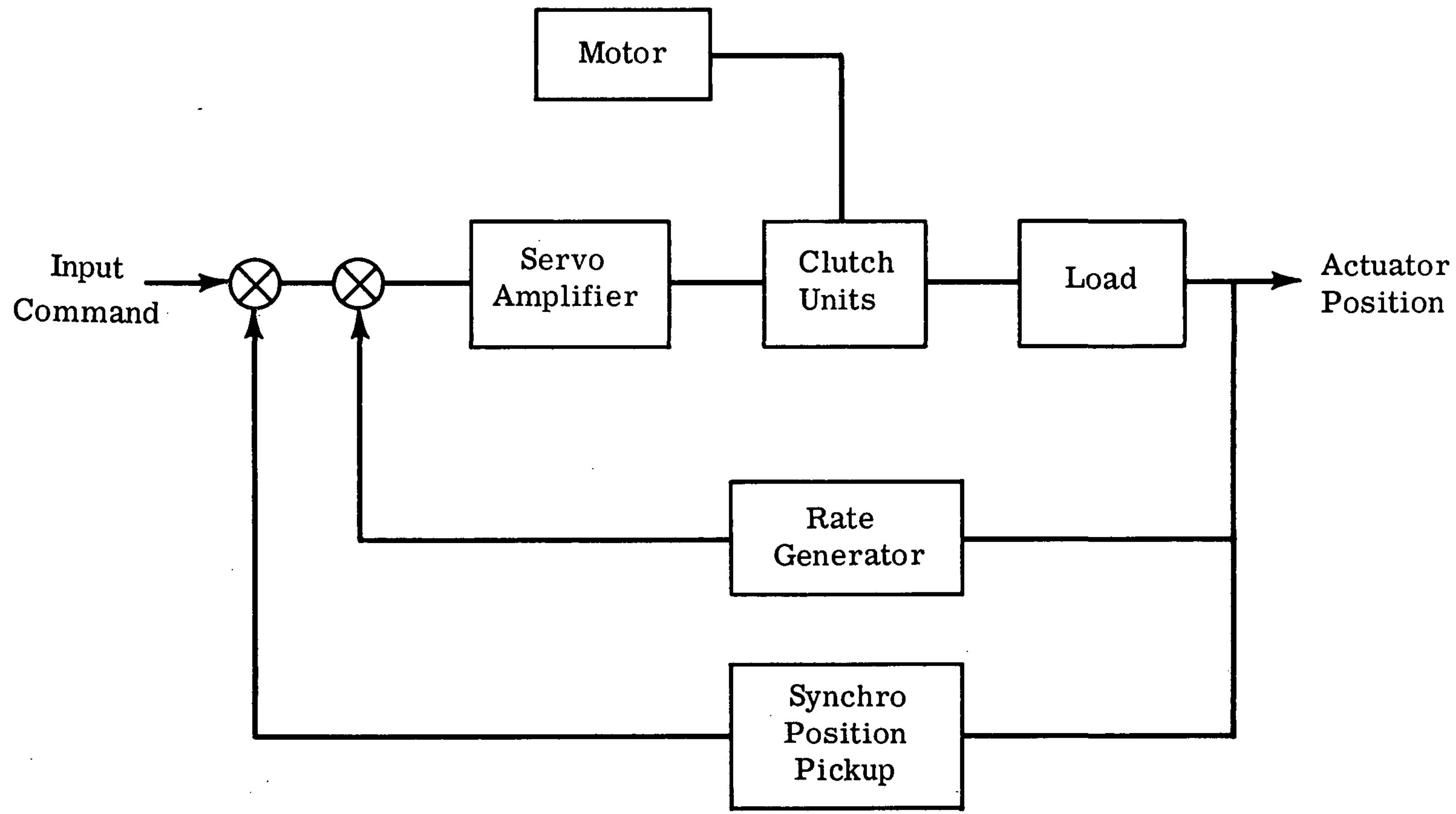


Figure VI-11. Gimbal System Schematic

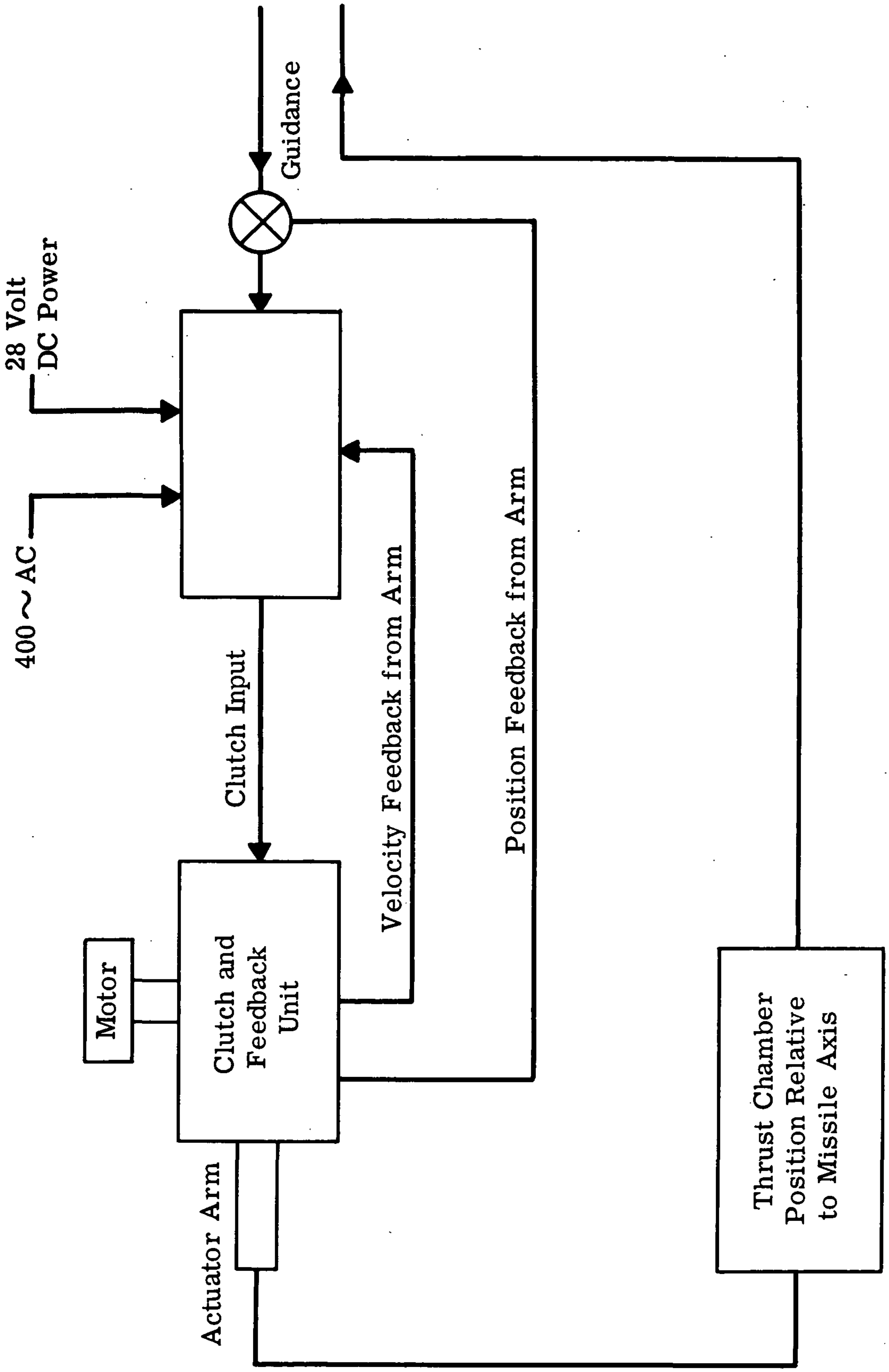


Figure VI-12. Gimbal Electrical System Diagram

SYSTEM DESIGN DATA

Moment of Inertia	70 slug-Ft^2
Maximum Deflection	$\pm 5^\circ$
Maximum Rate	10 deg/sec.
Maximum Acceleration	1 Rad/sec^2
Accuracy	0.1 degree
Maximum Stall Force	365 lb
Actuator Force at 1.7 in/sec	300 lb
Maximum Peak Power (2 Actuators)	350 watts
Average Power (2 Actuators)	70 watts

A more extensive design analysis will be necessary to investigate the static and dynamic requirements for the system. In addition, analog simulation studies should be conducted to determine the effects of actuator force and velocity limiting, clutch hysteresis, gimbal bearing friction, etc. Other parameters which were estimated for preliminary analytical purposes would require verification by actual hardware tests.

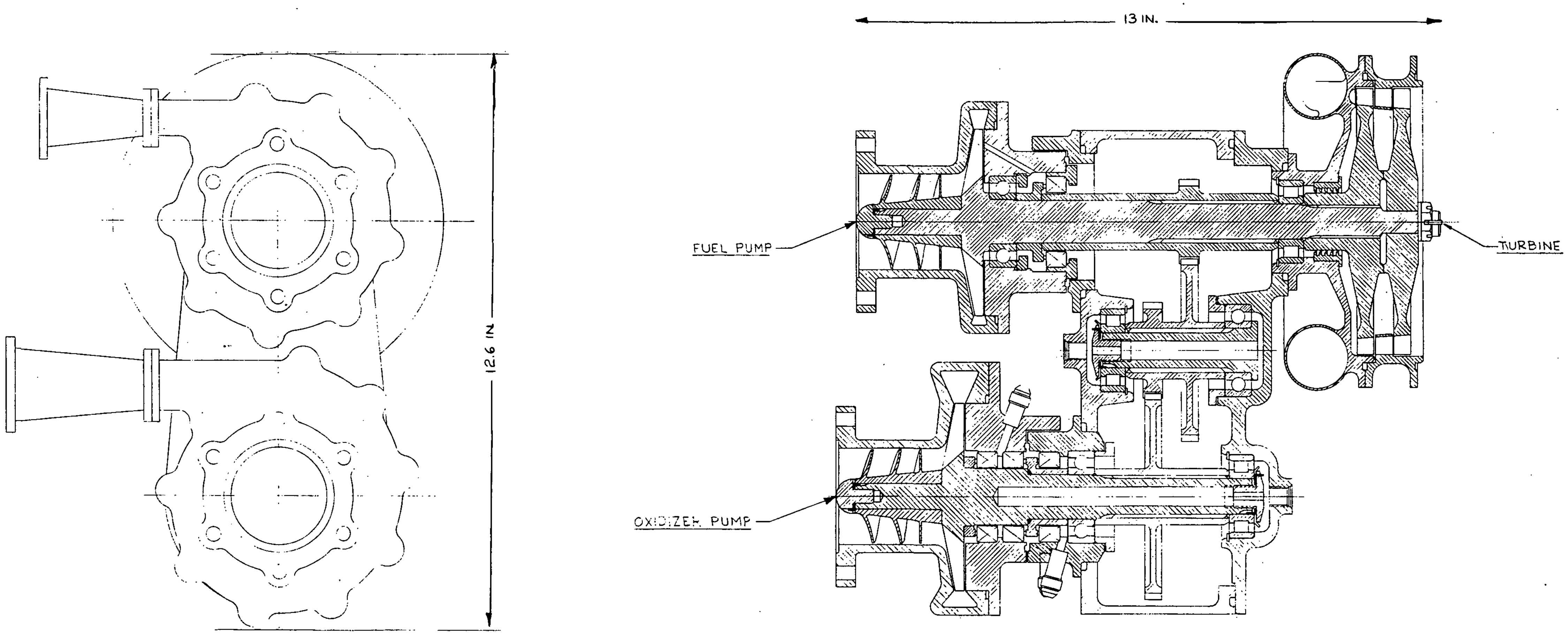
D. TURBOPUMP ASSEMBLIES

The turbopump shown in Figure VI-13 was sized to supply fluorine and hydrogen to the high performance thrust chamber. The design rated performance is tabulated in Table VI-4. The general test experience gained from the NASA program under contract NAS-w-28 as well as the industry "state-of-the-art" have been incorporated into the configuration. The hydro-dynamic characteristics of the BAC family of turbopumps have also been maintained.

The drive assembly consists of two propellant pumps and a turbine connected through a gear box.

1. Pumps

Both pumps are geometrically similar to the Bell Agena engine propellant pumps. The impeller consists of two stages which



12 K FLUORINE-HYDROGEN TURBINE PUMP ASSEMBLY
FIGURE VI-13

Figure VI-13. 12K Fluorine-Hydrogen Turbine Pump Assembly are directly coupled. An axial flow inducer stage feeds the straight radial unshrouded centrifugal stage. Both pumps are overhung from ball bearings. With both open eyed pumps mounted on the same side of the gear box, the best inlet conditions are provided for cavitation resistance.

The estimated cavitation characteristic of the pumps are shown in Figures VI-14 and VI-15 for the fluorine and hydrogen pump respectively. The characteristics are shown in terms of pump inlet head in feet above vapor pressure (Hsv) as it affects the discharge flow to the thrust chamber while holding speed constant. The cavitation performance at other speeds are determined by similarity relationships. Hence, for a constant flow factor, the volumetric flow is proportional to speed when the inlet head is proportional to the square of speed.

The cavitation performance of the fluorine pump was established on the basis of the Bell Agena Engine oxidizer pump. The same criterion applied to the hydrogen pump would lead to a very conservative design because of the strong fluid resistance to cavitation due to its thermodynamic properties. Data from NASA's Lewis Research Center demonstrates that axial flow inducers can operate in hydrogen with cavitation numbers in the order of 0.005 to 0.01. This is approximately five times lower than with normal fluids. The cavitation number is defined as:

$$K = \frac{(H_i - H_{vp})\, 2g}{W_i^2}$$

where: K is the cavitation number
H_i is the static pump inlet head in feet
H_{vp} is the fluid vapor pressure in feet
W_i is the fluid relative velocity at O°D° of the inducer inlet in fps.

The purpose of the axial flow inducer stage then is to permit high speed rotation without an excessive inlet pressure. The inducers have been referred to as cavitating inducers because they are designed to digest a sizable quantity of propellant vapors generated at the inlet and to recompress the vapors prior to discharge into the main stage centrifugal impeller. This is accomplished by providing high blade solidity by a few blades having a long chord and by having greater inlet area than exit area. Entrainment of inert pressurizing gases is a different phenomenon since they do not go into solution. A limited test on the Agena turbopump without inducers indicated the magnitude of the affect

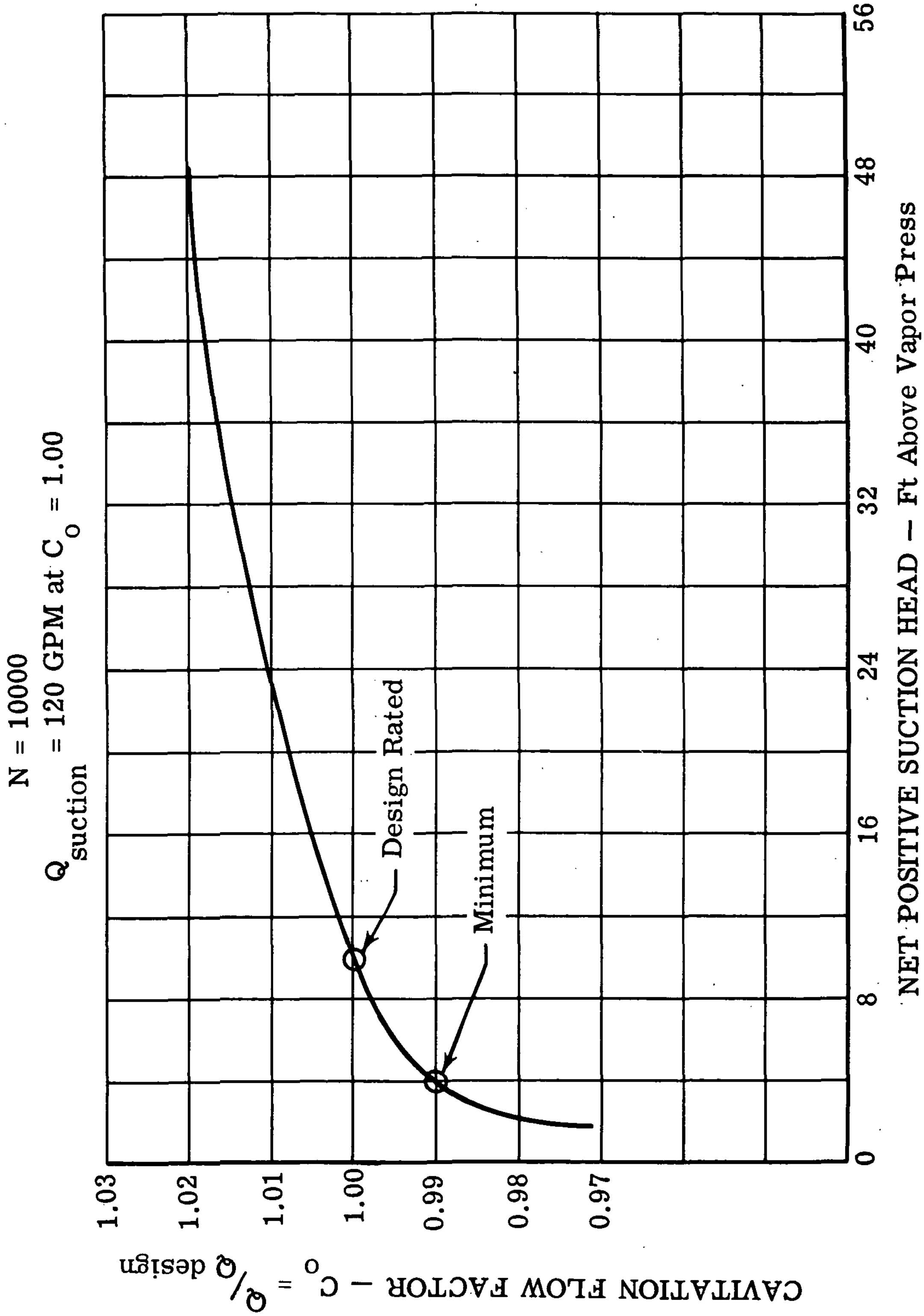


Figure VI-14. Fluorine Pump Cavitation Characteristic at Constant Speed

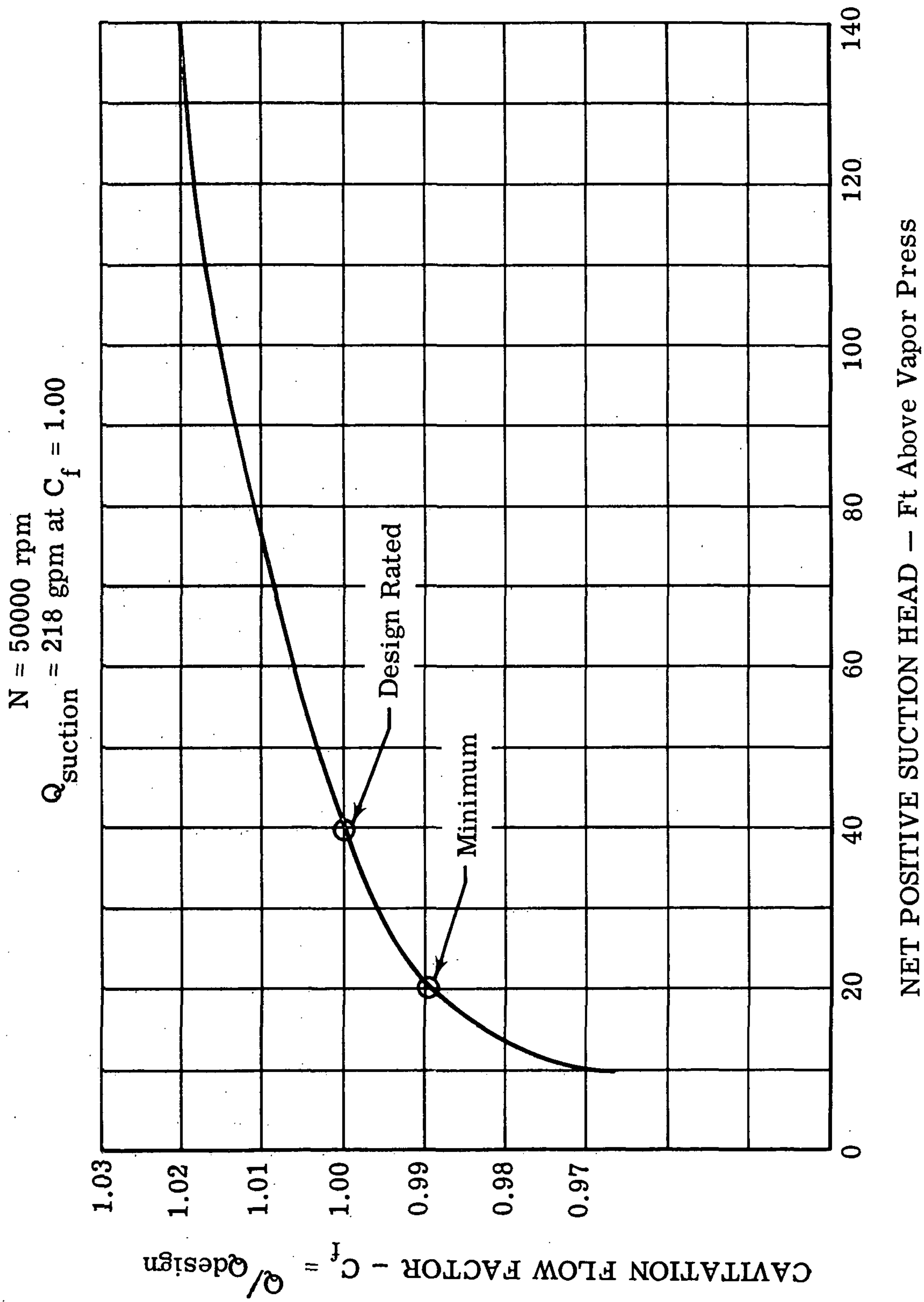


Figure VI-15. Hydrogen Pump Cavitation Characteristic at Constant Speed of helium injection into the pump suction line on performance. Up to 3 per cent helium-to-liquid volume at the fuel pump inlet and up to 6 per cent at the ox pump inlet had insignificant affect on turbopump performance.

The straight radial unshrouded centrifugal impeller produces the major proportion of the head rise. It discharges the flow into a vaneless volute where separate ports are provided to feed the thrust chamber and gas generator. The thrust chamber port is designed to provide a flow rate which is independent of the downstream system pressure requirements. A typical head capacity curve is shown in Figure VI-16. This flow control feature is one of the unique characteristics of the Agena turbopump. The governing equations for the discharge flows can be expressed as follows:

$$W_f = K_f C_f \rho_f N \qquad \text{Equation 1}$$

$$W_o = K_o C_o \rho_o N \qquad \text{Equation 2}$$

Where:
- W is the weight flow in lb/sec
- C is the cavitation flow factor
- ρ is the propellant specific gravity
- N is the turbine speed in RPM
- K is the constant of proportionality
- f is subscript for fuel
- o is subscript for oxidizer

From the above two equations, an expression for mixture ratio can be obtained which is independent of speed.

$$R = {}^{W_o}/_{W_f} = \frac{K_o C_o \rho_o}{K_f C_f \rho_f} \qquad \text{Equation 3}$$

To obtain an estimate of the effect of pump inlet temperature and pressure on engine mixture ratio, an analysis was made based on constant speed and the cavitation characteristics. The results of the analysis are plotted in Figure VI-17. To obtain the percentage change in engine mixture ratio, the changes in the individual pump flows are combined algebraically as follows:

$$\Delta R = \Delta W_o - \Delta W_f \qquad \text{Equation 4}$$

Where:
- ΔR is % change in ratio
- ΔW_o is % change in oxidizer flow
- ΔW_f is % change in fuel flow

	Design	Point
	F_2	H_2
Speed rpm	10,000	50,000
Imp Dia In.	4.26	4.26
Head Rise	616	15,350
Diffuser Flow gpm	120	198

$$\phi = \frac{2\ g\ (\text{Head Rise})}{(\text{Imp Tip Velocity})^2}$$

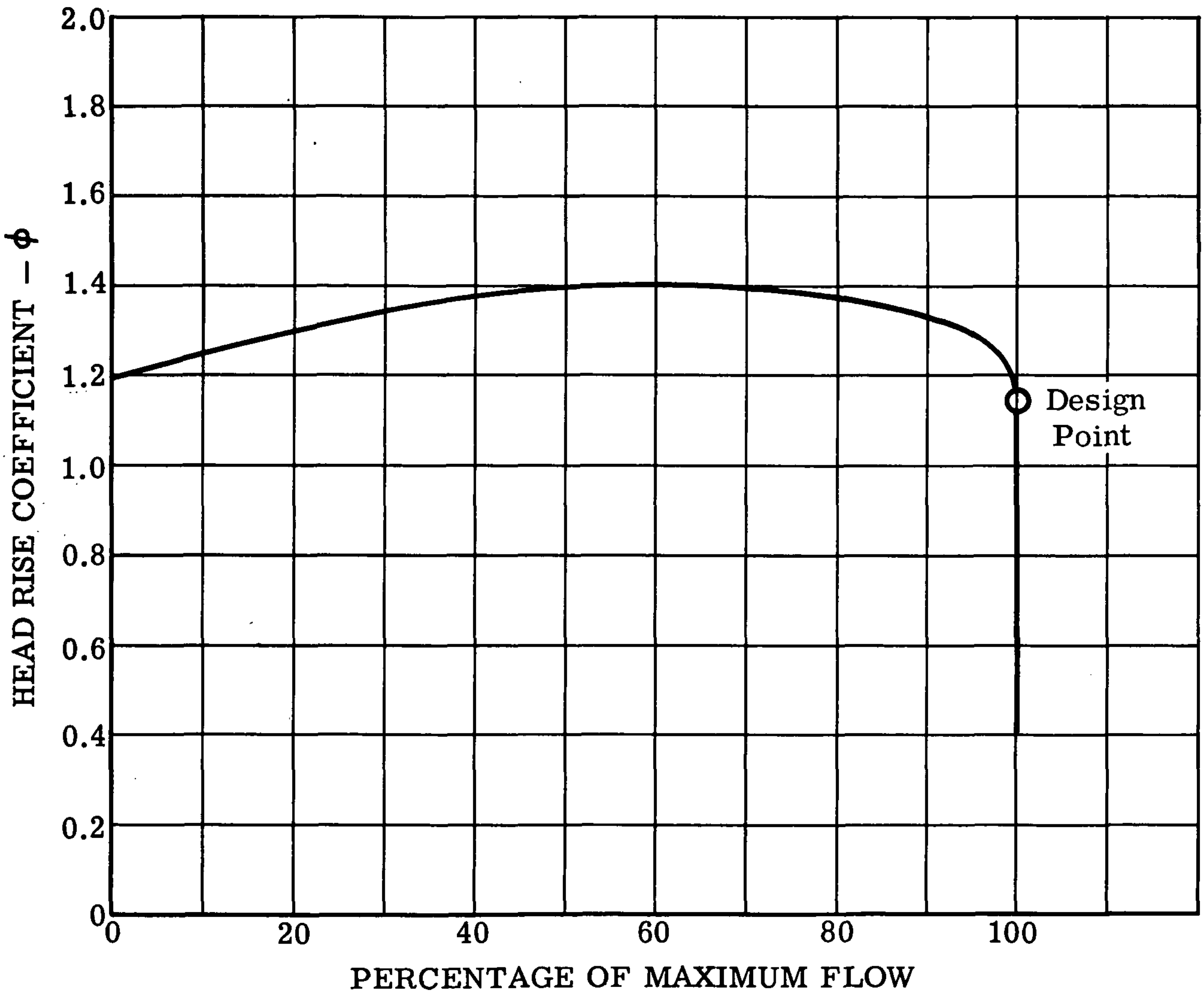


Figure VI-16. Head-Capacity Characteristic of Pump Diffuser for Fluorine and Nitrogen Pump

It is observed from the curves in Figure VI-17 that variations in fluorine pump inlet conditions have the strongest influence on mixture ratio changes.

2. Turbine

The turbine is mounted on the opposite side of the gear box from the pumps. It is a two stage velocity compounded partial admission impulse type with driving fluid obtained from a bi-propellant gas generator. The turbine is mounted on a common shaft with the hydrogen pump and overhangs the turbine bearing. Thus 70% of the horsepower is transmitted directly through the shaft and 30% through the gear train.

Conical nozzles cast into the manifold and equally spaced provide the high velocity gas to the turbine. An exhaust duct transports the gas overboard for exhaust thrust directional control.

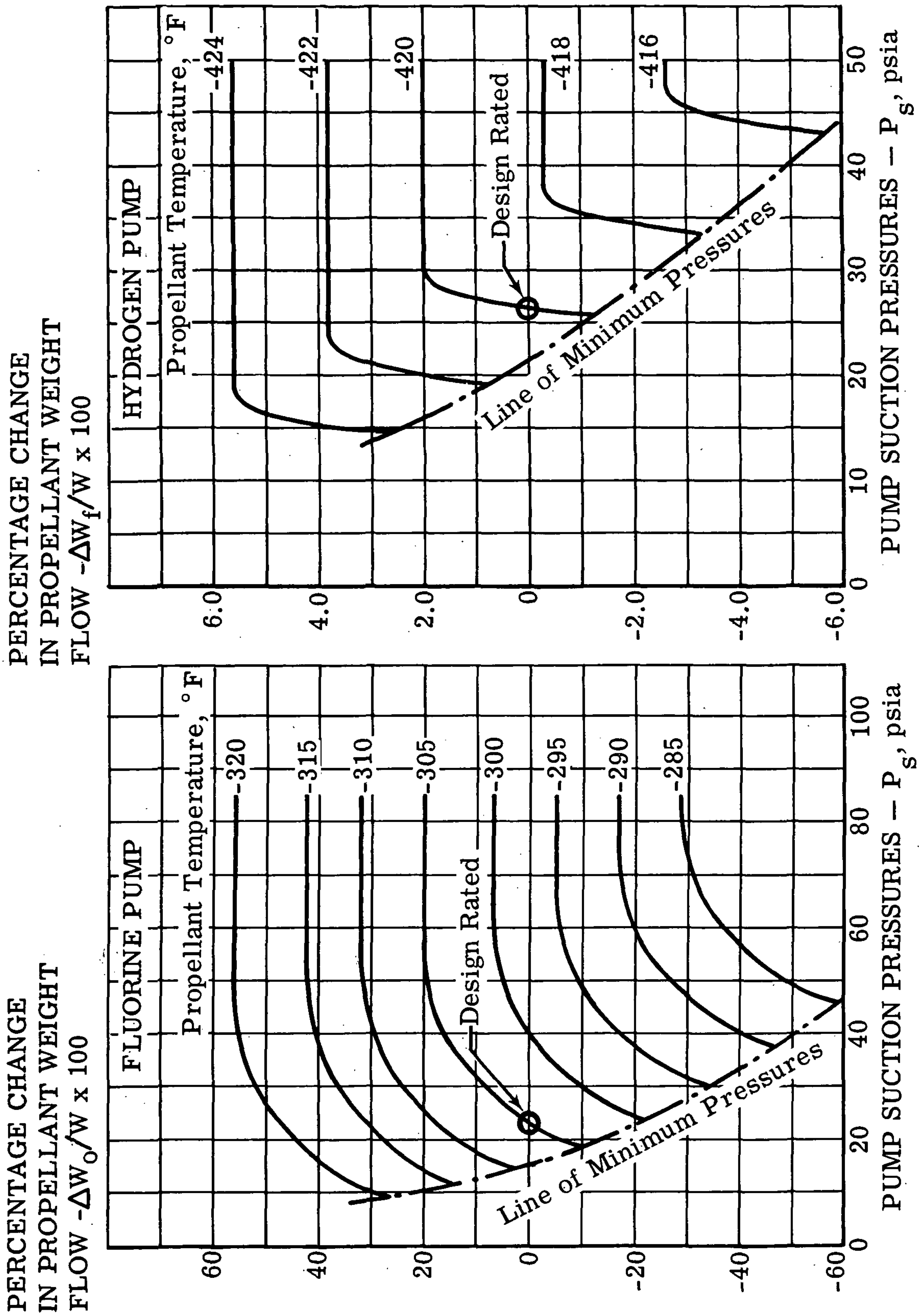
The two-stage turbine is expected to result in a 40% reduction in gas generator propellant flow over a single stage turbine. Justification of the two-stage turbine use is subject to further optimization of turbine weight and gas generator propellant flows.

3. Seals

A very reliable sealing concept for the fluorine pump has been developed at Bell Aerosystems Company under the NASA program. It consisted primarily of face type bellow seals arranged along the shaft to provide a fluorine cavity, an overboard leakage path, and a helium pressurized cavity next to the gear box to prevent the small amount of fluorine leakage from entering the gear box. A similar concept is utilized along the oxidizer shaft of the Agena engine. In both cases the concept has proven very reliable.

With due regard to material compatibility, a similar shaft seal is proposed. It consists of three 50/50 balanced face type bellows seals which allow the seals to be insensitive to the direction of pressure differential.

To insure the high reliability attainable by this seal configuration, the pressure to the helium cavity must be regulated to maintain a positive pressure differential towards the gear box and the fluorine vent cavity. The helium pressure must be maintained only when fluorine is admitted to the pump. When the turbopump is non-operational, the propellant supply to the pumps are closed by valves.



To Obtain the Percentage Change in Mixture Ratio (ΔR) Use the Following:

$$\Delta R = \Delta W_o - \Delta W_f$$

Figure VI-17. Effect of Inlet Pressures and Temperatures on Pump Weight Flow at Design Rated Speed

The hydrogen pump shaft seal has also been evaluated successfully during the NASA program under similar operating conditions as required here. One bellows seal similar to that used in the fluorine pump is required. The hydrogen seal leakage is allowed to pass directly into the gear box.

The turbine shaft seal is a simple labyrinth, through which all hydrogen gas used for cooling bearings and gears passes into the turbine exhaust duct.

4. <u>Gears, Bearings, and Lubrication</u>

Due to the great difference in head requirements between the two pumps, it was necessary to provide speed reduction through a gear box. This added the requirement of four gears, an idler shaft, and three additional bearings over a strictly "in-line" arrangement. Each of the three shafts are mounted in one antifriction ball bearing and one antifriction roller bearing. Due to the fluid temperature of the propellants, conventional lubricants will freeze without auxiliary means of temperature conditioning. A "state-of-the-art" development using cold hydrogen gas for cooling bearings and gears is well advanced. It is currently being used in the Centaur upper stage rocket engine and in the NAA Mark IX hydrogen pump.

In conjunction with hydrogen cooling, the heat generation is substantially reduced by the application of a solid film lubricant containing primarily molybdenum disulfide. Experimental data shows the coefficient of friction to be around .05 to .08 when dry film lubricant is added to uncoated parts in sliding contact. Under Air Force Contract No. AF33(616)-6960, the hydrogen cooling requirements for coated bearings and gears are being evaluated by NAA. A good degree of success is reported. As a result, hydrogen cooled dry film coated bearings were selected for this application. Due to the unshrouded pump impellers and the impulse turbine wheel, the axial thrust loads are negligible.

The high speed turbine and hydrogen pump bearings have DN values of 1.2×10^6 which is within the range of the investigation in the above mentioned contract. Monel cages piloted on the outside diameter are required for these bearings. Similar bearings have been found quite satisfactory at this DN values with oil lubrication during the NASA program. No cage rubbing of the outside race was noted.

The hydrogen pump bearing is located within the pump cavity. This has worked successfully in the Mark IX pump. Bell has also had considerable success with submerged bearings in N_2O_4. Currently consideration is being given to locating the fluorine pump bearing within the pump cavity. This arrangement has the advantage of decreasing the pump overhang, and providing self-cooling.

The spur type gearing is "run-in" on bench test in oil to remove roughness and establish a wear pattern. After this operation, the gears are cleaned, and coated with the dry film lubricant.

The precise amount of hydrogen gas for cooling bearings and gears is not known. An estimate of 0.02 lb/sec is made. The flow is metered from a port in the hydrogen pump housing by a cavitating venturi. The flow characteristic of the venturi is similar to the main pump diffuser. The flow is distributed to each bearing and the gear mesh points by a manifold. The coolant flow is collected in the gear box and passed through the turbine bearing into the turbine exhaust duct. A pressure relief valve is provided in the gear box to limit the maximum pressure.

5. <u>Materials of Construction</u>

The material compatibility with liquid fluorine has not been a problem area during the NASA program at BAC. The pump housings and seal and bearing supports machined from 6061-T6 aluminum bar stock has proven very satisfactory. Vacuum cast parts from aluminum should also produce the high purity, sound material required for fluorine service.

Similarly, the hydrogen pump housing, supports, and gear box are cast aluminum. The fluorine pump shaft, impeller, and both inducers are machined from forged aluminum while the hydrogen impeller-shaft is machined from stainless steel.

The turbine manifold and stator housing are cast from an Inconel alloy while the rotors are machined from forged Udimet 700.

The gears are machined from heat treatable steel. The bearings are constructed from 440C steel with Monel cages.

6. <u>Venturi Turbine Control</u>

Control requirements of the rocket engine involve primarily mixture ratio and thrust. The control first considered for this application is one of extreme simplicity which has been used successfully in the Agena engine. From Equations (1) and (2) previously discussed in relation to mixture ratio control, it can be seen that the total flow, and hence thrust, can be related to turbine speed, and pump inlet conditions.

$$W_t = W_f + W_o (K_f \delta_f C_f + K_o \delta_o C_o) N \quad \text{Equation 5}$$

For a given set of pump inlet conditions the thrust can be controlled by controlling speed. Equating turbine horsepower developed to pump horsepower required, a steady-state condition is satisfied.

$$\text{Pump Horsepower:} \quad BHP = BHP_o + BHP_f$$

and in terms of the pump characteristic and the above parameters:

$$BHP = K_1 (\rho_o C_o + \rho_f C_f) N^3 \quad \text{Equation 6}$$

7. <u>Turbine Horsepower</u>

$$THP = K_2 W_g C_g^* N \qquad \text{Equation 7}$$

Equations 6 and 7 are combined for steady state:

$$W_g C_g^* = \frac{K_1}{K_2} (\rho_o C_o + \rho_f C_f) N^2 \quad \text{Equation 8}$$

Equation 8 embodies the primary parameters related to speed control. For given pump inlet conditions, control of the gas generator flow (Wg) and the gas generator characteristic velocity (Cg*) will give the desired speed control.

Next, consider the use of cavitating venturis in the gas generator feed lines. Their characteristics are identical to the main pump diffusers, hence, the gas generator ratio and total flow follow the form of Equations 4 and 5. However, by adjusting the throat sizes, the ratio and total flow can be changed to trim turbine speed to any required value. This, coupled with a repeatable combustion process in the gas generator, provide all the necessary elements of control in a very simple, reliable package. More detailed study is required to establish the effect of pump inlet conditions on turbine speed and hence thrust.

Starting of the turbopump is accomplished by supplying propellants through the venturis to the gas generator from pressurized start tanks. The propellants ignite hypergolically and accelerate the turbine to a point where the increasing pump pressure takes over to bring the turbine to rated speed.

TABLE VI-4

TURBOPUMP DESIGN RATED PERFORANCE TABLE

	Fuel Pump	Oxidizer Pump
Fluid	Hydrogen	Fluorine
Head Rise; psi	465	400
ft	15350	616
Flow to Thrust Chamber, lb/sec	1.92	25
GPM	198	120
Speed, RPM	50,000	10,000
Impeller Diameter, in.	4.26	4.26
Inlet Head, psi	1.21	6.5
Above Vapor Pressure, ft.	40	10
Suction Specific Speed	47,000	20,000
Brake Horsepower	113	53.6
Efficiency	52.2	52.5

	Turbine
Speed, Rpm	50,000
Pitch Diameter, in.	5.50
Inlet Pressure, psia	300
Inlet Temperature, °R	1960
Exit Pressure, psia	12
Horsepower	167
Turbine Flow, lb/sec	0.26
Exhaust Thrust, lbs	65

8. F_2-H_2 Gas Generator

The gas generator consists of the combustion chamber, which is an integral part of the turbine manifold, and an injector which is welded into the combustion chamber (Figure VI-18). The generator will operate at a manifold pressure of 300 psia, an overall mixture ratio of 1.4 and a total propellant flow rate of 0.33 lb/sec. An actual characteristic exhaust velocity of 6150 ft/sec will be obtained at this mixture ratio. This represents a combustion efficiency of approximately 85 per cent, which has been obtained consistently with the Agena gas generator. The corresponding exhaust gas temperature will be approximately 1500°F. This temperature is compatible with the high temperature materials used in the gas turbine. Since no regenerative cooling of the chamber is required, the design of the combustion chamber is very simple and the fabrication problems are minimized. A plot of c* and temperature versus mixture ratio in the fuel rich region can be seen in Figure VI-19.

The generator injector configuration consists of a showerhead drilling pattern to provide for uniform entry of both propellants into the combustion chamber. Provisions for a fuel and oxidizer manifold have been made, keeping a minimum of volume prior to propellant entry into the chamber.

The problem areas to be investigated during the development program will be: (a) the effect of mixture ratio on performance, (b) combustion stability, and (c) the effect of chamber pressure on performance.

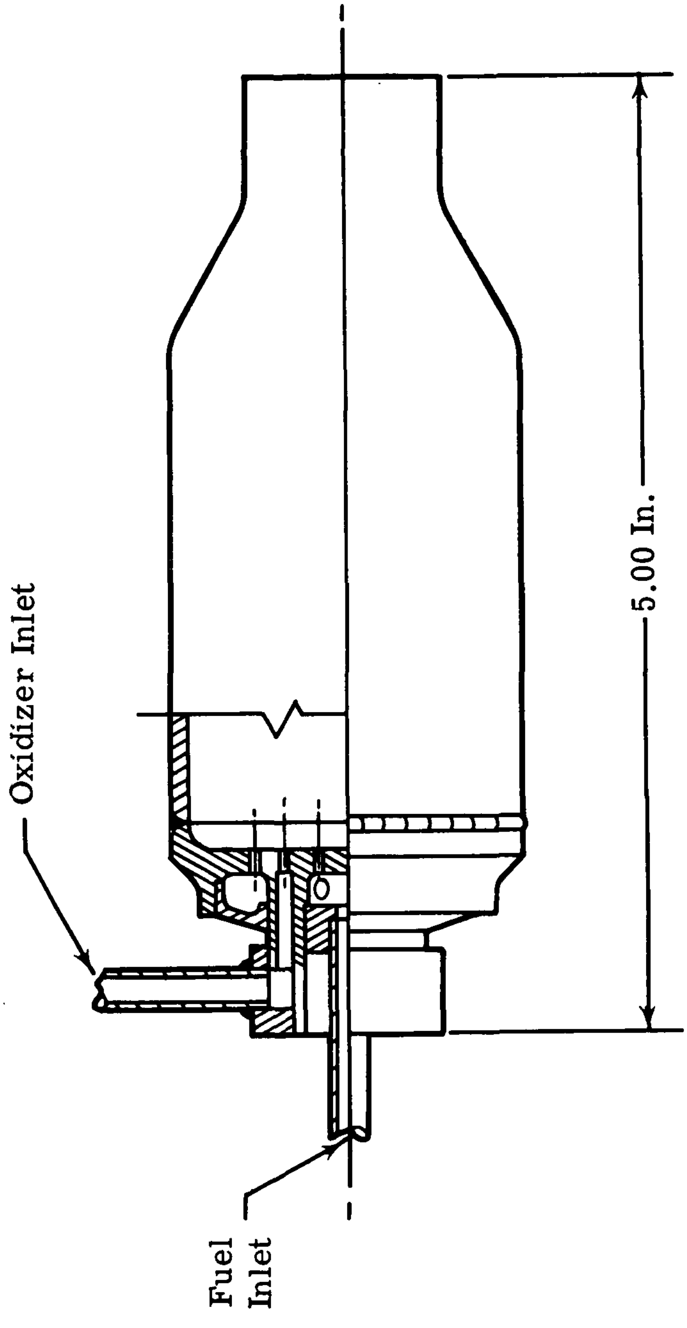


Figure VI-18. Gas Generator 12K — F_2-H_2 Engine

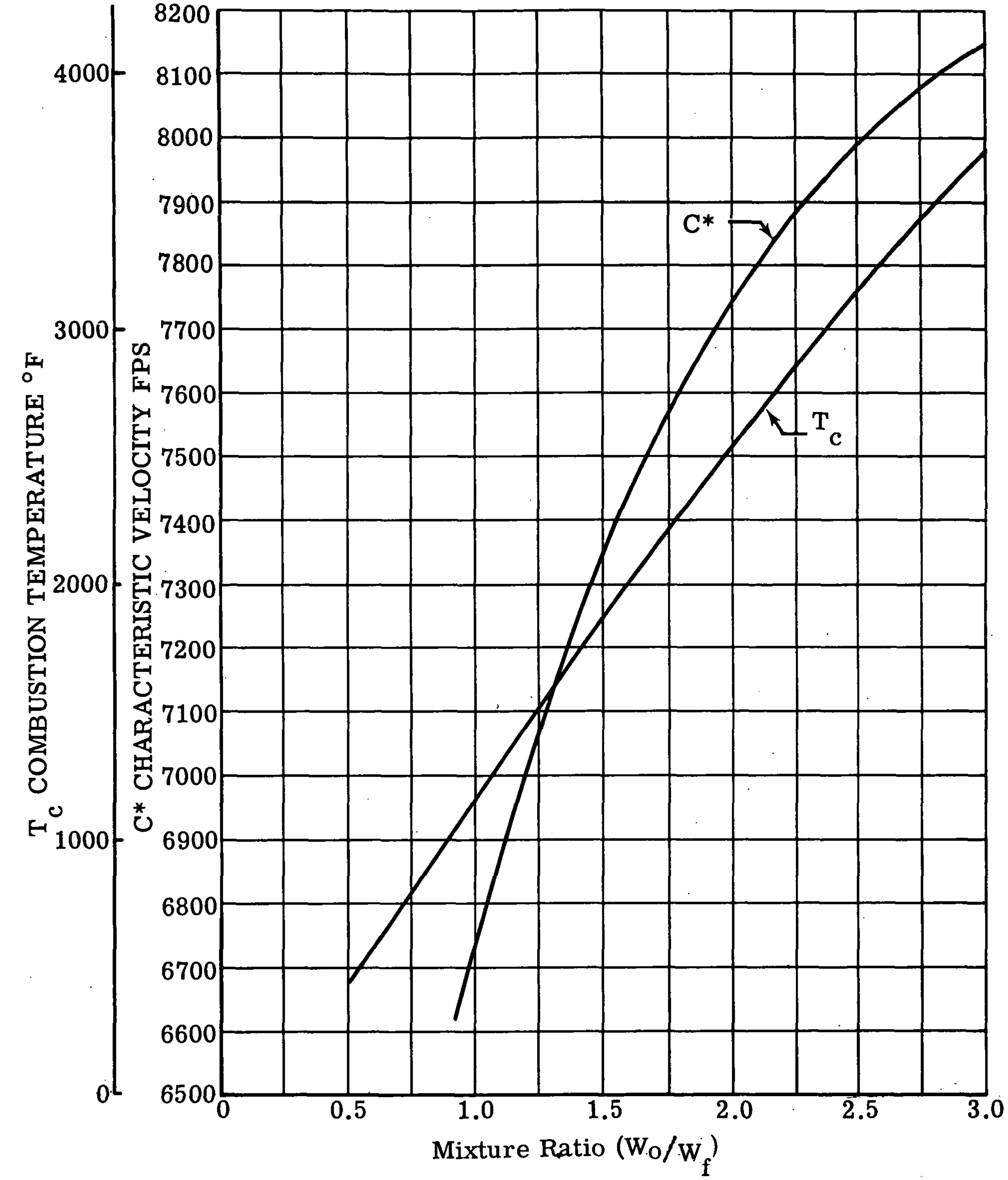


Figure VI-19. F_2-H_2 Theoretical Performance Versus Mixture Ratio (Gas Generator)

E. VALVING

1. Fluorine Valves and Components

Bell Aerosystems Company has a five-year backlog of experience in the actual test stand operation of fluorine oxidizer, high energy rocket engine systems. This experience has led to the use of certain types of metals and limited use of plastics in fluorine environments. The components developed and used include thrust chamber propellant valves, check valves, pressure switches, relief valves, pressure operated shut-off valves, purge valves, cooldown control valves, manual vent valves in addition to the basic thrust chambers and turbine pumping systems. This considerable fluorine experience has produced the following guide lines for the Apollo hardware.

For all parts in the fluorine system, no cast materials will be used. It has been found that it is impossible to guarantee non-porous cast material with absolute certainty of having no inclusions of foreign materials in the base metal. Therefore wrought alloys of aluminum, monel, 304 and 347 stainless steel and copper will be used.

On the thrust chamber propellant valves, the main body will be constructed from 6061-T6 wrought aluminum alloy. The poppet valve which controls the flow of fluorine to the thrust chamber is to be constructed of wrought 304 stainless steel. The shaft of the poppet assembly will be plated with electrolytic nickel and heat treated to a hardness of Rockwell C-56 to C-60. All static seals in this valve will be of 1100-0 pure aluminum and will use a flange type bolted design to achieve a high unit loading. In order to be sure of zero liquid fluorine leakage through the main shut-off seat, the stainless poppet contacts a 6061-T6 seat on a small area which is lapped lightly so as to achieve full contact over the entire 360° of the seat area. The seating force of the stainless poppet is maintained at a high level to utilize the Hertz stresses which give minute local deformations at the seat contact. On any parts where it is necessary to have a fool-proof seal with sliding contact, 347 stainless steel bellows are used.

The pressure switches that will be used for control of the engine components and their proper sequencing may see either liquid or gaseous fluorine at various times. Their construction will feature 347 stainless steel bellows and inlet fittings for those parts exposed to the propellant. Other parts will not directly contact the fluorine, and are 356-T6 aluminum microswitch housing, a 1095 steel spring, 2024 aluminum alloy electrical connector. The electrical wiring will all be covered with a 347 stainless conduit to prevent possible attack by fluorine vapors.

Valves for the test stand operation of the thrust chambers and turbines will be of monel wrought bars, stainless steel bodies with lapped valve seats and poppets. Teflon chevron packing will maintain zero external leakage of gaseous fluorine. This is typical of the type of construction that must be used on the various manual and pneumatic shut-off valves in order to maintain a high degree of reliability in the test-stand operation of fluorine systems.

2. <u>Hydrogen Valves and Components</u>

The use of both liquid and gaseous hydrogen on various Bell Aerosystems programs has resulted in the validation of a well documented set of design procedures for this service. The limits of applicability of various metals and plastics have been established.

Generally speaking, the conventional 300 series stainless steels, 2024-T6, 6061-T6 and 356-T4 aluminum alloys are usable with good results for the required thrust chamber fuel valves, check valves, pressure switches, pressure operated shutoff valves, purge valves, check valves, and manual vent valves.

Where plastic seals are to be used, mylar has been found to provide the best sealing at -425°F, although valve seats of Teflon also give satisfactory tight shutoff service in both liquid and gaseous hydrogen. Where sliding shaft seals will be used, mylar seals will be forced against 304 stainless steel shafting with a chrome plated, ground and polished surface. Static seals will be metal-to-metal types, using 1100-0 pure aluminum or spring loaded monel "K" type seals with either bolted flange loading or screw thread loading.

The pressure switches to be used for hydrogen are the same as those for fluorine service.

The various test-stand valving and components are similar to those to be used in fluorine except that a somewhat wider choice of materials is allowable. For example, 316 stainless steel castings and valve seats are highly satisfactory, and Kel-F seats can also be used.

Figure VI-20 shows a typical hydrogen fluorine propellant valve assembly.

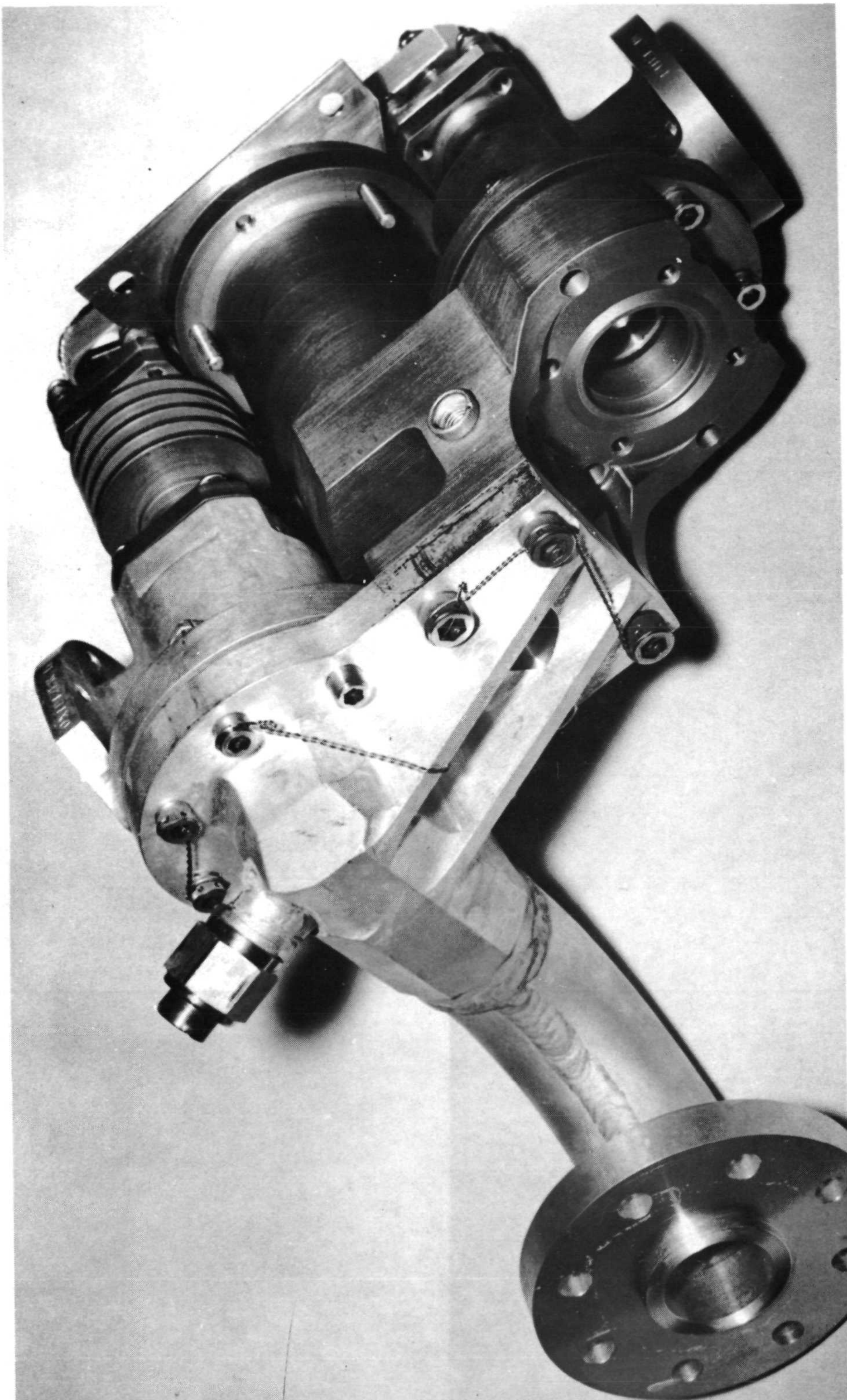


Figure VI-20. Fluorine-Hydrogen Propellant Valve Assembly

F. ENGINE

1. In-Board Profile

A preliminary in-board profile of a 12/4 K thrust engine is presented in Figure VI-21. The various components described in the preceding sections are included in the detail that they were defined. Overall dimensions are given. The turbine-pump assembly is mounted on the thrust chamber from supports about the throat section. The gas generator is mounted on the turbine inlet manifold. The gas generator exhaust duct is attached to the chamber at the chamber exit plane. The propellant inlet valves for pressure-fed and pump-fed operation, the gas generator propellant valves, and the gimbal mount and actuators are depicted.

2. Start Transient

The method of starting the engine is one of simplicity with unlimited start capability. After the pump cooldown has been accomplished during ullage rocket operation, the gas generator propellant valve is actuated, and propellants enter the gas generator at vehicle tank pressure and ignite hypergolically. Sufficient impulse is obtained to spin the turbine. The increasing speed further increases the gas generator flow to continue the bootstrap operation. An estimate of the start transient is plotted in Figure VI-22, for pressure-fed and both low pressure and high pressure bootstrap pump-fed operation. A typical Agena engine start transient is also included for comparison purposes.

3. Intermediate Thrust Levels

The defined main propulsion system provides 12,000 or 24,000 pounds of thrust to accomplish lunar orbit and disorbit. Pressure-fed firings at 4,000 pounds of thrust could be made after the pump-fed firings for fine velocity correction. If thrust levels between 4,000 and 12,000 pounds were desired for the large total impulse lunar orbit firings, minor modifications to the pump-fed side of the system would be required. Because the propellant flow to the turbine-pump gas generator determines the thrust level, provisions to change that flow would give thrust levels in the 4 to 12 K range. One method to provide intermediate thrust levels would consist of a second cavitating venturi and isolation valving in the fluorine feed line to one or both gas generators. The second venturi would be designed to restrict the F_2 flow during steady-state operation relative to that for 12,000 pounds of thrust reducing the $^{O}/F$ ratio of the gas generator combustion and consequently reducing the power available to drive the turbine.

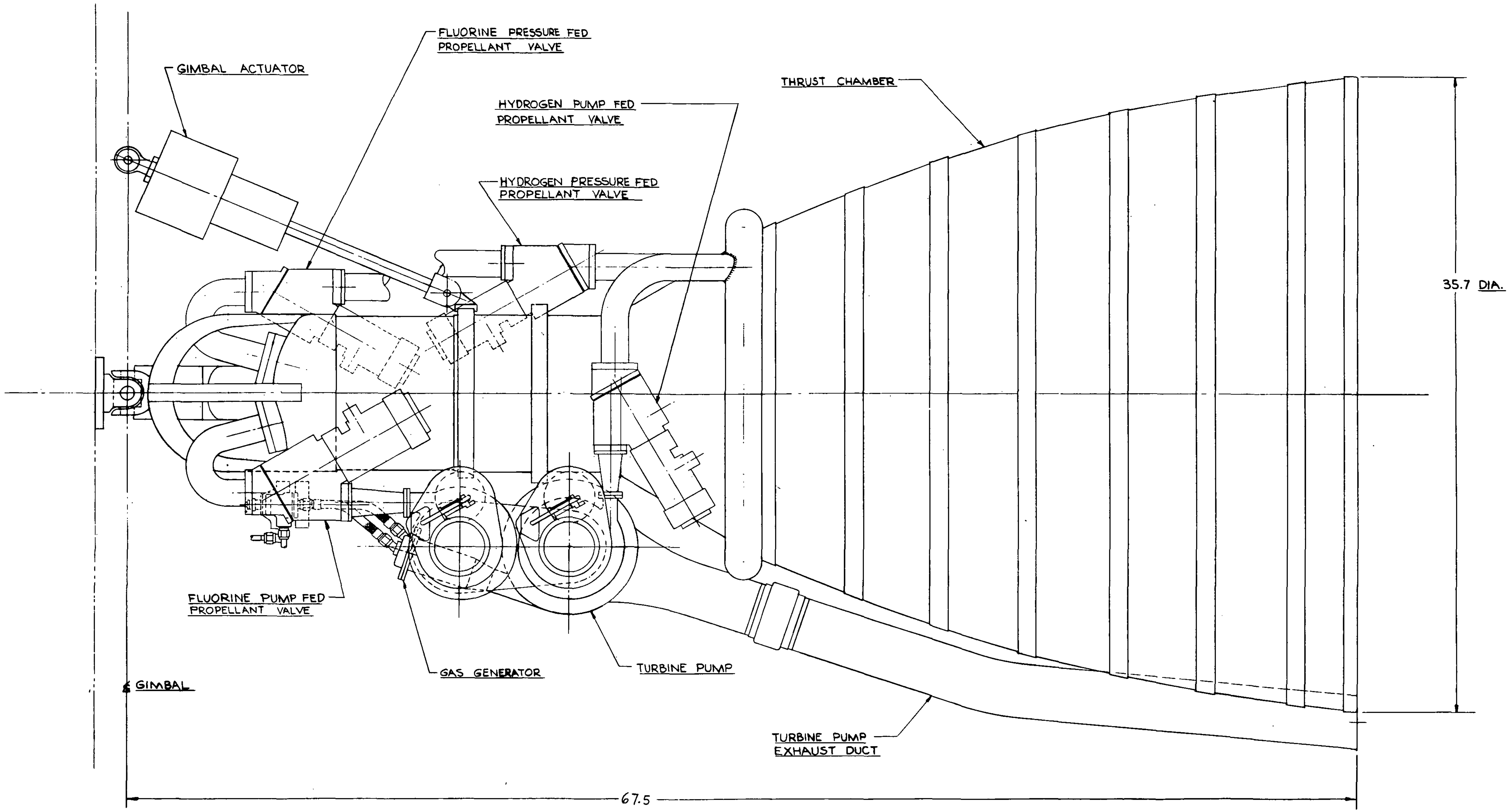


Figure VI-21. Major Components Installation – Rocket Engine Assembly (Sheet 1 of 2)

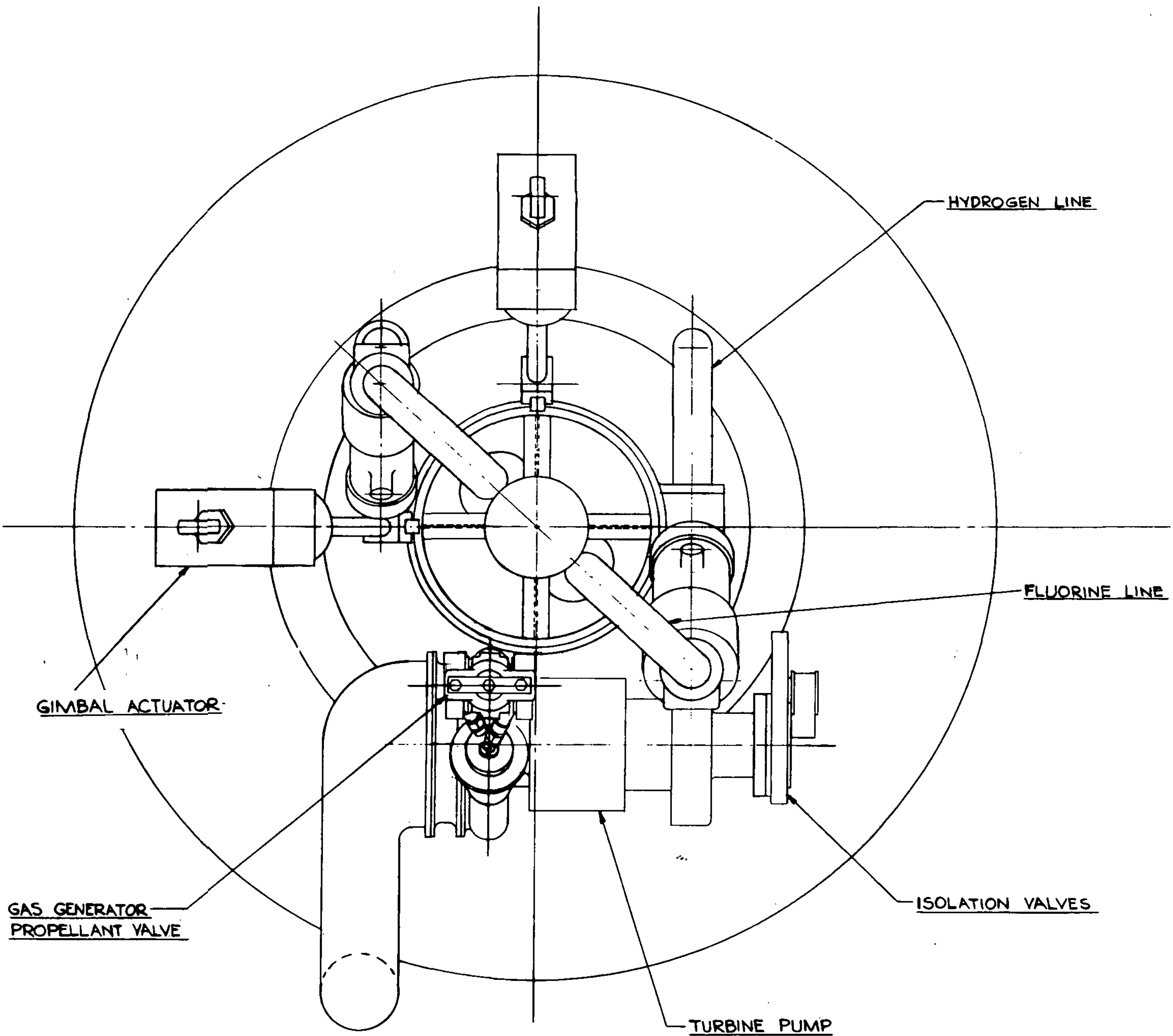


Figure VI-21. Major Components Installation – Rocket Engine Assembly (Sheet 2 of 2)

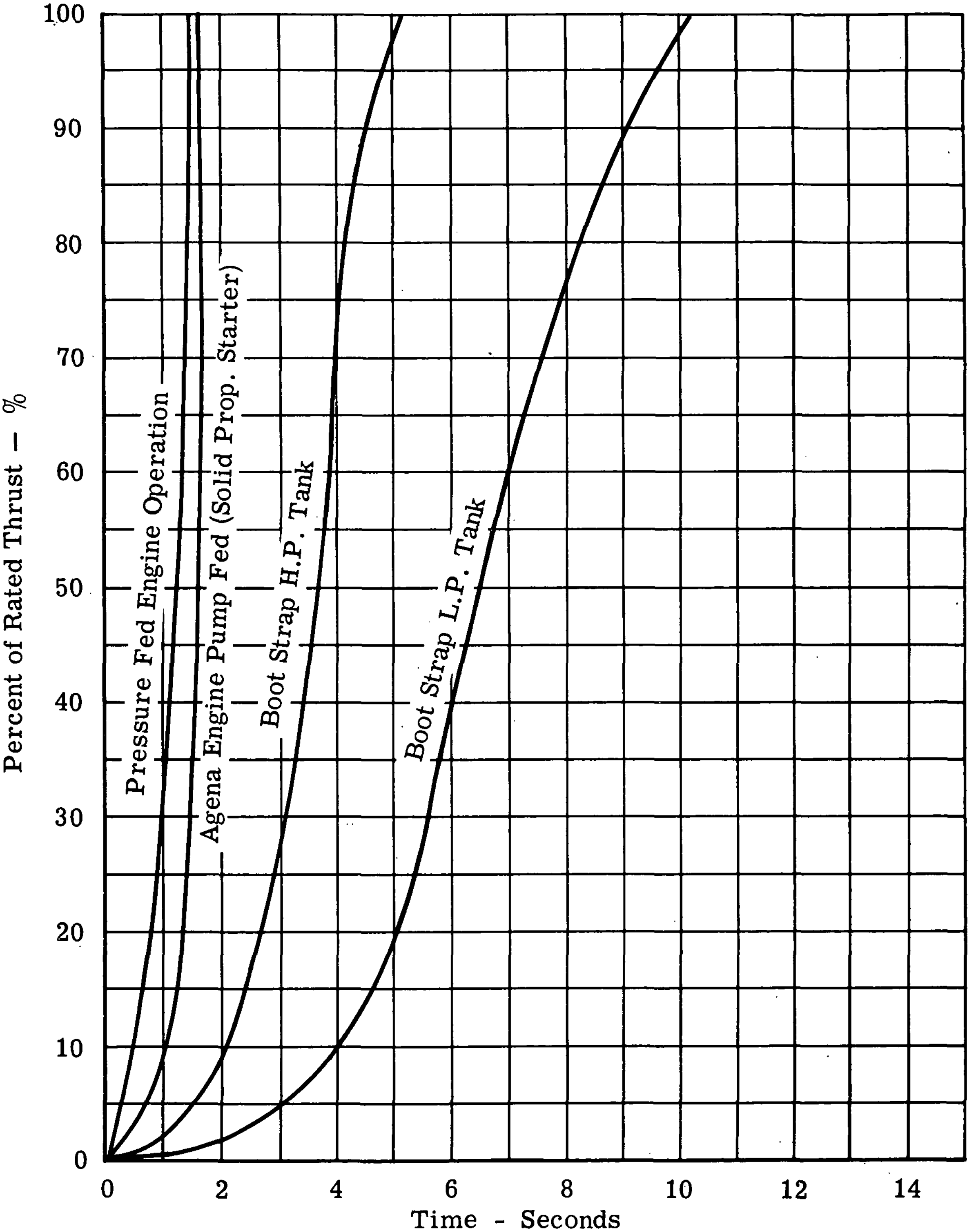


Figure VI-22. Start Transients Pump and Pressure Fed 12/4K Operation (Estimated)

4. Additional Thrust Capability

A preliminary analysis of the thrust chamber operation at very low chamber pressures (10 to 20 psia) indicates that the temperature rise of the hydrogen in the coolant jacket approaches 750°F at a mixture ratio of 10:1. Operation of F_2-H_2 thrust chambers with a H_2 coolant rise of 800°F has been accomplished at Lewis Flight Research Center

The capability of low chamber pressure operation is particularly important for the lunar disorbit. If there were a failure of both turbine pump assemblies, the propellant in the low pressure tanks can be utilized to accomplish lunar disorbit. The helium pressure already in the low pressure tanks is adequate to provide a chamber pressure initially at approximately 20 psia and dropping to 10 psia at propellant depletion.

If one turbine-pump assembly fails, the predicted probability of failure of the remaining turbine pump assembly during the two firings for lunar disorbit is approximately one in 2500 missions. If that probability is considered excessive, additions to the main propulsion feed systems are required to facilitate operation at 20 to 10 psia chamber pressure. Additions to the main propulsion system may ge necessary to insure a feed circuit for the low pressure propellants other than through the pumps. For example, isolation of the tanks to pump feed lines and cross-connection of those feed lines to the normal pressure-fed chamber lines may be then desirable. An additional set of cavitating venturies may be required for flow control in the added feed circuits. Modifications of the F_2 feed to the chamber injector would be required in order to utilize a small number of centrally located F_2 injection holes to maintain adequate F_2 injector pressure drop.

G. SYSTEMS CONTROLS

1. Automatic Safety Controls and System Monitoring

The basic automatic safety circuit used in conjunction with the main propulsion rocket engine electrical systems consists of a turbopump overspeed switch, a safety timer, gas generator chamber pressure and temperature switch and a safety relay. This circuitry is designed to provide automatic control for those conditions where pilot reaction time would prevent appropriate action. During the normal start sequence, the 28 V D.C. starting signal is applied to the automatic safety timer and relay. Unless this power is cut off by the action of the thrust chamber pressure switch within some pre-assigned period of time, the safety relay is closed by the timer, thus energizing a shutdown relay which is a part of the standard engine control system. The relay initiates the normal shutdown process. The overspeed switches are used to prevent an overspeeding turbine wheel. These switches sense both the oxidizer and fuel pump case pressures and when either one goes above a preset value, it closes and energizes the engine shutdown relay. The third possible initiation of automatic shutdown of the engine is based on a decay in thrust chamber pressure. Whether this drop in pressure may be due to propellant exhaustion, a feed system malfunction or even thrust chamber failure, the result is the energizing of the shutdown relay followed by a normal shutdown sequence. The fourth, and normal, shutdown procedure would result from an engine shutdown signal either from an integrating accelerometer in the vehicle control system or from a command given by a crew member. The engine control processes outlined above are only those which are connected with engine oriented shutdown problems and are normally included in the ground testing an unmanned stage engine fed from a single propellant supply system.

In the Apollo vehicle, a multiple feed system is used and thus there is the choice of using normal or emergency propellant feed circuits. Such choices as which gas generator and turbo-pump are to be used for a given firing, and which thrust chamber is to be used, are not normally involved in the operation of rocket engine powered vehicles. However, the Apollo crew can add greatly to the probability of completion of any given mission by making such choices. It is with this intent, enhancement of an inherently high system reliability through command decision, that the system monitoring provisions discussed below are proposed.

In order to maintain cognizance of the condition and operational readiness of the redundant Apollo propulsion system components, two essentials are required. First, as compared to a typical unmanned stage propulsion system, the Apollo will be much more completely instrumented. Secondly, in addition to monitoring the engine performance during firing the crew should be able to review records of the performance of all the components used in each firing. Many of the transient phenomena (starts and shutdowns, typically) in rocket engines occur in such a short time interval that they pass unseen in visual observation and can be inspected only by study of oscillograph records. The considerable time period which may elapse between certain firings of the Apollo engines makes it quite feasible to use this data for a detailed and complete on-board review of data, and analysis of component performances for the purpose of planning of the mode of propulsion system operation best suited to the next firing. This combination of complete instrumentation, automatic safety circuits and high speed recording of propulsion system performance data will greatly aid the crew in their mission. This combination will preclude certain types of malfunctions, allow evaluation of propulsion operation and enable the establishment of alternate modes of propulsion system operation. Of prime importance is the fact that it will provide the intelligence necessary for the crew to make sound command decisions regarding the remainder of the flight.

Because of the availability of alternate propellant feed systems, additional automatic protective devices will be included. For example, with a single feed system, a malfunction which decreases the fuel flow to the gas generator reduces the ability of the turbine to pump the propellants needed by the engine, so the flow drops off, chamber pressure is reduced and the engine shuts down in the limit. If the feed system is not operational, mission failure has occurred, so that the alternate possibility of a lean mixture ratio in the gas generator which would have overheated and damaged the turbine is of no importance. However, this is not true on the Apollo spacecraft. Because alternative feed systems can be used at will, it becomes profitable to protect the turbo-pump units from such harm in order that they can be used further. In short, a feed system malfunction cannot be allowed to damage an engine turbo-pump. In order to implement this rationale, the signal from an overtemperature pickup in each gas generator will be recorded, transmitted to the instrument panel and on exceeding a pre-set value (indicating an overtemperature), it will energize the engine shutdown relay. This in turn de-energizes the He valve pressurizing the gas generator propellant valve. When this valve closes, the overtemperature combustion ceases, due to propellant shut-off. Since the mixture ratio and the resulting gas generator combustion temperature are established by the pressures at which the propellants are delivered to the injectors, the possibility of turbine damage is reduced even further by limiting the maximum allowable pressure differential of the gas generator F_2 feed pressure over the gas generator H_2 feed pressure. Shutdown is signaled to the gas generator propellant valve if the output from the feed pressure pickups indicate that this differential has been exceeded, whatever the reason.

The gas generator temperature sensing system and the differential pressure limiting devices complement each other as safety devices. In the remote case of a partially plugged fuel injector orifice, the combustion temperature would rise without an oxidizer overpressure occurring, thus the shutdown signal would come only from the temperature sensor. Alternatively, a plugged oxidizer injector orifice or an enlarged fuel injector orifice would result in cold running at an abnormally high fuel rate and a shutdown impulse would come from the pressure differential switch only. In any case which produces a F_2 pressure rise or a H_2 pressure drop so to cause overheating, both systems will give shutdown signals. Thus, depending on the nature of the malfunction, a certain redundancy exists in this safety circuit. At the same localization of potential problem areas.

Figure VI-23 shows schematically the propellant feed system, and the location of all the various sensors which will be required to monitor the propulsion system operation. The sensor numbers in this figure correspond to those listed in the "Engine Performance Inputs", (Table VI-5). The output from all the pick-ups listed therein will be recorded during all firings and at appropriate intervals during coasting. Certain of these outputs will be displayed directly on the flight engineers panel; some will be suitably processed first and then displayed; and some will be recorded or telemetered only. Those engine performance sensor outputs which need be processed before display include only the propellant mass sensor signals. These outputs will be both displayed and recorded simultaneously as well as fed through a time differentiating circuit to produce a total fuel flow rate and a total oxidizer flow rate. These will be both displayed and recorded also.

Although all the data pick-ups listed in Table VI-5 will be necessary for in-flight analysis of the propulsion system, Table VI-6 lists those which will be displayed during firing.

Tables VI-7, VI-8, VI-9, and VI-10 note the type, number, and operating range required of the sensors listed in Table VI-5.

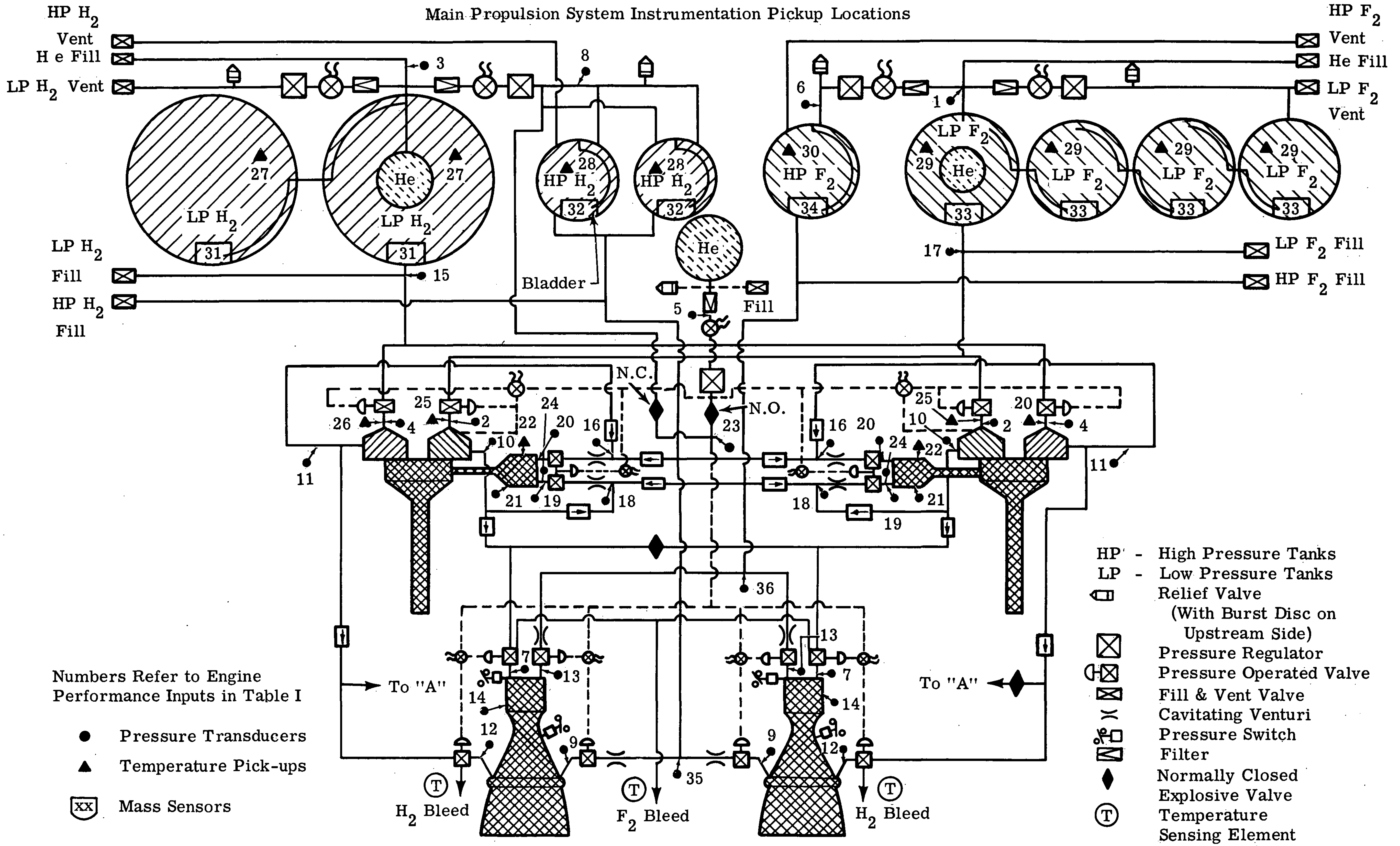


Figure VI-23. Main Propulsion System Instrumentation Pickup Locations

TABLE VI-5

ENGINE PERFORMANCE INPUTS

NUMBER		SOURCE
1.	LP F_2 Tank He Pressure	
2.	LP F_2 Feed Pressure	(No 1 & No 2 Engines)
3.	LP H_2 Tank He Pressure	
4.	LP H_2 Feed Pressure	(No 1 & No 2 Engines)
5.	Control He Tank Pressure	
6.	HP F_2 Tank He Pressure	
7.	T.C.-Pumped F_2 Feed Pressure	(#1 & #2)
8.	HP H_2 He Pressure	
9.	T.C. HP H_2 Feed Pressure	(1 & 2)
10.	Turbo-Pump F_2 Pressure	(1 & 2)
11.	Turbo-Pump H_2 Pressure	(1 & 2)
12.	T.C. Pumped H_2 Feed Pressure	(1 & 2)
13.	T.C. HP F_2 Feed Pressure	(1 & 2)
14.	Pc Main Engines	(1 & 2)
15.	LP H_2 Line Pressure	
16.	G.G. H_2 Line Pressure	(1 & 2)
17.	LP F_2 Line Pressure	
18.	G.G. F_2 Line Pressure	(1 & 2)
19.	Gas Generator F_2 Feed Pressure	(1 & 2)
20.	Gas Generator H_2 Feed Pressure	(1 & 2)
21.	Gas Generator Chamber Pressure	(1 & 2)
22.	Gas Generator Chamber Temperature	(1 & 2)
23.	Control He Pressure	
24.	Propellant Differential Pressure	(1 & 2)
25.	LP F_2 Feed Temperature	(1 & 2)
26.	LP H_2 Feed Temperature	(1 & 2)
27.	LP H_2 Tank Temperature	(1 & 2)
28.	HP H_2 Tank Temperature	(1 & 2)
29.	LP F_2 Tank Temperature	(1, 2, 3 & 4)
30.	HP F_2 Tank Temperature	
31.	LP H_2 Weight	(1 & 2)
32.	HP H_2 Weight	(1 & 2)
33.	LP F_2 Weight	(1, 2, 3 & 4)
34.	HP F_2 Weight	
35.	HP H_2 Line Pressure	
36.	HP F_2 Line Pressure	
37.	Turbo-Pump Speed	(1 & 2)
38.	Power Supply Volts & Amps	

TABLE VI-6

PROPULSION SYSTEM ITEMS MONITORED

		Pick-Up	Indicated	Recorded
1.	Total Fuel Flow	$\frac{d\ 31}{dt}$ or $\frac{d\ 32}{dt}$	PI	x
2.	Total Ox. Flow	$\frac{d\ 33}{dt}$ or $\frac{d\ 34}{dt}$	PI	x
3.	Gas Gen. Fuel Flow	16	-	x
4.	Gas Gen. Ox. Flow	18	-	x
5.	Thrust Chamber Fuel Flow	9 or 12	-	x
6.	Thrust Chamber Ox. Flow	7 or 13	-	x
7.	Turbine Speed	37	I	x
8.	Gas Gen. Fuel Venturi Inlet Pressure	16	I	x
9.	Gas Gen. Ox. Venturi Inlet Pressure	18	I	x
10.	Turb. Man. Press.-i.e. G.G. Pressure	21	-	x
11.	Thrust Chamber Pressure	14	I	x
12.	Ox. Pump Inlet Pressure	2	I	x
13.	Ox. Pump Inlet Temperature	25	I	x
14.	Fuel Pump Inlet Pressure	4	I	x
15.	Fuel Pump Inlet Temperature	26	I	x
16.	Engine Power Supply Voltage	38	I	x
17.	Engine Power Supply Amperage	38	I	x
18.	Turbine Inlet (G.G.) Temperature	22	I	x
19.	Fuel Mass Aboard	31+32	I	x
20.	Oxidizer Mass Aboard	33+34	I	x

I - Sensor Signal Indicated

PI - Sensor Signal Processed Then Indicated

i - Sensor Number in Table VI-5

TABLE VI-7

PRESSURE PICK-UPS REQUIRED

NUMBER	FLUID	PRESSURE RANGE	ENVIRONMENTAL TEMPERATURE
3	He	0-3000	-423°F
3	F_2	0-75	-310°F
4	F_2	0-600	-310°F
3	H_2	0-100	-423°F
1	He	0-200	-310°F
1	He	0-250	-423°F
3	H_2	0-250	-423°F
4	H_2	0-500	-423°F
4	F_2	0-450	-310°F
3	H_2	0-250	-423°F
4	F_2	0-200	-310°F
4	Hot Gas	0-400	Combustion Temp.
4	H_2	0-600	-423°F
1	He	0-600	Ambient

TABLE VI-8

TEMPERATURE PROBES

NUMBER	FLUID	OPERATING TEMPERATURE	ENVIRONMENTAL PRESSURE
4	HF, H_2	1500°F ± 200°F	0-300 PSIA
12	F_2	-275→ -310°F	0-150 PSIA
10	H_2	(-400→ -423°F)	0-200 PSIA

TABLE VI-9

PROPELLANT MASS SENSORS

		RANGE
2	Bogue Elec. Ultrasonic H_2 Wt. Pickups	0-261 lbs.
2	Bogue Elec. Ultrasonic H_2 Wt. Pickups	0-26 lbs.
4	Bogue Elec. Ultrasonic F_2 Wt. Pickups	0-1440 lbs.
1	Bogue Elec. Ultrasonic F_2 Wt. Pickups	0-402 lbs.

All Designed For 0-100 PSIA.

Include Computers & Read-out Indicators.

Include Diff. Circuit For $\frac{d\ F_2}{dt}$ & Indicator.

Include Diff. Circuit For $\frac{d\ H_2}{dt}$ & Indicator.

TABLE VI-10

MISCELLANEOUS ELECTRICAL ELEMENTS

		REQUIRED
1.	Turbine Tach. Gen. and Indicator (0-60,000 RPM)	(2)
2.	Engine Power Supply Ammeter (0-20 Amps)	(2)
3.	Engine Power Supply Voltmeter (0-32V)	(2)
4.	Differentiating Circuit - Fuel Mass To Rate	(2)
5.	Differentiating Circuit - Ox. Mass To Rate	(2)

(Item 4 will have to sum up all the fuel in 4 tanks and differentiate with respect to time)

(Item 5--------Similarly in 5 Oxidizer Tanks)

6. Control Panel Indicators for all "I" & "PI" Items in Table VI-6 Ranges are as indicated in Tables VI-7, VI-8, and VI-9.

7. All required wiring harnesses, amplifiers, switches, controls etc., for items listed in Tables VI-5 & -6 (Fluorine resistant stainless steel covering required on all such items.)

8. 24 channel quick developing oscillograph, complete with mounts, control systems, power supply.

In the Apollo powerplant usage, several different modes of operation are possible. They are as follows:

1. High Pressure Tank Feed - Engine No. 1
2. High Pressure Tank Feed - Engine No. 2
3. Turbo-Pump No. 1 Feed - Engine No. 1
4. Turbo-Pump No. 1 Feed - Engine No. 2
5. Turbo-Pump No. 2 Feed - Engine No. 1
6. Turbo-Pump No. 2 Feed - Engine No. 2
7. Propulsion System Condition Watch

During each of the above modes of operation, certain sensor signals will be monitored and others will not. These are listed and discussed here. Also listed are those which will be recorded for post-firing study and evaluation.

MODE 1 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank He Pressure	
2	3.	LP H_2 Tank He Pressure	
3	5.	Control He Tank Pressure	
4	6.	HP F_2 Tank He Pressure	
5	8.	HP H_2 Tank He Pressure	
6	35.	HP H_2 Line Pressure	
7	36.	HP F_2 Line Pressure	
8	13.	T.C. HP F_2 Feed Pressure	(Engine No. 1)
9	9.	T.C. HP H_2 Feed Pressure	(Engine No. 1)
10	23.	Control He Pressure	
11	15.	LP H_2 Line Pressure	
12	17.	LP F_2 Line Pressure	
13	14..	Pc Main Engine	(Engine No. 1)
14	31.	LP H_2 Weight	(Tanks 1 & 2)
15	32.	HP H_2 Weight	(Tanks 1 & 2)
16	33.	LP F_2 Weight	(Tanks 1, 2, 3 & 4)
17	34.	HP F_2 Weight	

MONITORED ITEMS

1	$\frac{d\ 32}{dt}$	Total Fuel Flow	
2	$\frac{d\ 34}{dt}$	Total Oxidizer Flow	
3	14.	Thrust Chamber Pressure	(Engine No. 1)
4	6.	HP F_2 Tank - He Pressure	
5	8.	HP H_2 Tank - He Pressure	
6	9.	T.C. - HP H_2 Feed Pressure	(Engine No. 1)
7	13.	T.C. - HP F_2 Feed Pressure	(Engine No. 1)

MODE 2 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank He Pressure	
2	3.	LP H_2 Tank He Pressure	
3	5.	Control He Tank - Pressure	
4	6.	HP F_2 Tank He Pressure	
5	8.	HP H_2 Tank He Pressure	
6	35.	HP H_2 Line Pressure	
7	36.	HP F_2 Line Pressure	
8	13.	T.C. HP F_2 Feed Pressure	(Engine No. 2)
9	9.	T.C. HP H_2 Feed Pressure	(Engine No. 2)
10	23.	Control He Pressure	
11	15.	LP H_2 Line Pressure	
12	17.	LP F_2 Line Pressure	
13	14.	Pc - Main Engine	(Engine No. 1)
14	31.	LP H_2 Weight	(Tanks 1 & 2)
15	32.	HP H_2 Weight	(Tanks 1 & 2)
16	33.	LP F_2 Weight	(Tanks 1, 2, 3 & 4)
17	34.	HP F_2 Weight	

MONITORED ITEMS

1	$\frac{d\ 32}{dt}$	Total Fuel Flow	
2	$\frac{d\ 34}{dt}$	Total Oxidizer Flow	
3	14.	Thrust Chamber Pressure	(Engine No. 2)
4	6.	HP F_2 Tank He Pressure	
5	8.	HP H_2 Tank He Pressure	
6	9.	T.C. - HP H_2 Feed Pressure	(Engine No. 2)
7	13.	T.C. - HP F_2 Feed Pressure	(Engine No. 2)

MODE 3 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank He Pressure	
2	2.	LP F_2 Feed Pressure	(Pump No. 1)
3	3.	LP H_2 Tank - He Pressure	
4	4.	LP H_2 Feed Pressure	(Pump No. 1)
5	5.	Control He Tank Pressure	
6	7.	T.C.-Pumped F_2 Feed Pressure	(Engine No. 1)
7	10.	Turbo-Pumped F_2 Pressure	(Pump No. 1)
8	11.	Turbo-Pumped H_2 Pressure	(Pump No. 1)
9	12.	T.C.-Pumped H_2 Feed Pressure	(Engine No. 1)
10	14.	Pc-Main Engine	(Engine No. 1)
11	15.	LP H_2 Line Pressure	
12	16.	Gas Gen.-H_2 Line Pressure	(Pump No. 1)
13	17.	LP F_2 Line Pressure	
14	18.	Gas. Gen.-F_2 Line Pressure	(Pump No. 1)
15	19.	Gas. Gen.-F_2 Feed Pressure	(Pump No. 1)
16	20.	Gas. Gen.-H_2 Feed Pressure	(Pump No. 1)
17	21.	Gas Gen. -Chamber Pressure	(Pump No. 1)
18	22.	Gas. Gen.-Chamber Temperature	(Pump No. 1)
19	23.	Control He Pressure	
20	25.	LP F_2 Feed Temperature	(Pump No. 1)
21	26.	LP H_2 Feed Temperature	(Pump No. 1)
22	31.	LP H_2 Weight	(Tanks 1 & 2)
23	33.	LP F_2 Weight	(Tanks 1, 2, 3 & 4)
24	37.	Turbo-Pump Speed	(Pump No. 1)

MONITORED ITEMS

1	$\frac{d\ 31}{dt}$	Total Fuel Flow	
2	$\frac{d\ 33}{dt}$	Total Oxidizer Flow	
3	14	Pc Main Engine	(Engine No. 1)
4	15	LP H_2 Line Pressure	
5	17	LP F_2 Line Pressure	
6	22	Gas Gen.-Chamber Temp.	(Pump No. 1)
7	37	Turbo-Pump Speed	(Pump No. 1)

MODE 4 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank He Pressure	
2	2.	LP F_2 Feed Pressure	(Pump No. 1)
3	3.	LP H_2 Tank He Pressure	
4	4.	LP H_2 Feed Pressure	(Pump No. 1)
5	5.	Control He Tank Pressure	
6	7.	T.C.-Pumped F_2 Feed Pressure	(Engine No. 2)
7	10.	Turbo-Pumped F_2 Pressure	(Pump No. 1)
8	11.	Turbo-Pumped H_2 Pressure	(Pump No. 1)
9	12.	T.C.-Pumped H_2 Feed Pressure	(Engine 2)
10	14	Pc-Main Engine	(Engine 2)
11	15	LP H_2 Line Pressure	
12	16.	Gas Gen.-H_2 Line Pressure	(Pump 1)
13	17.	LP F_2 Line Pressure	
14	18.	Gas Gen.-F_2 Line Pressure	(Pump 1)
15	19.	Gas Gen.-F_2 Feed Pressure	(Pump 1)
16	20.	Gas Gen.-H_2 Feed Pressure	(Pump 1)
17	21.	Gas Gen.-Chamber Pressure	(Pump 1)
18	22.	Gas Gen.-Chamber Temperature	(Pump 1)
19	23.	Control He Pressure	
20	25.	LP F_2 Feed Temperature	(Pump 1)
21	26.	LP H_2 Feed Temperature	(Pump 1)
22	31.	LP H_2 Weight	(Tanks 1+2)
23	33.	LP H_2 Weight	(Tanks 1+2+3+4)
24	37.	Turbo-Pump Speed	(Pump 1)

MONITORED ITEMS

1	$\frac{d\ 31}{dt}$	Total Fuel Flow	
2	$\frac{d\ 33}{dt}$	Total Oxidizer Flow	
3	14	Pc - Main Engine	(Engine No. 2)
4	15	LP H_2 Line Pressure	
5	17	LP F_2 Line Pressure	
6	22	Gas Gen.-Chamber Temperature	(Pump No. 1)
7	37	Turbo-Pump Speed	(Pump No. 1)

MODE 5 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank - He Pressure	
2	2.	LP F_2 Feed Pressure	(Pump 2)
3	3.	LP H_2 Tank - He Pressure	
4	4.	LP H_2 Feed Pressure	(Pump 2)
5	5.	Control He Tank Pressure	
6	7.	T.C.-Pumped F_2 Feed Pressure	(Engine 1)
7	10.	Turbo-Pumped F_2 Pressure	(Pump 2)
8	11.	Turbo-Pumped H_2 Pressure	(Pump 2)
9	12.	T.C.-Pumped H_2 Feed Pressure	(Engine 1)
10	14.	Pc -Main Engine	(Engine 1)
11	15.	LP H_2 Line Pressure	
12	16.	Gas Gen.-H_2 Line Pressure	(Pump 2)
13	17.	LP F_2 Line Pressure	
14	18.	Gas Gen.-F_2 Line Pressure	(Pump 2)
15	19.	Gas Gen.-F_2 Feed Pressure	(Pump 2)
16	20.	Gas Gen.-H_2 Feed Pressure	(Pump 2)
17	21.	Gas Gen.-Chamber Pressure	(Pump 2)
18	22.	Gas Gen.-Chamber Temperature	(Pump 2)
19	23.	Control He Pressure	
20	25.	LP F_2 Feed Temperature	(Pump 2)
21	26.	LP H_2 Feed Temperature	(Pump 2)
22	31.	LP H_2 Weight	(Tanks 1+2)
23	33.	LP F_2 Weight	(Tanks 1+2+3+4)
24	37.	Turbo-Pump Speed	(Pump 2)

MONITORED ITEMS

1	$\frac{d\ 31}{dt}$	Total Fuel Flow	
2	$\frac{d\ 33}{dt}$	Total Oxidizer Flow	
3	14	Pc-Main Engine	(Engine 1)
4	15	LP H_2 Line Pressure	
5	17	LP F_2 Line Pressure	
6	22	Gas Gen.-Chamber Temperature	(Pump 2)
7	37	Turbo-Pump Speed	(Pump 2)

MODE 6 ENGINE SURVEILLANCE

Recorded Sensor Signals

ITEM	SENSOR NO.		
1	1.	LP F_2 Tank He Pressure	
2	2.	LP F_2 Feed Pressure	(Pump 2)
3	3.	LP H_2 Tank He Pressure	
4	4.	LP H_2 Feed Pressure	(Pump 2)
5	5.	Control He Tank Pressure	
6	7.	T.C.-Pumped F_2 Feed Pressure	(Engine 2)
7	10.	Turbo-Pumped F_2 Pressure	(Pump 2)
8	11.	Turbo-Pumped H_2 Pressure	(Pump 2)
9	12.	T.C.-Pumped H_2 Feed Pressure	(Engine 2)
10	14.	Pc-Main Engine	(Engine 2)
11	15.	LP H_2 Line Pressure	
12	16.	Gas Gen.-H_2 Line Pressure	(Pump 2)
13	17.	LP F_2 Line Pressure	
14	18.	Gas Gen.-F_2 Line Pressure	(Pump 2)
15	19.	Gas Gen.-F_2 Feed Pressure	(Pump 2)
16	20.	Gas. Gen.-H_2 Feed Pressure	(Pump 2)
17	21.	Gas Gen.-Chamber Pressure	(Pump 2)
18	22.	Gas. Gen.-Chamber Temp.	(Pump 2)
19	23.	Control He Pressure	
20	25.	LP F_2 Feed Temperature	(Pump 2)
21	26.	LP H_2 Feed Temperature	(Pump 2)
22	31.	LP H_2 Weight	(Tanks 1+2)
23	33.	LP F_2 Weight	(Tanks 1+2+3+4)
24	37.	Turbo-Pump Speed	(Pump 2)

MONITORED ITEMS

1	$\frac{d\ 31}{dt}$	Total Fuel Flow	
2	$\frac{d\ 33}{dt}$	Total Oxidizer Flow	
3	14.	Pc-Main Engine	(Engine 2)
4	15.	LP H_2 Line Pressure	
5	17.	LP F_2 Line Pressure	
6	22.	Gas Gen.-Chamber Temperature	(Pump 2)
7	37.	Turbo-Pump Speed	(Pump 2)

MODE 7 ENGINE SURVEILLANCE

Control Panel Monitored

ITEM	SENSOR NO.			
1	1.	LP F_2 Tank He Pressure		
2	3.	LP H_2 Tank He Pressure		
3	5.	Control He Tank Pressure		
4	6.	HP F_2 Tank He Pressure		
5	8.	HP H_2 Tank He Pressure		
6	15.	LP H_2 Line Pressure		
7	17.	LP F_2 Line Pressure		
8	27.	LP H_2 Tank Temperature		(Tanks 1&2)
9	28.	HP H_2 Tank Temperature		(Tanks 1&2)
10	29.	LP F_2 Tank Temperature		(Tanks 1, 2, 3 & 4)
11	30.	HP F_2 Tank Temperature		
12	31.	LP H_2 Weight	"Push-to-Check"	(Tanks 1+2)
13	32.	HP H_2 Weight	"Push-to-Check"	(Tanks 1+2)
14	33.	LP F_2 Weight	"Push-to-Check"	(Tanks 1+2+3+4)
15	34.	HP F_2 Weight	"Push-to-Check"	
16	38.	Power Supply - Volts & Amps-		

2. Propulsion System Power Requirements

The 28 V D.C. power level required for the various modes of propulsion system operation naturally vary from mode to mode.

During Modes 1 and 2, the requirements are identical. The engine activation will require 3.4 amps over the firing period. The associated engine surveillance pickups and indicating meters total less than 0.5 ampere. The propellant mass sensors and computer require 7.35 amps.

For Modes 3, 4, 5 and 6 the engine activation power requirements rise from the 3.4 amps of Modes 1 and 2 to 3.9 amps.

From just before the initiation of firing in any mode, through engine start-up, firing and shutdown, the 24-channel oscillograph will be recording and will require 4 amps of electrical power.

Therefore, the maximum power required during engine firing (Modes 1, 2, 3, 4, 5 or 6) is:

Mass Computer and Pick-Ups	7.35 A
Engine	3.90 A
Pick-Ups and Meters	0.50 A
Recorder	4.00 A
Firing Total	15.75 Amps

Steady-state Mode 7 coasting will demand a constant 0.5 ampere current at 28 V D.C.

Firing each of the explosive valves, if required, will take a 2 ampere pulse of only a few milli-seconds duration.

H. DAMAGE FROM EXTERNAL SOURCES

1. Nuclear Radiation

In this section consideration is given to the effects on materials and propellants of the propulsion system of the Apollo vehicle due to particulate radiation encountered during a typical mission. A typical lunar mission, lasting a total of 12 days, involves 3.5 days to reach lunar orbit; 7 days in lunar orbit; and 3.5 days to return to earth. The materials and propellants to be considered are:

F_2	Kel-F
H_2	Teflon
MON	Mylar
UDMH	Aluminum Alloys Stainless Steels

The integrated dose of radiation which these materials and propellants will receive is dependent upon the following factors:

a. Mission trajectory
b. Amount of shielding provided
c. Solar activity during the mission
d. Number intensity and energy spectrum of the incident radiation

The vehicle trajectory is such that it will be subjected to the radiation from the Van Allen zones and to the solar proton radiation near the geomagnetic poles, or if a flare occurs while in the vicinity of the moon, to a lesser intensity of this type of radiation. Present knowledge of the energy, spatial, and flux spectra of these sources of exo-atmospheric radiation is far from complete.

For most engineering considerations, such as the determination of the radiation life time of a particular material to be used in the vehicle, the most important single parameter is the amount of energy deposited in unit volume of the material, i.e., the dose. One then hopes to establish a correlation between this dose and an effect upon the physical properties of the material.

The flux of particles to which the vehicle will be subjected is important only to the extent that it establishes a limiting environmental condition, i.e., the unshielded rate of energy input to the vehicle skin by radiation. Since the vehicle walls attenuate the electrons and protons in different ways, the radiation intensity inside the satellite will be different due to each component. Wall thicknesses, dictated by present structural considerations, i.e., about 1/2 to 1 gram/cm^2 of low Z (atomic number) material, should be sufficient to attenuate the electron component and its resulting Bremmstahlung radiation by several orders of magnitude. The proton component is far more penetrating. The important feature of the proton flux is that it exponentially approaches a limiting value with an infinitely thick wall. More practically, the weight penalty which accrues by attempting to shield beyond a wall thickness, X, is prohibitive for the small reduction achieved in the proton intensity. For X of the order 1 g/cm^2, the proton intensity beyond X, i.e., to which the interior of the vehicle will be subjected, results in a dose rate between 10^4 and 10^5 roentgens per year for a 2000 nautical mile polar orbit. To specify the resultant dose rate to better than a range covering an order of magnitude for this mission or any other implies a better knowledge of the energy and number spectra than is presently available.

Further, the effect upon a given material under irradiation is dependent upon the nature and energy of the radiation, and very often on the rate at which the radiation deposits energy in the material. Most empirical data of radiation material damage are the result of nuclear reactor irradiations. Using such data to predict the extent of damage in a given material which has been subjected to radiation different from those encountered in a reactor has been successful to a degree. However, this assumes "an equal energy absorbed - equal damage" relationship applies to this material and to the different conditions of irradiation. Such an assumption may legitimately apply to this case; however, it is still an assumption. In the case of plastics and other covalent-bonded material, this is a reasonable assumption to make.

From the above considerations, it appears that the maximum anticipated dose which the Apollo propulsion section will receive is of the order of 10^3 to 10^4 rads. Further, this assumes that at least one solar flare occurred during the course of the mission.

Metals or metal alloys begin to show appreciable changes in physical properties after a dose of 10^{10} to 10^{11} rads. Plastics and elastomers have a threshold dosage considerably less, of the order of 10^4 to 10^6 rads. Teflon, one of the least resistant plastics to radiation damage, does not begin to lose properties until a dose of about 4×10^4 rads. Kel-F and Mylar are more resistant to damage than Teflon.

A current development program at BAC has shown that MON does not decompose even after receiving 2.3×10^7 rads. UDMH does decompose, however. The principal decomposition products are dimethylamine and methylene dimethyl-hydrazine with traces of ammonia, hydrogen, nitrogen and methane. The results of gamma ray exposure on UDMH is reported by Food Machinery Company (Report No. 4-484-R, October 1960). The report shows the percent decomposition of UDMH at 1.4×10^5 reps is less than 1%. (1 rep = 0.93 rads).

Elemental substances, such as H_2 and F_2 whether in liquid or gaseous form, should be quite inert to radiation damage of the amount and intensity to be encountered during the mission considered.

Finally, the maximum reported particle flux in the Van Allen zones corresponds to a surface radiation dose rate of about 10^6 rads/hr. Although an unrealistic case, if the Apollo honeycomb structure were to "see" such a radiation skin dose rate for the full 14 days, the integrated dose would be less than the dose level for gross depreciation of properties for the Al skin. In summary, a very conservative analysis assuming maximum radiation doses indicates no problem areas in propulsion system materials.

2. <u>Micrometeorites</u>

The penetration by micrometeorites of the honeycomb skin structure of the propulsion section of the Apollo vehicle has been considered. GE/MSVD data (Reference 10) indicates that the probability of propulsion system malfunction during a 14 day lunar mission due to micrometeorite penetration is relatively low. However, it should be mentioned that the selection of mission dates should include consideration of minimizing the probability of encountering meteoritic showers.

The honeycomb structure is considered to be sandwiched between two sheets of 0.010 inch aluminum and is from 0.5 inch to 0.8 inch thick overall. A conservative estimate of the number of micrometeoritic penetrations through the honeycomb structure can be made by assuming the structure is represented by a single sheet of 0.020 inch aluminum stock. Figure 2 of the Reference 15 shows that approximately seven (7) holes per square meter of exposed surface will be made through 0.020 inch of aluminum during a typical 14 day mission. The bottoms of some of the craters barely coming through the aluminum sheet and the larger crater producing larger holes. The upper limit of geocentric velocity of 73 Km/sec is assumed for these meteorites.

In order to penetrate to the interior of any of the propellant tanks the micrometeorites must pass through a minimum of one (1) inch of tank insulation as described in earlier sections. This insulation is constructed of 60 layers/inch of Al foil separated by fiberglass threads. As a conservative assumption that the insulation can be represented by 60 layers of 0.001 inch foil; ignoring the stopping power of the fiberglass. For the case of 0.080 inch of Al (the honeycomb sandwich plus tank insulation), Figure 2 (Reference 10) shows that approximately 2.5 holes per square meter of exposed surface will be made to the interior of the tanks per year, or less than 0.1 hole per square meter per mission.

Although particle diameters below one micron do exist in the upper atmosphere, a conservative estimate of the cratered area is based on the assumption of a mean diameter of 10 microns. On that assumption the ratio of cratered to exposed area becomes 5.5×10^{-10} for the case of penetrating only the honeycomb structure; and 8×10^{-12} for penetrating to the interior of a propellant tank.

The assumptions and extrapolations (velocity and size) which were necessarily made in the Reference 10 are such that the calculation of depth of crater is expected to be correct within a factor of 2 or 3 even for an untested material. A further assumption is that the skin of the satellite behaves as a semi-infinite medium; in other words, the craters in a thin skin are the same as those on a thick plate and the skin is penetrated when the depth of the crater exceeds the thickness of the skin.

The multiple tanks for the pump-fed firings and the separate small tanks for the pressure-fed midcourse corrections reduce the probability of tank puncture. Explosive isolation valves between the two low pressure H_2 tanks and between the first 2 and second 2 low pressure fluorine tanks would decrease the probability further. Secondary pressurization lines to the low pressure tanks incorporating normally closed explosive valves would be required to complete the pump feed tank isolation.

The thrust chambers are protected from micrometorite damage by the propellant tanks and the module tank and chamber support structure. The tanks and chambers are further protected by the heat shield covering the bottom of the propulsion module approximately in the plane of the thrust chamber exits. The number and sizes of holes that the thrust chamber can withstand is a finite number but requires further study. The heat shield also gives some measure of protection from shrapnel in the event of catastrophic malfunction of the third booster stage during firing.

VII MISSION ULLAGE AND ATTITUDE CONTROL SYSTEM

A. GENERAL REQUIREMENTS

The integrated, auxiliary propulsion system described in the following is designed to provide attitude control thrust and ullage orientation thrust during the mission phase of the Apollo flight as distinguished from the re-entry phase. This propulsion system must be capable of operating in a space environment, including radiation levels encountered between the earth and moon over a period of 14 days. The system must also exhibit sufficient capability in the combined areas of performance, cycle life, radiation resistance and leakage minimization to assure high reliability for the Apollo mission and adaptability to other space vehicles. Particular consideration must be given to the problem of minimizing the danger of meteorite puncture and its consequences on the safety of the system. For the manned flight application, highest reliability is the prime requirement.

The attitude and ullage control system is to consist of two independent, redundant subsystem sets, each capable of maintaining full control. The subsystem set is to consist of two packages, identical except for pressure control devices. The packages are to contain the fuel and oxidizer propellants, pressurization gas, bipropellant thrust chambers, valves, controls, mounting and subsystem interconnection provisions. The packaging and the assembly of the subsystem components shall provide an efficient, compact, self-contained and functional unit. Provisions for thermal control of the package are to be incorporated. Furthermore, instrumentation is to be provided for pilot indication to monitor the essential functional parameters of the system.

The attitude control thrust chambers are required to operate for time pulses of various frequencies and durations. Additional system description, design and performance parameters are as follows:

Number of Thrust Units (per subsystem)	
50 lb unit (ullage orientation)	2
10 lb unit (pitch)	2
5 lb unit (roll, yaw)	4

Number of Starts Per Unit	
50 lb unit	16
10 lb unit	3000
5 lb unit	3000

Operational Life — 6 months

Total Usable Impulse (in 2 Redundant Subsystems)

For ullage orientation	14,000 lb sec
For attitude control	60,000 lb sec
Total	74,000 lb sec

Maximum System Weight, Loaded (2 Redundant Subsystems) — 507 lb

Temperature Environment

All components except thrust chambers	-8°F to +100°F

Tankage Structural Design Ratios

Propellant tank - Proof/Max. working	1.5
Propellant tank - Ultimate/Max. working	2.0
Gas storage tank - Proof/Max. working	1.5
Gas storage tank - Ultimate/Max. working	2.0

Minimum Deliverable Impulse

50 lb unit	5.0 lb sec
10 lb unit	0.5 lb sec
5 lb sec	0.25 lb sec

Minimum Specific Impulse at +60°F

50 lb unit	300 sec
10 lb unit	268 sec
5 lb unit	260 sec

Pulse Frequency

10 lb unit	3 cps
5 lb unit	3 cps

Max. Electrical Current Drain (at 28V dc)

50 lb unit	1.0 amp
10 lb unit	0.5 amp
5 lb unit	0.5 amp

B. SYSTEM DESCRIPTION AND OPERATION

1. Propellant Selection

From the viewpoint of a minimum weight system and propellant servicing it would be desirable to utilize the main propulsion system propellants for the attitude control system. This concept, however, was excluded for the following reasons:

1. Cryogenic types of propellants are difficult to control at the very small flow rates required on the low thrust levels in the attitude control system. Guarantee of mixture ratio and thrust in short pulses would require elaborate environmental control.

2. Reliability would be low because in case of failure of the main propulsion pressurized section, attitude control would be lost.

3. No low thrust combustion chamber hardware is available at present for liquid F_2 and H_2 and would require new developments. Therefore, medium energy, storable combinations were evaluated to determine their suitability in this application. Among the storable combinations the group of mixed oxides of nitrogen and hydrazine type fuels promise the highest performance. Because the environmental temperature limits in the engine compartment could not be ultimately defined at this time the propellant combination could not yet be optimized in respect to desirable freezing point. Therefore, for this study the following combination was selected.

Oxidizer: Mixed oxides of nitrogen

89.3% N_2O_4+ 10.7% NO

Fuel: Unsymmetrical dimethylhydrazine

The oxidizer has a freezing point of -11°F and the fuel -71°F. The proposed system design provides sufficient insulation for tanks and propellant lines to prevent the oxidizer from freezing under present assumptions of engine compartment temperatures during a mission. However, when need of lowering the freezing point should arise the nitrogen oxide content in the oxidizer may be raised with no sacrifice in performance.

2. Description

Figure VII-1 represents the over-all system arrangement in the vehicle. The system consists of two subsystem sets mounted in planes perpendicular to each other so that the six attitude control thrust chambers in each subsystem set can provide full three-axis control. However, the attitude control system does not provide balanced force couples in pitch and yaw in reference to the vehicle center of gravity because this would greatly complicate the installation of the system. The small translatory accelerations of the vehicle produced by this arrangement can be considered to be averaged in various attitude control cycles along a mean trajectory.

Each subsystem set also contains two main propulsion system propellant settling or ullage thrust chambers operated in pairs. These thrust chambers could also provide high, emergency, pitch and yaw control when operated individually, however, such use is not considered in the present system with regard to the complications and additions involved in the guidance and electrical system. In like manner, a redefinition of the mission phases could include their use for very small velocity corrections.

Figure VII-2 shows the propulsion section schematic. The packages in each subsystem set contain their own fuel, oxidizer and pressurization gas tanks. This multiplicity minimizes the effects of meteorite puncture and bladder failure, and since spherical tanks are used throughout, little weight penalty results. Pressurization gas flow control is built into one pack of each set only to guarantee equal tank pressures and consequently equal thrust output within each subsystem set.

The total amount of propellants carried in the two subsystem sets is not redundant for the full mission. Therefore, cross-coupling provisions are made between all tanks to utilize all of the propellants in either subsystem set.

Figure VII-3 presents the assembly and installation details of the package.

The package assembly is constructed of six major subassemblies to minimize joints and seals and to provide for ease of assembly and replacement. These subassemblies are:

1. The oxidizer tank, including expulsion diaphragm and associated plumbing and components.

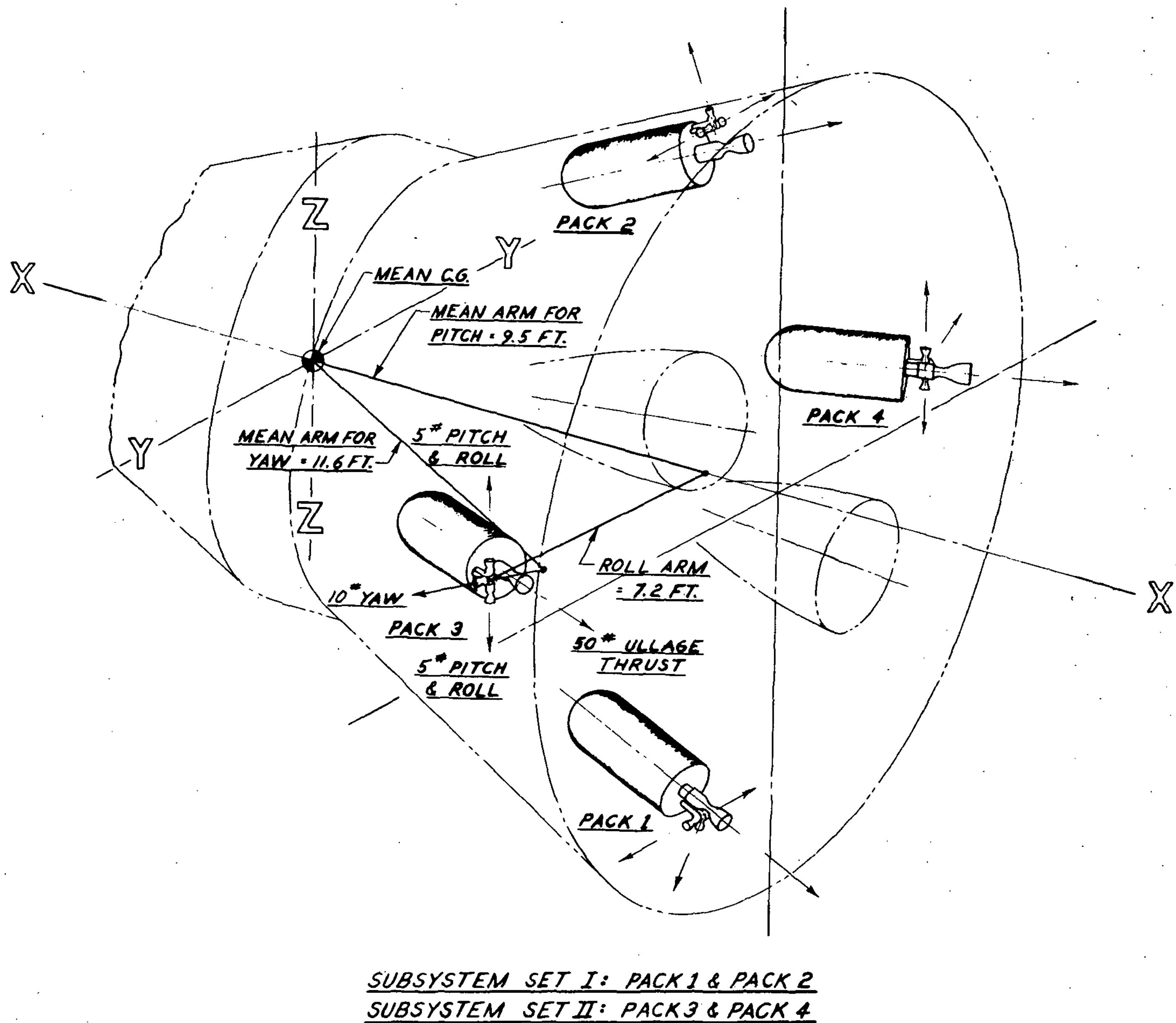


Figure VII-1. Mission Ullage/Attitude Control Propulsion System Vehicle Installation

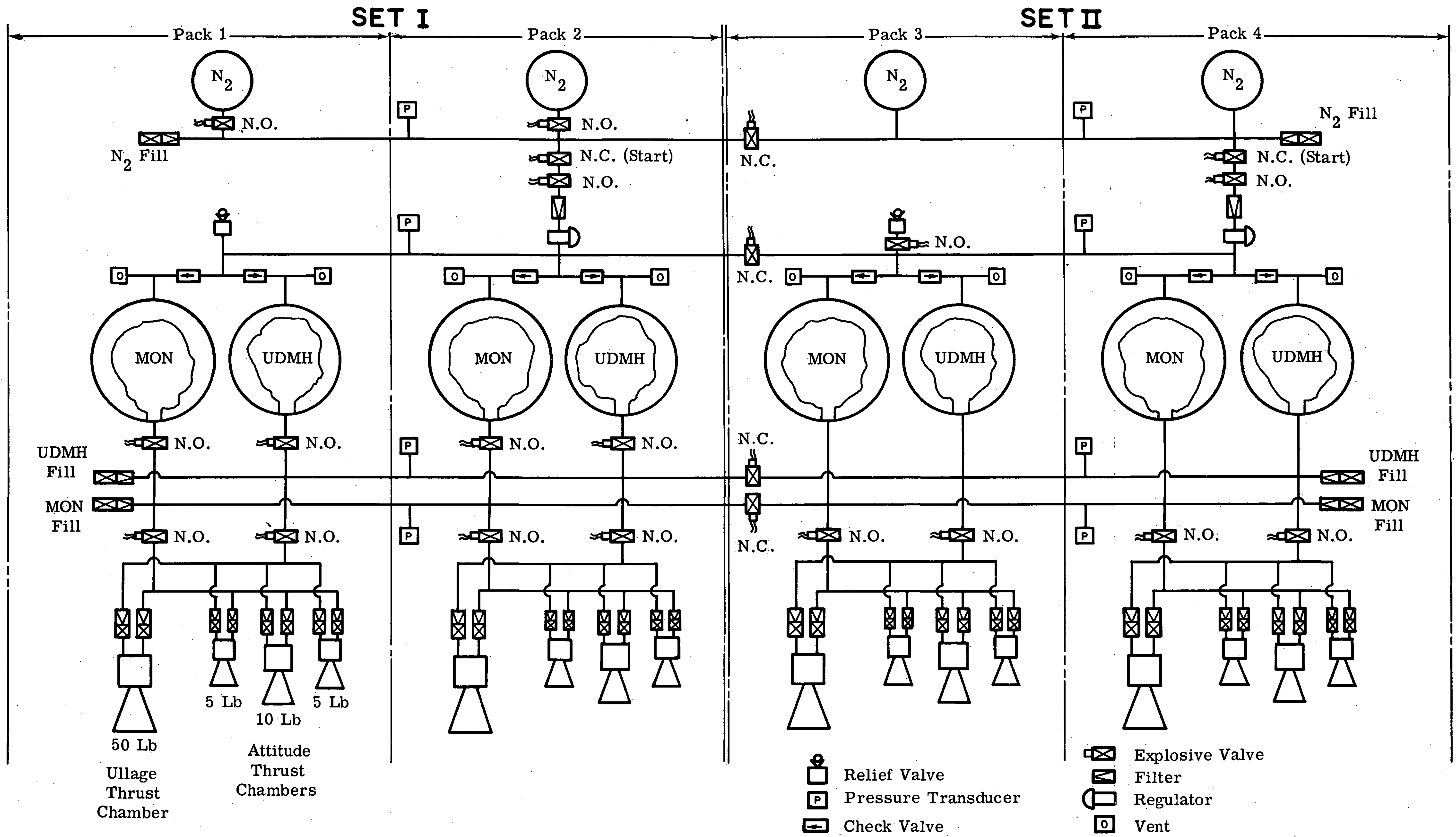


Figure VII-2. Ullage/Attitude Control Propulsion System Schematic

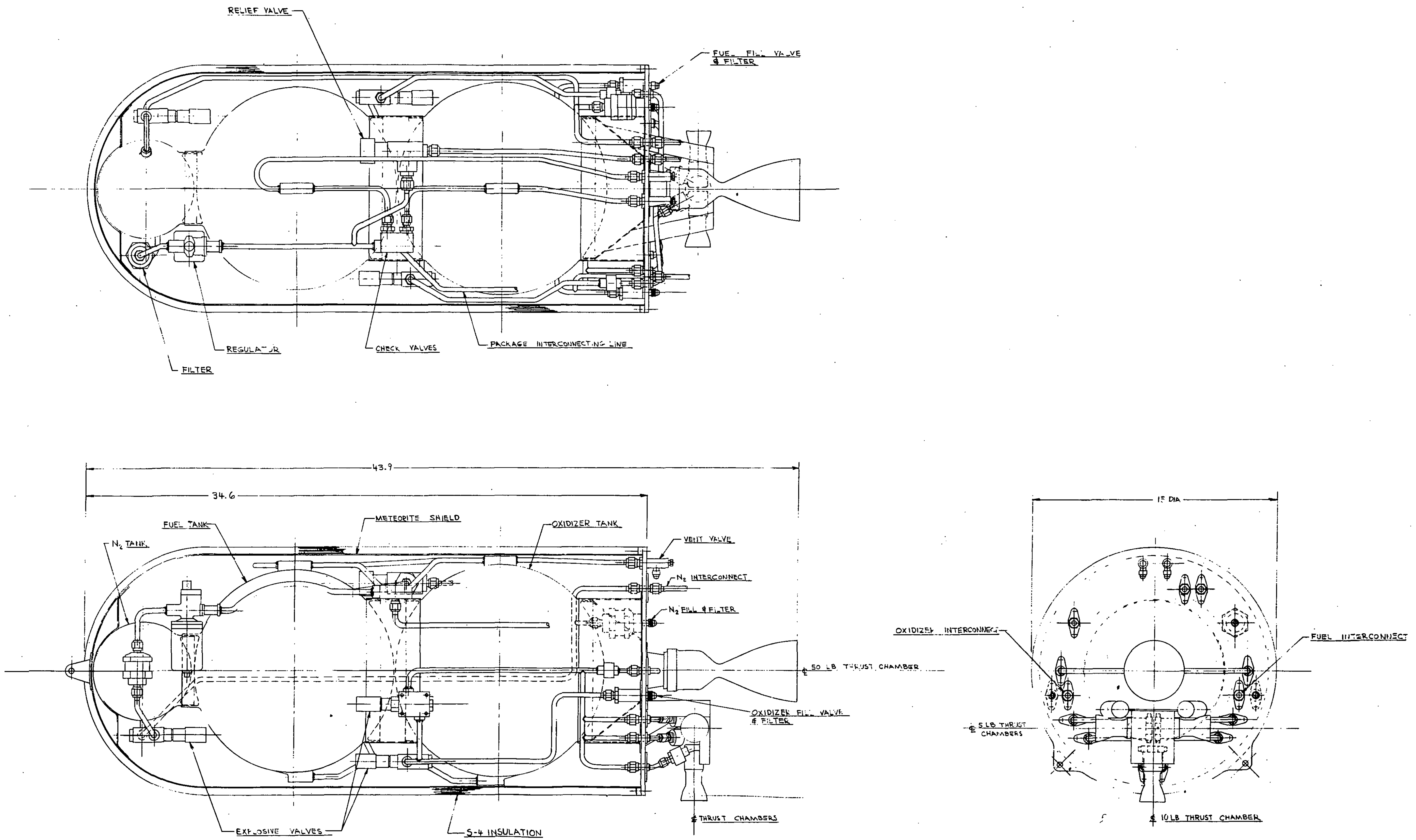


Figure VII-3. Propulsion Package Assembly

2. The fuel tank-nitrogen tank, including expulsion diaphragm and associated plumbing and components.

3. A ground plate carrying the tank assemblies, fill and interconnect lines and components and thrust chamber package mounts.

4. The 50-pound thrust chamber including propellant inlet valves.

5. The 10-pound, 5-pound thrust chamber package.

6. Thermal insulation and meteorite shield cover.

A detailed description of these subassemblies and components is presented in Section C.

3. Operation

The two subsystem sets are designed to be operated normally, in sequence I ⟶ II. Consideration has been given to the mode of operating the two redundant systems alternately which would theoretically slightly increase the system mean time between failure. However, since additional switching networks would be required and it is undesirable to have both systems pressurized continuously, the system and its malfunction features have been based on the concept of normal operation in fixed sequence. In the case of any malfunction within a subsystem set, transfer of propellants in the intact set can be achieved, and the normal operational sequence may be reversed.

The operational sequence in a subsystem set follows. Propellants and pressurization gas are loaded through their respective fill ports, which incorporate filters. The system is armed when desired by energizing the squib actuated gas start valve, allowing gas flow from the high pressure storage system through the start valve, filter and tank pressure regulator. The high pressure gas is regulated to the required tank pressure of 235 ±5 psia. The regulated gas pressure is also used to pressurize the propellant tanks in package No. 2 through the low pressure line interconnecting the two propulsion packages. Any of the set's eight thrust chambers may now be fired by applying an electrical signal input to the control box.

4. Malfunction Provisions:

To maintain the system operational in case of certain component failures provisions have been incorporated to isolate the failed section and cross connect to redundant components. The following components were considered to have the highest possibility of failure: the gas regulators, gas relief valves, thrust chamber inlet valves. For malfunction indication and monitoring purposes pressure transducers have been incorporated in the high and low pressure nitrogen lines and the propellant feed lines of both subsystem sets.

In Subsystem Set I normally open isolation valves are added: (1) at the N_2 tank outlet to prevent gas backing up into these tanks in case they are empty and N_2 sources from Set II are cross connected to Set I; (2) on the high pressure side of the gas regulator; (3) at the propellant tank outlet for the same reason as under (1); (4) in the main propellant feed lines of each pack.

Fewer additional separation valves are required in Subsystem Set II because the normal operating sequence is Set I-Set II. Normally closed separation valves are installed in the gas and liquid interconnect lines of the two subsystem sets. Some typical malfunction cases are as follows:

Case 1: After Set I has depleted 1/2 of its propellants:

a. Malfunction or damage occurs in the thrust chamber circuits of Set I.

Indication: Pressure monitoring light in main feed lines shows red.

Action: Close all isolation valves in main feedline branches of pack 1. Switch control to pack 2. Open main feed line and gas side separation valves between Set I and Set II. Thus utilize propellants from Set I in Set II.

b. Regulator in Set I fails:

Indication: Low pressure gas line monitoring light shows red.

Action: Close isolation valve upstream of regulator in pack 1. Open high and low pressure isolation valves between Set I and Set II. Arm start valve in Set II. Continue use of Set I until propellant exhaustion. Complete mission on Set II.

Case 2: Set I has depleted all of its propellants but is intact. While Set II is being operated, 5 lb thrust unit fuel valve fails to close.

Action: Close isolation valves of main feed lines in Set II. Close tank isolation valves in Set I. Open feed line interconnect valves between Set I and Set II. Resume operation on thrust chambers in Set I utilizing propellants from Set II.

The malfunction provisions shown here represent a weight increase of approximately 13.2 pounds.

5. Performance and Weights

Table VII-1 presents the detail performance parameters of the proposed system. Table VII-2 presents the system weight estimate.

TABLE VII-1

DETAIL SYSTEM PERFORMANCE AND OPERATING PARAMETERS

SYSTEM

Overall system specific impulse	260 sec
Total deliverable impulse	74,000 lb sec
Ullage orientation impulse/engine start	875 lb sec
Number of engine starts	16
Total deliverable impulse for attitude control	60,000 lb sec
Nominal torque pitch	116 ft lb
Nominal torque yaw	95 ft lb
Nominal torque roll	72 ft lb

50 LB. THRUST CHAMBER

Vacuum thrust (lb.)	50 ± 2.5
Vacuum I_{sp} continuous operation (sec.)	304
Minimum vacuum I_{sp} at 5 sec pulse length (sec)	290
Minimum impulse bit (lb. sec.)	5
Maximum continuous operation (sec)	30
Maximum number of 8.75 sec. firings	100
Start response from valve signal to 90% thrust (sec)	0.020
Mixture ratio O/F	2.25 ± 5%
Chamber pressure nominal (psia)	125
Nozzle area ratio	50

10 LB. THRUST CHAMBER

Vacuum thrust (lb)	10 ± 1
Vacuum I_{sp} continuous operation (sec)	284
Minimum vacuum I_{sp} at 0.050 sec. pulse length (sec)	268
Minimum impulse bit (lb sec)	0.5
Maximum continuous operation (sec)	20
Maximum number of 0.10 sec pulses	5000
Maximum pulse frequency per second	3
Valve signal to 90% thrust (sec)	0.015
Residual impulse (lb sec)	0.10
Mixture ratio O/F	2.25 ± 5%
Chamber pressure (psia)	125
Nozzle area ratio	40

5 LB. THRUST CHAMBER

Vacuum thrust (lb)	5 ± 0.5
Vacuum I_{sp} continuous operation (sec)	284
Minimum vacuum I_{sp} at 0.050 sec. pulse length (sec)	260
Minimum impulse bit (lb sec)	0.25
Maximum continuous operation (sec)	20
Maximum number of 0.10 sec pulses	5000
Maximum pulse frequency per sec	3
Start response	
Valve signal to 90% thrust (sec)	0.015
Residual impulse (lb sec)	0.05
Mixture ratio O/F	2.25 ± 5%
Chamber pressure (psia)	125
Nozzle area ratio	70

OXIDIZER TANK

Nominal working pressure (psia)	235 ±5
Relief pressure (psia)	275
Burst pressure (psia)	480
Provision for ullage and mixture ratio mismatch (%volume)	+10

FUEL TANK

Nominal working pressure (psia)	235 ±5
Relief pressure (psia)	275
Burst pressure (psia)	480
Provision for ullage and mixture ratio mismatch (% volume)	+10

NITROGEN TANK

Maximum working pressure (psia)	3600
Burst pressure (psia)	7200

TABLE VII - 2 WEIGHT ESTIMATE

	REQ PKG.	PKG. 1 WT.LBS.	PKG. 2 WT.LBS.	PKG. 3 WT.LBS.	PKG. 4 WT.LBS.
Thrust Chamber Assembly - 50 Lb. (including solenoids)	1	6.3	6.3	6.3	6.3
Thrust Chamber Assembly - 10 Lb. (including solenoids)	1	1.5	1.5	1.5	1.5
Thrust Chamber Assembly - 5 Lb. (including solenoids)	2	3.0	3.0	3.0	3.0
Propellant System		(14.1)	(15.1)	(14.7)	(14.7)
Tank - MON	1	5.9	5.9	5.9	5.9
Tank - UDMH	1	5.0	5.0	5.0	5.0
Valves	AR	3.2	4.2	3.8	3.8
Pressurization System		(3.0)	(7.1)	(2.9)	(6.8)
Tank - N_2	1	2.5	2.5	2.5	2.5
Valves	AR	0.5	4.6	0.4	4.3
Supports	AR	9.5	9.9	9.5	9.9
Plumbing and Fittings	AR	1.7	2.0	1.7	2.0
Insulation	1	5.0	5.0	5.0	5.0
Electrical System		(3.4)	(0.7)	(3.4)	(0.7)
Wiring and Plugs	AR	1.0	0.5	1.0	0.5
Control Box	1	2.0	-	2.0	-
Supports	AR	0.4	0.2	0.4	0.2
PACKAGE WEIGHT		47.5	50.8	48.0	49.9

Total Weight of (4) Packages	196.0 Lbs.
Plumbing, Fittings, Clips - (interconnecting)	12.0 Lbs.
Wiring, Plugs, Clips - (interconnecting)	3.0 Lbs.
TOTAL REACTION CONTROL SYSTEM - DRY	211.0 Lbs.

Total Reaction Control System - Dry	211.0 Lbs.
Loadable/System (4 required)	(296.0) Lbs.
Loadable/Package	74.0 Lbs.
Oxidizer - MON	49.5 Lbs.
Fuel - UDMH	22.0 Lbs.
N_2 Gas	1.5 Lbs.
TOTAL REACTION CONTROL SYSTEM - LOADED	507.0 Lbs.

C. COMPONENTS

1. Thrust Chambers

a. General Characteristics and Problem Areas of Small Pulse-Type Thrust Chambers

In adapting the concept of the liquid bipropellant combustion chamber to very low thrust-level application and the particular modes of operation required for attitude control purposes, some general characteristics and problem areas become apparent. The thrust chamber must deliver very precisely metered and controlled amounts of energy at high rates with high efficiency. This short-time release of energy occurs in the fashion of a small, controlled explosion. This means that at very short pulse durations the release of energy occurs only during transient states and the process may not attain thermal and dynamic equilibrium. Therefore, performance parameters under such operating conditions will differ from steady-state operating conditions. The degree of these deviations is a very complex function of time, temperature, and dynamic factors of the system.

The absolute amount of energy release per pulse required is very small. Therefore, due to the high energy content of the propellant, the absolute amount of propellant and the flow rate per pulse is extremely small. It becomes evident that metering and control of these small, volumetric flows to accuracies of a few percent constitutes a major problem area. Precision in design, fabrication and testing, and miniaturization is required to guarantee proper functioning.

The heart of the combustion chamber is the injection system. This system is governed mainly by the problems associated with the extremely small propellant flows which have to be properly mixed, atomized, and evaporated to provide highly efficient combustion. The injection problems at thrust levels down to 10 pounds are not unusual and can be solved with conventional injector design methods. However at lower thrust levels, problems with very small orifices arise. Losses in performance have to be traded against reliability.

For pulse mode duty cycles, attitude control thrust chambers encounter no unusual cooling problems and in general cooling can be achieved satisfactorily by means of radiation and heat sink. However, a serious thermal problem may arise during shutdown periods in the form of heat flux soaking back into the injection and feed system where stagnant propellants are trapped and are susceptible to evaporation or thermal decomposition.

Thermal isolation of such areas from the hot combustion chamber, of heat conduction by appropriate paths, must be provided under certain conditions to maintain reliable operation and proper performance.

If the pulsing is sufficiently rapid that the heat is not dissipated from the engine by radiation to space, and conduction to the surrounding structure during the down period, the chamber continues to heat up until it reaches a quasi-steady-state condition. If the thrust chamber is not sufficiently well isolated from its valves, there will be an excessive heat soak back to the valve.

Two particular design approaches are being considered, and undoubtedly they will be used in conjunction with each other. First, the valve and thrust chamber are isolated from each other by short, thin walled heat barrier tube and the structural supports between the two are minimized. Second, a short conduction path to a capacitor, such as the fuel tank, from the injector will be incorporated. The former of course is desirable because it tends to isolate the hot chamber from the valve. The purpose of the latter is two fold. During firing the injector is operated at normal temperatures and no significant heat is being conducted to or away from it to the capacitor. However, immediately after a firing, heat can flow from the hot chamber to the injector and thence to the capacitor and thus reduce the peak temperature. Later on as the engine cools down due to the space environment, the capacitor provides a heat source to maintain the injector at a temperature above the propellant freezing point.

Also, the valves and propellant feed lines will be insulated to minimize radiant heating from the hot chambers. This will be done rather than installing radiation shields around the thrust chambers since it is desirable to radiate away as much heat as possible from these high-performance, radiation-cooled engines.

During the preliminary and detailed system design studies, a thorough analysis of this possible problem area will be conducted on each of the three thrust chamber assemblies. The analyses will be supplemented by actual fire test data from the initial component firings to supplement and/or confirm the analyses. The final design of the thrust chamber-valve assembly and installation will be evolved by this method.

High-temperature material problems encountered on small, uncooled thrust chambers are severe when long-duration, steady-state operation is required. However, they can be solved with existing, commercially available alloys and other high-temperature materials.

b. Discussion of Requirements

The thrust chamber designs proposed for use in the bipropellant attitude and orbit control system emphasize fulfilling the requirements outlined in the following paragraphs.

(1) General

All thrust chambers proposed for use in the bipropellant attitude and ullage control system are designed to meet the following requirements:

(a) Mixture Ratio

The nominal mixture ratio for all thrust chambers shall be the same. Permissible maximum variation from the value selected shall be 3% with propellants at 60°F. The mixture ratio selected is 2.25, O/F.

(b) Life Cycle

The life cycle shall be three times the duty cycle of each thrust chamber as described below. Deviation from specified performance shall not be experienced throughout the life cycle.

(c) Operational Life

The operational life of the thrust chambers shall be six months under orbit condition without deviating from specified performance.

(d) Environment

The attitude and orbit adjustment thrust chambers shall meet the requirements of environment as described in another section of this report pertaining to the entire system.

(2) Performance of Bipropellant Attitude and Ullage Control Thrust Chambers

(a) Five-Pound Thrust Chambers

1) Impulse Control

The output impulse in vacuum with propellants at 60°F shall follow the signal pulse duration in accordance with the following relationship:

$Y = (X-C)\ 0.005 \pm 10\%$

Y = output impulse in lb-sec

X = input signal in milliseconds

C = transport lag to 80% full thrust (not to exceed 15 milliseconds) to be reproducible within ±5 milliseconds. Thrust decay impulse shall not exceed the thrust rise impulse by more than 50%.

2) Thrust

The steady-state vacuum thrust of the five-pound thrust chamber shall be 5.0 ±0.5 lb. with propellants at 60°F.

3) Minimum Impulse

The minimum impulse delivery capability shall be 0.25 lb-sec.

4) Minimum Specific Impulse

Minimum specific impulse shall not be less than 260 seconds with propellants at 60°F and a pulse of 0.25 lb-sec.

5) Duty Cycle

The duty cycle after acceptance testing shall be:

1000 cycles, 50 milliseconds duration at 3 cps.
1000 cycles, 200 milliseconds duration at 1 cps.
950 cycles, 600 milliseconds duration at 0.2 cps
50 cycles, 20 seconds duration every 15 minutes.

1) Impulse Control

The output impulse in vacuum with propellants at 60°F shall follow the signal pulse duration in accordance with the following relationship:

$$Y = (X-C)\ 0.005\ \pm 10\%$$

Y = output impulse in lb-sec

C = transport lag to 80% full thrust (not to exceed 15 milliseconds) to be reproducible within ±5 milliseconds. Thrust decay impulse shall not exceed the thrust rise impulse by more than 50%.

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4) Minimum Specific Impulse

Minimum specific impulse shall not be less than 260 seconds with propellants at 60°F and a pulse of 0.25 lb-sec.

5) Duty Cycle

1000 cycles, 50 milliseconds duration at 3 cps.
1000 cycles, 200 milliseconds duration at 1 cps.
950 cycles, 600 milliseconds duration at 0.2 cps.
50 cycles, 20 seconds duration every 15 minutes.

6) Maximum Pulse Width

The five-pound thrust chamber shall be capable of producing one 20-second pulse without deviating from specified performance.

(b) Ten-Pound Thrust Chamber

1) Impulse Control

The output impulse in vacuum with propellants at 60°F shall follow the signal pulse duration in accordance with the following relationship:

$$Y = (X-C)\ 0.01 \pm 10\%$$

Y = output impulse in lb-sec

X = input signal in milliseconds

C = transport lag to 80% full thrust (not to exceed 15 milliseconds) to be reproducible within ±5 milliseconds. Thrust decay impulse shall not exceed the thrust rise impulse by more than 50%.

2) Thrust

The steady-state vacuum thrust of the ten-pound thrust chamber shall be 10 ±1 pound with propellants at 60°F.

3) Minimum Impulse

The minimum impulse delivery capability shall be 0.50 lb-sec.

4) Minimum Specific Impulse

Minimum specific impulse shall not be less than 260 seconds with propellants at 60°F and a pulse of 0.50 lb-sec.

5) Duty Cycle

The duty cycle after acceptance testing shall be:

1000 cycles, 50 milliseconds duration at 3 cps.
1000 cycles, 200 milliseconds duration at 1 cps.
950 cycles, 600 milliseconds duration at 0.2 cps
50 cycles, 20 seconds duration every 15 minutes.

6) <u>Maximum Pulse Width</u>

The ten-pound thrust chamber shall be capable of producing one 20-second pulse without deviating from specified performance.

(c) <u>Fifty-Pound Thrust Chamber</u>

1) <u>Impulse Control</u>

The thrust decay impulse (residual impulse) of the 20 pound chamber shall not exceed 10 ±1 lb-sec.

2) <u>Thrust</u>

The steady-state vacuum thrust of the fifty-pound thrust chamber shall be 50 ±3 pounds.

3) <u>Minimum Impulse</u>

The minimum impulse delivery capability shall be 10 lb-sec.

4) <u>Minimum Specific Impulse</u>

The minimum specific impulse shall not be less than 300 seconds with propellants at 60°F.

5) <u>Duty Cycle</u>

The duty cycle after acceptance testing shall consist of the following sequence:

Thirty 10-second duration firings, with 300 seconds off time between firings.

6) <u>Maximum Continuous Firing Time</u>

The twenty-pound thrust chamber shall be capable of producing one 30 second run without deviating from specified performance.

c. General Design Approach Proposed to Fulfill These Requirements

During several years Bell Aerosystems reaction control work on various types of pulse mode thrust chambers has been analyzed and experimentally investigated. Two types were considered in detail in regard to their capabilities to fulfill the requirements specified:

(1) From the viewpoint of most accurate propellant metering and flow control, a positive-displacement, pump-injection, frequency-modulated thrust unit appeared to be most favorable to fulfill the requirements of impulse control, thrust, minimum impulse, mixture ratio. However, this type thrust unit could not be adapted to the requirement of maximum pulse width without adding considerable complexity, thus impairing reliability of the end product. Therefore, this approach was abandoned.

(2) A pressure-fed, solenoid valve controlled, propellant-injection system was considered adequate to fulfill all basic requirements of impulse control, thrust tolerances, mixture ratio, minimum impulse, and requirements for frequency and pulse width modulation, in the simplest possible manner. This is the design approach which is taken to fulfill the above requirements and is detailed in subsequent sections of this proposal.

d. Theoretical Performance of Proposed Propellant Combination

The 89.3% N_2O_4 plus 10.7% NO with UDMH propellant combination has received extensive study at Bell Aerosystems Company with respect to performance and transport properties. The theoretical computations are obtained with a high-speed digital computer and are based on equilibrium conditions through expansion through the rocket nozzle. Results of these computations are plotted in Figures VII-4 through VII-9.

The performance calculations were made with the assumption that the expanding gases are ideal. In addition, it is assumed that the gases have zero velocity in the chamber, that the gases at each station in the nozzle are at complete thermodynamic equilibrium, that the flow is one-dimensional, isentropic and that no heat is transferred to the nozzle walls. The specified variable is the static pressure in the nozzle. The equations used to calculate performance are summarized as follows:

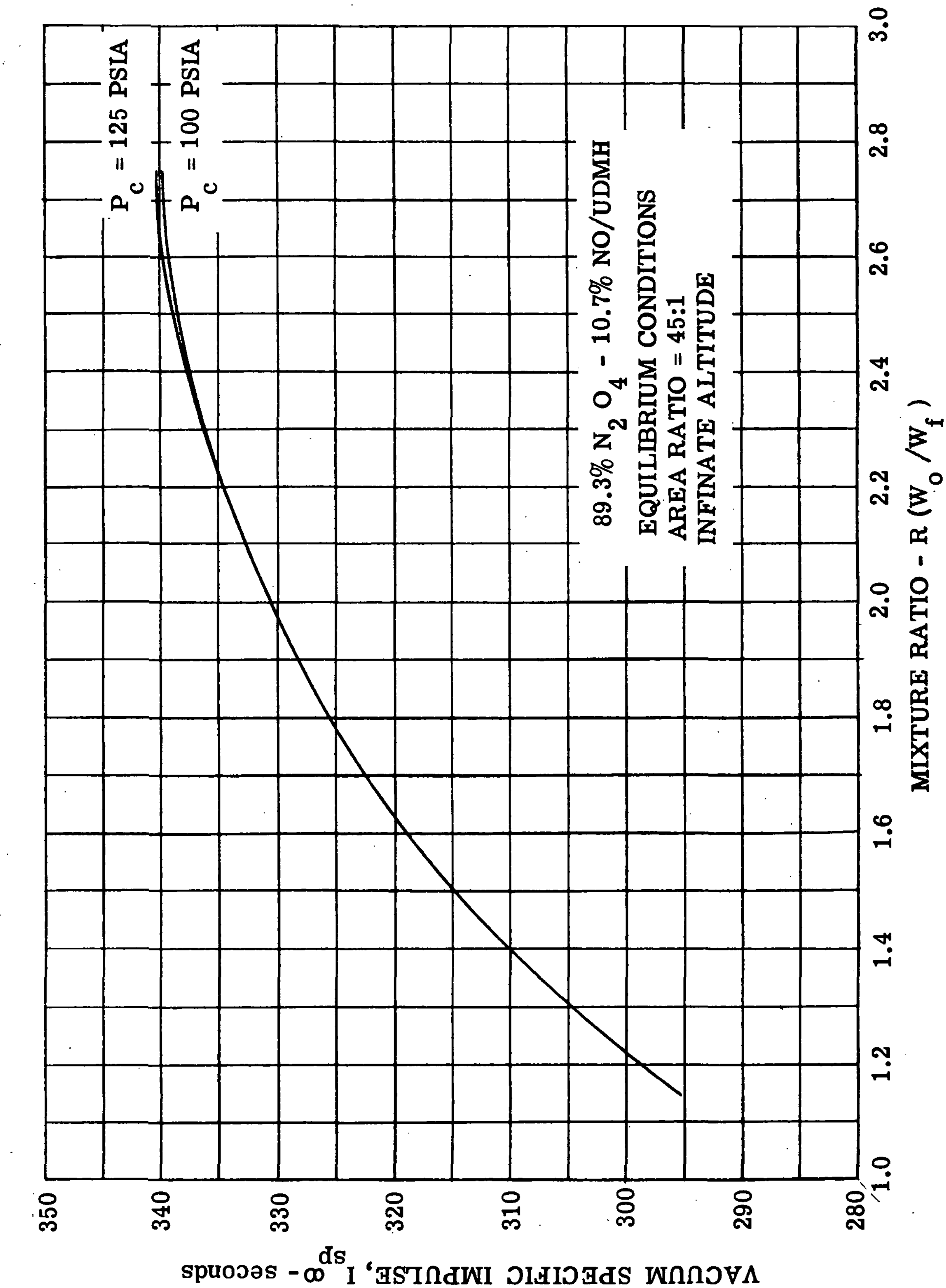

Figure VII-4. Vacuum Specific Impulse for MON-UDMH

[redacted]

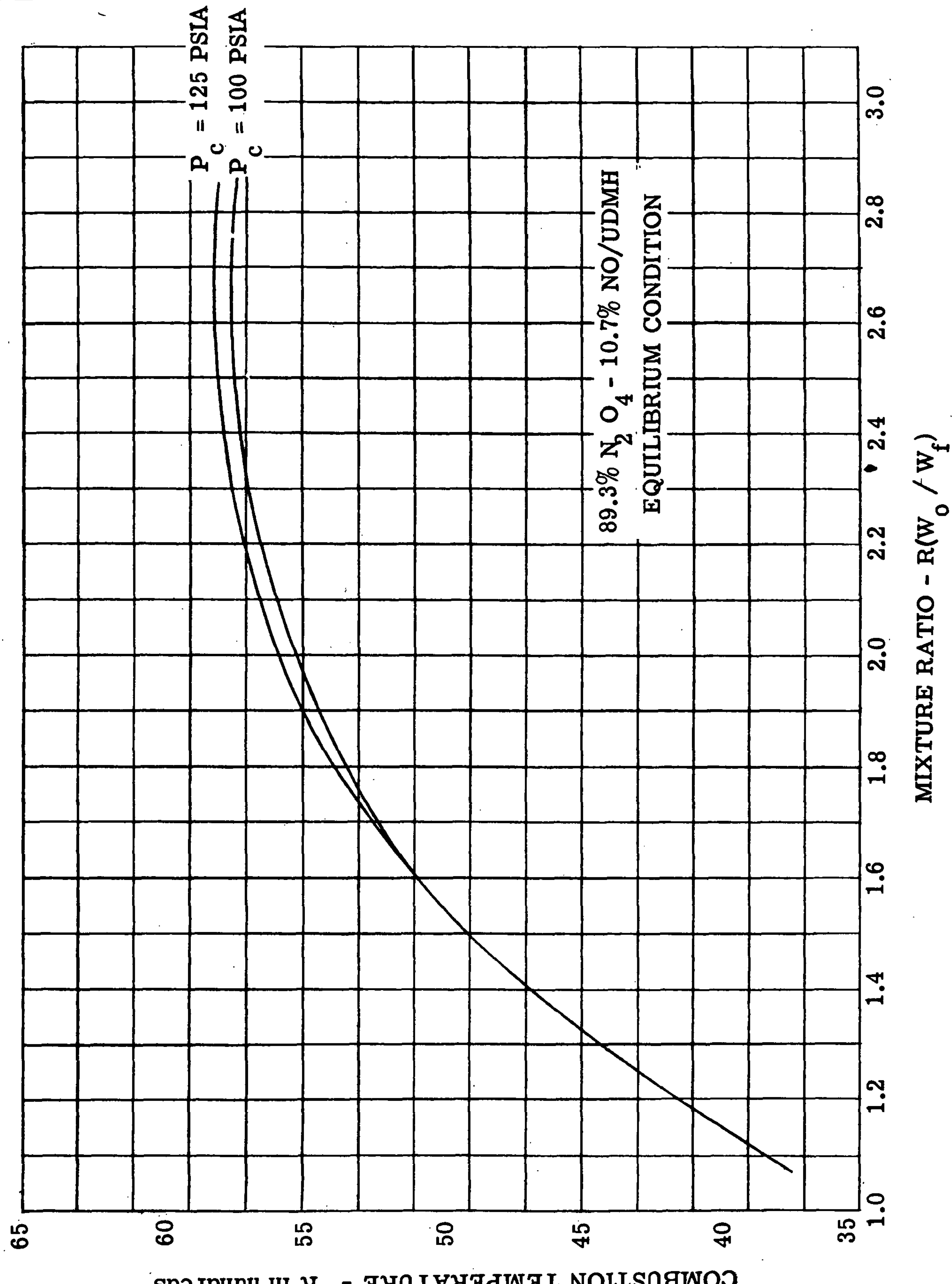

Figure VII-5. Combustion Temperature for MON-UDMH

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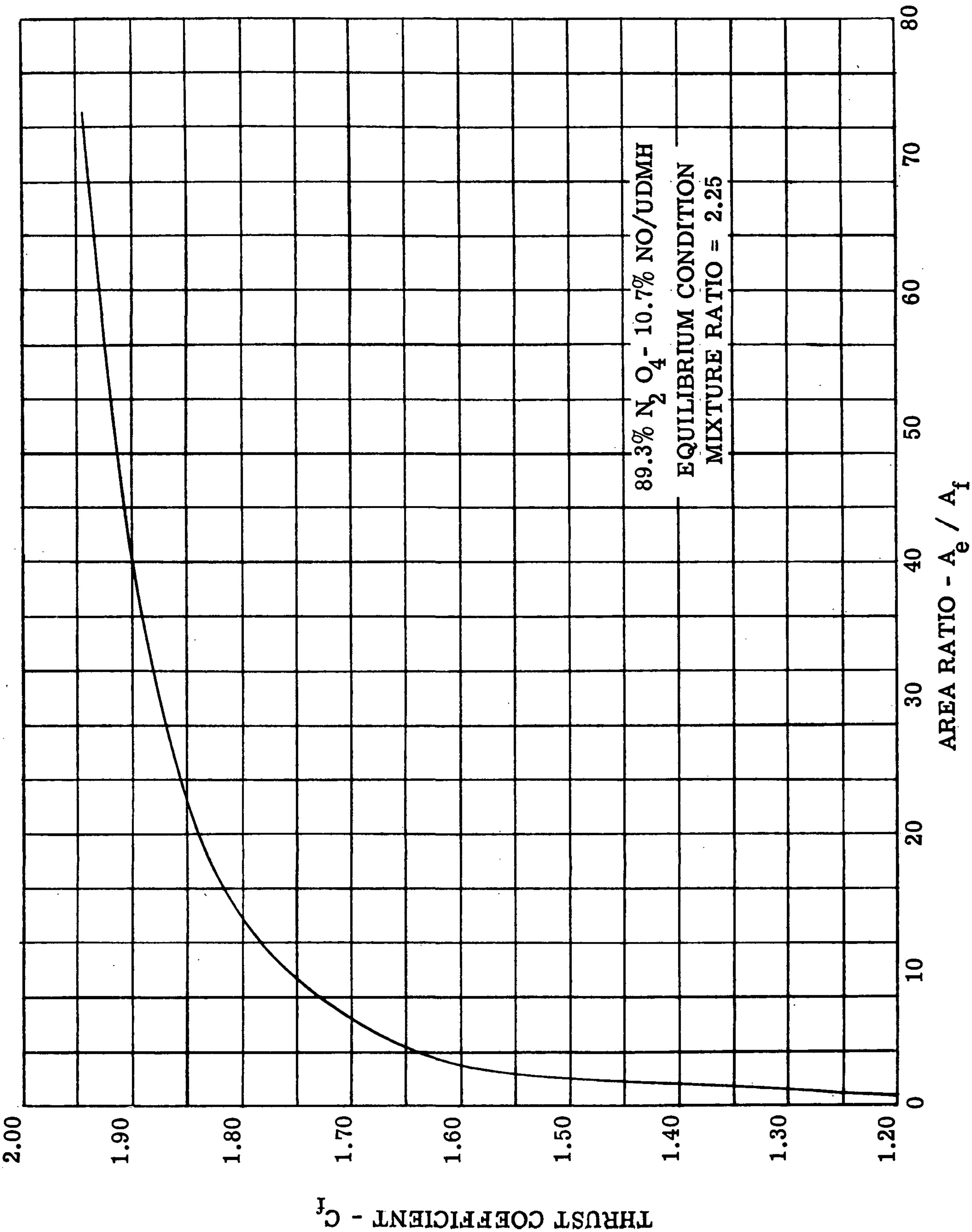


Figure VII-6. Thrust Coefficient for MON-UDMH

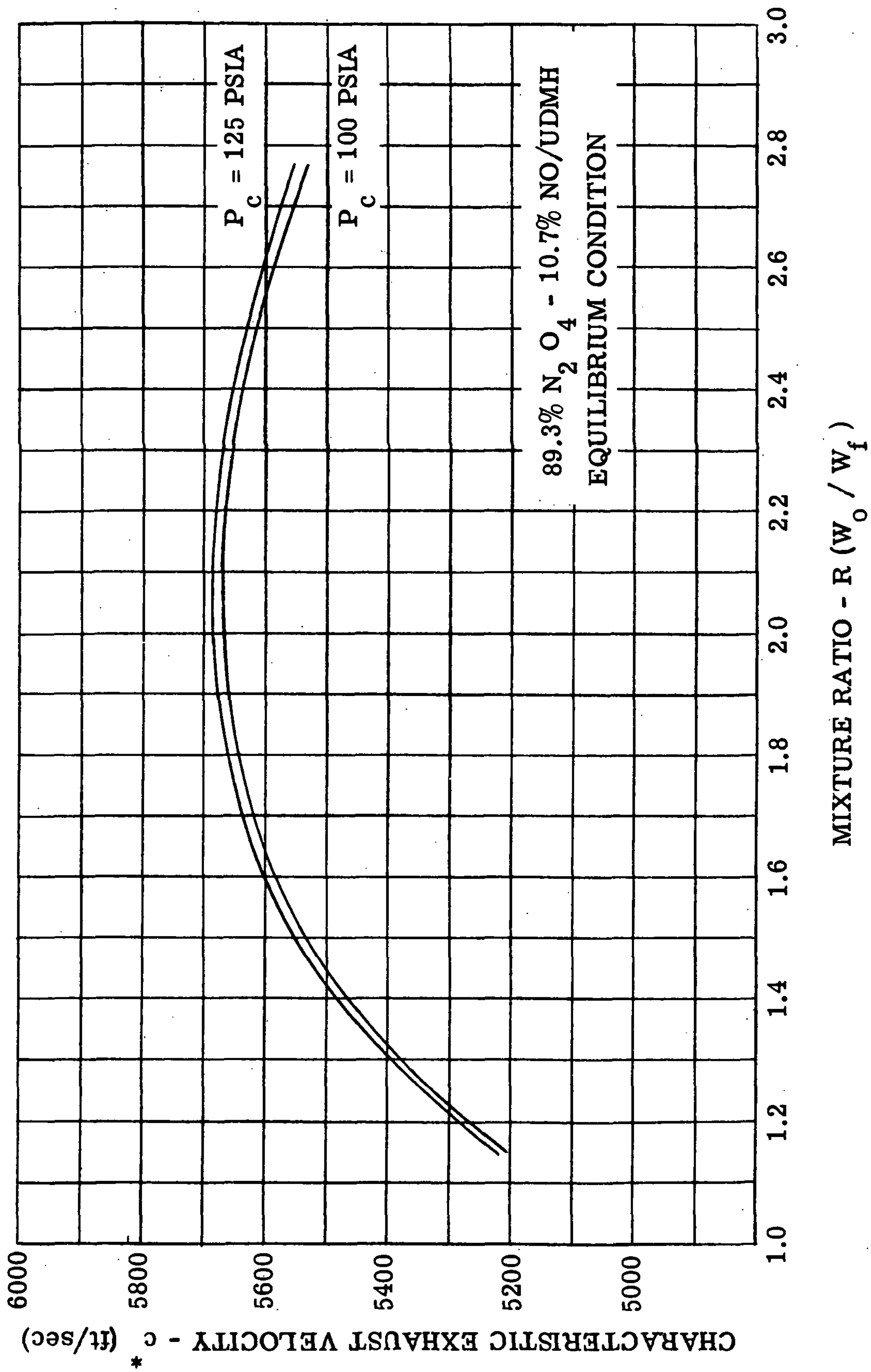


Figure VII-7. Characteristic Exhaust Velocity for MON-UDMH

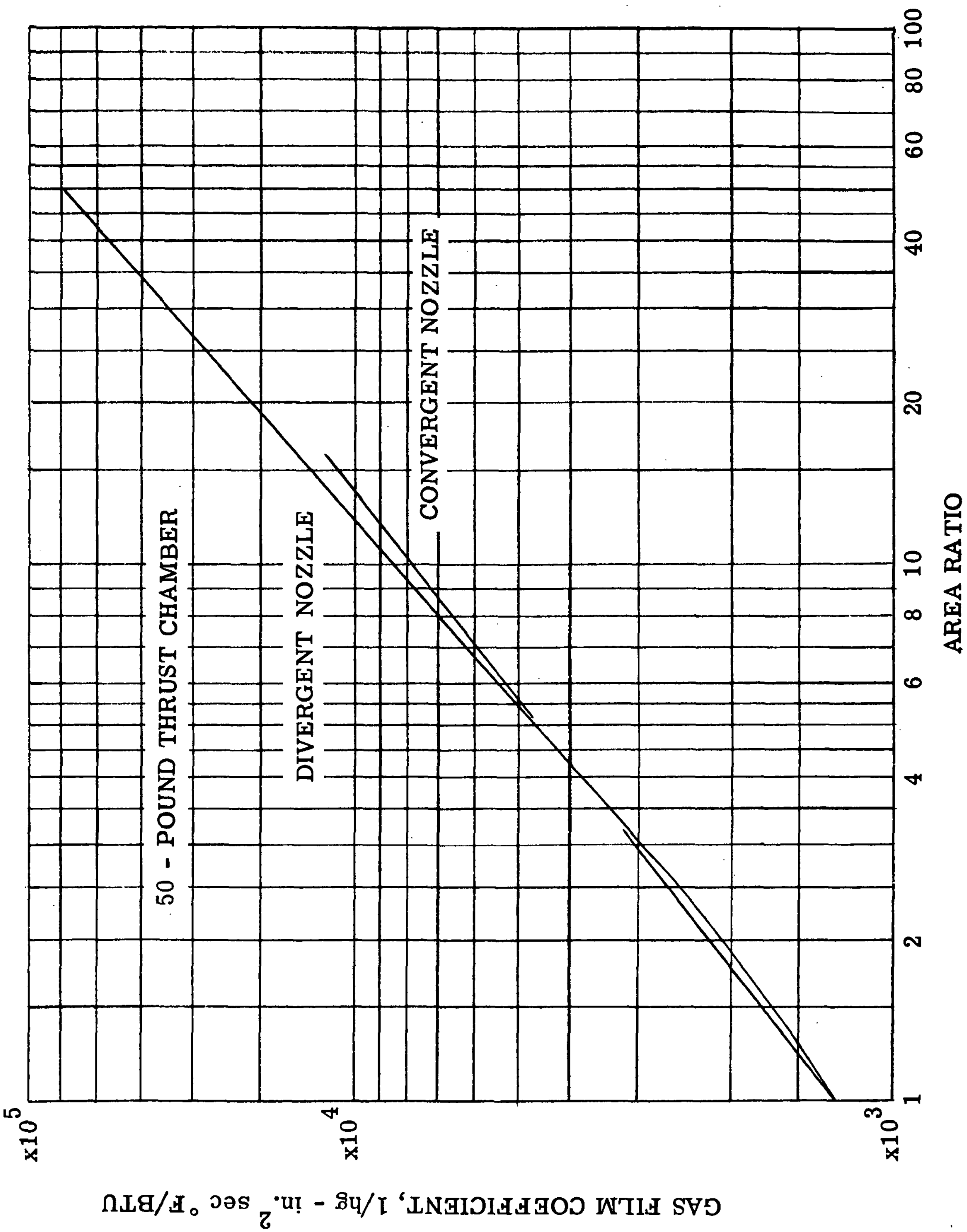


Figure VII-8. 50-Pound Thrust Chamber Gas Film Coefficient Versus Area Ratio

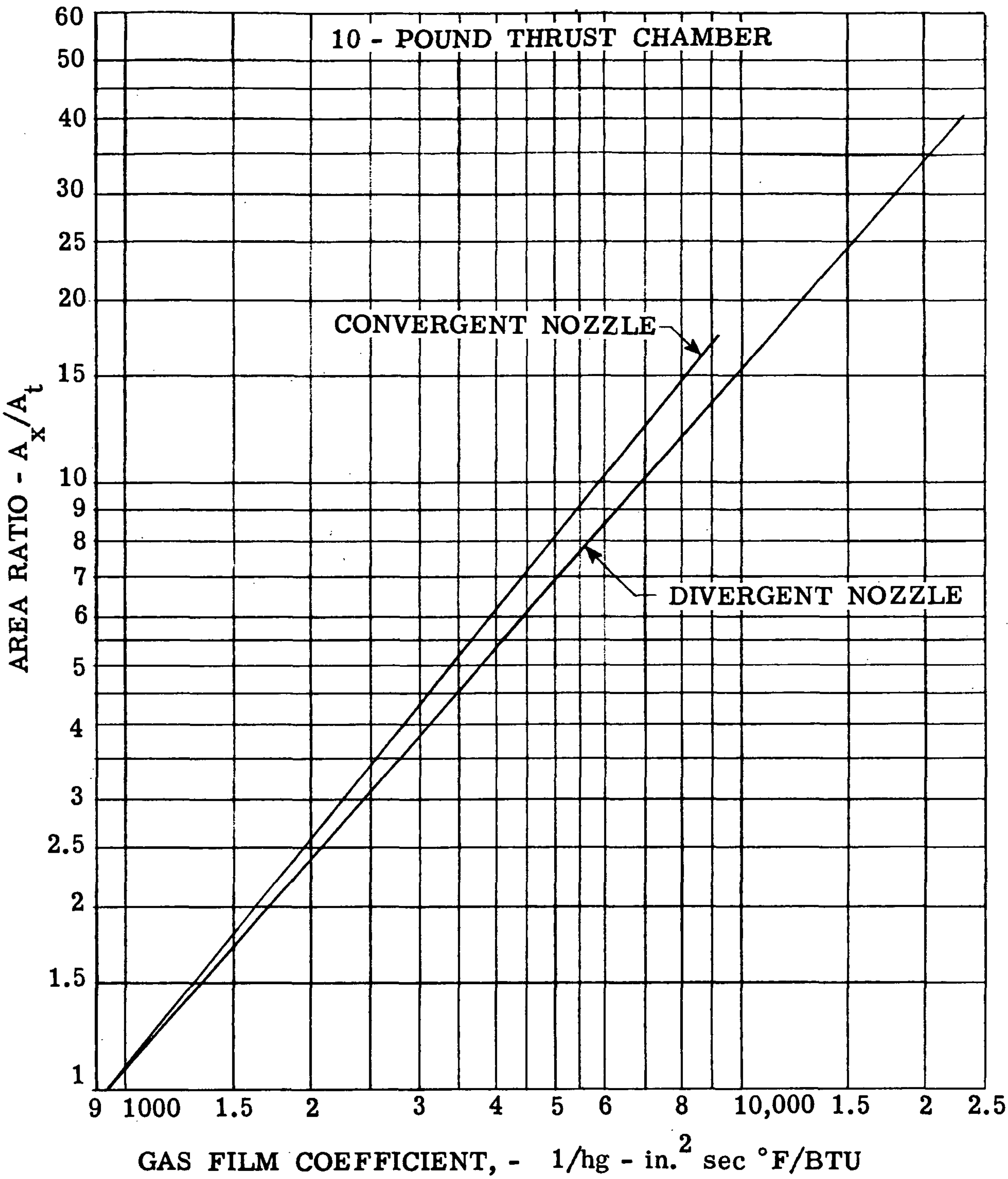


Figure VII-9. 10-Pound Thrust Chamber Gas Film Coefficient Versus Area Ratio

Specific Impulse (I_{sp}) = $9.3281 \sqrt{h_c - h_1}$, lb-sec/lb

h_c = enthalpy of combustion gases in the chamber, cal/gm

h = enthalpy of combustion gases in the nozzle, cal/gm

Velocity (V) = $300.14 \sqrt{h_c - h}$, ft/sec

Characteristic exhaust velocity (c^*) = $\dfrac{P_c}{(\rho V)_t}$

P_c = chamber pressure, psia

$(\rho V)_t$ = mass velocity at the throat station, slugs/ft^2-sec

Thrust coefficient (C_F) = $I_{sp}\ g/c^* \quad V/c^*$

Area ratio (A_X/A_T) = $\dfrac{(\rho V)_t}{(\rho V)_X}$

$(\rho V)_X$ = mass velocity at the nozzle station at specified pressure, slugs/ft^2-sec

All the constituents are known by mole fractions. Therefore, gas film coefficients for the heat transfer analysis are also obtained. The basic relationship, advocated by McAdams, for turbulent flow inside tubes used in this program is:

$$N_u = 0.0225\ R_e^{0.8}\ P_R^{0.4}$$

where N_u = Nusselt number, $h_g \dfrac{D}{K}$

R_e = Reynolds number, $\rho \dfrac{DV}{\mu}$

P_R = Prandtl number, $\dfrac{C_P \mu}{K}$

h_g = gas film coefficient, BTU/ft^2 $^\circ$F sec

D = wetted diameter, feet

K = gas conductivity, $\dfrac{\text{BTU ft}}{\text{ft}^2 \text{ sec } ^\circ\text{F}}$

ρ = gas density, lb/ft^3

C_P = specific heat of gas, BTU/lb $^\circ$F

μ = viscosity of gas, lb/ft sec

By rearrangement, the basic equation is put in the more convenient form

$$h_g D^{0.2} = 0.0225 \, C_P \quad (\rho \, V) \left(\frac{\rho V}{\mu}\right)^{-0.2} \quad P_R^{-0.6}$$

The Prandtl number can be expressed in two ways

$$P_R = \frac{C_P \mu}{K} = \frac{4\gamma}{9\gamma - 5}$$ in which γ is the ratio of

specific heats and where the second form, computed by the kinetic theory of gases, is a very close approximation of the original definitive form. It has the decided value of eliminating the use of the gas conductivity which is at best an area of questionable test data at the high temperatures that are attendant in rocket combustion processes.

The viscosities of the combustion gases are calculated at the selected pressure stations for which compositions had been calculated. A theoretical method outlined by Hirschfelder (Reference 11) is used.

It should be noted that the effect of radiation has been ignored as treated in Reference 12.

Gas film coefficients at the throat where the heat flux is maximum is adjusted to the particular station in the chamber, convergent nozzle, and divergent nozzle.

The equation for the gas film coefficient can be cast into another form, i.e.,

$$h_g = 0.0225 \, \frac{P_c}{c^*} \, C_P \, \frac{A_T}{A} \, R_e^{-0.2} \, P_R^{-0.6}$$

which adjusts for area ratio through the nozzle.

The computed gas film coefficient in the computer calculation is then available at each pressure as

$h_g D^{0.2}$; BTU/ft^2 sec ^{o}F x ft$^{0.2}$

This number is then used for specific size nozzles. Use of this coefficient is explained in the section covering radiation cooling characteristics.

e. <u>Selection of Chamber Pressure and Mixture Ratio</u>

The choice of chamber pressure and mixture ratio of the chambers for this application was dictated by requirements and limitations of the following:

(1) The space envelope and minimum weight requirement

(2) Minimum specific impulse: 260 seconds (pulsing) for the 5-pound and 10-pound units and 300 seconds for the 50-pound unit

It was evident from the performance requirements that a mixture ratio of at least 2.25 would be necessary. It then was a case of using the compromises of computing the probable wall temperature for a given chamber pressure level and a nozzle expansion ratio which would extend the performance of the rather low chamber pressure thrust chambers to meet the performance requirements. The operating duty cycle was also considered in the wall temperature limitations as the transient heating of the 5-pound and 10-pound units would permit higher chamber pressures than if they had to operate at continuous firing as does the 50-pound unit.

Estimated combustion efficiencies were used in the computations to match chamber size with performance. Combustion efficiencies vary inversely with physical size as affected by the diminishing number of impinging injector orifices. Single pairs of injectors coupled with small chamber volumes are here considered mandatory to meet the starting transient times. A higher combustion efficiency for the 50-pound unit was estimated because six pairs of impinging jets would be used.

Relative lengths of the divergent nozzles were set at an identical value - 80% of the length of a 15^{o} half-angle conical nozzle of the same area ratio. This value is very close to that length at which an optimum expansion area ratio is found. To elaborate, for these lengths, attempts to increase performance by increasing area ratio would actually result in a decrease in performance because of the non-axial momentum losses attendant with excessively large divergent nozzle exit wall angles. Longer nozzles would have greater efficiencies of one-dimensional thrust coefficients but it is felt that the length trade-off with chamber length was balanced in a proper manner. For a given contraction ratio, combustion efficiency varies with chamber length but is less sensitive than the relationship of nozzle efficiency with nozzle length.

Identical mixture ratios for all chambers is beneficial systemwise as one set of propellant tanks is used. This assures that both oxidizer and fuel tanks have the same percentage of loaded volume at all times.

A parametric study was performed for each individual case where the major interrelated parameters are:

(1) Throat diameter
(2) Contraction ratio
(3) Emissivity
(4) Material temperature
(5) Combustion gas temperature
(6) Gas film coefficient.

Additional parameters are:

(1) Chamber pressure
(2) Mixture ratio
(3) Propellant combination
(4) Combustion efficiencies.

The primary limitations of the design are the physical properties of the thrust chamber materials and the space envelopes available. Obviously, past experience in similar thrust chamber designs is a guide for the judicious selection of the combustion and nozzle efficiencies and construction details.

The design approach and design details are presented in the following discussions.

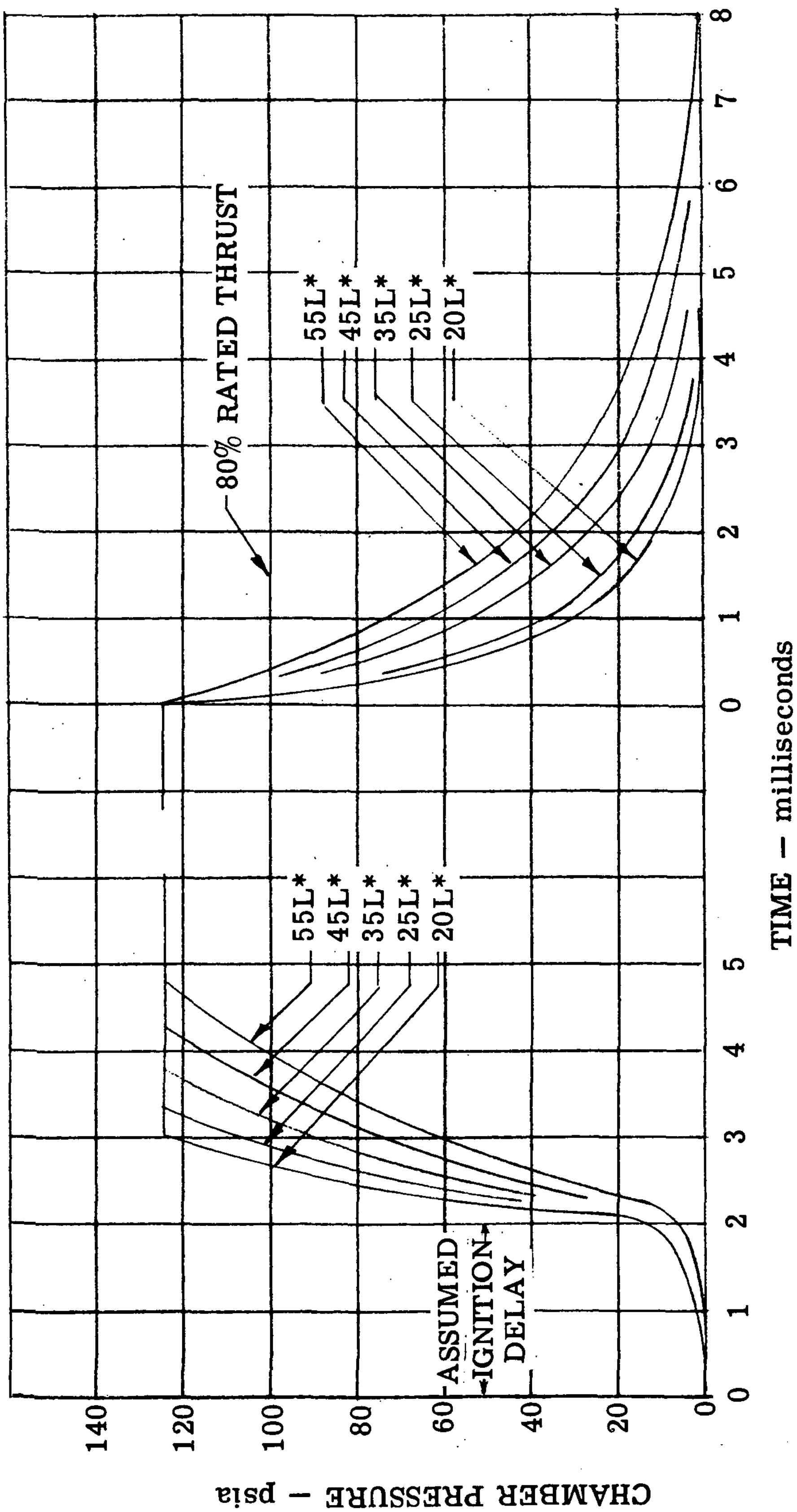
f. <u>Impulse Control</u>

It is anticipated that the impulse control requirements detailed for the 5 and 10-pound thrust units can be met during the development period ending in mid 1963. This conclusion is based on the development of an analytical method for predicting the thrust transients, the use of fast response valves and actuation methods and substantiation by actual test firings of 10-pound thrust units. A detailed discussion of the analytical method and a discussion of the test results follows. A prediction of the transients for the 10-pound chambers is also made based on the analytical method.

(1) <u>Analytical Results</u>

(a) <u>Ten-Pound Thrust Chamber</u>

The results of the analysis predicted start and shutdown transients are shown in Figure VII-10. The start transients were calculated assuming the following:



Calculated start and shutdown transients at infinite altitude for 10-pound thrust chambers using MON (10% NO) and UDMH.

Static feed pressure 210 psia, 2 millisecond ignition delay assumed.

Figure VII-10. 10-Pound Thrust Chamber Start and Shutdown Transient

1) Operation at infinite altitude

2) A two-millisecond ignition lag after the propellants entered the chamber

3) The pressure in the chamber reached the vapor pressure of the propellants at the given mixture ratio before ignition took place.

4) The characteristic overshoot in chamber pressure at ignition was the result of an excess of propellant flow into the chamber during the transient. (The chamber pressure overshoot amounted to a maximum of 10% of rated chamber pressure.)

5) Heat loss was negligible. (This was substantiated by analysis.)

6) The nozzle flow was also sonic at infinite altitude and gas flow leaving it was proportional to the chamber pressure.

7) The flow of propellants into the chamber was proportional to the square root of the pressure drop across the valve, orifice and injector.

The calculation was also made for various values of L*, as shown in Figure VII-10. As expected, the larger values of L* took a larger time increment to reach rated chamber pressure because of the larger physical chamber volume involved. The shutdown transients shown in Figure VII-10, also for various values of L*, were computed assuming combustion of all propellants prior to venting, and adiabatic venting. These theoretical curves indicate that the shutdown time and total impulse can be made extremely small (of the order of 0.008 lb-sec for an L* of 20, to 0.018 lb-sec for an L* of 55) if the residual volume of propellant downstream of the valves is kept small. The designs herein proposed have kept these volumes to a minimum. The analytical results in general indicate that a chamber designed with a low value of L* is desirable to obtain short start and shutdown transients.

The injector is fabricated from 304L stainless steel and welded in place at the head end of the combustion chamber. A single pair of injection orifices are sized to deliver equal-momentum propellant streams at rated flow. The injector is machined on the outer surface to accommodate a flange-mounted valve at each orifice location, providing a short coupled fluid path from the valve seat to the orifice.

The exterior surface of the thrust chamber is coated, after the installation of the injector, with "Preferred Pyrochrome" to increase the emissivity of the surface and aid radiation cooling.

The primary consideration in the design of the injector was to maintain volume of propellants downstream of the valve seat. The rapid fill time for a pulse type of operation not only requires rapid operation of the propellants valves, but also requires a minimum of void area that the propellants have to fill before actual injection of propellants into the thrust chamber is accomplished. This criteria was accomplished on this design by maintaining almost a straight tube between the valve and the injector orifices and also by the use of single injection orifices, thus negating the requirement for distribution manifolds in the injector.

(2) <u>Fifty-Pound Thrust Chamber</u>

The fifty-pound thrust chamber design shown in Figure VII-11 incorporates many of the design features developed for the units of a current contract at BAC. The combustion chamber and nozzle are fabricated from a forging of 90% tantalum 10% tungsten alloy. The combustion chamber is designed for operation at 125 psia chamber pressure and has an L* of 35. The nozzle, an 80 percent Bell, contoured, is expanded to an area ratio of 50:1.

The injector, made entirely of tantalum, is attached to the thrust chamber by welding. The material selection of the injector was based on the weld compatibility with the thrust chamber material. Although other materials would be equally suitable from a temperature and propellant compatibility viewpoint the choice is dictated by weldability.

The basic configuration of the injector is similar to the impinging-showerhead design used on the Model 8101 twenty-pound thrust chamber except that the orifice sizes are modified for the difference in mixture ratio and flow rate. The oxidizer enters the chamber through orifices drilled in the central dome, while the fuel enters through the outer fuel manifold which is suitably baffled to prevent stagnation and local boiling. The baffle requirement has been conclusively demonstrated on the Model 8101 Program.

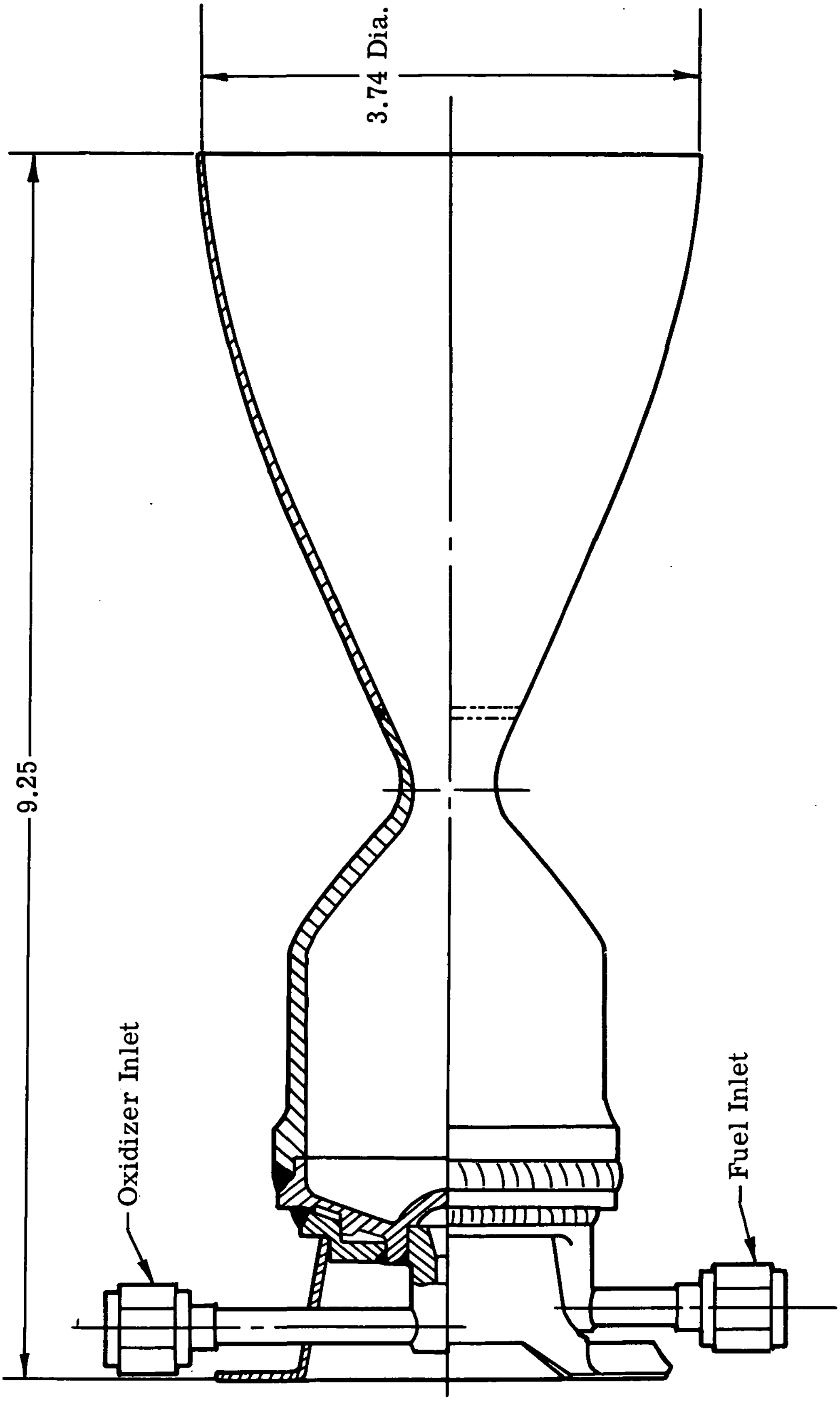


Figure VII-11. 50-Pound Thrust Chamber

(2) Test Results

(a) Ten-Pound Thrust Chamber

Figure VII-12 shows a portion of a test record obtained during pulsed operation of a test unit. Pulses are approximately 50 milliseconds long and 50 milliseconds apart resulting in a duty cycle pulsing rate of 10 cycles per second. It can be seen that starting is extremely fast and that rated chamber pressure is reached in 4.5 to 5 milliseconds from start signal. It also appears that ignition delay is shorter than the two milliseconds assumed in the analytical prediction of chamber pressure rise, however, the initial pulse in this series exhibited an ignition lag of the order of two milliseconds. The shutdown of each pulse is also smooth and reproducible. It is obvious that reproducibility of pulses is excellant. The total impulse of each of these three pulses varies only a total of ±4% from the average.

g. Thrust Chamber Design - Attitude Control

The thrust chamber designs presented in this study, for use in the attitude control system, are based on experience in design, fabrication and testing of small radiation-cooled rocket thrust chambers both monopropellant and bipropellant. The engineering and manufacturing background applied herein offers a high degree of confidence to deliver a design consistent with the system performance definition for the 1963 version of the spacecraft.

(1) Five and Ten-Pound Thrust Chambers

The thrust chambers proposed for the system are illustrated in Figures VII-13 and VII-14 and are designed to operate at a chamber pressure of 125 psia at the 5- and 10-pound nominal thrust levels. In each case the thrust chambers consist of a 21L* combustion chamber with an integral contoured nozzle and a welded injector. The area ratio of the 5-pound thrust chamber nozzle is 70:1 and that of the 10-pound unit is 40:1. The chamber and nozzle sections are machined from Haynes Alloy No. 25 bar stock. Although this alloy is not considered capable of extended continuous operation, substantially reduced temperatures will be experienced with the pulse-type operation, therefore the material can be used. If temperatures higher than anticipated are encountered a throat insert of a 90% tantalum-10% tungsten alloy can be installed. This requirement is not anticipated since a continuous pulse test for nine minutes has been conducted on a thrust chamber of this design without overheating or erosion of the internal metal surfaces.

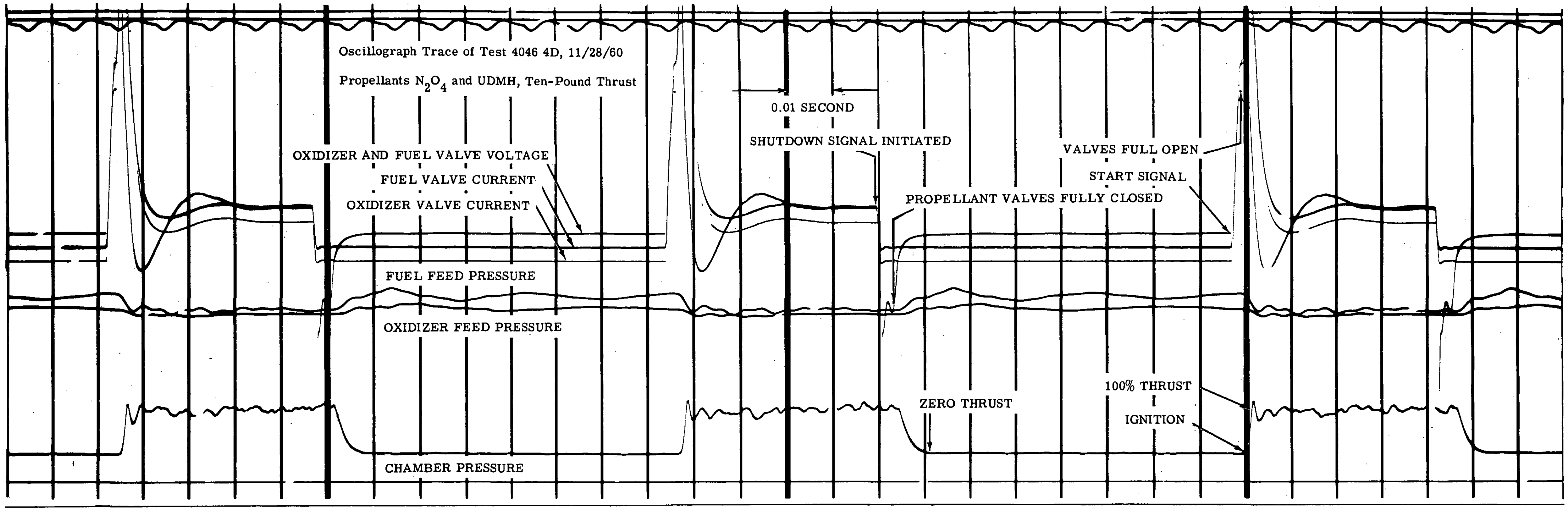


Figure VII-12. Oscillograph Trace of Test 4046 2D – MON-UDMH 10-Pound Thrust Chamber

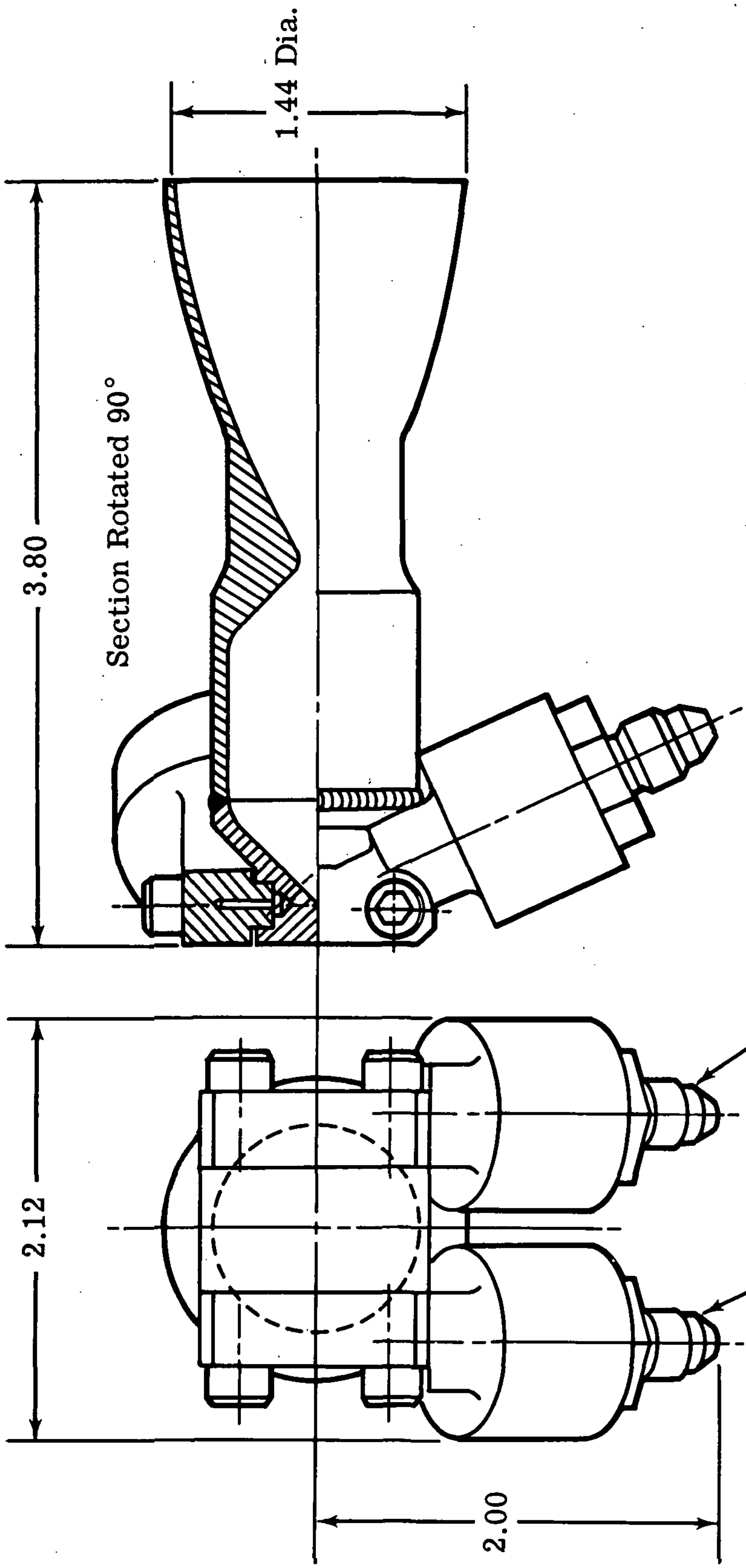


Figure VII-13. 5-Pound Thrust Chamber

[redacted]

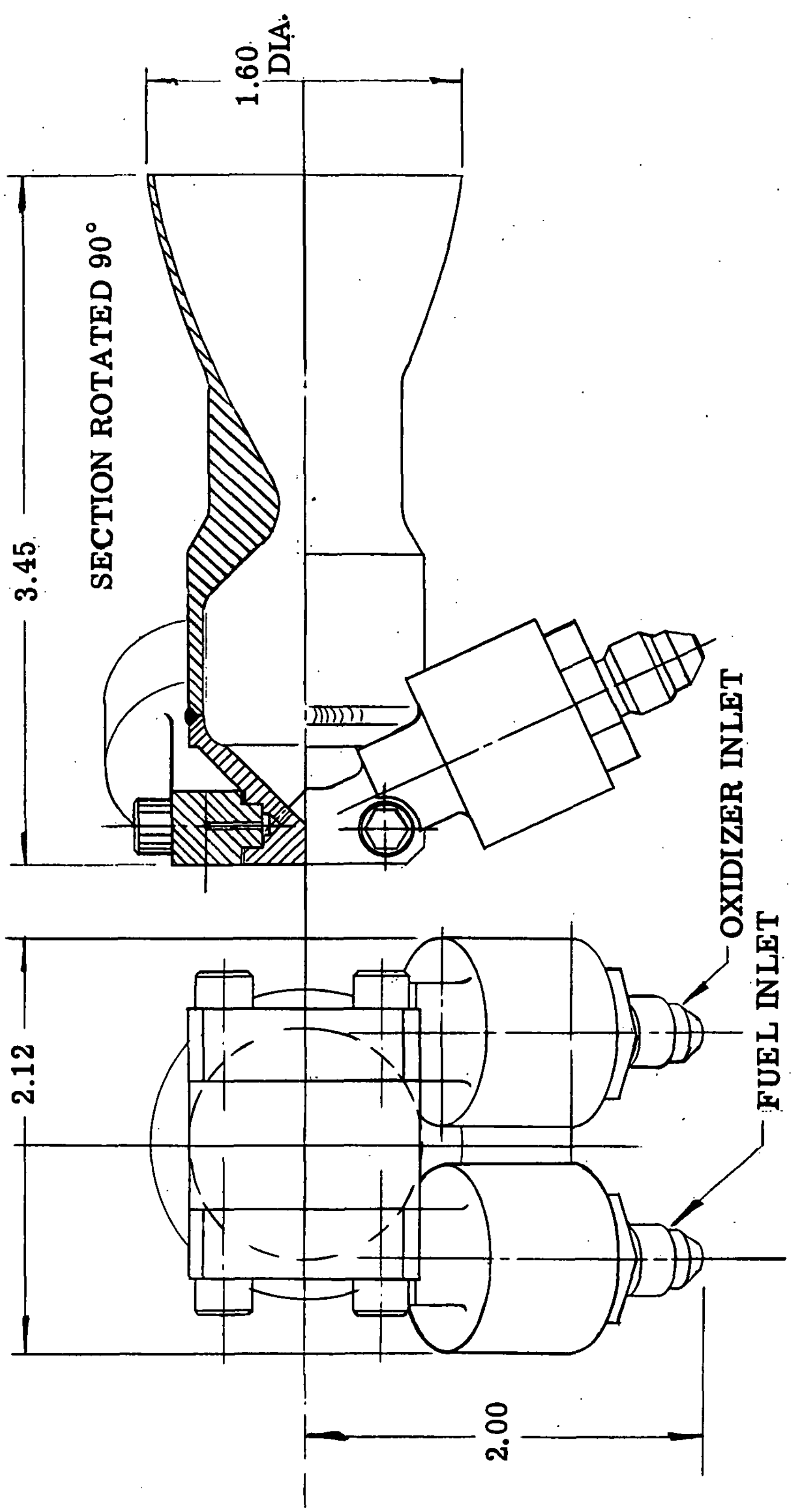


Figure VII-14. 10-Pound Thrust Chamber

[redacted]

The external surface of the thrust chamber will be coated with platinum black which will serve as a high emissivity coating. The chromium oxide coating employed on the Model 8101 thrust chamber and the Model 8096 nozzle extension is unsatisfactory in this application since its vapor pressure is too high for operation at temperatures above 2300°F in a vacuum environment.

Selection of materials for use in this radiation-cooled thrust chamber design involved the consideration of elevated temperature mechanical and physical properties, thermal conductivity, oxidation resistance, emissivity and durability, as well as availability and fabrication requirements.

Verification of the suitability of the material selection has been established in the development and test program on the "Agena Orbital Adjust System" thrust chambers which operate under similar conditions as will be required for the proposed hardware.

h. Estimated Steady-State Performance of Proposed Designs

The steady-state performance estimates for the reaction control and ullage adjust thrust chambers are based on experience with bipropellant and monopropellant thrust chambers in the one-pound to 20-pound thrust range. Results of extensive testing of small nozzles under altitude conditions indicates that, with careful design and rigorous quality control, nozzle efficiencies equal to that obtainable with engines of the 15,000-pound thrust class can be realized with the small units. Attainment of the high combustion efficiencies at the 5 and 10-pound thrust levels appear difficult to achieve. Recent testing of a 10-pound thrust bipropellant unit has resulted in combustion efficiencies of 91 percent with a single pair of impinging streams. The combustion efficiency of the 5-pound thrust unit using 0.015 inch diameter streams will be somewhat lower since the mixing and atomization of the low velocity injection streams will inherently reduce the combustion efficiency. Test results on a 10-pound thrust chamber are shown in Figure VII-15.

The estimated steady-state performance and principal design parameters of the proposed thrust chambers are presented below.

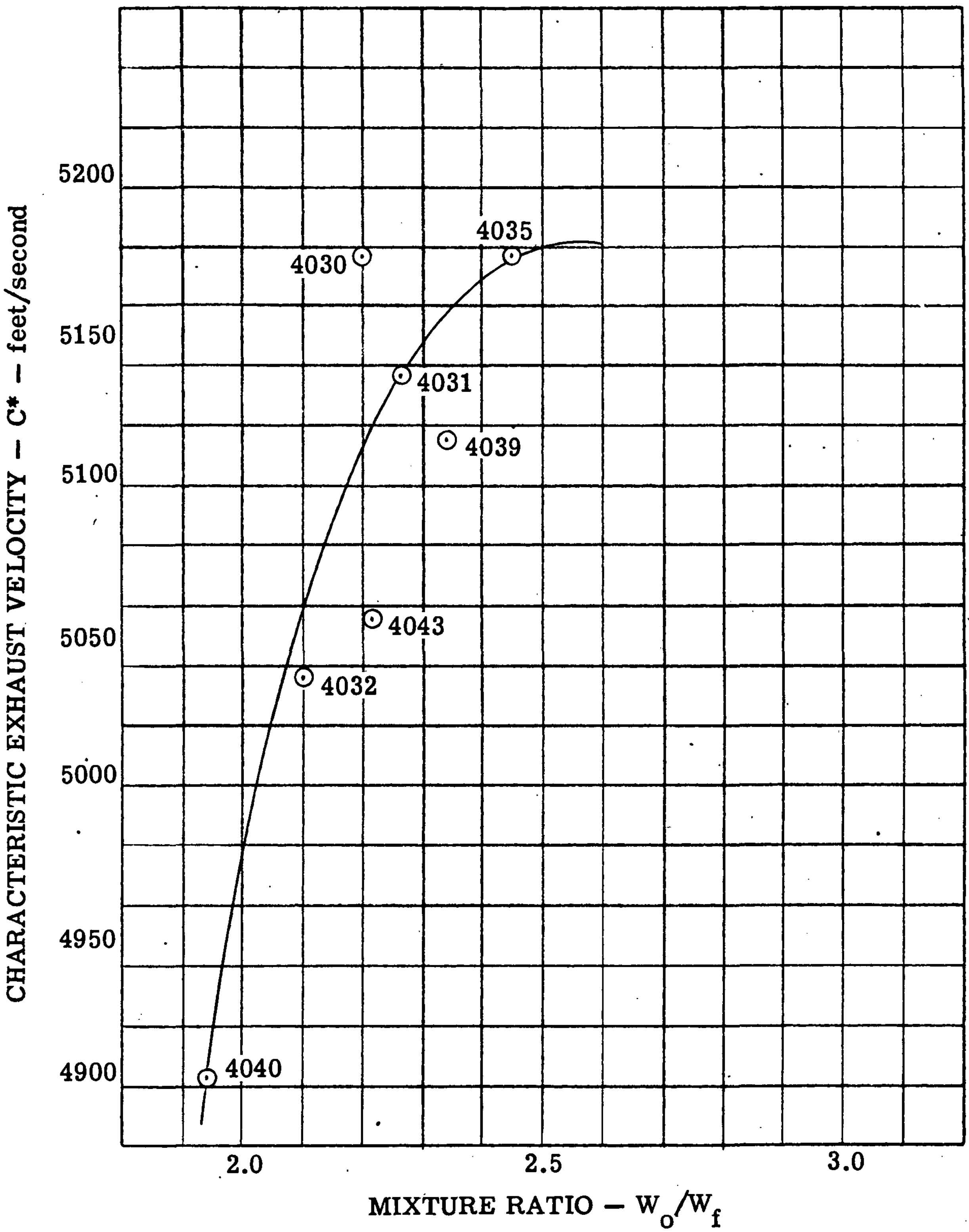


Figure VII-15. Performance Test Data Obtained With Experimental 10-Pound Thrust Chamber

Vacuum thrust (pounds)	5	10	50
Rated chamber pressure (psia)	125	125	100
Rated mixture ratio	2.25	2.25	2.25
Nozzle area ratio	70	40	50
Characteristic length (in.)	21	20.8	35
Theoretical c* ft/sec	5675	5675	5660
Estimated c* ft/sec (Avg.)	4900	5000	5310
% of theoretical c*	86.4%	88%	94%
Theoretical $C_{f\infty}$	1.94	1.90	1.916
Estimated $C_{f\infty}$	1.868	1.828	1.84
% theoretical $C_{f\infty}$	96.2	96.2	96.2
Theoretical $I_{sp\infty}$	342	335	337
Estimated $I_{sp\infty}$ (Avg.)	284	284	304

i. Estimated Performance Under Pulse Mode Operation

The steady-state performance of the proposed thrust chambers can be readily defined in terms of characteristic velocity and thrust coefficient, where the characteristic velocity defines the efficiency of the combustion process and the thrust coefficient defines the efficiency of the nozzle in converting the thermal energy of the combustion gases into kinetic energy. Energy losses due to the transients on start and shutdown have no significance in long duration firing and need not be considered. However, in short duration pulsing mode, the transient times on start and shutdown occupy a significant part of the total operating time and the efficiency during these periods must be taken into consideration.

It can be said that in vacuum operation, the nozzle efficiency is practically independent of chamber pressure, and therefore the thrust coefficient can be considered constant during all chamber pressure transients. However, combustion efficiency undergoes changes with chamber pressure and also, for instance, during the ignition phase where the characteristic velocity cannot be normally defined. Other losses occur during chamber pressure build-up in an individual pulse where heat is being transferred at a high rate to the cold thrust chamber wall and detracted from producing chamber pressure.

The magnitude of these losses is listed below for the various transient portions of the pulse, based on experience on a one-pound and 1.5-pound thrust unit.

	Transient Phase	Equivalent Loss in c* for Mean Length of Transient
1.	Ignition Delay	10%
2.	Chamber Pressure Build-up	5%
3.	Chamber Pressure Stabilization	3%
4.	Shutdown Transient and Down Time	25% of residual propellant volume

The relatively high loss after shutdown is mainly due to the fact that after valve closing some fuel is retained in the small feed holes between the valve poppets and injection orifices by capillary forces. During the down time in vacuum this residual evaporates without producing useful thrust. Taking these fixed, equivalent propellant losses per pulse into account, the specific impulse obtained at varying pulse width as been calculated and is shown in Figure VII-16 for the proposed thrust chamber designs.

2. Valves and Controls

a. Thrust Chamber Propellant Valve Assemblies

The thrust chamber propellant valves proposed for all the thrust units are modifications to conventional in-line solenoid valves. Modifications may include the use of a metal-to-metal seat design. All internally wetted surfaces of these valves are constructed of stainless steel to eliminate corrosion due to electrolytic action of dissimilar metals over the required operating life of the component, or due to any possible contamination of the propellants with water. Figure VII-17 shows the design of the 50-pound fuel and oxidizer valve incorporating this construction method. The valve can include a position indicator as shown if desired. The 5 and 10-pound fuel and oxidizer valves will incorporate the same internal configuration but will be physically smaller. The coil assemblies are hermetically sealed for vacuum operations. Sample valves of this design have been built and evaluated in hydrogen peroxide service at Bell Aerosystems. Satisfactory results from these tests indicate that long-term contamination will not present a problem in this design.

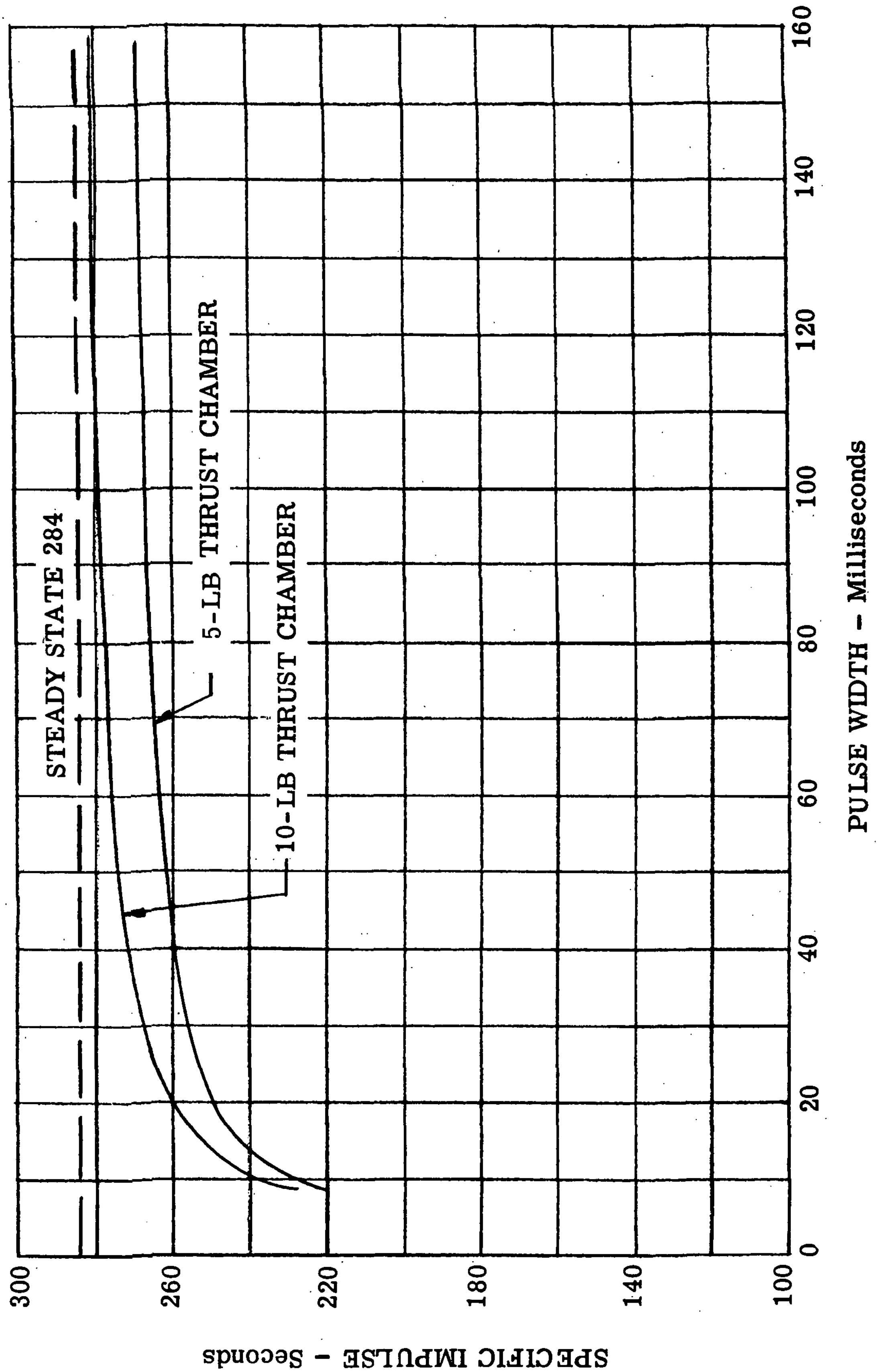


Figure VII-16. Estimated Specific Impulse in Pulse Mode

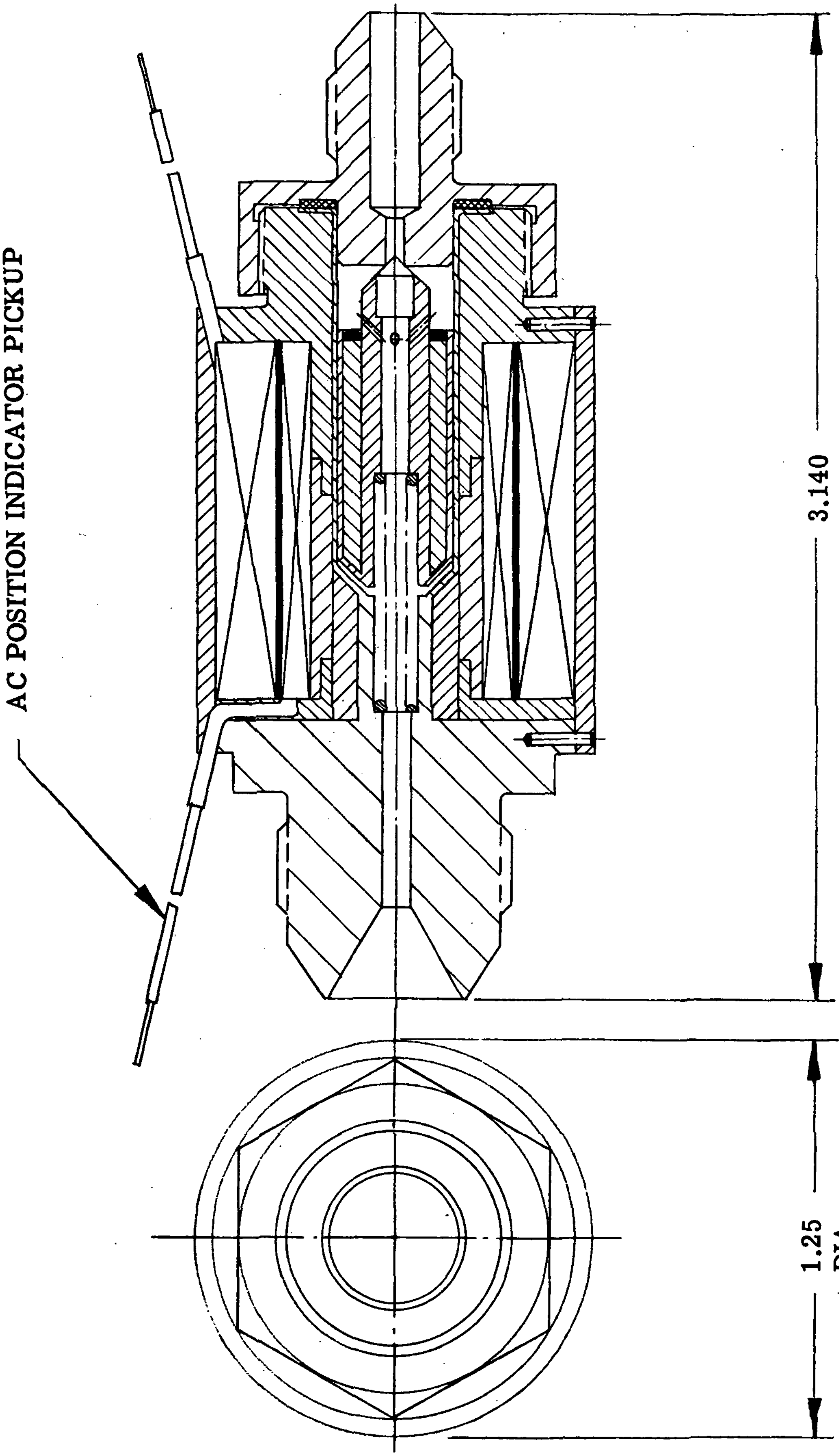


Figure VII-17. Propellant Valve with Position Indicator

Filters are provided at the inlets to each propellant valve to protect the valve and thrust chamber orifices from contamination or clogging. The filter elements will be constructed of stainless steel woven mesh. Sintered filters are not considered due to the possibility of particles separating from the element and becoming a contaminant. Filter particle removal is 98 percent of all five micron particle sizes, or larger. The remaining 2 percent can be a maximum of 15 micron. Each valve also includes an orifice assembly used to obtain the desired propellant flow rate for thrust and mixture ratio calibration in each assembly.

All static seals within the valves will be of crush metallic design.

Two basic seat designs will be considered. Metal-to-metal designs using high unit stress levels based on Hertz Mathematical Deformation Theory are the main approach. Plastic or elastomer materials will be considered as a second approach.

b. Gas Regulator

The gas pressure regulator proposed for the attitude and orbit adjust system is shown in Figure VII-18. The unit is of spring loaded, single-stage design to regulate the source pressure to the required tank pressure. The device incorporates the internal pressure sensing design concept with a metal-to-metal seat and a metal bellows as the diaphragm operator. The main seat design uses a high seat force to produce local deformation and obtain leak-tight shut-off during the long "off" periods. The leakage under these conditions must be held to values below a 3 cc/hr. Present "state-of-the-art" in regulator designs is 1 cc/hr. The development objectives of the design shown, which is currently being investigated, is 1 cc/24 hr. No major advancement is therefore required to meet the leakage rates defined for this system.

The high seat forces are obtained by the use of the mechanical linkage shown in the cross-section drawing. The internal moving links are steel. As in the propellant valve designs, all static seals are of the metallic crush type.

c. Gas Filter

A 5-micron-capacity-type filter is installed in the main gas line ahead of the regulator. The filter protects the remaining gas components.

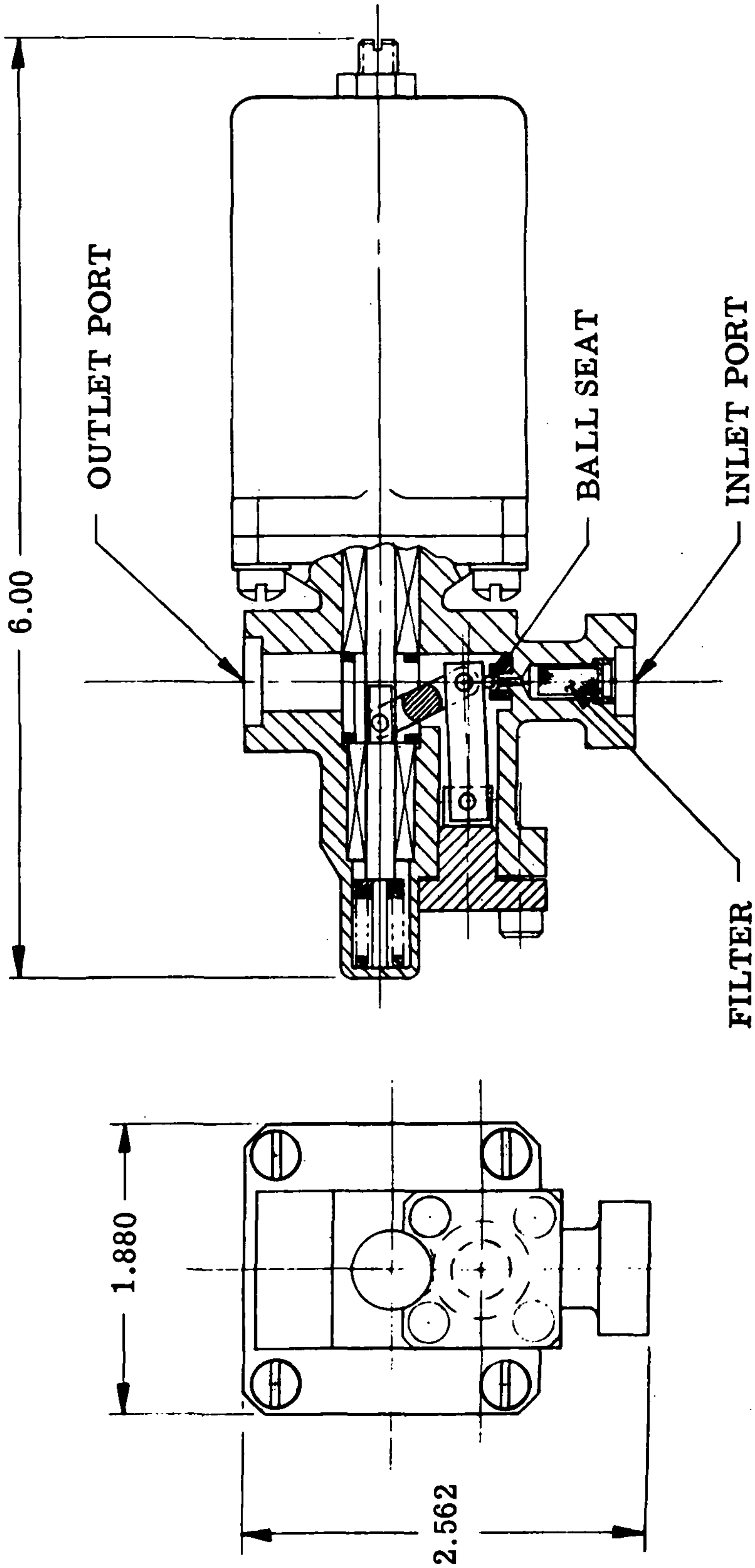


Figure VII-18. Nitrogen Gas Regulator

d. Tank Relief Valves

Figure VII-19 presents the proposed tank pressure relief valves for both the oxidizer and fuel tank installations. It will be noted that this design is not the conventional type, but uses the sealing principles described for the gas regulator. In conventional designs, problems always exist due to the low unit loading applied to the seat. As the inlet pressure approaches the relief setting, the low loading allows leakage which is a constant source of gas loss even during normal regulator operation. In addition, repeatable reseat values compatible with system requirements and gas loss minimization are difficult to achieve. As a consequence, propulsion system thrust tolerances and system weight are penalized by setting the relief pressure higher than optimum.

The proposed design makes use of the mechanical advantage of a linkage system to apply high unit loads to the seat. In addition to the use of metallic design for radiation capability its advantage is that closer pressure tolerances between cracking, full flow and reseat values are possible. All gas to the valve is filtered by an integral filter assembly, thus protecting the seat and moving components. As in the previous valves, metallic crush seals are used for static applications. The internal sensing bellows is stainless steel and the mechanical linkage is of high-strength steel. A leakage requirement of 1 cc/hr presents no area for development as 1 cc/hr is state-of-the-art and designs are currently being developed with objectives of obtaining 1 cc/24 hours.

e. Fill Valve Assemblies

The nitrogen gas and oxidizer and fuel fill valve assemblies shown in Figure VII-20 incorporate integral filters as shown. The fill valves are of conventional designs, incorporating metallic static seals and plastic dynamic seals on the poppets. Plastics may be used in these applications since any deterioration in sealing properties due to radiation effects will not effect the system. External leakage is prevented by capping after fill procedure, while the filter prevents system contamination due to decomposition of the seals.

f. Check Valves

The check valves are provided between the propellant tanks to protect the regulator from corrosion in the event of bladder leakage, and to prevent intermixing of propellants in the event of dual bladder leakage. This check valve assembly will be of all metallic construction similar to a valve assembly currently in use for a secondary propulsion system contract at BAC.

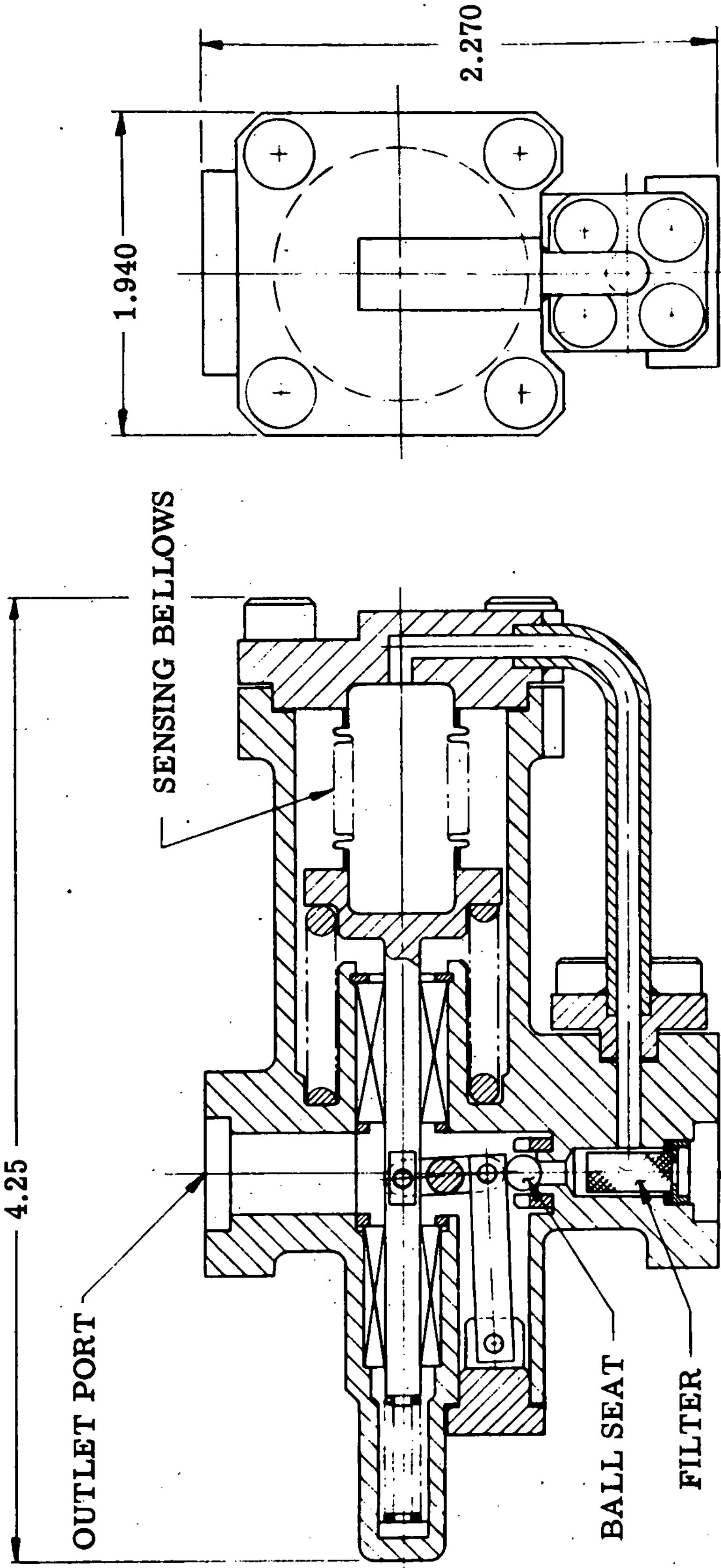


Figure VII-19. Propellant Tank Relief Valve

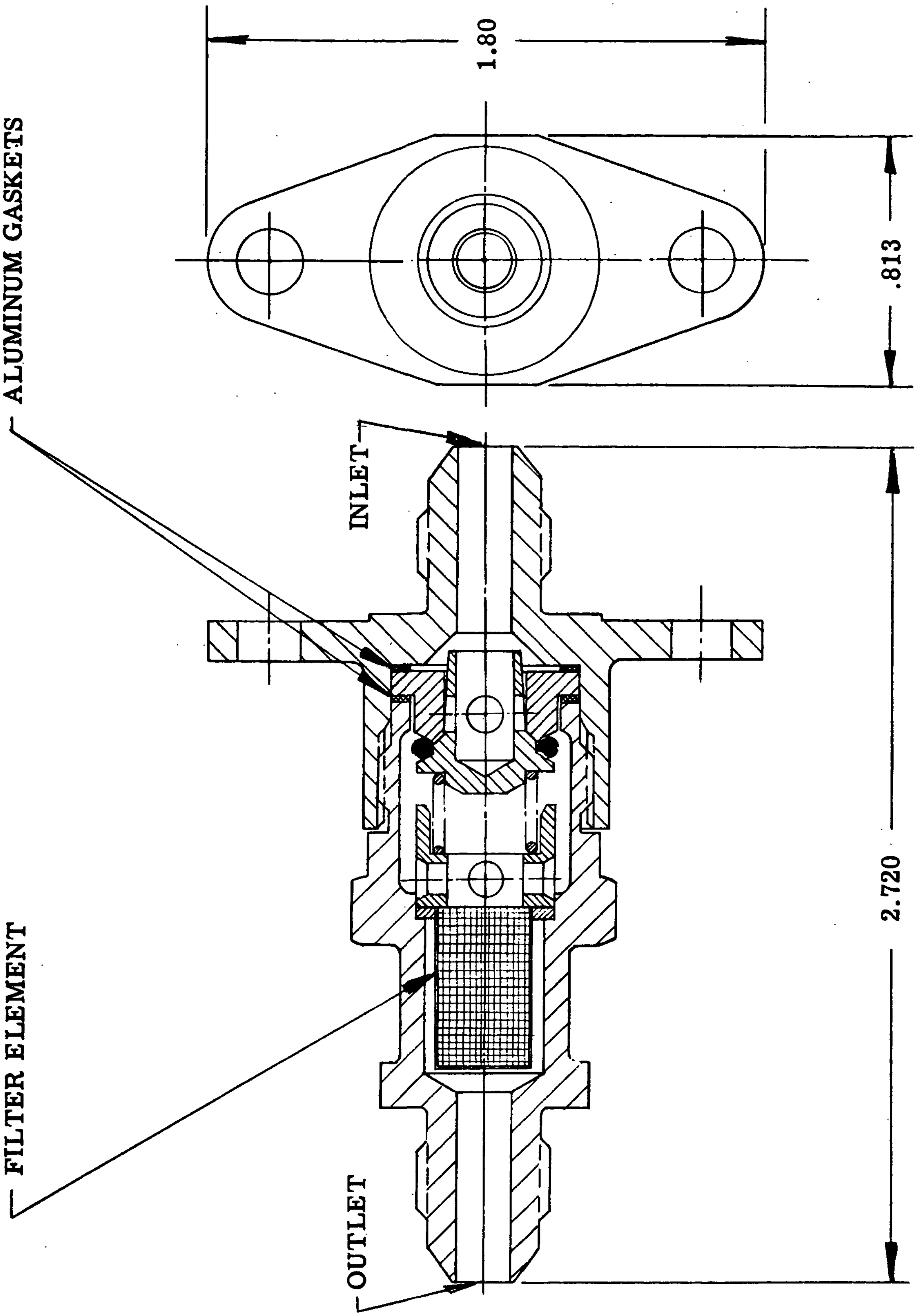


Figure VII-20. Fill Valve Incorporating Filter

3. Tankage Preliminary Design

The propellant tanks are spherical in shape with two bosses located diametrically opposite to each other. One boss is for the propellant discharge and the other is for the pressurization gas inlet. The bosses will be designed so that the inlet and outlet tubing can be welded in place to insure a leak-tight joint. Mounting of the tank can be accomplished in a variety of ways. One way is by butt welding an integral knuckle ring with a skirt into the tank shell. Other methods include mounting into a saddle and strapping or suspending the tank from mounting lugs. Each of the four MON and UDMH propellant tanks of the mission attitude/ullage control system will be identical in size. The oxidizer tank outside spherical diameter, excluding bosses and mounts, would be 13.05 inches for the required total impulse. The fuel tank outside spherical diameter would be 11.92 inches. The calculated weights of the tanks are 5.9 pounds and 5.0 pounds respectively.

The propellant is contained between one-half of the tank shell and a thin aluminum diaphragm which conforms to the other half of the tank. This diaphragm will provide positive expulsion under all gravity conditions and in all tank attitudes. The propellant is expelled by introducing pressurized gas between the tank shell and the diaphragm, through the gas pressure port causing the diaphragm to deflect forcing the propellant out of the discharge port.

The operating pressure of these tanks is nomianlly 235 psia. All material design considerations are base on Specification MIL-HDBK-5 with a 93% weld efficiency and pressure factors based on MIL-T-5208 for manned applications. The material selected is type 6061 aluminum alloy because of its good forming characteristics and resistance to corrosion. The tanks are welded in the T-4 condition and aged to the T-6 condition.

The pressurization portion of the tank shell is attached to a center splice ring by butt welding. After the welds are cleaned up on the penetration side, the diaphragm is permanently attached to the tank shell by sandwiching it between the tank shell and a special ring roll seam welded to the tank wall. The diaphragm is attached at approximately the mid-point of the tank. The outlet portion of the tank shell is then butt welded to the center splice ring using the diaphragm ring as a back-up.

The metal diaphragm configuration described above was selected rather than a plastic diaphragm or bladder because of its lighter weight and simplicity of design. Back-up approaches would include a flanged tank with a replaceable metal or plastic diaphragm, and also a plastic bladder attached in a manner similar to the proposed aluminum diaphragm.

Considerable testing has been accomplished to date at BAC on bladders (metal, plastic, and rubber) and testing is being started on diaphragms. The tank configuration and the environmental requirements for an existing contract with cylindrical tanks are such that the bladder testing is not directly applicable to this particular design. However, the results from the diaphragm testing on this existing contract are applicable and would be utilized in the development of spherical diaphragm tanks for the Apollo mission/ullage attitude control system.

VIII. RE-ENTRY VEHICLE ATTITUDE CONTROL SYSTEM

A. System Requirements

The auxiliary propulsion system described herein is designed to provide attitude control thrust for the specific purpose of correcting the attitude of the re-entry vehicle after mission module separation to insure proper approach to the atmosphere before the re-entrant phase. It must be capable of operating with maximum efficiency in a space environment, as well as in the transitional interval which will occur before the capsule sinks into the outer atmosphere to a point where sufficient air density is present to permit use of aerodynamic control surfaces to provide attitude correctional forces. The system must demonstrate sufficient capabilities in the combined areas of performance, cycle life, radiation resistance and minimization of leakage. The possibility of meteorite damage to the system must be taken into account, as well as the consequence of such an occurrence on the safety of the system, to insure the highest possible reliability essential to manned flight.

The attitude adjustment system includes fuel and oxidizer propellant tanks, pressurization gas container, bipropellant thrust chambers, valves, controls and associated tubing and component mounting hardware. The packaging and assembly of the system components provides an efficient, compact, self-contained and functional unit. The thrust chambers are mounted in recessed containers which are in turn installed in the capsule skin in such a manner that the integrity of the hermetical seal of the re-entrant module cabin shall not be imperiled. The propellant tanks, situated in the cabin proper, are encapsulated by a second tank and the space between the two is vented overboard to prevent propellant leakage into the cabin. All oxidizer lines within the cabin area shall be similarly shrouded and vented for the same reason.

The system, so installed, will be capable of operating in a pulsing mode consisting of the pulses of various frequencies and durations. Electrical control impulses may be generated either manually or by a guidance system.

B. Propellant Selection

The properties and characteristics of propellant combinations which would lend themselves most advantageously to the system requirements under consideration are that they should be:

1. Hypergolic and Non-Cryogenic

These contribute to the simplicity of the system by eliminating the need for mechanical ignition devices and protective insulation.

2. Storable

Mission requirements make it necessary for the system to be maintained in constantly operable state over an extended period of time.

3. Capable of High Specific Impulse

This property minimizes weight and tank sizes in the capsule where space is at a premium.

The propellant combination, mixed oxides of nitrogen and UDMH, proposed for the mission attitude control system meets these requirements and, to simplify ground handling and servicing, these propellants are also proposed for the re-entry vehicle system.

While the oxidizer has a comparatively high freezing point (-11°F), this does not appear to be a limiting factor in this application since the tanks are located in the cabin where temperatures well above this point will be maintained. Detailed performance analysis of the selected combination is presented in Section VII-B-5 of the mission attitude control system.

C. System Description

The limiting design factors which control the installation and dictate the general locations of the thrust chamber assemblies in the modified lentricular re-entry vehicle are:

1. The unavailability of the ablative surface on the under side of the module.

2. The hermetically sealed cabin of honeycomb construction.

The first limitation precludes the possibility of employing the attitude control thrust chambers in force couples. The alternative is to install all the units in a vertically upward position and to employ them as unbalanced forces to attain the desired attitude. The second limiting factor requires that the thrust chambers be installed in recessed compartments which, when built into the skin structure, must not compromise the integrity of the hermetically sealed cabin nor weaken the adjacent structure.

Figure VIII-1 depicts the system in schematic form. The propellants and pressurizing gas are contained in spherical tanks which affords maximum weight savings. These tanks are jacketed with suitable overboard vents of the jacketed space to prevent the introduction of propellants into the cabin area should a leak occur. The oxidizer lines to all thrust chambers are routed along the cabin walls and are also jacketed with adequate overboard venting.

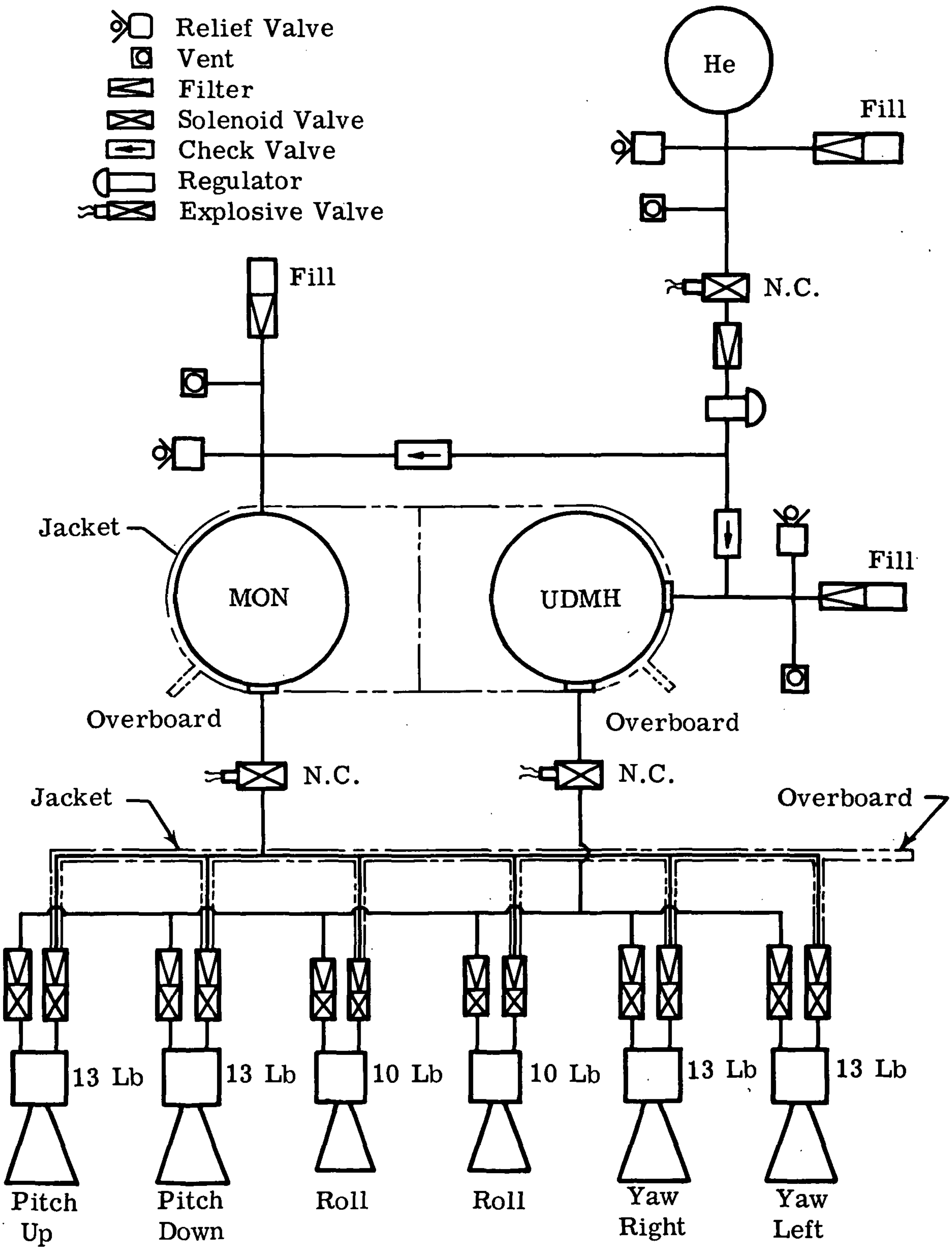


Figure VIII-1. Re-entry Vehicle Attitude Control System Schematic

The pressurization of the system is accomplished with stored high-pressure nitrogen gas regulated to attain proper tank pressure. This high-pressure system is equipped with suitable venting, filtering and pressure relieving devices to afford maximum protection.

The thrust chambers are actuated by solenoid valves incorporating filters at the inlet to protect valve seats and injection orifices from contamination or plugging.

Additional system data is presented in Table VIII-1. A detailed weight breakdown is contained in Table VIII-2.

D. Operation

Propellants and pressurization gas are loaded through their respective fill ports through filters incorporated in these devices. Arming of the system is accomplished by energizing the squib actuated gas start valve allowing gas to flow from the high-pressure storage system through the start valve and filter to the tank pressure regulator. This device regulates the gas pressure to the required tank pressure of 235 psia. With the propellant tank pressurized, the squib actuated valves in the propellant lines downstream of the tanks are energized. This allows pressurized propellants to fill the system to the thrust chamber propellant valves and completes the arming of the system. Since some gas is entrapped in the lines between tank and propellant valves, provisions are made to energize all propellant valves simultaneously for a very short interval to bleed the system of this trapped gas. Bleeding the system by this method eliminates the need to perform this complex procedure prior to flight; it also reduces, by a minimum of six, the number of attachment points where leaks may occur and results in the complete isolation of both pressurizing gas and propellants until actual system operation.

Following the "bleed" burst, the system is ready to perform its designed function with the thrust chambers being capable of any pulse mode necessary to attain proper re-entry attitude.

E Components

The re-entry vehicle attitude control system contains essentially the same components as the mission attitude control system as described in Section VII. Thrust chambers are the ten-pound units as depicted in Figure VII-14. The thirteen-pound thrust chamber for yaw has a modified throat and is otherwise identical with the ten-pound thrust chamber for pitch and yaw. Performance ratings shown in Section VII for the ten-pound thrust chamber also apply to the thrust chambers of this system.

TABLE VIII - 1

SYSTEM PERFORMANCE

Estimated Vehicle Moments of Inertia:

I_{pitch} = 1900 slug ft^2

I_{yaw} = 2125 slug ft^2

I_{roll} = 1900 slug ft^2

Desired Angular Accelerations Are Estimated to be:

Pitch - 1.5°/sec^2

Yaw - 1.75°/sec^2

Roll - 1.5°/sec^2

Torque:

Pitch - 50 ft lb

Yaw - 65 ft lb

Roll - 50 ft lb

Vehicle Torque Arm:

$\approx$5 ft for all axes

Thrust:

Pitch - 10 lb

Yaw - 13 lb

Roll - 10 lb

Minimum System Specific Impulse	- 260 sec
System Total Impulse	- 7000 lb sec
Minimum Pulse Length, All Thrust Chambers	- 0.050 sec
Maximum Pulse Length, All Thrust Chambers	- 20 sec

TABLE VIII - 2

ESTIMATED WEIGHT
RE-ENTRY VEHICLE ATTITUDE CONTROL SYSTEM

	No. Req.	Total Wt. lbs
Thrust Chamber Assembly - 10 lb (Incl. Solenoid)	2	3.0
Thrust Chamber Assembly - 13 lb (Incl. Solenoid)	4	7.0
Propellant System		(23.0)
Tank - Fuel	1	2.9
Tank - Oxidizer	1	3.2
Fill Valve Incl. Filter	2	.2
Relief Valve	2	.4
Squib Valves (N.C.)	2	.8
Jacket for Tanks	1	5.0
Plumbing and Fittings	AR	7.0
Supports	AR	3.5
Pressure System		(5.8)
Tank - N_2	1	1.5
Check Valve	2	.7
Filter	1	.3
Start Valve	1	.6
Regulator	1	1.0
Fill, Vent and Filter Unit	1	.4
Relief Valve	1	.3
Plumbing and Fit.	AR	.7
Supports	AR	.3
Electrical System		(1.7)
Control Box	1	.7
Wiring	AR	1.0
TOTAL SYSTEM - DRY		(40.5)
Oxidizer - MON (Loadable)		20.55
Fuel - UDMH (Loadable)		9.12
Gas - (N_2)		.8
TOTAL SYSTEM - LOADED		(70.97)

IX. WEIGHT BREAKDOWN

A. 1963 Version

The complete estimated weight breakdown for the 1963 version of the propulsion module is presented in Table IX-1. The total propulsion module weight loaded (including all structure, attitude control system) is 9461 pounds. Additional component weight information is contained in Sections VI-B-1 and VII.

B. 1966 Version

Weight reduction programs, advances in the state-of-the-art, and redefinition of some of the main propulsion operating parameters and requirements will provide reduced propulsion module weight for the 1966 spacecraft version. It is the intent of the weight reduction program to retain the basic design parameters and components in order to retain the reliability demonstrated by the earlier version of the propulsion systems. Areas for weight reduction and the estimated weight saving are as follows:

(1) Main engines including valving; use of one set of propellant valves rather than two for the pump-and pressure-fed side; turbine pumps weight reduction/ thrust chamber weight reduction; gimbal system weight reduction. 40 lb

(2) Tank insulation; advances in the state-of-the-art, repositioning of solar collector and heat dissipation surfaces providing a lower average propulsion module temperature. 50 lb

(3) Reduction of thrust level for pressure-fed main engine firings from 4000 pounds to approximately 2500 pounds; reducing high pressure tank weight; helium and helium tankage weight; valves in pressure-fed firing circuit. 30 lb

(4) Utilization of pump bleed propellants for useful velocity change through the addition of thrust chamber devices to combust the bleed propellants. 45 lb

(5) Use of bleed propellant in (4) above to reduce ullage propellant in the mission attitude control system; state-of-the-art advances in cryogenic bladders reducing backup ullage rocket propellants. 40 lb

(6) Reduction of attitude control system thrust chamber chamber pressure; reducing MON and UDMH tank weight and N_2 pressurizing circuits' weight. 50 lb

(7)	Propulsion module: reduction in support structure weight through optimization and advances in the state-of-the-art for materials fabrication.	75 lb
(8)	Reduction in low pressure (pump fed) tankage helium requirements through (2) and pump development to operate at lower NPSH	40 lb
	TOTAL	370 lb

The total propulsion module weight loaded, including the mission attitude system, support structure, 10% propellant reserves, etc., is estimated as 9091 pounds for the 1966 version.

TABLE IX-1

ESTIMATED WEIGHT

		Req. /Sys.	Total Wt. lbs
Thrust Chamber Assembly		2	(250.0)
Propellant System			(899.0)
Low Pressure			(301.7)
Tank - H_2 (Incl. Insulation)	60" ID	2	191.2
Tank - F_2 (Incl. Insulation)	37.6" ID	4	104.0
Valves		AR	6.5
High Pressure			(62.3)
Tank - H_2 (Incl. Insulation)	27.6" ID	2	45.6
Tank - F_2 (Incl. Insulation)	24.5" ID	1	11.3
Valves		AR	5.4
Plumbing		AR	110.0
Supports (Preliminary Estimate). (Includes propulsion module structure, tank supports, thrust chamber supports; does not include separation structure)		AR	425.0
Pressurization System			(126.9)
Low, High Pressure - Propellant System			(98.5)
Tank - He	24.5" ID	1	47.0
Tank - He	14.1" ID	1	24.0
Valves		AR	27.5
Valve Operation - Control Valves			(6.4)
Tank - He	6.5" ID	1	2.5
Valves		AR	3.9
Plumbing		AR	15.0
Supports		AR	7.0

TABLE IX-1
(Continued)

	Req. /Sys.	Total Wt. lbs
Control Valves		(114.7)
Turbine Pump Valves	AR	39.2
Thrust Chamber Valves	AR	60.5
Plumbing	AR	15.0
Turbine Pump Assembly		(104.0)
Drive	2	100.0
Gas Generator	2	4.0
Engine Mount - Universal	2	(14.0)
Engine Installation		(72.4)
Electrical Wiring and Plugs	2	8.0
Control Box	2	8.0
Exhaust Duct - Turbine Pump	2	15.4
Actuators - Elect. Mechanical (Incl. Amplifier)	4	36.0
Oxidizer Line - Turbine Pump to Thrust Chamber	2	2.5
Fuel Line - Turbine Pump to Thrust Chamber	2	2.5
TOTAL PROPULSION AND CONTROL SYSTEM - DRY		(1581.0)

TABLE IX-1
(Continued)

	Req. /Sys.	Total Wt. lbs
Loadable		(6784.9)
Low Pressure System		(6313.0)
H_2 60 in. ID	2	531.0
F_2 37.6 in. ID	4	5782.0
High Pressure System		(456.3)
H_2 27.6 in. ID	2	39.5
F_2 24.5 in. ID	1	370.8
He Tanks		(61.6)
He 24.5 in. ID	1	55.5
He 14.1 in. ID	1	4.6
He 6.5 in. ID		1.5
TOTAL MAIN PROPULSION SYSTEM - LOADED		8365.9
Mission Ullage/Attitude Control System		507.0
Propulsion Module Skin Structure		563.0
Instrumentation Pickups		25.0
Total Weight Propulsion Module (1963)		9460.9

X PROPULSION MODULE WEIGHT VS. VELOCITY INCREMENT

Figure X-1 presents the weight of the complete propulsion module as a function of the total velocity increment capability at various spacecraft total weights. The propulsion module weight reflects values presented for the 1966 version and includes all structure, skin, propellants, supports, tanks, mission ullage/attitude control system, thrust chambers etc., but does not include the weight of the solar collector package. The solar collector package is in the "space not available" area shown in Figure III-1. In addition, the weight of usable propellants is based on the required velocity increment with zero reserve supply.

Figure X-1 was constructed as follows:

1. Total Fixed Weight (1966 Version) W_1

Thrust Chamber Assemblies	515.1
Valves, Low and High Pressure Tankage Valves, Helium Systems' Valves, He Supply Tank for Valve Operation	45.8
Propellant lost on Shutdown, Pump Fed Operation	23.4
Mission Ullage and Attitude Control System	417.0
Structure Weight 0.5 (350)	175.0
Instrumentation Pick-ups	25.0
Skin Weight	563.0
Plumbing for Propellant Tanks and He Systems	125.0
W_1 =	1889.3

2. Total Usable Propellants W_{pu}

W_{pu} = Usable Propellants + 10% Reserves Table VI-2

= 6566 Lbs.

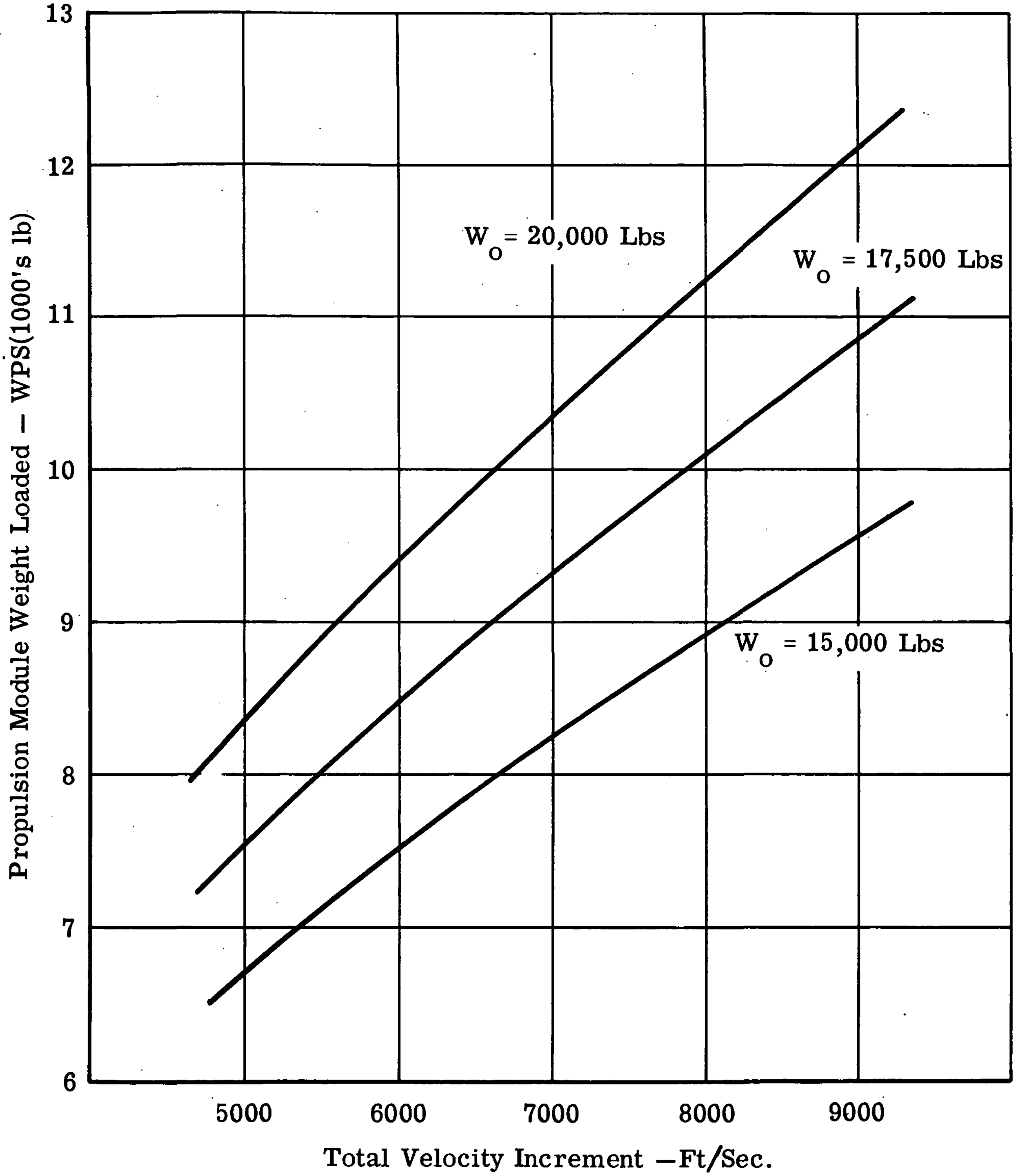


Figure X-1. Propulsion Module Weight Versus Total Velocity Increment Required (1966 Version)

3. Propulsion Module Weight Loaded, WPS = 9090.9

   WPS - W_1 = 7201.6

4. Ratio, R, Variable Weight to Usable Propellant,

$$R = \frac{7939.6}{W_{pu}} = 1.097$$

5. New Usable Propellant, W_{pu}'

$$W_{pu}' = W_o - \frac{W_o}{\text{anti ln } \left(\frac{\Delta V}{I_{sp}\ g} \right)}$$

where: W_o = new total vehicle weight at 3rd stage burnout

ΔV = velocity increment

$I_{sp}\ g$ = average specific impulse 446 seconds x 32.2 ft/sec^2

6. WPS' for Figure X-1

$$= R\ W_{pu}' + W_1$$

$$= 1.097\ W_{pu}' + 1889.3$$

based on an estimated direct proportionality between usable propellants and the weight of tankage, insulation, residual propellants, mismatch, and 50% of module structure and tank support weight.

XI RELIABILITY ANALYSIS

A. MAIN PROPULSION SYSTEM

An analysis has been conducted on the proposed system for the purpose of determining an optium trade off on reliability and safety consistent with the Apollo mission requirements.

The probability of completing the mission is $R_M = R_E \cdot R_{U/A}$ where R_E is the reliability of the main propulsion system and $R_{U/A}$ is the reliability of the Ullage/Attitude Control. The following discussion will first, present the reliability and safety of the engine, second, derive the reliability of the Ullage/Attitude Control, and third, combine the two for computing Apollo mission reliability.

To determine the above reliability of the main propulsion system the schematic illustrated in Figure VI-1 has been orientated into a reliability block diagram for clarity and to pin point the interaction of subsystem redundancies during various phases of the mission. Table XI-1, "Estimated Failure Rates of Engine Components", presents a detailed breakdown of the components that make up the various subsystems in the reliability block diagram and contains the summation of the failure rates applicable to each subsystem. This table has been designed to illustrate the estimated failure rates for all components based on Bell Aerosystems Company's extensive experience and historical data on rocket engines and attitude control systems and to present those components whose failure rate is a function of cycles imposed upon them and those components whose failure rate is independent of cycles. As an example, the Helium tank failure rate is independent of the number of starts required by the mission. This is, basically, because its failure rate is a function of time, not cycles. On the other hand, the propellant valve failure rate is a function of the number of cycles imposed upon it because each cycle (un-seat, flow, reseat) will make the valve less reliable than the first cycle. This cycle failure rate is contained in each "block" of the reliability block diagram, preceded by the (x) multiplier, which is the number of starts for each phase of the mission.

The mission phases and method of completing them are as follows:

Phase I - "From booster burnout to prior to Lunar Orbit". This is accomplished by 5 starts of the pressure side of the engine and failure of this side constitutes an aborted mission. The safety involved in returning to earth therefore, is the reliability of the pump fed side of the engine.

<u>Phase II</u> - "Lunar Orbit". The 2 starts of the pump fed side and 1 start of the pressure fed side of the engine can only accomplish this phase. If a failure occurs, the mission is aborted. The safety here is a function of the reliability of the pressure fed side for the return to earth if the orbit is not attempted. Once committed to orbit however, failure will constitute an unsafe condition and complete failure of the mission.

<u>Phase III</u> - "Lunar De-orbit". Only the 2 starts of the pump fed side of the engine can accomplish this. Failure during this phase will constitute an unsafe condition and complete failure of the mission.

<u>Phase IV</u> - "Return to Earth". Like Phase I, the pressure fed side of the engine will accomplish this in 5 starts. However, the failure of this side of the engine will not affect the mission since the pump fed side can complete the mission. This complete redundancy enhances safety to the extent that both sides must fail to affect the safety of the crew.

Figure XI-1 presents the reliability block diagram during Phase I. It should be noted that a redundancy exists in the Valve Actuation Pressure block through the use of a Normally Open and a Normally Closed explosive valves crossover network. In the event the Valve Actuation Pressure (Block 5B) should fail, the system would be isolated by actuating the normally open explosive valve (Block 5A) and a supply of pressure would be furnished the 3-Way Solenoid valves by the Helium pressure located in the low pressure tanks subsystem (Block 1) and regulation from applicable components in the propellant feed subsystem (Block 6). Utilizing the failure rate in each applicable block, the reliability for Phase I with five (5) starts is computed to be block

$$1 + 2 + (3A + 4A)\,(3B + 4B) + (1 + \text{Applicable } 6 + 5a)(5b) = 237 \times 10^{-6}$$

or 4219 missions between failure.

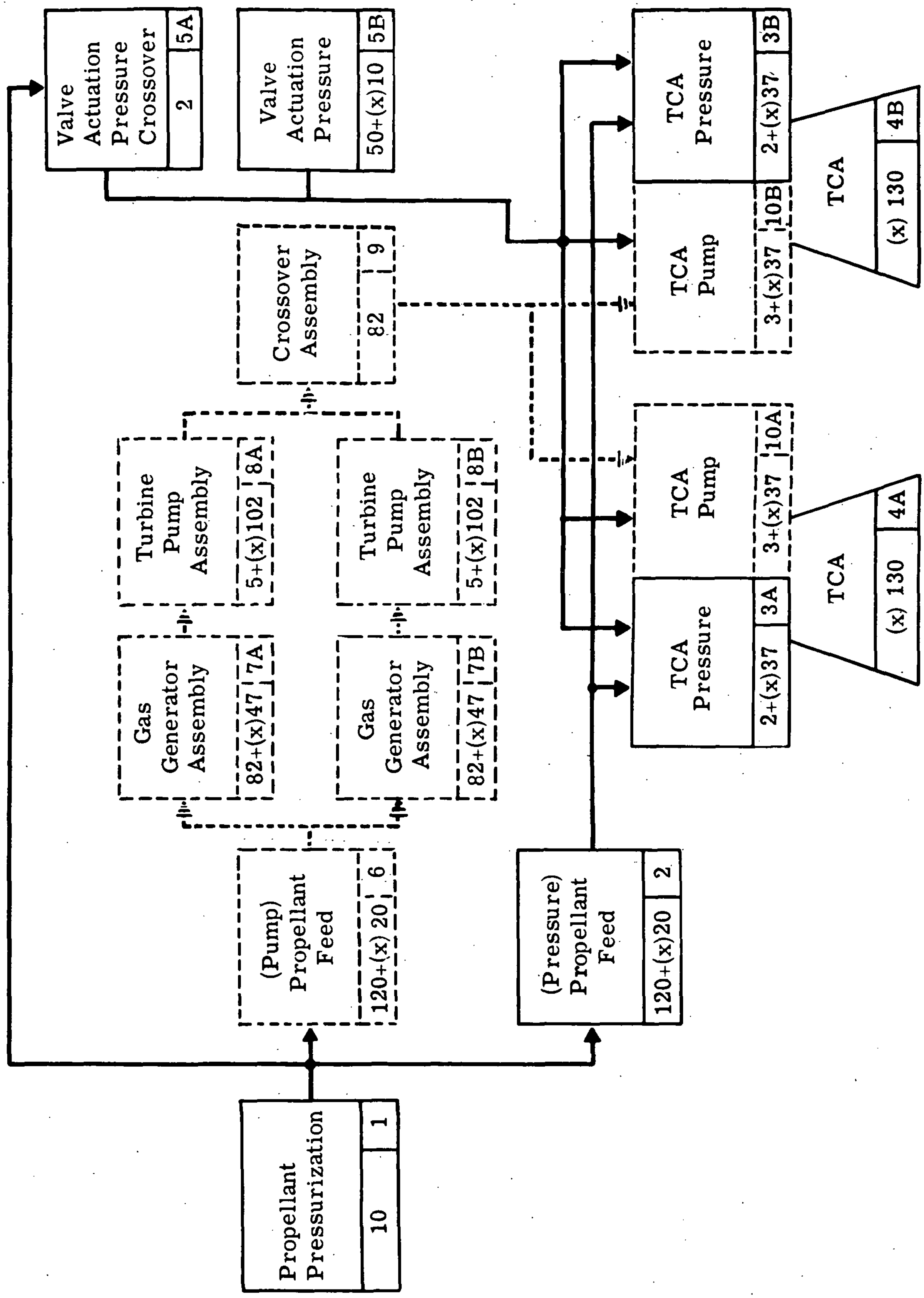


Figure XI-1. Phase I Reliability Block Diagram

During Phase II, as illustrated in Figure XI-2, the reliability for two (2) starts of the pump fed side and one (1) start of the pressure fed side without double counting mission dependent failure rates from Phase I is block.

$$1 + 2 + (3A + 4A)(3B + 4B) + 6 + (7A + 8A)(7B + 7B) + 9 +$$

$$(1 + \text{Applicable } 6 + 5a)(5b) + (10A + 4A)(10B + 4B) = 265 \times 10^{-6}$$

or 3774 missions between failure.

Since only the pump fed side can de-orbit Apollo, Figure XI-3 reflects this case and presents the failure rates without double counting mission dependent failure rates from the previous phases. The estimated reliability for this phase of the mission is block

$$1 + 6 + (7A + 8A)(7B + 8B) + 9 + (1 + \text{Applicable } 6 + 5a)(5b) +$$

$$(10A + 4A)(10B + 4B) = 40 \times 10^{-6}$$

or 25,000 missions between failure.

The last phase, the return to earth, is contained in Figure XI-4. This reliability block diagram illustrates complete redundancy of the pump and pressure fed side and disallows the mission dependent failures that have been accounted for in previous phases. Phase IV reliability is computed to be block

$$1 + 6 + (7A + 8A)(7B + 8B) + 9 \quad 2 + (1 + \text{Applicable } 6 + 5a)(5b)$$

$$(3A)(10A) + 4A \qquad (3B)(10B) + 4B = 5 \times 10^{-6}$$

or 200,000 missions between failure.

Therefore, the total failure rate for the four phases are summarized below.

	Estimated Failure Rate
Phase I	237×10^{-6}
Phase II	265×10^{-6}
Phase III	40×10^{-6}
Phase IV	5×10^{-6}
Total Mission	547×10^{-6}

(1828 Mission Between Failure)

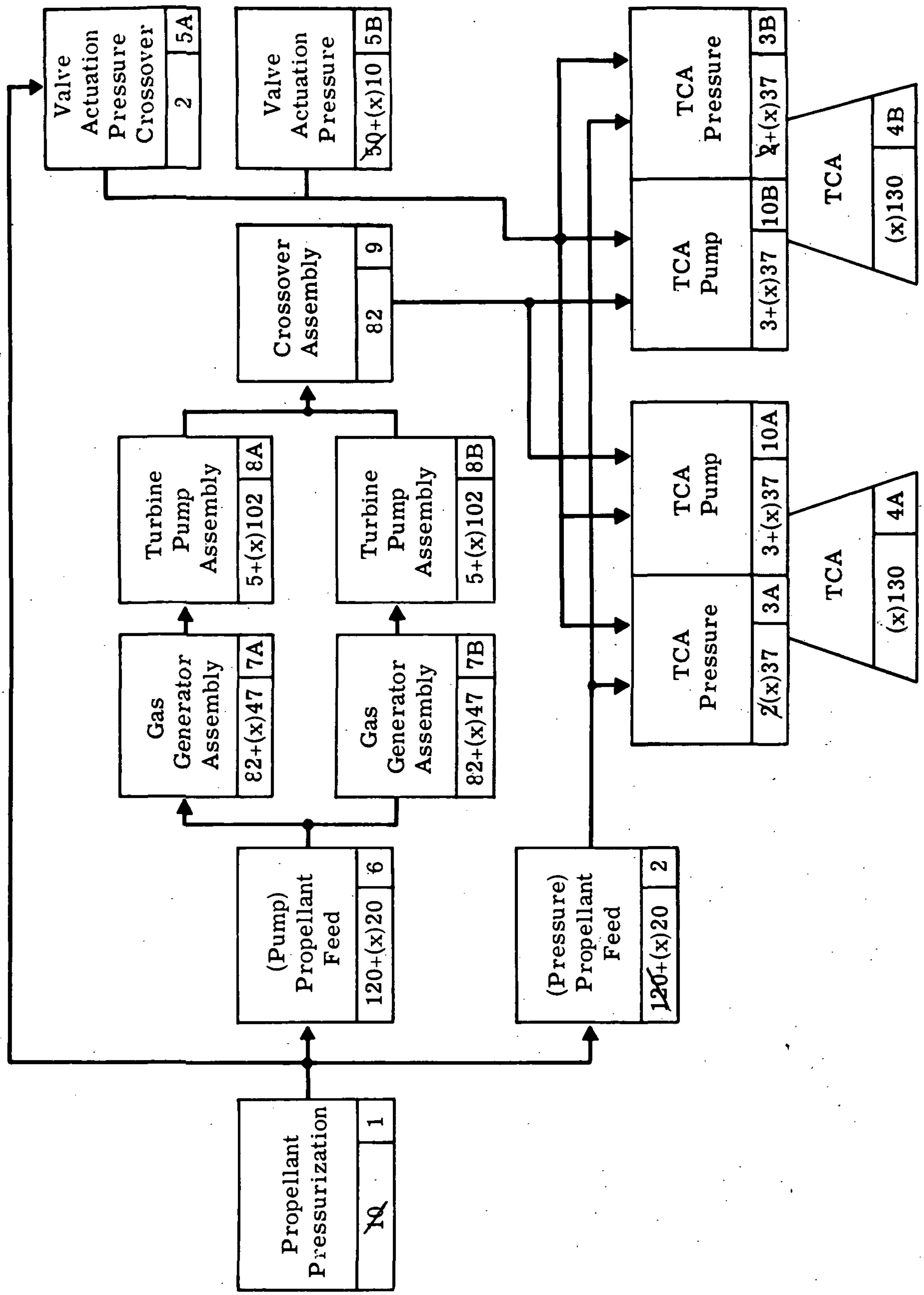


Figure XI-2. Phase II Reliability Block Diagram

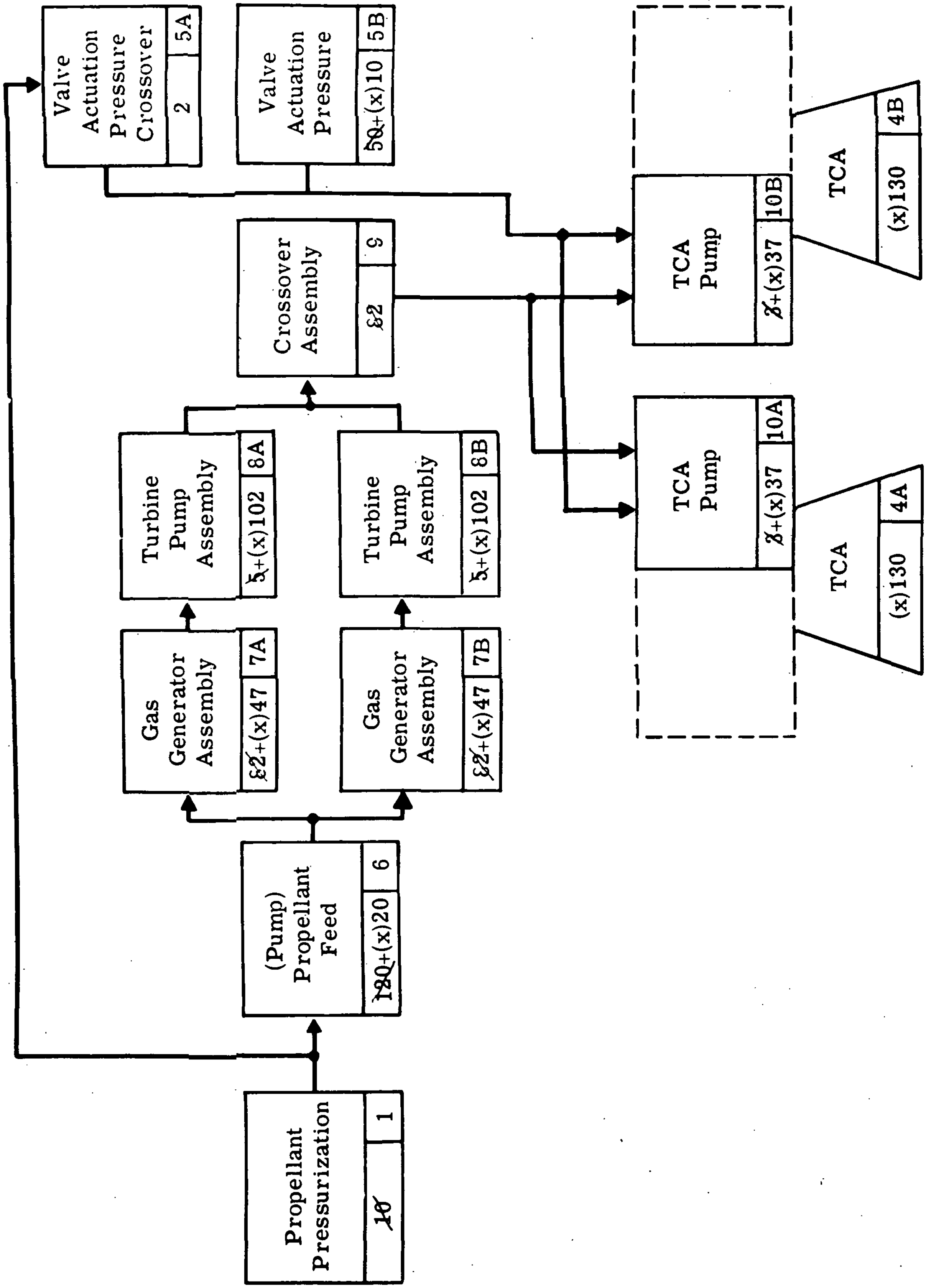


Figure XI-3. Phase III Reliability Block Diagram

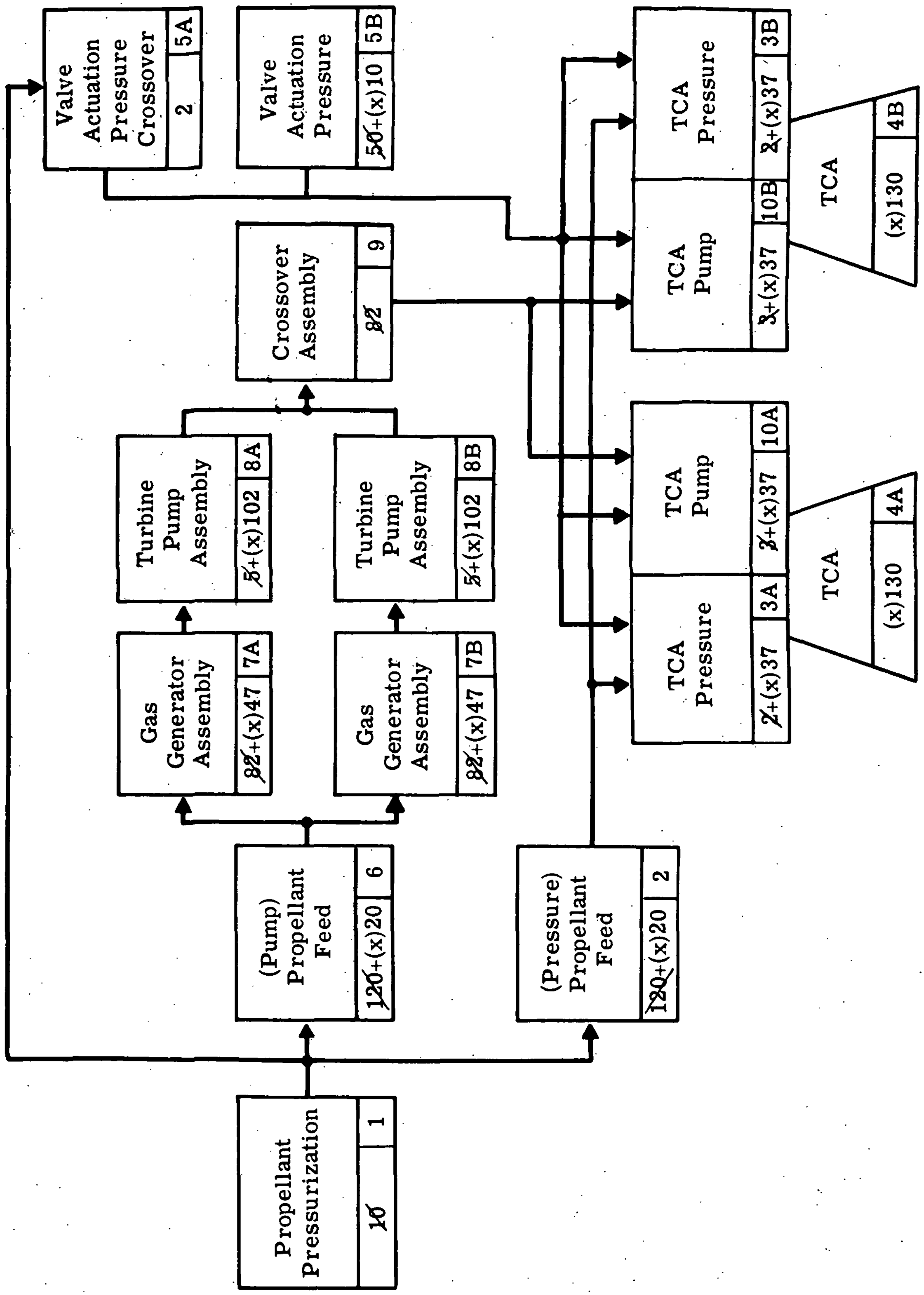


Figure XI-4. Phase IV Reliability Block Diagram

The 1828 missions between failure reliability has been estimated with the mission defined in Phase I through IV. Reliability would be improved if a fewer number of cycles are utilized to complete the mission, and conversely, reliability would be degraded if more cycles were utilized to complete the mission.

For any number of cycles to complete the mission, the reliability can be computed by re-working the above calculations and substituting the new number of cycles in the multiplier (X) indicated in the reliability block diagrams.

TABLE XI-1

ESTIMATED FAILURE RATES OF
F_2-H_2 PUMP/PRESSURE FED ENGINE
COMPONENTS

Item	Number Used (n)	Estimated Failure Rate (X 10^{-6}) Mission Dependent (md)	Cycle Dependent (cd)
Propellant Pressurization			
Helium Fill Valve	2	*	-
Helium Tank	2	5	-
Σn (md)		10	
Propellant Feed			
(Pump Side)			
Propellant Fill Valve	2	*	-
Propellant Vent Valve	2	*	-
Propellant Tanks	6	5	-
Helium Filter	2	5	-
Solenoid Start Valve	2	-	3
Pressure Regulator	2	-	7
Relief Valve	2	40	-
Σn (md) or Σn (cd)		120	20
(Pressure Side)			
Propellant Fill Valve	2	*	-
Propellant Vent Valve	2	*	-
Propellant Tank Assembly	3	10	-
Helium Filter	2	5	-
Solenoid Start Valve	2	-	3
Pressure Regulator	2	-	7
Relief Valve	2	40	-
Σn (md) or Σn (cd)		120	20
Valve Actuation Pressure			
Helium Tank	1	5	-
Helium Fill	1	*	-
Relief Valve	1	40	-
Helium Filter	1	5	-
Solenoid Start Valve	1	-	3
Pressure Regulator	1	-	7
Σn (md) or Σn (cd)		50	10

TABLE XI-1 (Cont'd)

ESTIMATED FAILURE RATES OF
F_2-H_2 PUMP/PRESSURE FED ENGINE
COMPONENTS

Item	Number Used (n)	Estimated Failure Rate (X 10^{-6}) Mission Dependent (md)	Cycle Dependent (cd)
Valve Actuation Pressure Crossover Redundancy			
N.O. Explosive Valve	1	1	-
N.C. Explosive Valve	1	1	-
Σn (md)		2	
Turbine Pump Assembly			
Pressure Operated Propellant Valves	2	-	17
3-Way Solenoid	1	-	3
Propellant Pump	2	-	30
Turbine Drive	1	-	5
Exhaust Duct	1	5	-
Σn (md) or Σn (cd)		5	102
Gas Generator Assembly			
Cavitating Venturi	2	1	-
G.G. Propellant Valve	2	-	17
3-Way Solenoid Valve	1	-	3
Check Valve	4	20	-
Gas Generator	1	-	10
Σn (md) or Σn (cd)		82	47
Crossover Assembly			
N.C. Explosive Valve	2	1	-
Check Valve	4	20	-
Σn (md)		82	

TABLE XI-1 (Cont'd)

ESTIMATED FAILURE RATES OF
F_2-H_2 PUMP/PRESSURE FED ENGINE
COMPONENTS

Item	Number Used (n)	Estimated Failure Rate (X 10^{-6}) Mission Dependent (md)	Cycle Dependent (cd)
Thrust Chamber Assembly (Pump Side)			
Cavitating Venturi	2	1	-
Pressure Operated Propellant Valve	2	-	17
3-Way Solenoid Valve	1	-	3
Temperature Sensing Elements	1	1	-
Σn (md) or Σn (cd)		3	37
(Pressure Side)			
Cavitating Venturi	2	1	-
Pressure Operated Propellant Valve	2	-	17
3-Way Solenoid Valve	1	-	3
Σn (md) or Σn (cd)		2	37
(Thrust Chamber Group)			
Pressure Switch	2	-	5
Thrust Chamber	1	-	100
C.G. Seeking Actuator	2	-	10
Σn (cd)		-	130

* Non-functional during mission - Capped and safety-wired after servicing.

B. ULLAGE/ATTITUDE CONTROL

As described in the Ullage/Attitude Control section, this configuration is designed to the highest reliability possible through the use of malfunction detection and explosive crossovers to achieve redundancy in the gas regulation and the thrust chamber assemblies.

This subsystem will be sequenced such that package 1 and 2 will be fired to propellant exhaustion; or approximately one-half of the required cycles, and then package 3 and 4 will complete the mission.

However, if a malfunction occurs, the mission will be aborted and the Apollo vehicle will return to earth utilizing the operating set of Ullage/Attitude Controls and whatever redundancy is avilable from the malfunctioned set. (Refer to Figure VII-2, schematic of the subsystem).

For a standard mission, the Ullage/Attitude Control subsystem will require three thousand (3000) cycles and will be distributed as follows:

	Number of Cycles
Yaw - Set 1	600
Roll - Set 1	600
Pitch - Set 1	298
Ullage - Set 1	2
Ullage - Set 2	2
Pitch - Set 2	298
Roll - Set 2	600
Yaw - Set 2	600
Total Mission	3000

Table XI-2 "Estimated Failure Rates of Ullage/Attitude Control Components", contains the details that are summarized in the "Reliability Block Diagram - Ullage/Attitude Control", Figure XI-5. Utilizing these estimated failure rates and the redundancies described above as indicated in the reliability block diagram, the computed reliability is block

$$1.1 + 5 + (1.2)(2.2) + 1.3 + 6 + (1.4)(2.4) + 2.1 + 2.3 = 1213 \times 10^{-6}$$

or 825 missions between failure.

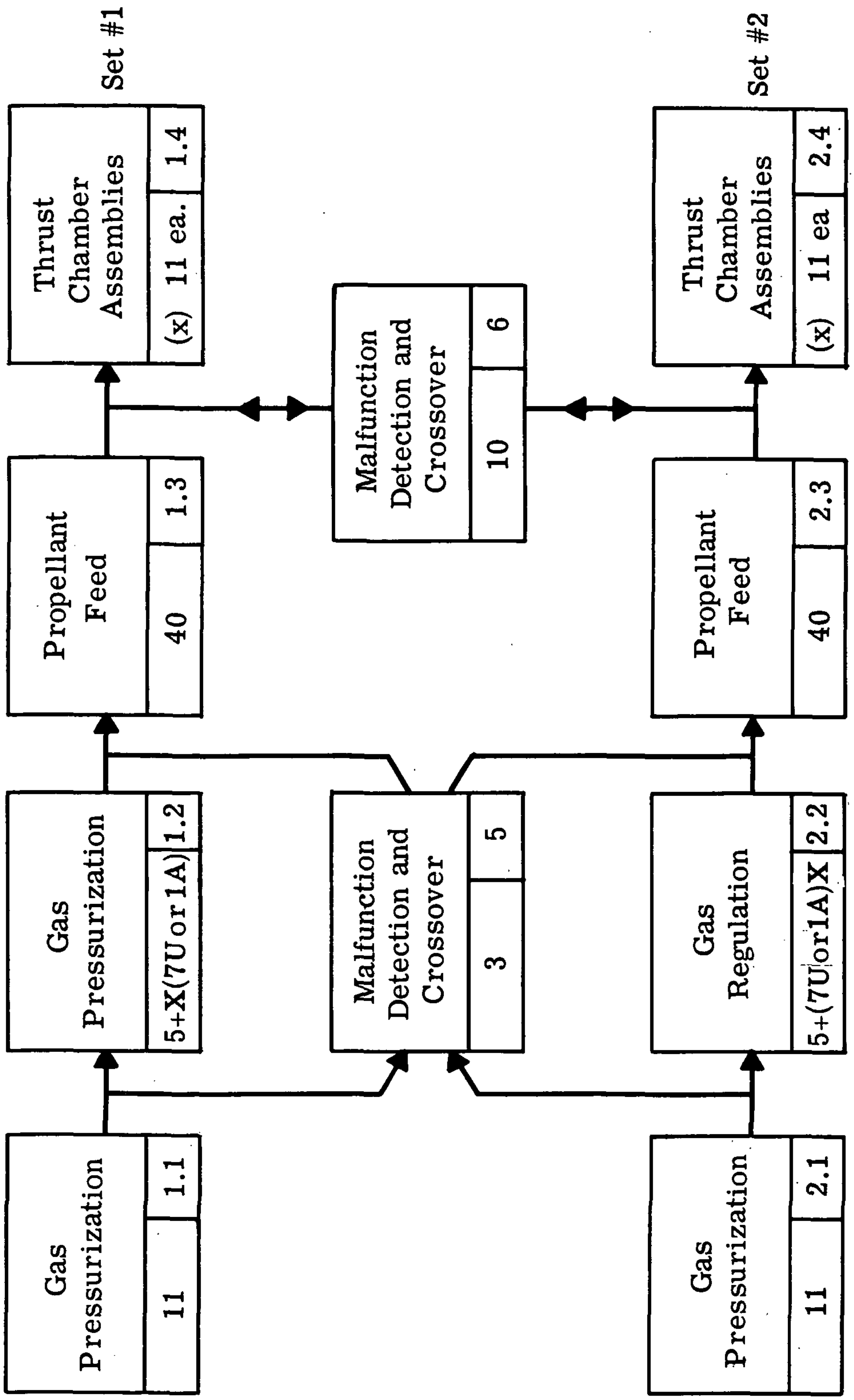


Figure XI-5. Reliability Block Diagram Ullage/Attitude Control

TABLE XI-2

ESTIMATED FAILURE RATES OF
ULLAGE/ATTITUDE CONTROL
COMPONENTS

Item	Number Used (n)	Estimated Failure Rate ($X\ 10^{-6}$) Mission Dependent (md)	Cycle Dependent (cd)
Gas Pressurization			
Gas Tank	2	5	-
Gas Fill-Filter Valve	1	*	-
Check Valve (Set #1 only)	2	1	-
N.C. Explosive Start Valve	1	1	-
Σn (md)		11	
Gas Regulation			
Gas Filter	1	5	-
Gas Pressure Regulator 2	1		
Ullage		-	7
Attitude		-	1
Σn (md) or Σn (cd)		5	7 or 1
Propellant Feed			
Check Valve Assembly	2	1	-
Fill Vent Valve	4	*	-
Pressure Relief Valve	2	1	-
Propellant Tank Assembly	4	10	-
Σn (md)		40	
Malfunction Detection and Crossover - Gas Regulation			
Transducer	2	1	-
N.O. Explosive Valve	1	1	-
Σn (md)		3	

TABLE XI-2 (Cont'd)

ESTIMATED FAILURE RATES OF
ULLAGE/ATTITUDE CONTROL
COMPONENTS

Item	Number Used (n)	Estimated Failure Rate ($X 10^{-6}$) Mission Dependent (md)	Cycle Dependent (cd)
Malfunction Detection and Crossover - Thrust Chambers			
Transducer	2	1	-
N.O. Explosive Valve	8	1	-
$\sum n$ (md)		10	
Thrust Chamber Assemblies			
Thrust Chamber	1 ea	-	5
Filtered Solenoid Prop. Valve	2 ea	-	3
$\sum n$ (cd)			11

* Non-functional during mission - capped and safety wired after servicing.

1 Safety components - used only to prevent catastrophic failures.

2 Regulator cycles are 1:1 for Ullage cycles and 1:7 for Yaw, Pitch, or Roll cycles.

Since safety is in reality identical to reliability, the safety of the crew for a 3000 cycle mission is one complete loss of the Ullage/Attitude Control subsystem every 825 missions.

However, in addition, the safety of the crew is a function of the type of failure and number of cycles remaining with the ability of the operating set to return the crew safely to earth. Since there are 98 components that could fail during four phases of operation and any where from one to three thousand cycles in the two sets of Ullage/Attitude Control subsystem, no attempt will be made to compute the safety of the crew for these 1,176,000 possibilities.

Safety and reliability would be increased in proportion to the reduction in cycle requirements, therefore, an optimum number of cycles should be utilized per mission to enhance reliability and safety.

C. MISSION RELIABILITY

The preceding section on the main propulsion system and the Ullage/Attitude Control system has estimated their respective mean missions between failure as 1828 and 825. Therefore, the probability of completing the mission of 3000 Ullage/Attitude Control starts and 15 F_2-H_2 Pump/Pressure-Fed Engine starts on just this propulsion system is 99.8241%. This reliability indicates that the complete inability of the propulsion to operate would occur once in 569 missions. If committed to do a complete mission as described above, the safety of the crew is such that they would not return once in 569 "flights". If the total number of mission firings were reduced, the safety would be increased. For example, for a mission consisting of 2 pressure-fed, 2 pump-fed and 2 pressure-fed main propulsion system firings and 1500 attitude control firings, the safety increases to 1188 missions between failure. However, since the crew can abort the mission during the various phases, the safety of the crew will be a function of the probability of the "still-operating" equipment to bring them to earth. No attempt has been made to determine the mission safety since there are 1,176,000 possibilities of the Ullage/Attitude Control and 2520 possibilities of the F_2-H_2 Pump/Pressure-Fed engine to "limp home".

D. ALTERNATE SYSTEMS

1. Pressure-Fed

The F_2-H_2 Pump/Pressure-Fed main propulsion system analyzed in the preceding discussion presents a mission reliability of 1828 missions between failure based upon the proposed configuration and phase operation. This combination of pump and pressure-fed operation presents an optimum trade-off in reliability - weight - and performance to complete the Apollo mission. However, from the reliability and safety aspects alone, the following analysis of a pressure-fed engine and a pump-fed engine is presented.

Applying the data contained in the F_2-H_2 Pump/Pressure-Fed system, a deletion and addition of components were made to make the engine a pure pressure-fed system. In addition, two (2) thrust chamber assemblies were added for the "superorbital velocity" requirements. This feature enhances reliability through an additional redundancy feature of the thrust chambers. The deletion consisted of all elements of the pump-fed side and the low pressure helium circuits.

Figure XI-6, "Phase I and IV Reliability Block Diagram - Pressure-Fed Engine", contains the adjusted failure rates ($X\ 10^{-6}$) for each subsystem and the computed reliability for a total of ten (10) midcourse corrections is block

$$1 + 2 + (3)(1 + \text{applicable } 2) + (4A)(4B)(4C)(4D) = 410 \times 10^{-6}$$

or 2439 missions between failure.

The lunar orbit and dis-orbit phase is presented in Figure XI-7. This reliability block diagram disallows mission failure rates accounted for during Phase I and IV and adds the mission failure rate of the Propellant Feed tankage utilized only during this phase of the mission. For the lunar orbit and disorbit phases, consisting of five starts, the reliability is block

$$1 + 2 + 3 + (4A)(4B)(4C)(4D) = 130 \times 10^{-6}$$

or 7692 missions between failure.

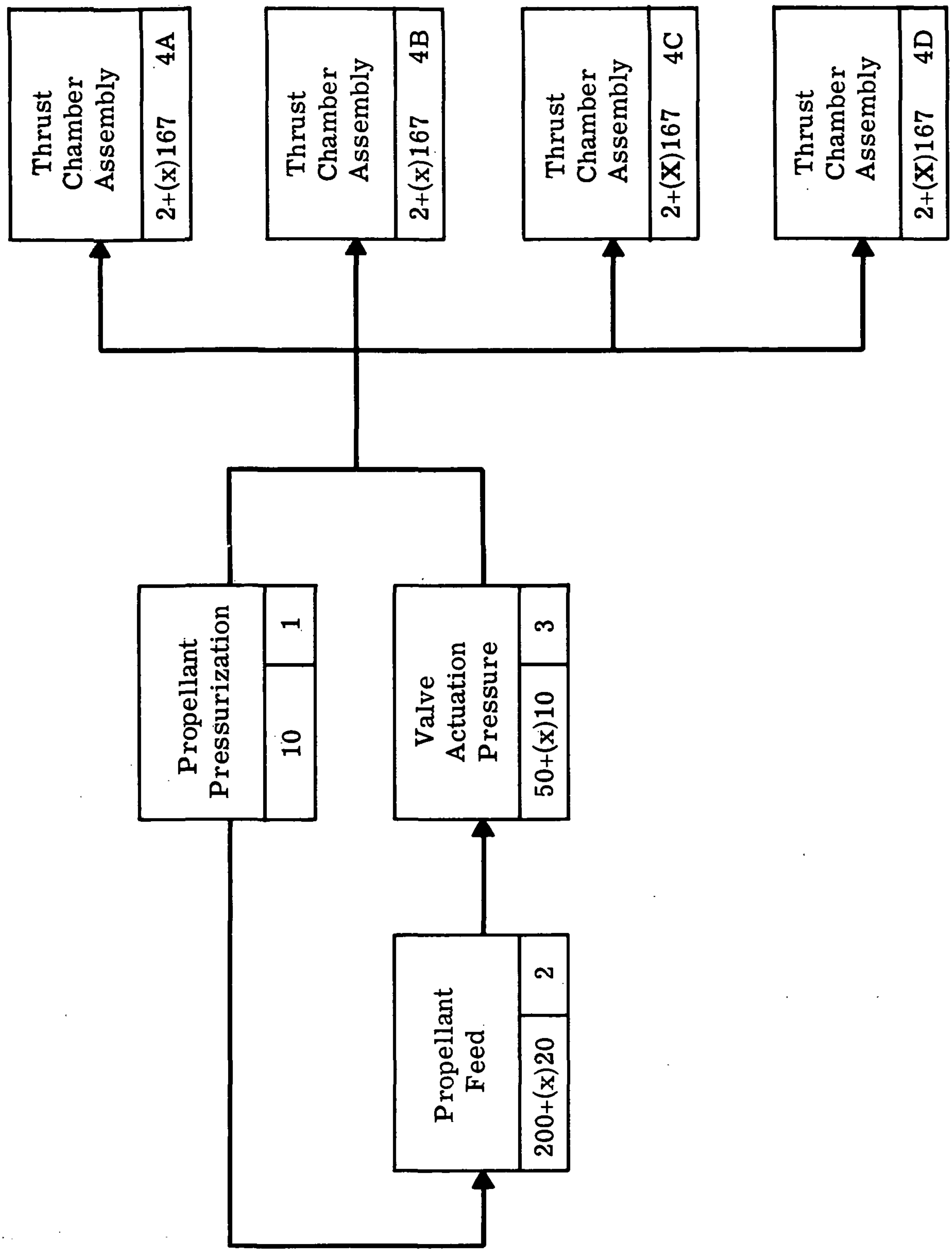


Figure XI-6. Phase I and IV Reliability Block Diagram Pressure Fed Main Propulsion System

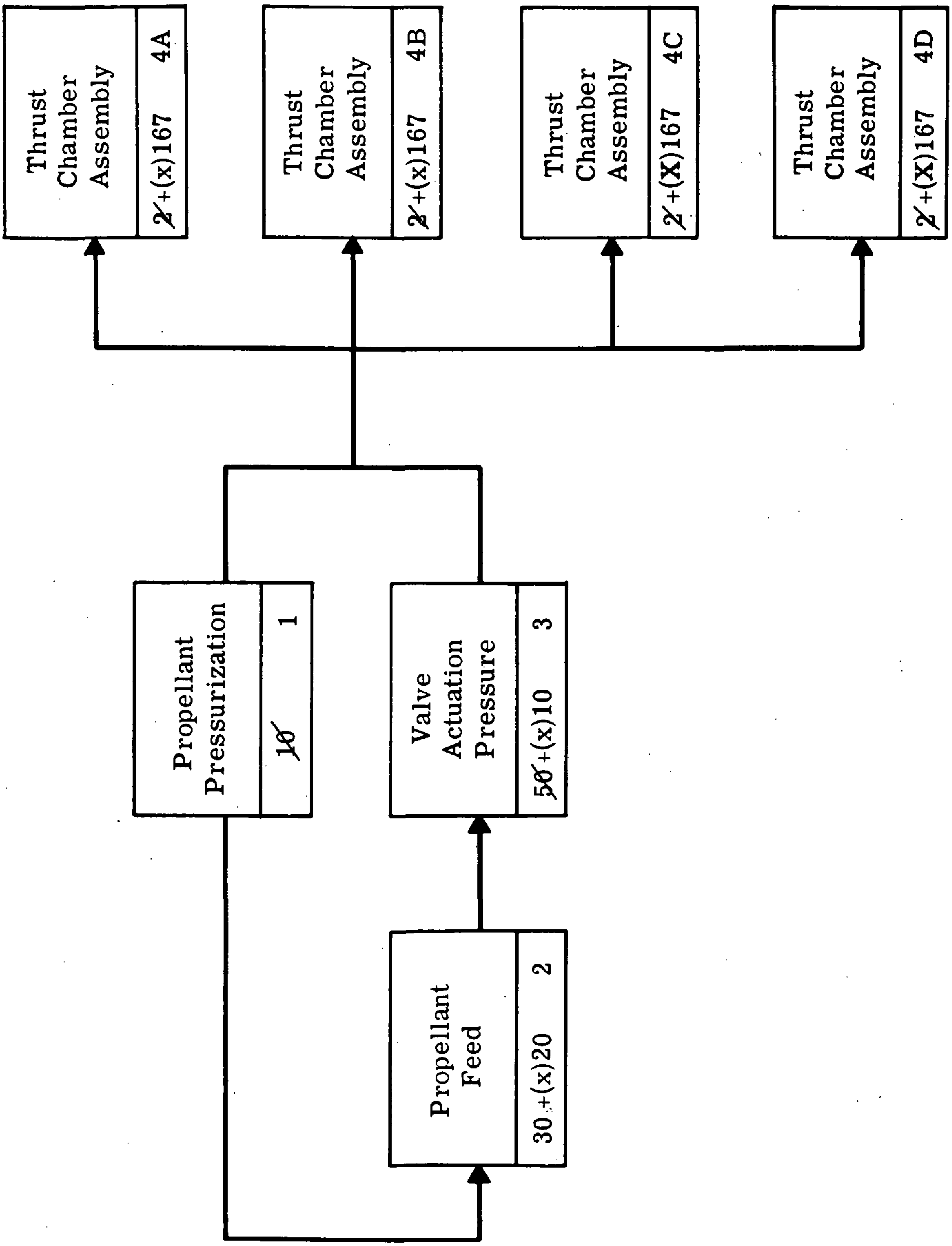


Figure XI-7. Phase II and III Reliability Block Diagram Pressure Fed Engine

The estimated reliability of a pressure-fed engine is as follows:

	Estimated Failure Rate
Phase I and IV	410×10^{-6}
Phase II and III	130×10^{-6}
	540×10^{-6}

This indicates that this system would have a failure in 1852 missions, or approximately a one percent (1%) increase in reliability over the proposed configuration. Therefore, as stated earlier, the F_2-H_2 Pump/Pressure-Fed system is selected because it has the weight advantage over the pressure-fed engine with a negligible change in reliability and safety.

2. Pump-Fed Engine

Figure XI-8, "Reliability Block Diagram - Pump-Fed Engine" reflects the failure rate changes to make the Apollo engine a pure pump-fed engine. Since this pump-fed configuration is identical for all phases of the mission, the estimated reliability for fifteen (15) starts is block

$$1 + 2 + (3A + 4A)(3B + 4B) + 5 + (6A)(6B) + (1 + \text{applicable } 2 + 7A)$$
$$(7B) = 633 \times 10^{-6}$$

or 1580 missions between failure.

The Apollo mission reliability for the pump-fed engine would be 1580 missions between failure or approximately a 14% reduction in reliability and safety. This reduction in reliability and safety is considered to be significant and the F_2-H_2 Pump/Pressure-Fed Engine has been chosen as the more reliable configuration.

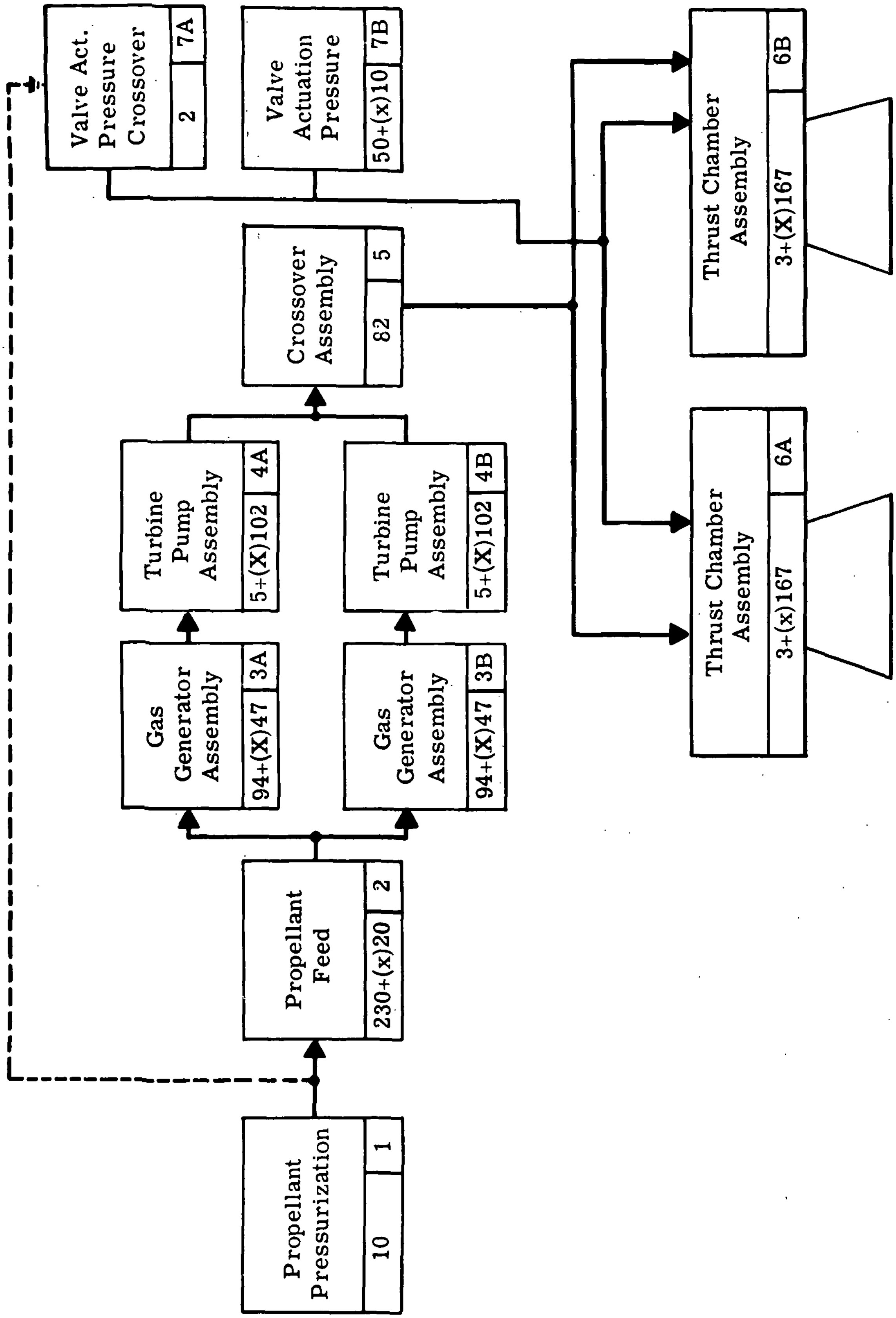


Figure XI-8. Reliability Block Diagram Pump Fed Engine

XII. PROGRAM PLAN

A. MASTER PROGRAM PLAN

Figure XII-1 shows the program plan for the design, development, test, fabrication and delivery of components, subsystems and systems for the Apollo spacecraft. This master program plan has been designed to provide for the most logical and expeditious development of the propulsion system, in keeping with the desired objectives. The milestone dates shown are predicated on the provision of final propulsion system requirements from GE to Bell Aerosystems by the end of the second month. The tooling cycle on the tanks is critical and can be integrated with the overall delivery requirements only if this advanced release is available. Final design selection would follow at the end of the third month, and from then on design releases would be accomplished as shown.

In order to achieve the schedule for delivery of flight article propulsion systems, concurrent efforts in several areas are required. One of the main goals of this plan, therefore, in addition to the design development and delivery requirements, is for strong program integration to control these efforts.

B. TEST PLAN THROUGH 1963

Development of the proposed propulsion system follows a progressive plan of component level testing, integration of components into minor assemblies or subsystems and combination for evaluation and combination of the major elements for system evaluation. Constant communication among all levels of test and design is maintained to insure coordinated efforts. Inherent in development is the evolution of the components and subsystems from "breadboard" or test item to prototype configuration. Past experience dictates that the required system configuration and conditions be simulated in all tests as accurately as possible commensurate with the state of development. This overall philosophy of development is applied, as well, to the engines, tankage and propellant system, pressurization systems, and reaction control systems which constitute the propulsion package. The master program schedule reflects the phasing of the major subsystems for integrated testing through the basic stages of breadboard and prototype testing. Some additional outline of the engine, pressurization system, tank and attitude control systems test plans follows. The test programs for the other propulsion system components will be similar, as required.

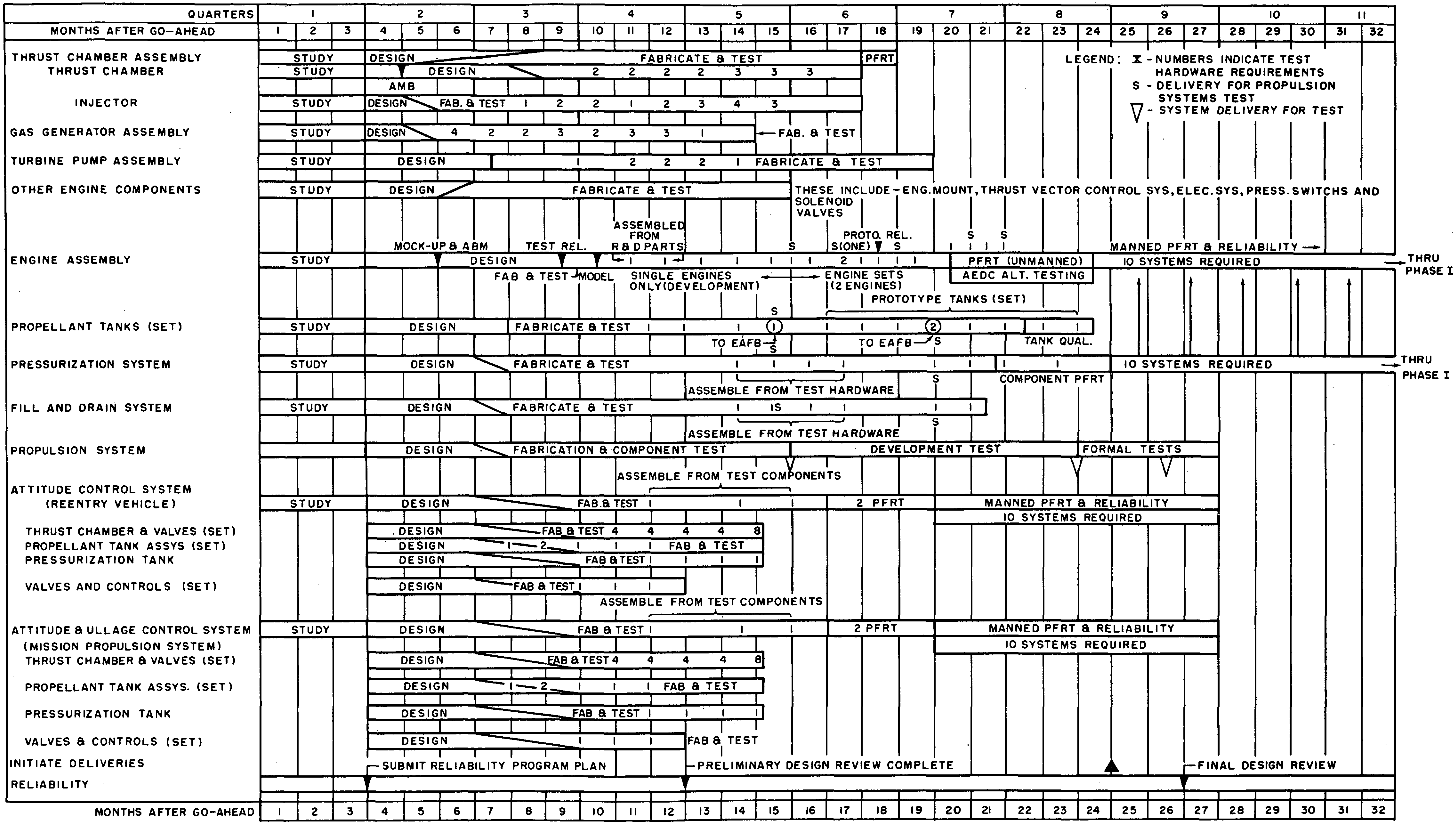


Figure XII-1. Program Schedule

1. Engine Test Plan

The purpose of this plan is to show the development through manned PFRT of a 12,000 pound thrust, multistart, gimballed rocket engine utilizing fluorine and hydrogen propellants. Engineering, manufacturing and testing schedules are shown in the master program plan along with the development schedule depicting the anticipated achievement of significant milestones. The program is based on the definition of the performance, characteristics, configuration and requirements of the engine and its components during the design study and analysis phase conducted in the initial three months of the program.

The manufacturing plan is formulated on the selection of all prototype hardware basically identical to test hardware evaluated early in the program. For example, prototype injector assembly fabrication would progress to the point of final assembly with the exception of the orifice plate detail. The optimum propellant stream configuration, based on initial test results, would then be incorporated with the prototype injector assembly delivery for test with a minimum of lead time.

The test program reflects in summary the following:

(1) Test cell construction and buildup is planned to accommodate the increasing frequency duration and complexity of tests. Existing test cells will be utilized to the greatest possible extent. More detailed facilities plans are included in Appendix 2.

(2) Injector evaluation consisting of short duration (5 and 10 seconds) fire tests in workhorse thrust chambers for the purpose of determining optimum performance stability and heat rejection characteristics begin the test effort on the major components.

(3) Thrust chamber testing consists of water cooled and fuel regeneratively cooled fire tests for the purpose of evaluating performance, heat rejection, durability, and configuration. These tests require longer run durations including full duration firings to insure hardware durability. Vacuum environment tests will be conducted to evaluate thrust chamber operation with full divergent nozzle flow.

(4) Gas generator testing consists of performance and stability evaluation over a mixture ratio range; determination of the effect of chamber pressure on performance and environmental tests.

(5) The turbine pump program is divided as follows: (a) water tests to determine performance and cavitation characteristics, (b) pump seal tests, (c) independent pump and turbine tests, and (d) composite turbine pump tests. The composite test unit is comprised of a turbine drive assembly, gas generator, control system, engine mount and associated propellant valves to simulate engine operation. These tests are planned to evaluate start and shutdown transients to demonstrate multi-start capability and to evaluate the control system and life or durability.

(6) Component (valves and controls) evaluation consists of flow, cycle, temperature, shock and vibration tests for each of the propellant valves; calibration of gas generator venturis, and testing of the thrust vector control (gimbal) system.

(7) Engine development testing is scheduled to begin with short duration tests on a single engine for evaluating control, stability and gimballing. Concurrently with the fire testing, the engine will be subjected to vibration tests. Subsequently dual engine testing will commence where the fire test duration will be increased, eventually to full duration, to prove durability. Altitude environment is required to prevent nozzle separation. Informal PFRT will be conducted during the latter part of the development program as a final evaluation of the development status prior to formal testing.

(8) Preliminary flight rating tests, generally in accordance with MIL-E-6626A2 will be conducted in two steps to demonstrate the capability of the design to meet requirements. Unmanned PFRT will be conducted prior to delivery of preflight systems and will be followed by manned rated PFRT and reliability testing through 1965. Altitude testing will be conducted at the Arnold Engineering Development Center, Tullahoma, Tennessee.

2. Tank Test Plan

The propellant tank static test program has been divided into two parts: (1) detailed tests of materials, structural elements, etc., to establish design information, and (2) static load, pressure and thermal tests on completed tanks to verify the strength and insulating characteristics.

a. Material and Element Tests

Material tests will be conducted, first on a small number of aluminum alloys, to aid in the final alloy selection, and then in more detail on the alloy actually selected. These tests will also be conducted in conjunction with the development of process specifications for various manufacturing operations such as welding and a sufficient number of tests will be run to cover variability in material characteristics and manufacturing operations.

Elemental vacuum insulation tests will be instituted to check insulation conductivity over a range of temperatures and external pressures, to check the load-deflection and deflection recovery characteristics at various external pressures, and to check the analytical predictions of heat conduction through tank supports and lines.

b. Static Load, Pressure and Thermal Tests

Following the completion of design, tooling, and initial fabrication, the first two sets of tanks will be used to verify the structural integrity of the design, and the proper functioning of the insulation system, and to demonstrate that the tanks meet the design criteria. Adequacy of the design will be considered established by satisfactory completion of the following program:

Thermal Test
Pressure Test
Joint Separation Test
Structural Load Test
Pressure Cycling (Fatigue) test

The thermal test to be conducted will evaluate the performance of the thermal insulation shell. Temperature measurements at various points on the shell will permit evaluation of heat path and insulation effectiveness and will serve as a check of heat transfer analysis on the subassembly level. Pressure tests will be conducted to evaluate the tanks structurally as pressure vessels. The primary objective will be to demonstrate the structural adequacy of the designs. After completion of the design pressure tests, the pressures will be increased to establish yield point of the shells and thus determine the upper limits permitted for propellant tank operation.

After determining the proof pressure level for the propellant tanks, pressures will be further increased until local shell failure occurs, as a verification of ultimate strength. To complete the structural integrity checks of the complete tank and propulsion module structure, external load at the engine mount and at the fairing and interstage connections will be applied. It is also necessary to reproduce the lateral acceleration conditions since, due to the asymmetry of this loading, it cannot be reproduced by internal pressure. In each of the preceding structural tests, incremental loading to limit load will be applied.

Fatigue tests will be conducted to simulate the repeated pressurization-depressurization cycles to which the flight configuration tanks will be subjected during checkout, ground tests and flight. Results of the pressure cycling test will verify tank life.

3. Pressurization System

The pressurization system test program is divided into three parts: (1) system component design fabrication and testing, (2) integrated system testing with the main propellant tankage and (3) component preliminary flight rating testing per MIL-E-6626A2. Component evaluation will consist of flow, cycle, temperature, shock and vibration tests. PFRT will consist of electrical and pressure sensitive subcontrols testing.

4. Main Propulsion System Development Tests

The initial testing series in the propulsion system development program will be conducted with the second set of test tanks and a development rocket engine. The tanks for this testing phase are the second set of "boiler-plate" tanks which will be available at the test site as indicated on the master schedule. Build-up time is allowed for installation of the tanks, instrumentation, test pressurization and propellant servicing systems. Initial tests, consisting of pressure checks, water and simulated propellant flow testing are followed by installation of the development rocket engine for hot firing. The configuration of the system for the initial hot firings consists of boiler-plate or tanks, a development rocket engine and test-stand equipment for propellant tank pressurization and propellant servicing. The initial hot firings in this series will be short duration tests - ten seconds to thirty seconds - to evaluate transients during the start-stop phase, the effects of tank pressure, vibration characteristics in the vertical position and engine servicing procedures.

Initial evaluation of the propellant system (fill and drain) and the pressurization components at the system level is to be phased into the water and simulated propellant flow tests being conducted on the first set of "boiler-plate" tanks. These components will be evaluated during the latter part of the program on this tankage.

The second complete set of prototype pressurization and fill and drain system components will be phased into the hot firing test series. Hot firing tests following integration of these prototype components will essentially duplicate those previously conducted - short duration to evaluate transient conditions.

The system testing will continue with progressively longer duration firings, evaluating system operation and performance. Firings with engine gimballing will be accomplished during this period. It should be noted that the design parameters of the prototype combustion chamber preclude full duration firings without the use of an aspirator or aspirating duct to accomplish full nozzle flow. The conditions of nozzle flow separation which will be experienced at sea level are predicted to have an adverse effect on the heat transfer conditions in the nozzle. Gimbal tests with the prototype chamber will, therefore, be limited to 10 seconds maximum in this phase. It is planned to conduct the longer duration gimbal firings using a modified prototype chamber which will have a nozzle expanded to sea level but incorporate a mass to simulate the prototype chamber with full expansion ratio.

The final series of tests will include total impulse firing demonstration, mission simulation, including engine restarts following various coasting duration, simulated acceleration head on the propellants and a series of malfunction tests. This testing phase will effectively explore and establish the extreme limits of the system operating regime. A second development engine and the first prototype engine are made available for the total effort in this development phase.

Qualification of the propulsion system for flight capability will be accomplished on a test program using prototype hardware. The fourth prototype propellant tank set is designated for the preflight test. This tank assembly will have completed inspection and acceptance tests including proof pressure test. A period of time for installation, instrumentation and checkout at the test site is scheduled. System testing in this series is also initiated with water and simulated propellant flow tests to checkout the fill and drain, propellant feed and pressurization systems operation. The system firing schedule for the preflight test program parallels that which was described for the development program above. Initial firings will be short duration, checking start and shutdown transients, effect of ullage volume and varying suction pressures. Twenty-five hot firings are scheduled for the preflight test phase, five to be formal qualification firings. The testing plan includes short duration gimbal firings with the prototype thrust chamber, long duration gimballing tests with the modified chamber, total impulse demonstration, fuel exhaustion shutdown, oxidizer exhaustion shutdown, engine restart tests and simulated acceleration head tests. Two prototype engines and a spare engine, a spare propellant tank set and pressurization system have been allocated for the flight rating demonstration program.

5. Reaction Control Systems Development

The reaction control systems development program which is based on past experience, requires a development effort that is followed by informal PFRT tests on components and systems for the purpose of demonstrating that a satisfactory state of development has been achieved for the successful completion of formal PFRT. The thrust chamber test program of the reaction control systems is almost identical to that of the main propulsion engine. Altitude testing of the thrust chambers will be done at Air Force Plant No. 38 (BTC). The reaction control pressurization subsystem and propellant tanks will be subjected to the same test program as the main propulsion package pressurization system and tankage. Unmanned preliminary flight rating tests will be followed by manned flight rating testing and reliability testing.

C. DEVELOPMENT 1963-1964

A weight reduction program will be initiated in 1963 and extend part way through 1964. The components and materials to develop lighter weight versions include the tank insulation, the high pressure tankage, the helium supply circuit to the high pressure tanks and the mission attitude control system and other components as discussed in Section IX of this report.

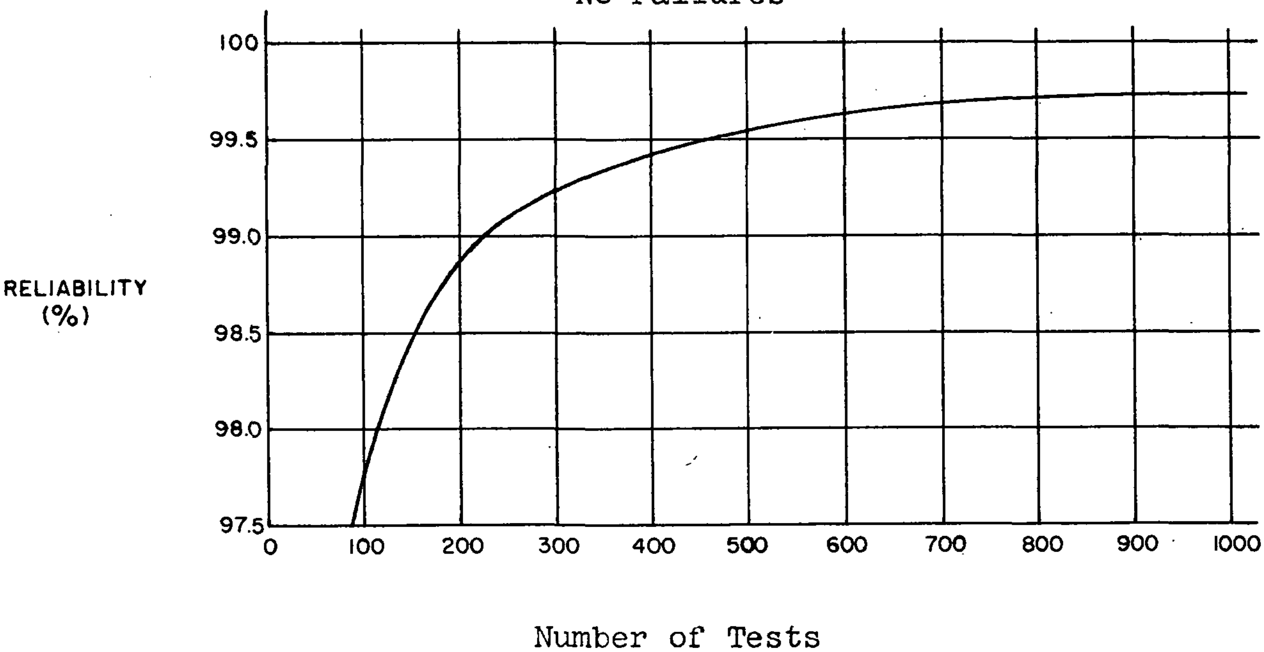
D. RELIABILITY TEST PROGRAM

1. General

The estimated mission reliability of the main propulsion system with the ullage/attitude control system as noted in the Reliability Analysis is 99.82%. For the main propulsion system alone, the estimated reliability is 99.95% for 15 starts; and for the ullage/attitude control system, 99.88% for 3,000 starts.

In developing a reliability test program to demonstrate reliability goals of a system certain basic concepts must be considered. The following diagram depicts the typical number of tests required to demonstrate reliabilities in the event no failures should occur during a reliability demonstration program.

Reliability vs Number of Tests
90% Lower Confidence Level
No Failures



It is rather evident that during the early phases of the test program relative changes in reliability are much greater for a specific number of tests than during later phases of testing. Consideration was given to the advisability of an extended program in order to achieve the demonstration of the extremely high predicted reliability. Further, the level of confidence in the reliability demonstration is a determinant in the selection of number of tests to be planned and is an assurance the reliability demonstrated has an excellent possibility of not being overly optimistic.

The recommended test program has been formulated to demonstrate a mission reliability of 98.90% for the main propulsion without tanks, and the mission ullage/attitude control system, at a 83.6% lower level of confidence as outlined in the following table:

TABLE

Item	Reliability	% LCL	MMBF	Cycles
*	99.95	95	44.64**	2475**
Valve Actuation Pressure	99.98	97.5	5,000	267,620(1)
Cross Over	99.992	97.5	12,195	45,000
Miscellaneous Components	99.996	97.5	(2)	(2)
Ullage/Attitude	99.0	95	100	900,000

* Gas Generator, Turbine Pump, Thrust Chamber
** For one half of the redundant system
(1) Explained in text
(2) Explained in text

The test program would be initiated in the last quarter of 1963.

2. Testing of Main Propulsion System

In order to demonstrate a reliability of 99.90% for the main propulsion system, a test plan has been formulated which will include system testing of those components which may display effects of interaction and supplemental bench testing of separate components for the remaining required tests on these components. The test plan as formulated may be reduced in magnitude by applicable data which may be accumulated through development testing and PFRT. In no event shall data be applied until the design of hardware has been finalized or under any conditions other than those specified by the mission planning. System testing of the main propulsion system during the reliability program will be conducted with a configuration which will simulate the flight system including all hardware with the exception of the tank group and the pressure fed lines to the thrust chambers.

Engineering testing of the tank configuration will be accomplished separately from system testing and will be supported by stress versus strength considerations and by historical data accumulated at both BAC and in industry in general. The tanks which will be used in the reliability demonstration program will have the same volume and form as flight hardware. The pressurization system will be identical to the flight hardware.

All components will be subjected to environmental testing as specified by military standards prior to initiation of the reliability test program.

The redundant systems of gas generator, turbine pump, and thrust chamber and all lines, valves and regulators will be simulated. The redundant character of this system is the basic reason the test program may be achieved economically at this level. The redundant systems will be fired both separately and simultaneously in order to further evaluate the interactions of components in the system and also to reduce the actual span of time for testing. As indicated in Table XI-1, the major component of the main propulsion system having the highest failure rate per mission is the thrust chamber assembly. The test plan, as developed, has a major goal of demonstrating the reliability of the combination of gas generator, turbine pump, thrust chamber and associated lines, valves and regulators. Since the main propulsion system contains completely redundant combinations, in order to demonstrate a reliability of 99.95%, it will only be necessary to demonstrate a reliability of 97.8% for one combination. Hence only 165 missions or 2475 cycles are required. Failure testing, PFRT and testing of the cross-over assembly will be conducted in order to demonstrate the complete redundancy of these combinations.

Each mission has a requirement of 15 cycles which include eleven 2-second bursts and four 30-second bursts. It is anticipated that temperature stabilization will be the major determinant of actual burst length; hence the 30-second firings may possibly be limited to a much shorter duration, as determined during development testing. Further testing of the valve actuation pressure assembly will be necessary to demonstrate the tabulated reliability of 99.98. This will require testing equivalent to 4835 missions or 267,620 cycles [1], a major portion of which will be accomplished through bench testing. Additional testing of the cross-over assembly will be accomplished through bench testing. Further applicable data on this assembly will be accumulated from historical data.

The mean missions to failure and cycles for the tabulated components[2] are explained next. The solenoid start valve and pressure regulator will, as a combination, reflect 50,000 missions between failures, a total of 2,029,500 cycles, each mission requiring a minimum of 11 cycles. Testing will be accomplished by cycling a series of these components an appropriate number of times through bench testing. The relief valve with a mission failure rate of one failure per 12,500 missions has no stated number of cycles per mission as it will be used to provide a safety feature. Therefore, in order to demonstrate this reliability with 97.5% lower confidence level, it is recommended that it be subjected to 46,125 cycles through bench testing.

3. Testing of the Ullage/Attitude Control System

For a standard mission this system will require three thousand cycles and these cycles are distributed over components so arranged to achieve redundancy. However, the reliability testing for the standard mission must be considered here due to the sequence of firing such that packages 1 and 2 will be fired to propellant exhaustion with packages 3 and 4 completing the mission and any malfunction will abort the mission. A mission reliability determination, for this assembly, of 99.0% would again require 300 missions, at a lower confidence level of 95%. Since each mission requirement is 3,000 firings, this would require 900,000 cycles with no mission failures. Of this number sufficient tests would be made at the system level to determine interactions in the system; the remaining testing would be bench testing, or accumulated from historical data. The number of each component type to be used in the reliability test plan will be predicated on the data accumulated during component and system development testing and PFRT. These tests will establish the service life of the system components through durability testing.

It is anticipated the reliability testing will be conducted over a period of two to three years. All failures or discrepancies will be reviewed as to cause and to required corrective action to make the failure nonrepetitive. If major design changes are dictated, a revision of the reliability test program will be instituted at that time. All data accumulated to date, and applicable to the configuration, will be maintained and considered in the revised test program.

4. Reliability of the Propulsion System

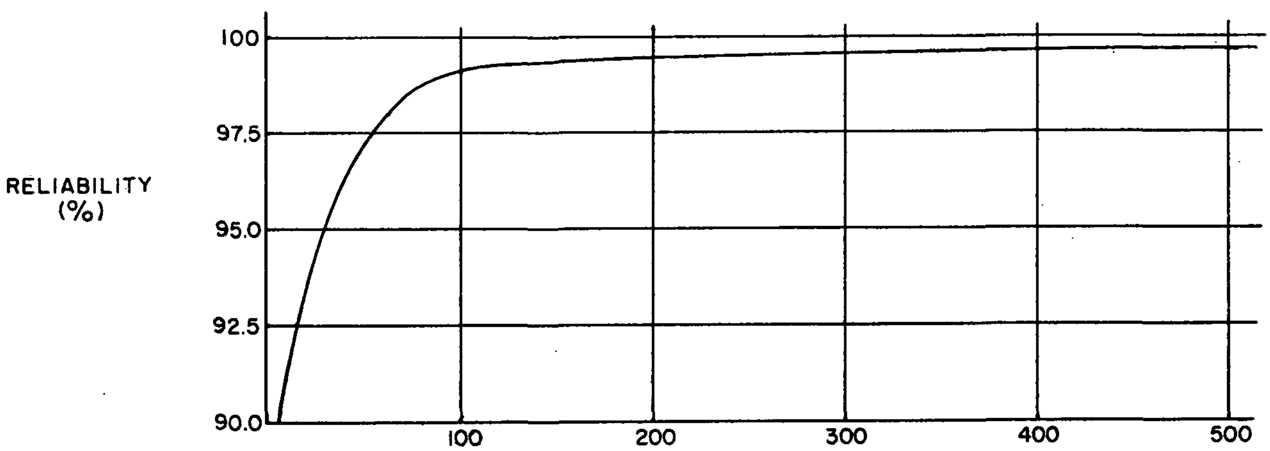
The reliability of the main propulsion system is the product of the reliabilities of each of the components in the system. In like manner the lower confidence level of the system is the product of the confidence levels of the component reliabilities. Therefore, the reliability of the main propulsion system upon successful completion of this reliability demonstration program is 99.9% with a lower confidence level of 88.1%. The mission reliability for the propulsion system, main propulsion system and ullage/attitude control system, is determined in similar manner and is 98.9% with a lower confidence level of 83.7%.

These reliability goals were chosen after due consideration of the weighing of the minor reliability gains which could be made with more extensive testing. It was concluded that the further expenditures in hardware, manpower and time consumed in testing, after this relatively high reliability demonstration had been achieved, would not merit consideration.

5. Propulsion System Reliability Growth

A predicted reliability growth curve for the Apollo propulsion system is shown in the succeeding chart. The milestones were selected from failure rates experienced on applicable projects and the anticipated growth during the reliability demonstration program.

Propulsion System Mission Reliability Growth



Number of Missions

The initial point indicates an estimate of 92.5% reliability at the completion of the PFRT program. Other milestones established are following the reliability demonstration program and an ultimate achievement. The reliability demonstration will provide 98.9% reliability after 80 equivalent missions and the ultimate reliability of 99.8% would be achieved after 500 equivalent missions.

E. BASE HEATING TESTS

As a part of the development plan, a test program will be initiated to evaluate the base heating effects of the main propulsion thrust chambers. Hot firings of a scale model of the aft propulsion section of the vehicle including scaled thrust chambers will be conducted at the maximum attainable altitude of the available test facilities. To more closely simulate the exhaust gases of the proposed vehicle, a propellant combination of fluorine and hydrogen may be used in the scaled rocket chambers. From these scaled model tests, sufficient data will be generated to more clearly define the effects of the complex interaction of the rocket exhausts. The resultant patterns of pressures, loading and heating that must be considered during the design of the full scale vehicle will be determined. The parameters will be utilized to design the necessary base shield of the propulsion module.

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APPENDIX I

BACKGROUND AND EXPERIENCE

A. FLUORINE ROCKET TESTS

1. Introduction

Bell Aerosystems Company has been actively engaged in evaluating liquid fluorine as a rocket propellant since May 1956 under United States Air Force, National Aeronautics and Space Administration and company sponsorship.

Over 570 thrust chamber fire tests have been performed, involving approximately 85,000 pounds of liquid fluorine in combination with hydrogen, hydrazine and ammonia fuels. These tests were concerned with the determination of performance, stability and heat rejection characteristics over a wide range of operating conditions (30 to 500 psia chamber pressure, 1,000 to 35,000 pounds of thrust). Successful regenerative thrust chamber operation has been demonstrated utilizing each of the aforementioned fuels. Verification of shifting equilibrium expansion and the evaluation of thermal barriers and uncooled nozzle extensions were accomplished with the fluorine-hydrazine propellant combination. The ammonia bulk boiling cooling concept which utilizes the heat of vaporization and hence allows optimum performance mixture ratio operation, was successfully demonstrated by a regeneratively cooled thrust chamber test series.

Rocket engine type fluorine pumping was successfully demonstrated initially under company sponsorship and subsequently under NASA contract. Liquid hydrogen pumping and composite fluorine hydrogen pump tests have also been performed. A fluorine hydrogen pump fed rocket engine has been successfully tested.

Details of the fluorine-hydrogen testing performed at BAC and related experience for the propulsion systems defined in the preceding sections of this report are presented in this appendix.

2. Fluorine-Hydrogen Thrust Chamber

Fluorine-hydrogen thrust chamber investigations were initiated in 1958 under company sponsorship. A program was performed which consisted of a series of thirteen fire tests utilizing gaseous hydrogen and liquid fluorine. The objective was to determine the effect of mixture ratio on performance and heat rejection characteristics. Combustion chamber pressure was 70 psia and mixture ratio was varied from 2.2 to 25.9. Combustion efficiencies of 99 per cent of the theoretical were obtained, and a characteristic velocity (C*) of 8320 ft/sec was measured at optimum operating conditions. Performance is shown in Figure I-1.

Under the sponsorship of NASA, the fluorine-hydrogen thrust chamber program was expanded to cover chamber pressures from 100 to 425 psia. Two thrust chamber configurations were used; one made of aluminum, developing 12,000 pounds of thrust at 160 psia chamber pressure with an expansion ratio of 15 to 1. The other chamber was rated at 12,000 pounds of thrust at 320 psia chamber pressure with an expansion ratio of 30 to 1. This thrust chamber has a thrust to weight ratio in excess of 220. The unit has an inner liner made by explosive forming, longitudinal coolant passages and an external shell formed by a continuous circumferential wrapping of high strength ribbon. The assembly is fabricated by furnace brazing. An uncooled nozzle extension is used in the divergent section. A photograph of the thrust chamber is shown in Figure I-2. In addition, several high chamber pressure tests were conducted utilizing surplus "Hustler" aluminum chambers. Test performance data are presented in Figure I-3.

Over 100 tests have been conducted including 38 which were regeneratively cooled with liquid hydrogen. Several thrust chamber tests were performed in an aspirator duct to evaluate low chamber pressure operation. A typical injector design is shown in Figure I-4.

Throughout all programs the start and shutdown transients have been smooth. The simplicity and reliability of the hypergolic combination is one of its most attractive assets and a major advantage over other propellant combinations requiring an ignition system.

3. <u>High Energy Propellant Pump Development</u>

Bell Aerosystems Company sponsored a rocket engine type fluorine pump feasibility program during 1958. The program was divided into three areas of investigation: (1) dynamic fluorine seals, (2) fluorine cavitation effects and (3) actual pumping tests.

A satisfactory dynamic seal configuration was developed. This design incorporates an oxygen buffer cavity contained between opposing face seals such that oxygen gas leaks into fluorine and liquefies should a seal opening occur. Inert gas pressure between an adjacent set of lip seals was utilized to prevent oxygen flow into the bearing cavity.

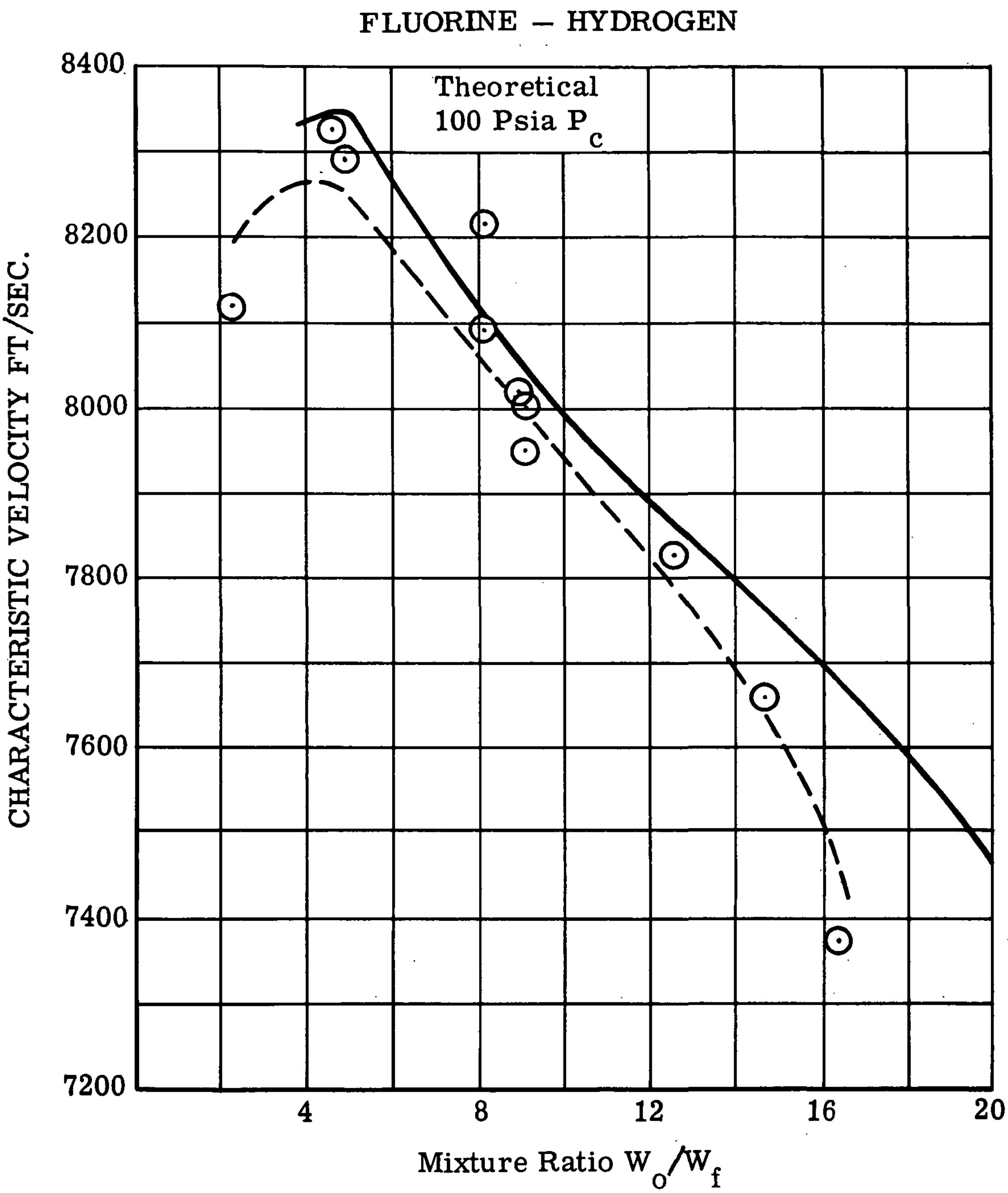

Figure AI-1. Characteristic Velocity Versus Mixture Ratio Injector FL-18, 70 Psia P_c, A_t = 17.15 in.^{2}

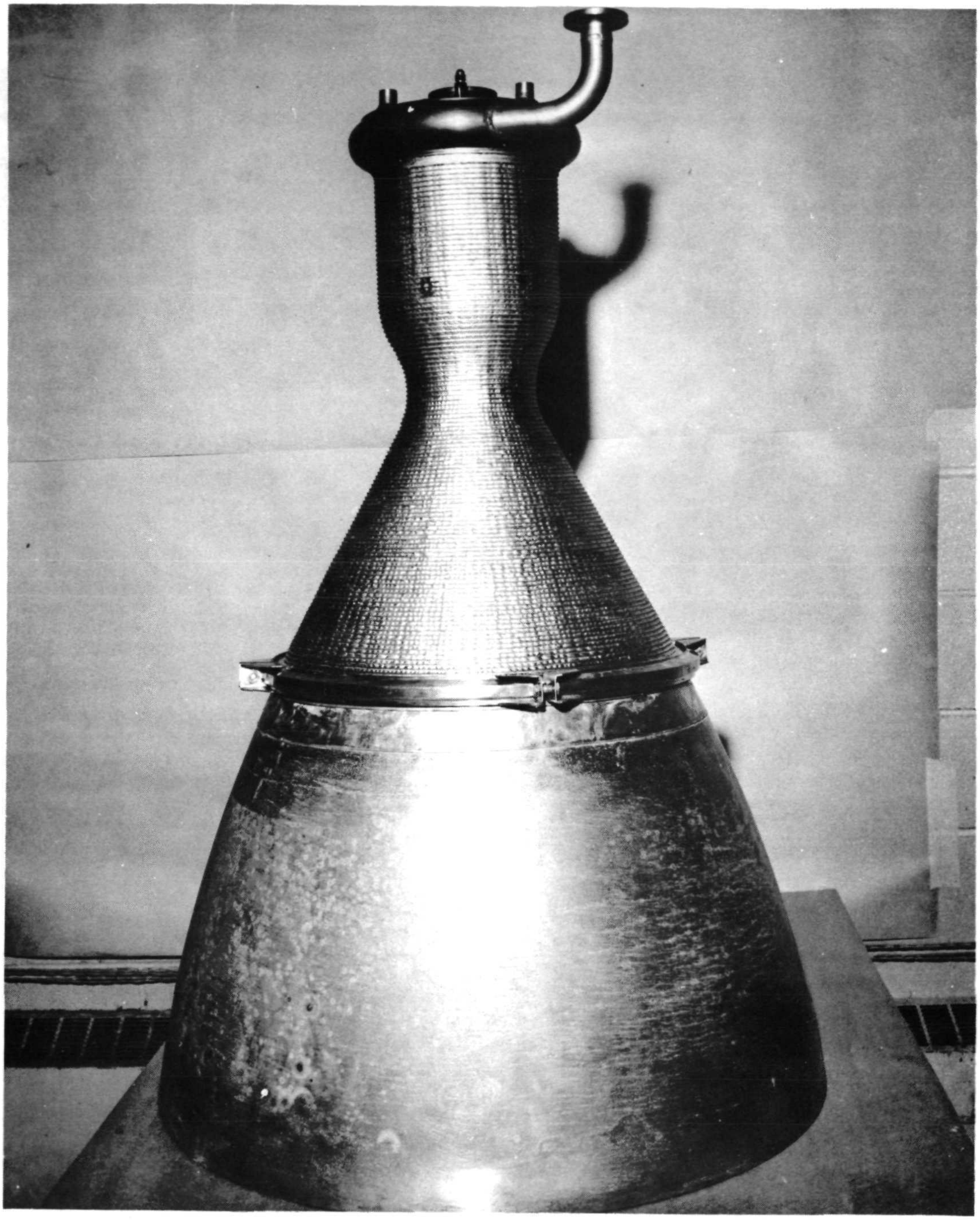

Figure AI-2. Fluorine-Hydrogen Brazed Thrust Chamber with Uncooled Nozzle Extension

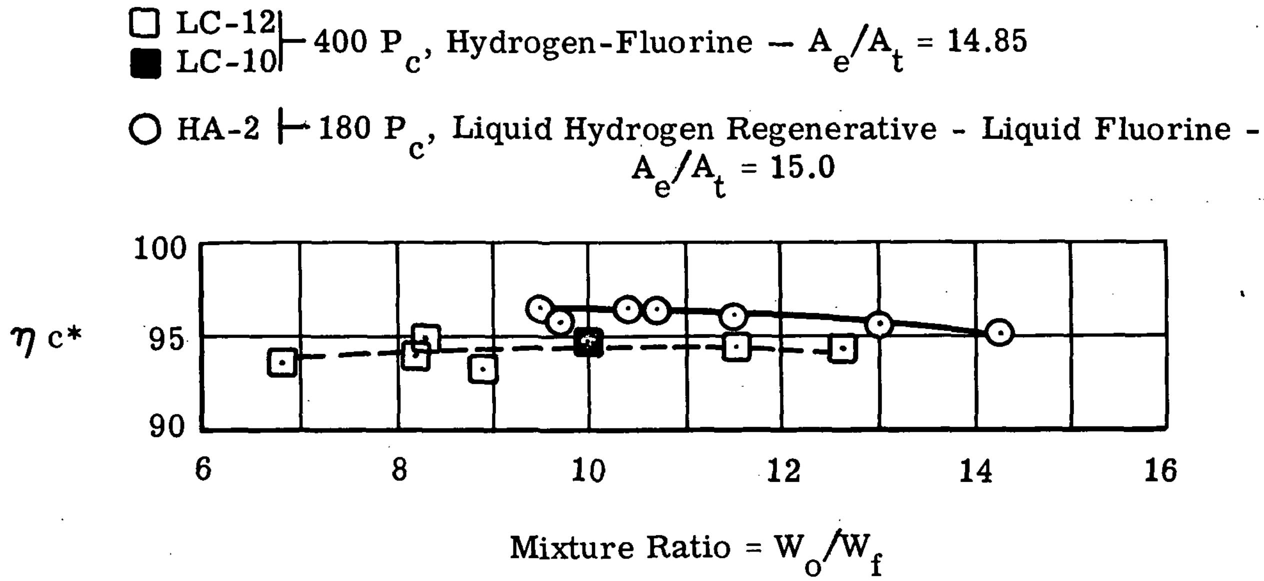

Figure AI-3. Efficiency of Injector FL-19 Versus Mixture Ratio

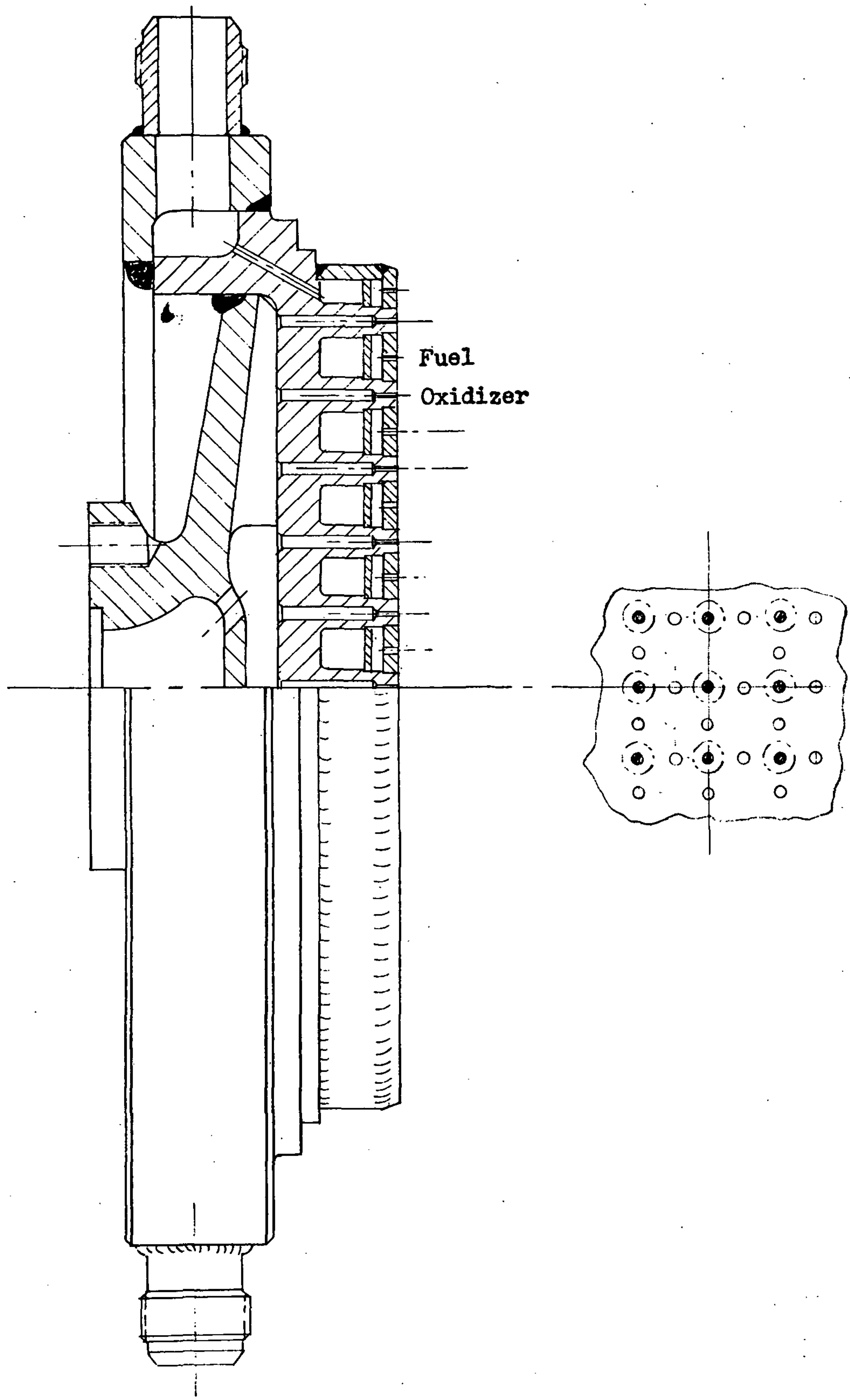

Figure AI-4. FL-18A and FL-19 Straight Showerhead Injector

To ascertain the fluorine cavitation effects, tests were made flowing fluorine at high velocity (approximately 250 ft/sec) through a cavitating venturi. There was no discernible reaction with monel or aluminum alloy metals.

Pumping tests were next conducted with liquid fluorine using both aluminum and monel pump housings. The pump design features an unshrouded, straight radial blade impeller with a tangential conical diffuser discharge. High velocity metal rubbing in liquid fluorine occurred during one of these tests, due to mechanical failure of a pump bearing; however, no reaction took place. These tests demonstrated the feasibility of pumping fluorine for rocket engine applications.

The program under National Aeronautics and Space Administration sponsorship was directed toward the development of a breadboard type turbo-pump for a 12,000 pound thrust fluorine-hydrogen rocket engine. A photograph of this turbo-pump unit without pump housings is shown in Figure I-5.

An inducer has been added to the fluorine pump which allows pumping with very low suction pressure - less than 10 psi above vapor pressure. A series of three, 300 second duration tests have been made within a two hour period. A single liquid fluorine seal has accumulated over 27 minutes running time in liquid fluorine. Several seal materials have demonstrated 900 second capability.

The pump described delivers 125 GPM of fluorine with a head rise of 450 psi and has a net positive suction pressure of less than 10 psi.

A pump of the same basic design has also been used in liquid hydrogen pumping operations. This unit delivers 265 GPM and also has a 450 psi head rise.

When coupled with a hydrogen gas driven turbine, the two above described pumps were operated as a turbo-pump assembly.

4. Propellant Valve

A dual fluorine-hydrogen thrust chamber valve was developed for 12,000 pound thrust application, and was utilized during thrust chamber, composite turbine pump and engine testing programs.

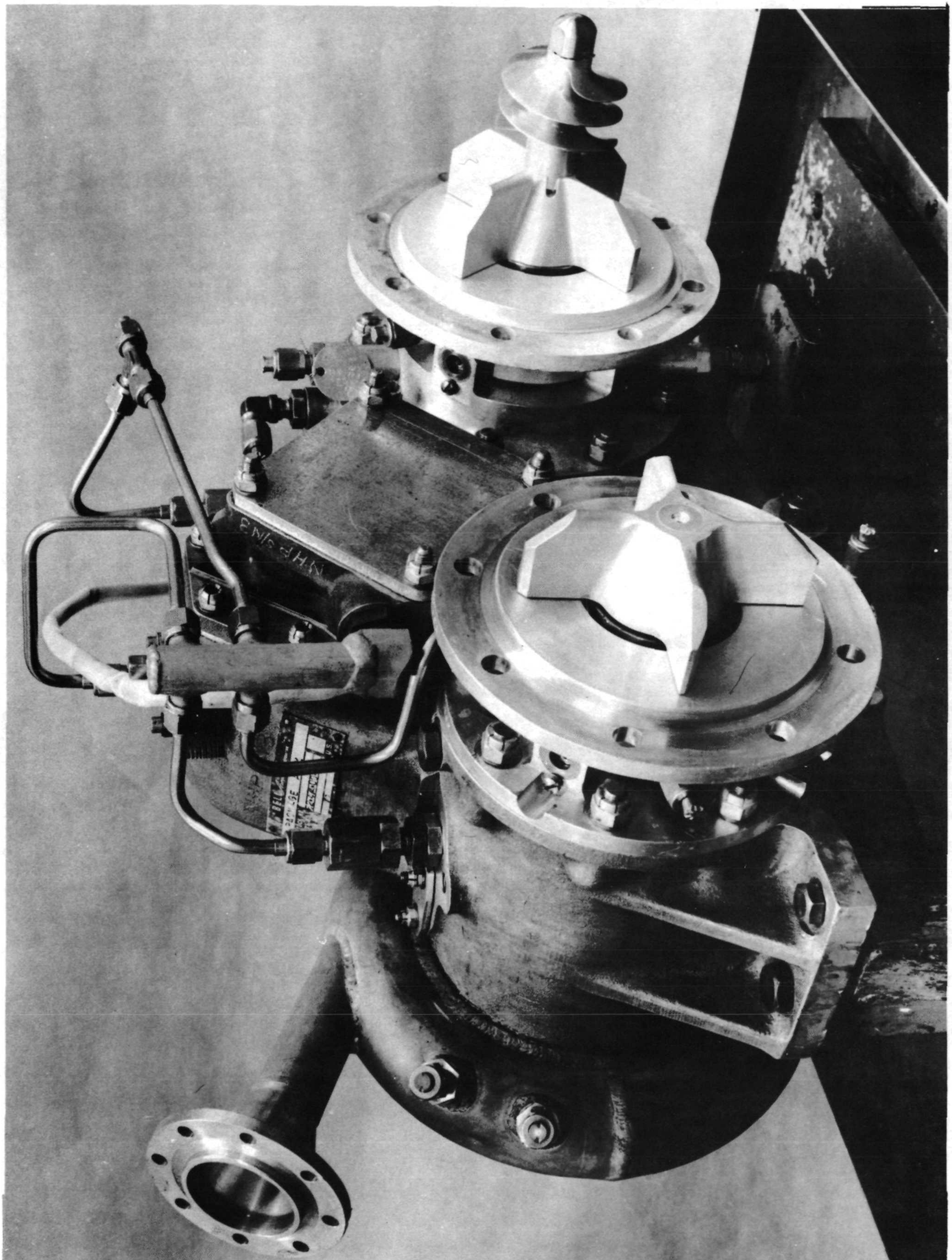

Figure AI-5. Fluorine-Hydrogen Turbopump

Valve housings, body, and fluorine seat were fabricated from aluminum alloy. Stainless steel valve poppets were utilized. A spun-in Teflon seat proved satisfactory for liquid hydrogen service. Simultaneous movement of the fuel and oxidizer valves was accomplished by a connecting yoke between the actuator and the valve stems.

The valve actuation and de-actuation time was 0.065 and 0.200 seconds respectively.

5. Engine Tests

The components previously described (turbine pump, aluminum thrust chamber and propellant valve) were integrated into an engine assembly, a photograph of which is shown in Figure I-6. A photograph of one of the engine firings is presented in Figure I-7.

Seventeen engine tests were conducted with one thrust chamber and turbine pump assembly. Satisfactory start, thrust chamber operation and shutdown phases resulted during these tests. Typical tests are discussed in the following paragraphs:

Run 1AW-352

This was a 15.5 seconds duration fire test on the complete engine assembly.

It should be noted that the fluorine pump assembly has been in use for eleven consecutive tests, including four composite pump tests, during a four month period without servicing.

The hydrogen pump assembly was overhauled prior to the initial engine test and again prior to Run 1AW-351 due to primary seal leakage. Displacement of the mating ring approximately 0.100 inch from its normal position was the cause of the abnormal leakage. A modification in the mating ring retaining design was incorporated to eliminate this condition.

Run 1AW-353 - 8-19-60

A spurious signal caused actuation of the overspeed safety circuit 2.7 seconds after start. Engine start, thrust chamber operation and shutdown were satisfactory, however, turbine speed attained only 90 per cent of its rated value.

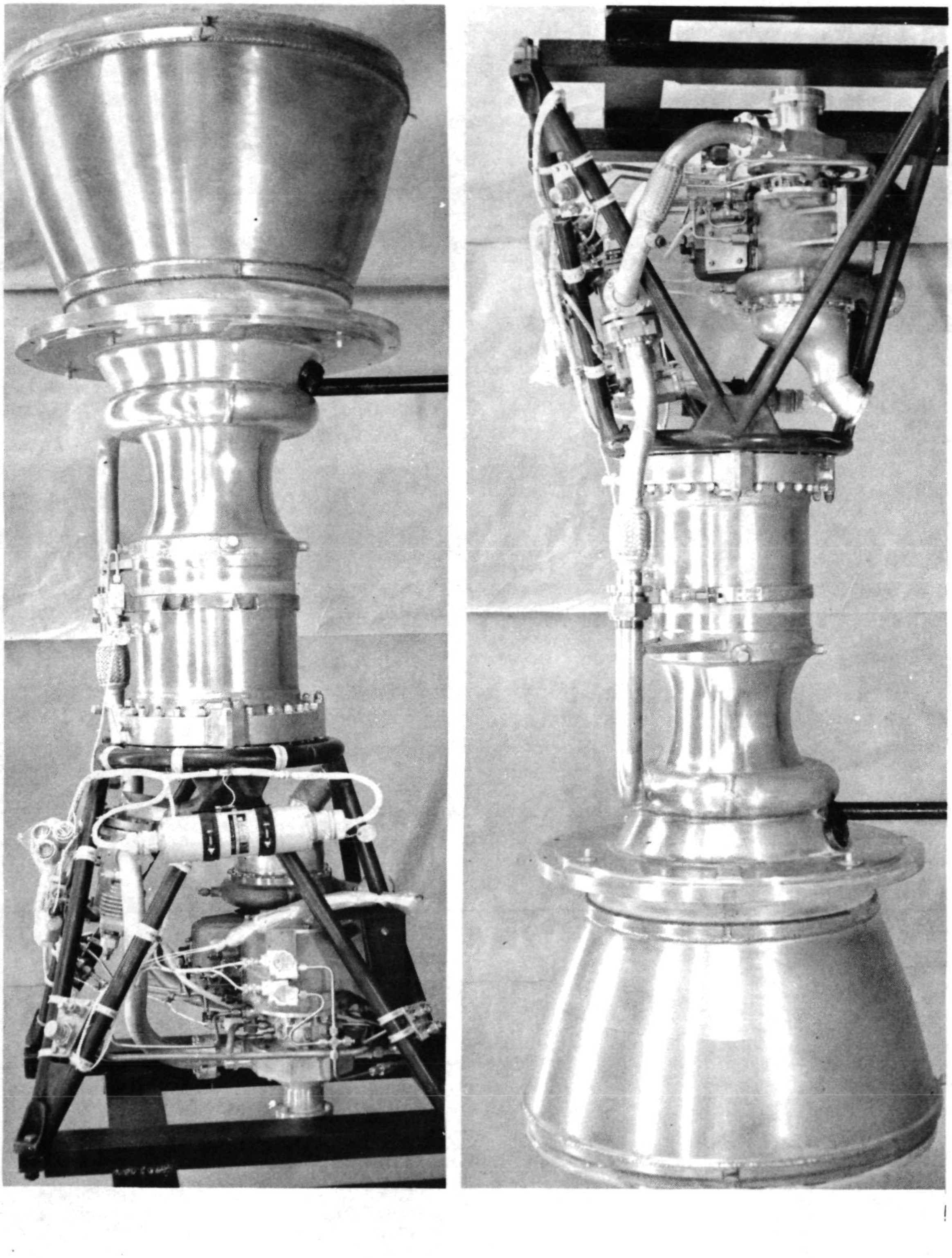

Figure AI-6. Side Views of the Fluorine-Hydrogen Engine

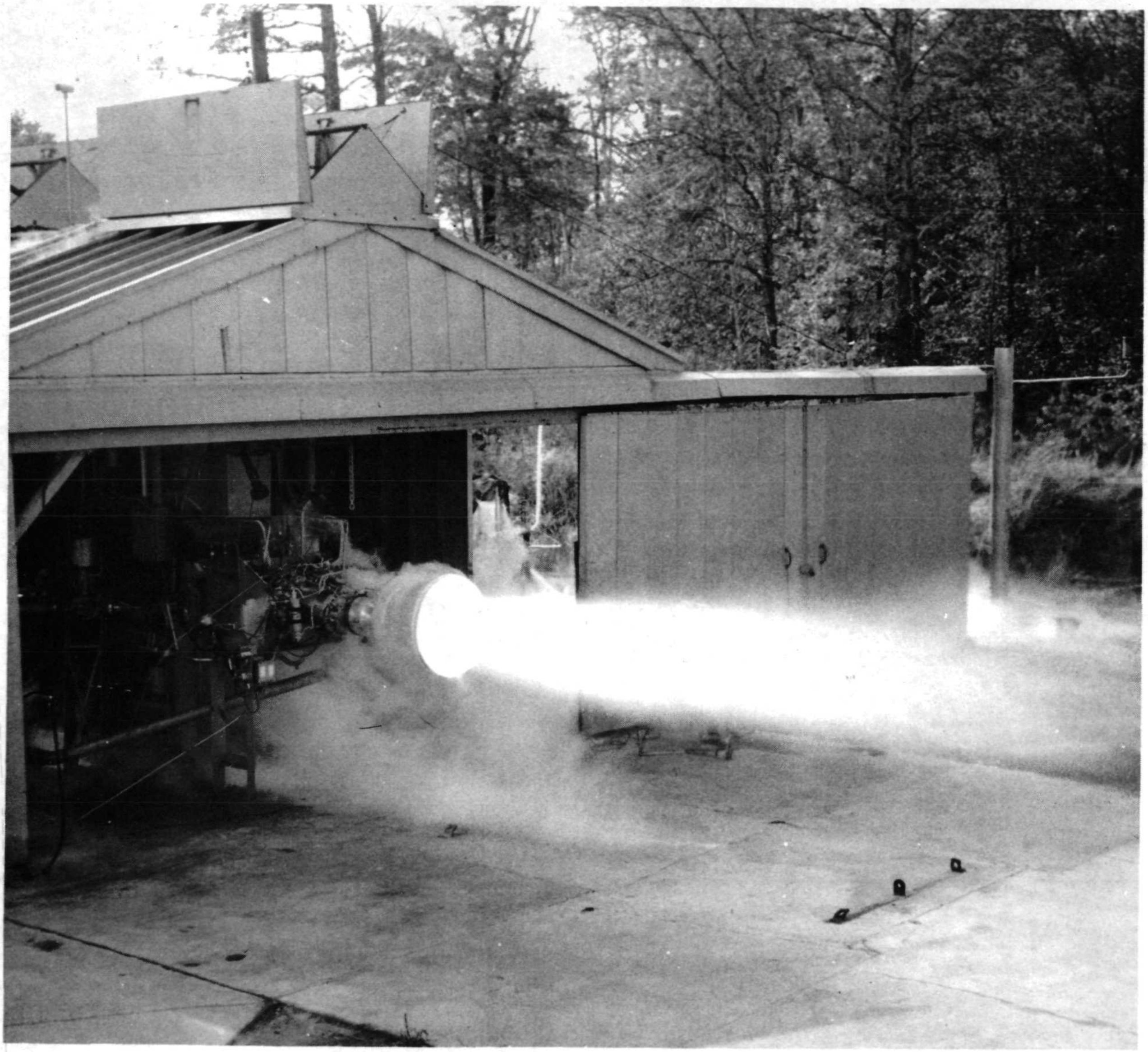

Figure AI-7. Fluorine-Hydrogen 12,000-Pound Thrust Engine Test

Run 1AW-354 - 8 19 60

This was a completely satisfactory 20.9 second duration test as programmed. Post-run pressure tests indicated excessive leakage past the fuel pump primary and shaft seals which dictated removal of the turbine pump for inspection and overhaul.

The fluorine pump was disassembled and inspected due to an increase in leakage rate past the oxygen gas seal. The fluorine primary seal was in excellent condition. This assembly had accumulated 110 seconds of operation during 12 tests on the composite pump and engine programs since April 16, 1960.

Inspection of the fuel pump disclosed that the primary seal mating ring had displaced approximately 0.100 inch from its normal position The cause of this movement was due to differential thermal contraction between the steel mating ring and aluminum shaft. The displacement occurs when the thrust chamber propellant valve closes producing surge pressures as a result of fluid deceleration. A cylindrical aluminum spacer was installed to axially retain the mating ring sleeve. Both carbon shaft seals were cracked and evidence of "shaft whip" was observed on the outboard seal body.

All pump and shaft seals were replaced.

The thrust chamber propellant valve fluorine assembly was inspected and overhauled due to stem leakage observed upon post-run pressure test. The poppet was slightly attacked at the stem seal area. An attempt to surface polish this area was unsuccessful (the electroless nickel plate was penetrated) therefore, a new poppet was installed.

Run 1AW-355 - 9-6-60

This was a satisfactory 20.2 second duration test as programmed. Post-run inspection indicated a recurrence of fuel pump mating ring displacement discussed in the previous test. The fuel pump was disassembled and the sleeve which receives the mating ring was staked to the mating ring at four places 90° apart. An interference fit of 0.0015 to 0.0025 inches was utilized between the mating ring and sleeve.

Run 1AW-356 - 9-9-60

A safety circuit actuated shutdown occurred after 18 seconds of the intended 30 seconds duration as a result of a decrease in thrust chamber pressure below the chamber pressure switch de-actuation setting. This decrease was the result of decreasing turbine speed and flow rate caused by a hydrogen pump shaft ball bearing failure. An analysis was conducted on the failed bearing by the vendor. It was concluded that a momentary stoppage of lubricant caused the failure.

Post-run inspection of the fuel pump showed evidence of impeller rub on the seal retainer plate as a result of the bearing failure. The shaft assembly was replaced, with the exception of the primary liquid seal, to permit complete inspection of the failed assembly. This seal was reinstalled without relapping.

Run 1AW-357 - 9-20-60

This run was terminated by the ground safety timer 2.9 seconds after the start signal. The turbine drive regulator solenoid valve failed to operate due to low voltage. Consequently, the start sequence did not proceed to the point where drive gas was admitted to the turbine.

Run 1AW-358 - 9-20-60

This run was shut down at 3.9 seconds by actuation of the oxidizer case pressure overspeed switch, although the turbine was operating close to rated speed. Drift in the pressure switch actuation setting from 518 psig to 503 psig, was the cause of shutdown. Inspection of the switch showed the microswitch coated with rust. A new pressure switch set to actuate at 550 ±10 psig was installed.

Run 1AW-359 - 9-29-60

This was a satisfactory 30 second duration test as programmed and represents maximum test stand duration. Turbine speed was approximately 4% lower than rated during this test. The breadboard type turbine drive control system used in this program has introduced inherent variation in turbine manifold pressure run-to-run reproducibility.

Run 1AW-360 - 9-29-60

This run was shut down after 23 seconds of an intended 30 second duration test by the control panel operator upon observation of a fire located at the propellant valve fluorine inlet joint. The feed line between the pump and propellant valve was removed to permit inspection of this joint. Evidence of slight localized discoloration appeared on the outermost portion of the serrated seal surface on the propellant line. Serration impression on the soft aluminum gasket in this area indicated nonuniform seal loading resulting from nonparallel surfaces. The serrated surface of the propellant line was remachined. The line was installed on the engine and pressure tested.

Run 1AW-361 - 10-13-60

This test was terminated at 1.87 seconds after start by erroneous actuation of the overspeed safety circuit. The instrumentation frequency integrator produced an overspeed indication although the actual turbine speed was only 50% of its rated value.

Post-run investigation showed the integrator to be intermittent in operation and it was removed from service.

Run 1AW-362 - 10-13-60

This was a satisfactory 20 second duration test as programmed and concludes engine testing.

The average vacuum specific impulse (ϵ = 45:1) of the runs which obtained steady-state operation was 444.4 seconds.

B. PROPELLANT HANDLING

1. Fluorine

Fluorine is transported from the chemical plant to the test site by trailer tanks of 5000 pound capacity. The tank is of triple wall construction - inner fluorine vessel; intermediate liquid nitrogen tanks; and, outer insulating container. Fixed storage tanks at the Bell test site are this type. This tank configuration affords loss free storage since fluorine is maintained in a subcooled state, i.e., near liquid nitrogen temperature (-320°F).

Pressurized transfer, using helium gas, is employed for test operations. The receiver tank is also equipped with a liquid nitrogen jacket to facilitate handling. Transfer lines are fabricated from copper tubing with AN flare type connections. Commercial type two-way valves are used to control flow and tank weight is monitored to indicate tanked propellant.

Fluorine vent gas is directed to a propane air burner where HF is formed and being less dense than air rises and disperses in the atmosphere. At Lewis Research Center of the National Aeronautics and Space Administration, fluorine gas is ducted to a charcoal reactor where inert CF_4 is produced and the effluent is discharged to the atmosphere.

A condensing system using liquid nitrogen could be employed to collect fluorine vapors.

2. Hydrogen

Liquid hydrogen is transported to the test site in roadable trailer tanks of 5,000 gallons capacity. Typical cryogenic tank fabrication concepts are applicable for hydrogen, i.e. a double tank arrangement with evacuated insulation space. The test site tanks are similar.

Pressurized transfer using hydrogen gas is employed. Vacuum insulated liquid hydrogen lines constructed of stainless steel are most commonly used.

Vent hydrogen gas is directed to an elevated stack (approximately 25 feet high) and discharged to the atmosphere.

C. SAFETY CONSIDERATIONS

Rocket oxidants, because of their physical and chemical nature offer more hazard potential in handling than fuels. Fluorine oxidizers, due to inherent characteristics, dictate that certain precautions be taken to assure safe handling. However, fluorine safety requirements are no more severe, or difficult in application, than those required for any other potentially hazardous propellant.

1. Personnel

a. Fluorine

Personnel safety is the most important aspect to be considered since the reactivity and toxicity of fluorine create limited health hazards.

The reactivity characteristic dictates remote conduct of passivation, transfer and vent gas disposal processes. This requirement is compatible with present launch pad operating procedures which dictate at least remote-controlled cryogenic propellant transfer.

Fluorine has a pungent, irritating odor detectable in very low concentrations. The inhalation of seriously toxic concentrations of gas is very improbable because of this characteristic. Safe, large-scale usage of fluorine has been demonstrated at the Oak Ridge Atomic Energy Commission Facility operated by Union Carbide Corporation, where a perfect safety record has been maintained for a period of 15 years.

As established by the American Congress of Government Industrial Hygenists in 1960 based on a 40 hour week, day after day, exposure periods, maximum allowable concentrations (MAC) for a number of common propellants are presented below:

Ammonia	100 parts per million (in air)
Nitrogen Tetroxide	5 ppm
Hydrogen Fluoride	3 ppm
Hydrazine	1 ppm
Dimethyl Hydrazine	0.5 ppm
Chlorine	1 ppm
Hydrogen Peroxide (90%)	1 ppm
Chlorine Trifluoride	0.1 ppm
Diborane	0.1 ppm

The literature concerning a similar value for fluorine is conflicting and varies from 0.1 ppm to greater than 1 ppm. Probably the most reliable information is available from the Union Carbide Corporation, Oak Ridge, Tennessee. Reports from this facility state that personnel have been exposed to 0.6 ppm annual average concentrations and maximum daily concentrations of 1.2 ppm with no adverse effects. This operator is using a MAC of 1 ppm. (Reference 13).

Nearly all the literature establishes concentration levels based on a continuous exposure period - 8 hour day, 40 hour week. This basis is not valid for personnel engaged in rocket propulsion work since exposure is intermittent. It appears that a greater MAC would be acceptable for this type operation. A reasonable approach might be to set a limit for intermittent operation based on parts per million concentration multiplied by the exposure time (ppm hours). Thus 40 ppm hours would be the weekly inhalation limit for personnel. For missile-type operation a level of 4 ppm might be tolerated for a period of 10 hours without harmful effects. However, personnel evacuation from the contaminated area or use of auxiliary breathing apparatus would be effected within a few minutes after detection of odor, since there is presently no immediate means of controlling or continuously determining the concentration level in the breathing zone.

During tests reported in Reference 14, two volunteers walked into a dispersed cloud of fluorine vapor and were exposed to a concentration of 25 ppm for a short time without lasting effects.

Periodic medical examinations of Bell Aerosystems Test personnel have shown no harmful effects from fluorine exposure.

Based on the above data and information the toxicity hazard associated with fluorine will not preclude its application in rocket propulsion systems.

b. Hydrogen

When mixed in the proper proportions with air, oxygen, or other oxidizers, gaseous hydrogen forms a highly flammable mixture. In air the explosive limits of hydrogen range from 4.1 to 74.2% hydrogen by volume. Recent experimental work (Reference 15) indicates that it requires a strong initiator to detonate an nonconfined mixture of hydrogen and air. The gas is not toxic. Adequate ventilation of enclosed places and elimination of ignition sources together with applicable combustible gas handling precautions should be effected.

2. Leaks and Spills

a. Fluorine

In the event of a leak or spill, the action required depends upon the nature and location of the defect. In general, the supply of fluorine should be shut off and the leaked fluorine allowed to evaporate. The cryogenic nature of liquid fluorine promotes rapid dispersion of leaks. This characteristic, together with its reactivity with atmospheric water vapor, (thus producing "lighter than air" HF which in turn rises and disperses), eliminates any toxicity hazard resulting from minor leaks. However, should gross spillage occur as a result of equipment failure, such as a tank, provision must be made to neutralize the discharged fluorine.

From a USAF-sponsored program in which large quantity fluorine spill neutralization methods were evaluated, it was shown that a simple water spray around the spill area was effective (Reference 14). A curtain sprayed vertically upward had the effect of reducing the ground level fluorine concentration at a sampling point 75 feet from the spill point from 10,000 maximum to an average of 20 parts per million. This method appears very practical and is recommended for launch pad spill neutralization.

b. Hydrogen

Minor liquid hydrogen leakage is not likely to occur. Hydrogen has such a low boiling point that any small flow of liquid will probably vaporize before or during its escape from the system.

In the event of equipment failure, such as tank rupture, the released liquid constitutes a safety hazard. If ignition occurs, large quantities of heat generated may result in widespread damage. However, this danger appears to be no greater than that associated with many combustible materials, such as gasoline and jet fuel. In point of act, hydrogen flames radiate much less heat than the flames of such fuels (Reference 16).

Since a small but finite probability exists that spontaneous deflagration will occur when hydrogen is vented to the atmosphere, remote venting will be required. A high-level disposal stack extending from the umbilical tower will be utilized for this purpose at the launch site.

D. FLUORINE INDUSTRIAL USE AND MANUFACTURE

1. General

Over 75 million pounds of fluorine gas and liquid have been produced by Allied Chemical and Union Carbide Companies during the past 10 years. The major part of this has been used by the AEC. It is important that no lost-time accidents have been experienced during the production of these large quantities. Liquid fluorine was first produced in this country in 1945 by Dr. Hans Neumark of Allied Chemical. The cost of fluorine for large tonnage quantities is less than $1 per pound.

2. Manufacture

The largest current plant producing liquid fluorine is that of the Allied Chemical Company at Metropolis, Illinois. This plant has a capacity of several million pounds of fluorine per year.

Fluorine is produced by over forty individual cells which production method has been found to be the most efficient and suitable. These cells are roughly the size of two office desks end to end. This incremental type of production provides for rapid plant expansion. Allied Chemical has indicated they can produce fluorine in any desired quantities with an 8-10 month lead time. The basic raw materials of fluorine are fluorispar and sulphuric acid which are combined to make the intermediate, which is hydrofluoric acid. The hydrofluoric acid in turn is electrolytically treated to produce fluorine gas which is then purified and condensed resulting in liquid fluorine. In the production of fluorine the workers wear simple rubber-type coat with a celluloid plastic face mask and rubber gloves. Bell has used similar protective covering for test work during our extensive rocket testing.

As regards transportation of fluorine, liquid fluorine is today transported in trailer trucks going through the city of Cleveland and the country being distributed to Bell, NASA, Edwards, and Rocketdyne. The trailers have a capacity of 5,000 pounds and are of the multiple tank construction type. The inner tank containing the fluorine which is in turn surrounded by liquid nitrogen and then surrounded by a insulation jacket. Fluorine is placed in the truck trailer and sealed. Under this arrangement the fluorine tank has a stay time of 30 days with the fluorine sealed and no replenishment required of the liquid nitrogen. The trailers have covered over 75 thousand road miles with no incidents and have ICC approval. Additional designs have been carried out for road trailers of 20,000 pound capacity and rail cars of 50,000 pounds capacity. The trailers have been in operation since 1954.

E. FLUORINE USE AT EXISTING TEST SITES

Now let us consider the modifications which would be necessary to a current vehicle launch site due to the introduction of fluorine. Detailed studies have indicated that only four additional items of equipment would be required. It is important to note that these items have been in use during the conduct of rocket tests and fluorine experiments for a considerable period of time. The first is a ready storage vessel which would be filled from the trailers or rail cars and can be either of the no loss type utilizing a liquid nitrogen jacket around the fluorine as is used for the trailer trucks or a type using a vacuum multiple air insulation approach. Bell has a storage vessel of the former type which has been in use at our Bell Test Center for the past nearly five years and has required no replacement of the parts coming in contact with fluorine. Allied Chemical has completed the design of 50,000 and 100,000 pound liquid fluorine storage vessels of this same general type. The second item of equipment is the conditioning equipment. This is required in order that the lines and tank may be conditioned with (or pacified with) gaseous fluorine by passing it through the system at a low pressure in order to consume under controlled conditions any moisture or dirt particles that might be present prior to filling the system with liquid fluorine. The conditioning equipment thus is that merely required to convert liquid fluorine to a gaseous state together with the associated valving. This conditioning has been performed by all organizations who have used liquid fluorine to date, and is relatively simple. The next item of equipment is that for disposing of the passivation gas and is a vent gas disposal item. Several approaches may be used to accomplish this function. At our Bell Test Center fluorine is burned in propane, however, it is our feeling that an open flame would be undesirable at the launch site. Thus, either a reflux system wherein the fluorine is recondensed into liquid may be used or the gas may be passed through a charcoal bed absorber. Or for example the fluorine may be bubbled through liquid nitrogen thereby contaminating the nitrogen and disposing of it following launch. The fourth item would be the installation of a neutralization system using water fog, similar to that which was tested for large spills at the Edwards Air Force Base as discussed above. Two concentric circles of spray nozzles are recommended one perhaps with a radius of 50 feet and the other on the order of 150 feet.

F. LOW THRUST STORABLE BIPROPELLANT SYSTEM

In December 1959 a contract was awarded to the Bell Aerosystems Company for the design and development of an ullage and orbital adjust system (secondary propulsion system) for the Agena B vehicle. This system is in addition to the main propulsion system of this stage which is also supplied by Bell (Model 8096, 15K engine). This system utilizes N_2O_4/UDMH as the propellant combination and is required to function reliably over a one-year period in exposed space, while remaining unaffected by cosmic and Van Allen radiation dosages. Thrust units of 20 pounds and 200 pounds thrust respectively are currently under development. A photograph of these chambers is shown in Figures I-8 and I-9. These chambers are completely radiation cooled and have adequately demonstrated this capability during a recently completed test program; one of several milestones already achieved. In all tests to date, thrust chamber injectors have repeatedly demonstrated combustion efficiencies as high as 92%. Preliminary tests results of the bipropellant valves show them to be in compliance with the design specifications. This program is well underway and is proceeding according to schedule. Figure I-10 is a photograph of one of the propellant tanks and bladders being developed for this program.

Figure AI-8. 20-Pound Thrust Uncooled Bipropellant Thrust Chamber Assembly

Figure AI-9. 200-Pound Thrust Uncooled Bipropellant Thrust Chamber Assembly

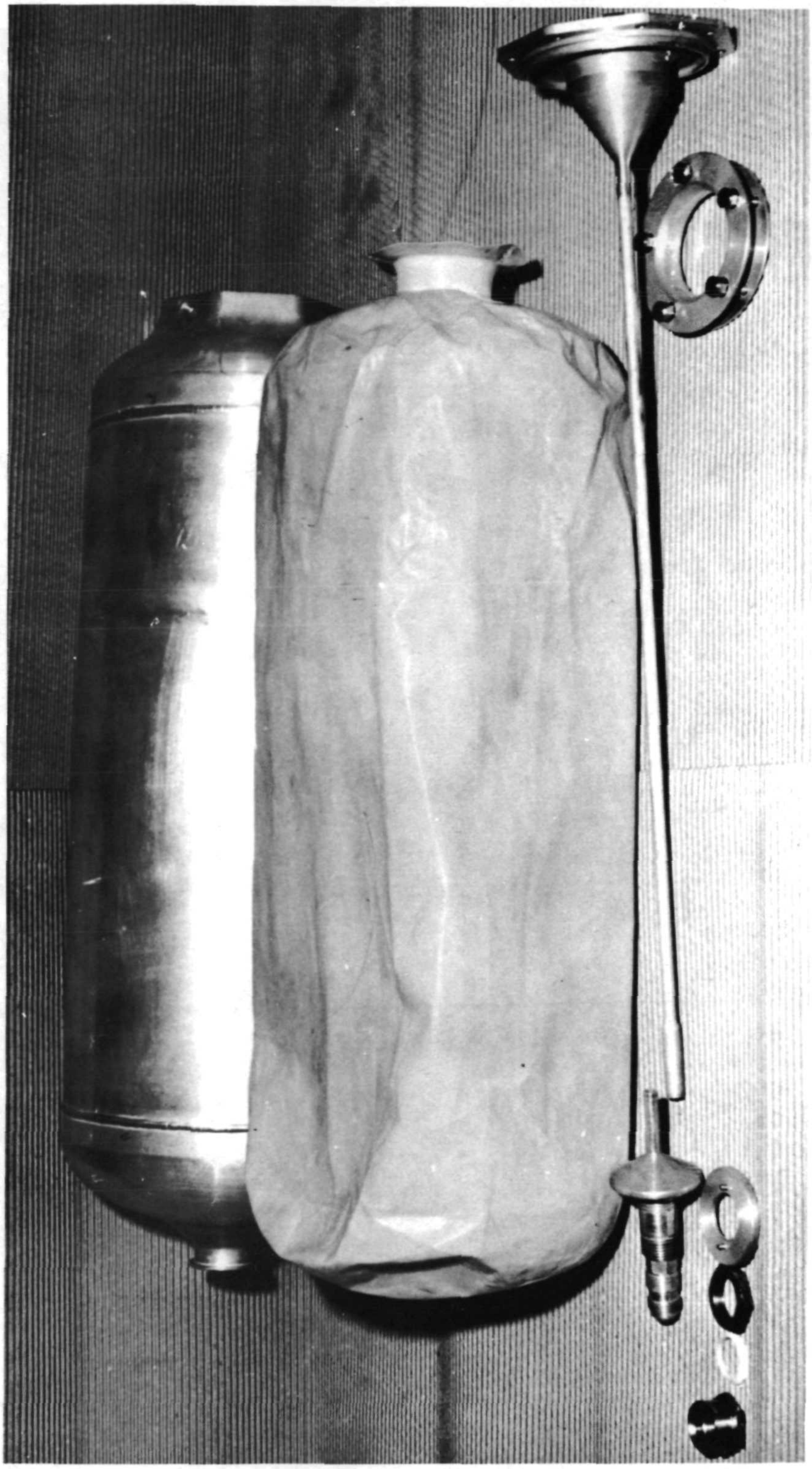

Figure AI-10. Secondary Propulsion System Bladder Assembly

APPENDIX II

FACILITIES

A. PRESENT CAPABILITY AT BAC

1. General

Efficient facility utilization is essential to the satisfactory conduct of the program defined in this report. The nature of such a program places a premium on the proximity of the functions of Engineering, Manufacturing and Test. The Wheatfield Plant of Bell Aerosystems Company and Air Force Plant No. 38 operated by Bell, provide many of the necessary facilities for the program. Air Force Plant No. 38 is a completely integrated rocket test facility with isolated test areas, support buildings, shops, propellant storage areas, roadways and utilities with a security fenced area.

The Wheatfield facility has 32 test cells with a capability to 25,000 pounds of thrust. Nineteen of these test cells contain environmental control equipment to permit both low and high temperature tests. In addition, there are instrument rooms, motor assembly areas, a valve test laboratory, water flow stands, a propellant research laboratory, a vibration laboratory, a machine shop and other facilities necessary to support the operation of these test cells. The two central instrumentation rooms at Wheatfield house 180 strip chart recorders, eleven 18 channel and two 36 channel recording oscillographs. A digital data acquisition system was installed which receives converted transducer data and converts these analogs to digital data, thereby eliminating the delay of manual data reduction. This system also provides conventional teletype code punched tape for remote transmission of data to the IBM-704 digital computer which then yields the desired performance calculations. The present digital data acquisition system has a capacity of 100 channels.

In addition to the main plant facility, the company operates Air Force Plant No. 38 (Bell Test Center) where 21 rocket test cells including altitude test facilities are available. One environmental test cell can accommodate a 30-foot missile, while other environmental cells are used for engine and component testing. This facility is also supported by a shop assembly area where final component and system assembly is accomplished. Instrumentation includes 225 strip chart recorders, sixteen 18 channel and three 36 channel recording oscillographs. Four test stands are instrumented with digital data acquisition system which have been in operation since 1956. The largest data system has a capability of recording 45 test parameters simultaneously at a repetition rate of 30 words a second. A small data monitoring system has the capability of recording 32 separate test parameters simultaneously. Figure II-I is a photograph of Air Force Plant No. 38. Figure II-2 is a view of the complete Rascal missile being test fired in the attitude tower. This tower is capable of changing missile attitude while the engine is firing. Attitude test firings of the Agena-Discoverer stage have been tested here. Figure II-3 presents a view of a typical test complex consisting of an ambient test shed and a control-instrumentation bunker. Figure II-4 shows an automatic digital data acquisition system installation in the bunker.

The Electronic Data Processing Center is located at the main Bell Plant at Wheatfield, New York. The entire building is air-conditioned and the computer room has additional temperature and humidity controls. Equipment includes an IBM-704 magnetic tape computer, a high speed machine with magnetic tape units and 32,000 words of core storage. Associated equipment includes the key punch, verifier, interpreter, numerical collator, alphabetical collator, sorter, reproducer, tabulator and the calculator. The data center handles computation of problems involving rocket engine combustion analysis, rocket nozzle shifting equilibrium axisymmetric flow analysis, missile trajectory analysis, structural analysis, aeroelastic analysis, flutter analysis, and the reduction of data from wind tunnel and flight testing. The data center has also been used for the computation of problems involving propellant characteristics and heat transfer analysis.

The Wheatfield facility is the center of manufacturing activities of the Bell Aerosystems Company. A modern machine shop of 104,000 square feet includes a complete dust free, humidity and temperature-controlled precision machine shop and assembly area.

To support its variety of programs, Bell maintains a group of technical laboratories. They are divided into three main categories:

(1) Materials

(2) Equipment

(3) Instrumentation

The materials section is composed of the metallurgical and chemical laboratories. This section furnishes data on properties of metals, finishes and coatings as well as a selection of materials

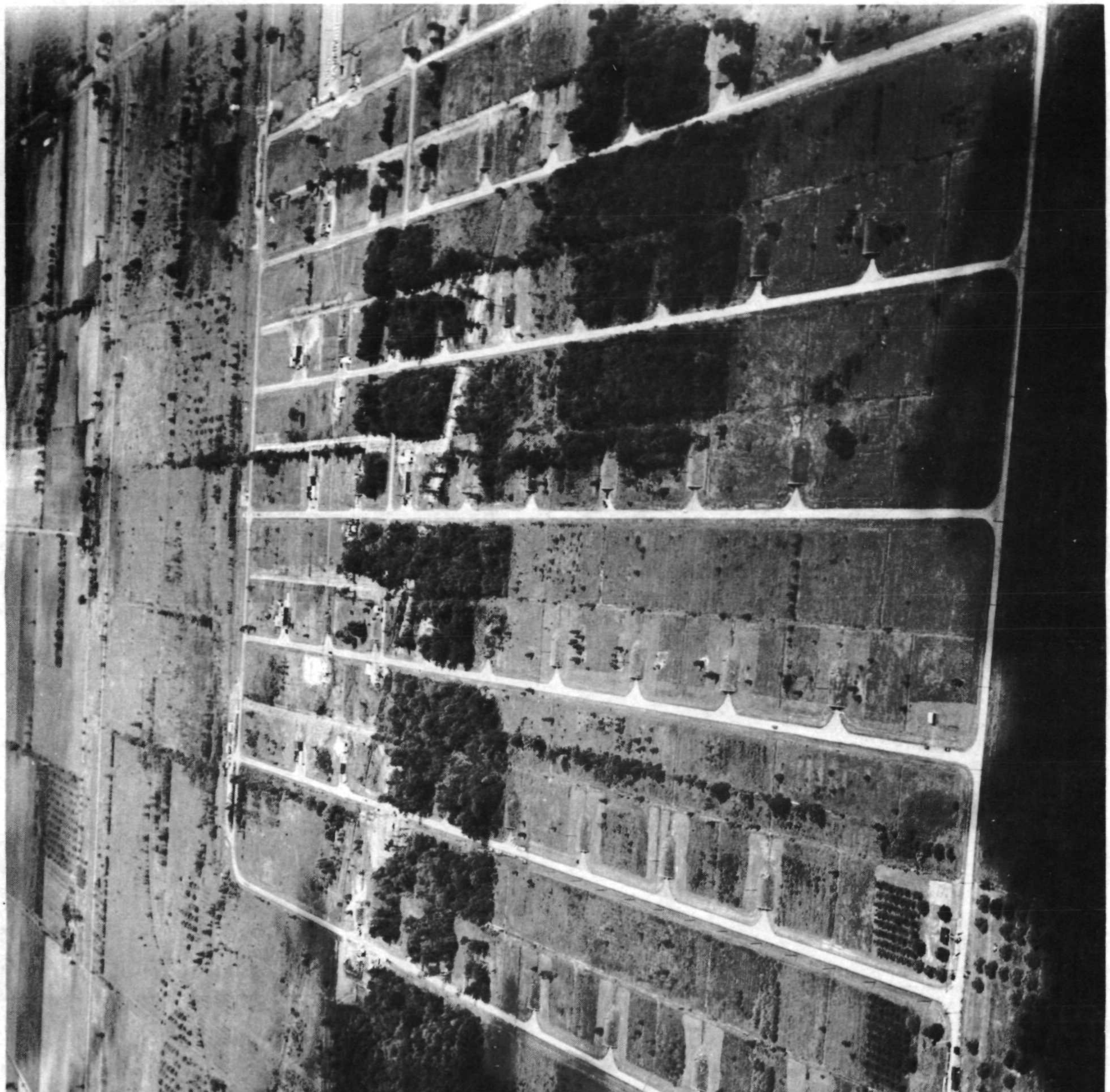

Figure AII-1 Rocket Test Cells, Air Force Plant No. 38

Figure AII-2. Thrust Stand for Attitude Testing

Figure AII-3. Test Stand and Instrument Control Room

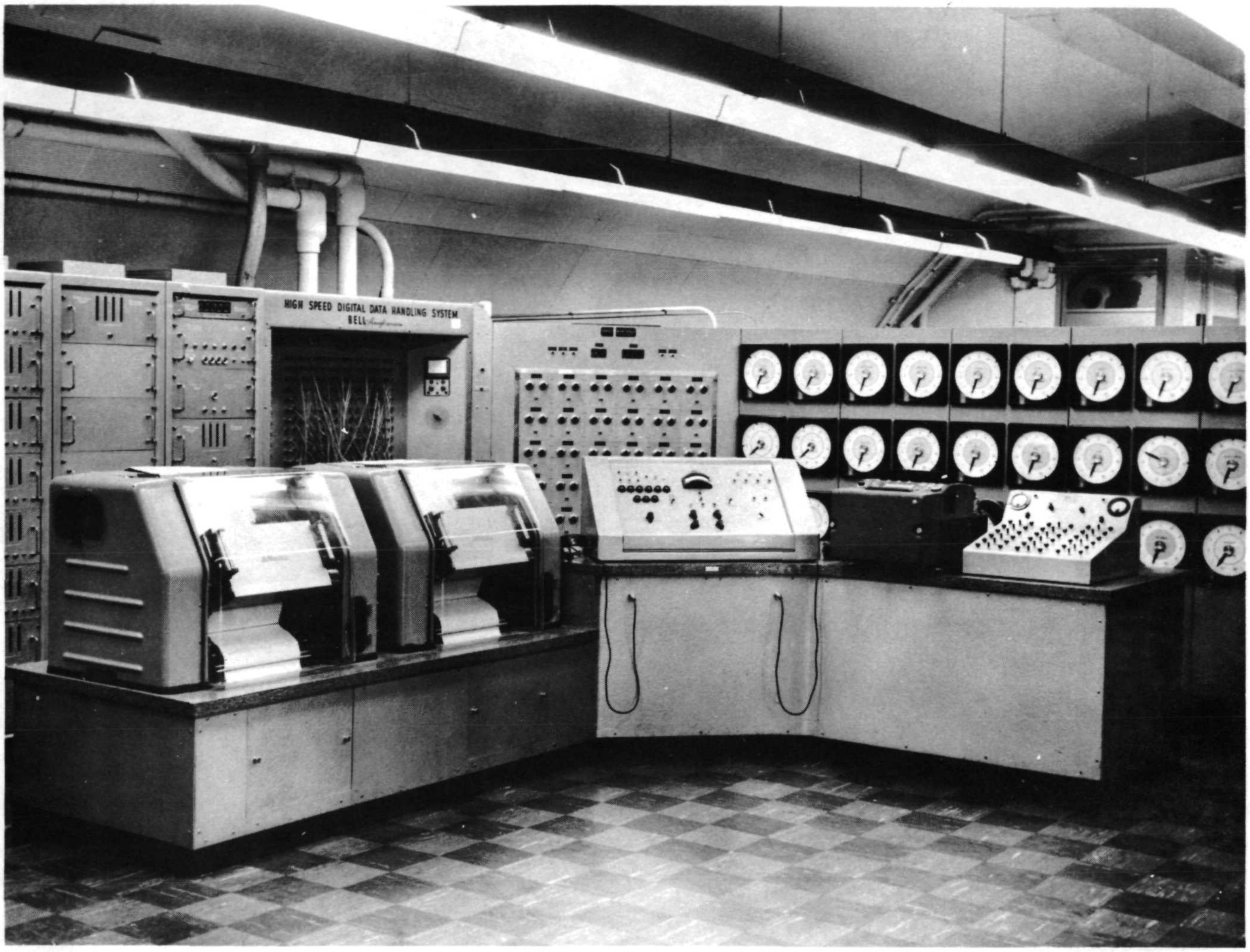

Figure AII-4. Automatic Digital Data Acquisition System for specific requirements. In addition, this section renders evaluations in the areas of heat treatment, hardness tests, weld analysis, tensile, yield and elongation tests on basic materials and monitors various manufacturing processes. Efficiency of the plating, cleaning and heat treating operations are reviewed daily. The chemical laboratory also furnishes information on the use of plastics, rubber, non-ferrous alloys, fuels, oxidizers and gases.

The equipment section, subdivided into mechanical, electrical, pressure testing, vibration and fatigue and structural laboratories, supplies research and evaluation services for prototype and full-scale manufacturing programs. Here environmental and qualification testing is conducted on structures, systems, accessories, and test equipment necessary to provide the foregoing services.

2. High Energy Propellants

All high energy propellants test firings are conducted at the Bell operated Air Force Plant 38 (Bell Test Center). The present high energy propellant thrust chamber test complex is shown in Figure II-5. Two firing sites are provided with a common control center. Data is recorded in the control center as well as in a separate remotely located instrumentation building. Supporting liquid nitrogen, helium, liquid fluorine and other services necessary to the conduct of fire tests, are common to both firing sites.

Firing site 1-A West has been utilized for testing with liquid fluorine/hydrazine, liquid fluorine/boiling ammonia, and the company sponsored liquid fluorine/liquid hydrogen program.

The test cell consists of a prefabricated metal building covering the test stand, thrust pad and propellant tank pits. Each tank pit is eight feet square and six feet deep. The forward end of the building has wide sliding doors on three sides affording excellent visibility from the control center as well as providing desirable ventilation characteristics. The building and pad are surrounded by a totally enclosed earth embankment with a minimum height of five feet. This embankment will retain spilled oxidizer in the event of an accident and also serves to deflect the rocket engine exhaust gases upward thereby aiding to rapid fume dispersal.

Test complex 3-B, shown in plot plan form in Figure II-6, is a typical cryogenic turbine pump test cell. This area is equipped with the same basic services and earth barrier as previously described for the thrust chamber facility. In addition, there are pump calibration test cells that are equipped with Scripps and Chrysler

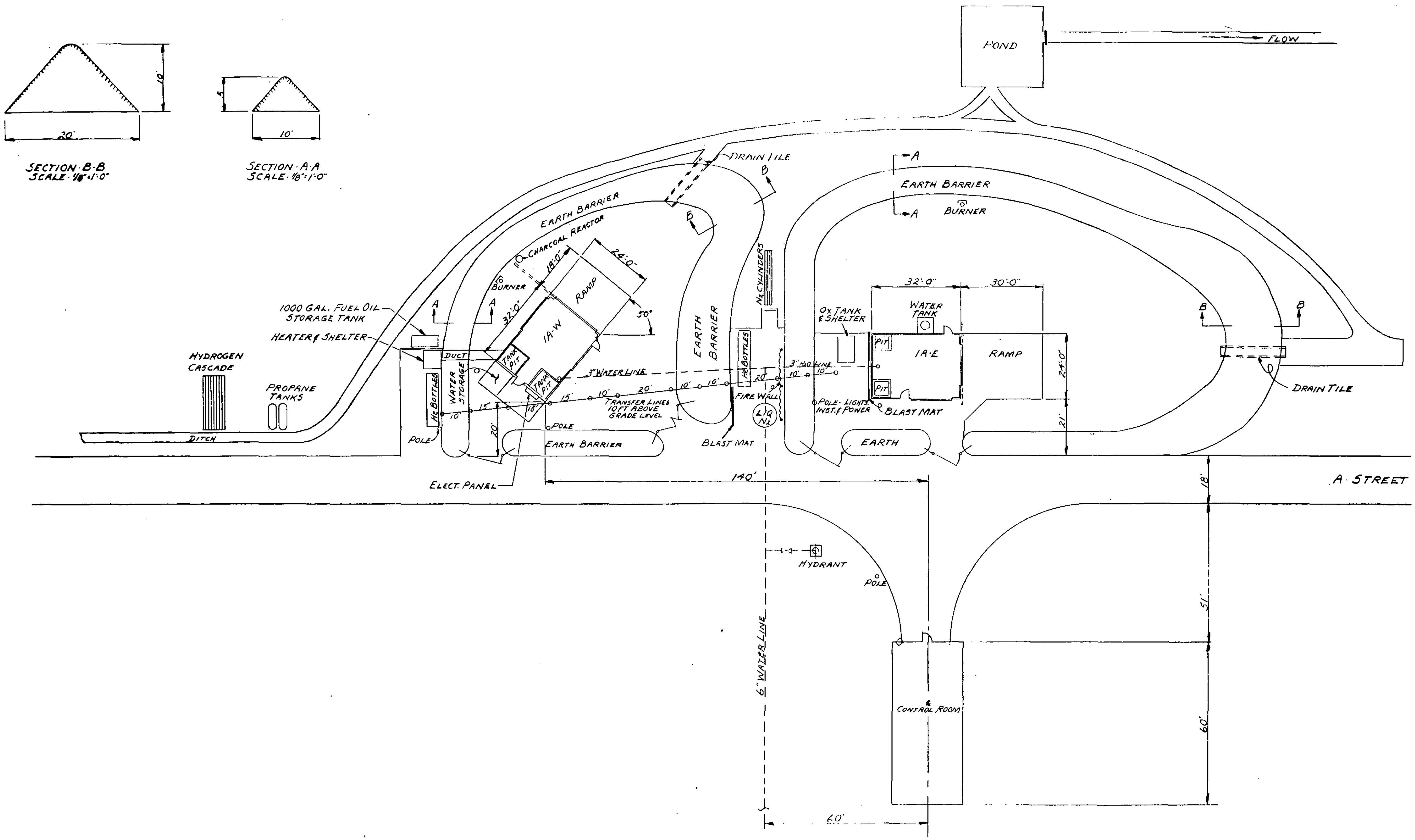

Figure AII-5. Plot Plan Cell 1-A, Air Force Plant No. 38

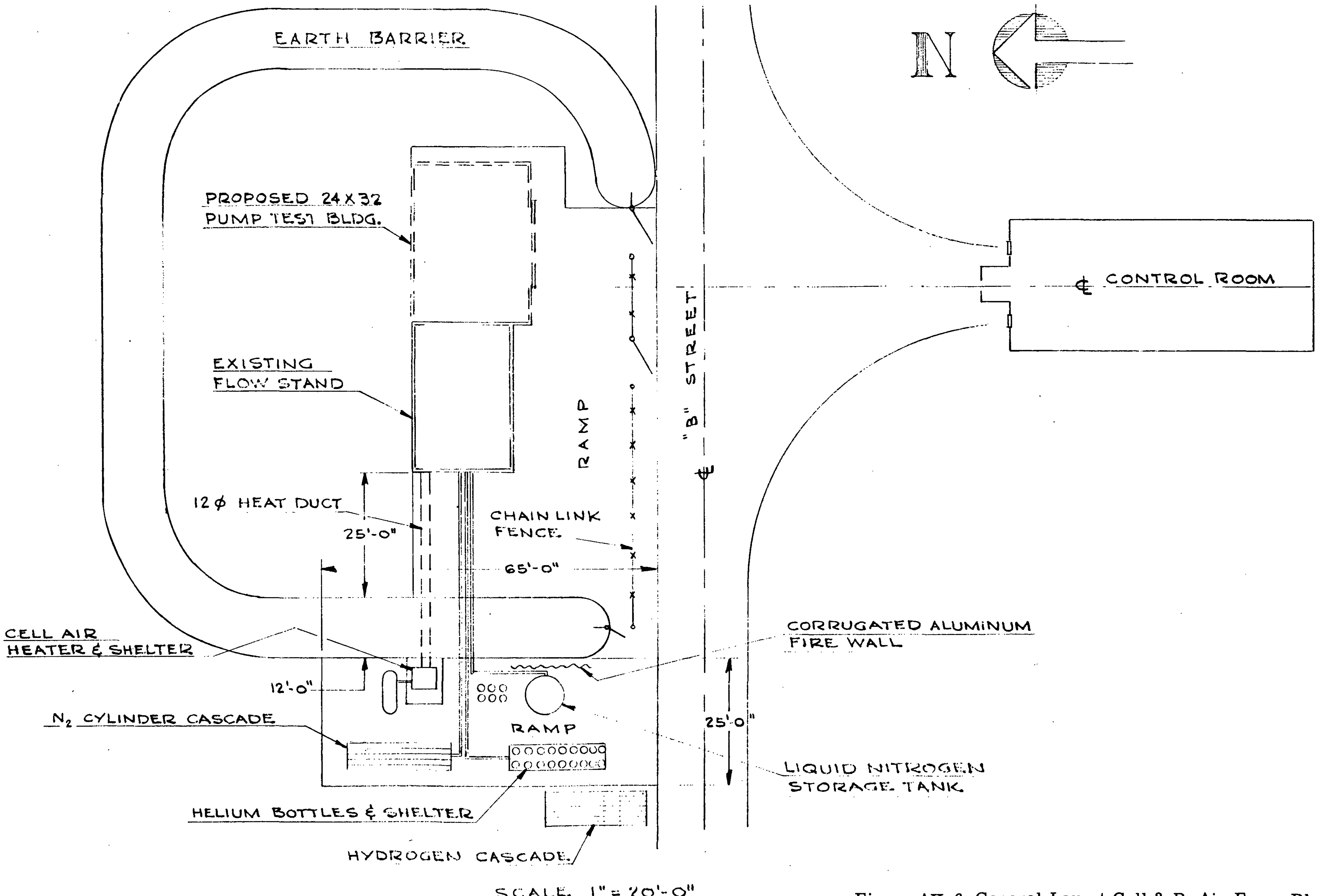

Figure AII-6. General Layout Cell 3-B, Air Force Plant No. 38

engines which are used to provide the drive for pumps undergoing investigation of performance characteristics.

3. Altitude Test Facilities (Attitude Control Systems)

One test cell at Air Force Plant 38 is equipped to conduct rocket firing tests at simulated altitudes ranging up to 110,000 feet. The ambient pressure at simulated altitudes is attained with steam aspirators. Hot gas from a thrust chamber of approximately 400 pounds thrust can be aspirated to a simulated altitude of 95,000 feet for ten minutes with a gas temperature of 1200°F. A smaller thrust chamber of 20 pounds thrust can be fired for longer periods using a smaller aspirator system that is capable of continuous operation. Both of these systems have three stages of aspirators with inter-condensers.

The altitude chamber of this test cell is nine feet in diameter and twenty-seven feet long. One end of the chamber is readily removable for the installation of test hardware. A four foot door is also provided for access of personnel. A thrust stand has been installed in the chamber that is capable of thrust levels to the limit of the aspirator capacity. Thrust can be measured from 20 to 200 pounds and can be calibrated by remote control with a dead weight tester while the thrust chamber is being fired at simulated altitude.

The altitude test chamber is complete with two 100 gallon high pressure MON and UDMH propellant tanks, valves and lines. In the event of a fire during a test, CO_2 and water deluge have been provided.

This altitude test facility has been in operation since mid-1960 in which thrust chambers of 20 and 200 pound thrust levels have been tested.

B. PROGRAM FACILITIES

1. General

The program for the fluorine hydrogen propulsion system could utilize manufacturing and test areas at Bell Aerospace Company's Wheatfield facility, and Air Force Plant No. 38 located 15 miles to the north of the Wheatfield Plant, for the development of the engine, flow and structural evaluation of the tankage and propellant system, development of the pressurization system and reaction control systems. Propulsion system development will be conducted at a new test site, assumed to be at Edwards Air Force Flight Test Center for planning purposes. Although past experience, and air sampling data, have indicated that the exhaust gas toxicity problem is not severe at the defined thrust levels, the potential toxic concentrations following an instantaneous reaction of the propellant quantities aboard the vehicle, such as might attend a catastrophic malfunction, dictate the use of a suitably remote site for system testing. In contrast to the tankage of the vehicle, tankage at the engine test stands has never presented a serious problem inasmuch as the individual test stand tanks discharge through stand pipes and are without bottom connections. These are installed in pits in the earth, in a barricaded test area with provisions for neutralization of minor spillage. In this instance gradual dispersion of minor spills associated with engine testing can be expected and will be tolerable. The required facilities are discussed briefly in subsequent paragraphs.

2. Engine Test Facilities

The development of the rocket engine as planned will entail the use of several test cells and installations for testing of the engine and its major components, as well as instrumentation and data recording centers. Oxidizer and fuel storage areas will be provided at the Bell Test Center at Air Force Plant No. 38. In addition, valve and control development will utilize existing stands and areas at Wheatfield. Environmental testing will be conducted on the components and engine in the Engineering and Research Laboratories at the Wheatfield Plant. A typical facilities plan for the engine development program at BAC is summarized in the accompanying table (Table II-1).

To illustrate the typical thrust chamber and engine test stands required for the fluorine-hydrogen engine program, the following brief description of a planned thrust chamber stand is presented.

A new test stand will be constructed to provide full duration thrust chamber horizontal firing capability. The control room will be a modified concrete powder magazine, supplied with all utilities, monitoring instrumentation, test area controls and firing console.

The test stand will be constructed about 200 feet north of the control station and will be surrounded by a ten-foot high earth embankment. A triple walled high pressure oxidizer tank and a jacketed fuel tank will be installed in below-grade pits and separated from each other and the firing stand by concrete walls. A prefabricated metal building will enclose the thrust stand and tank areas.

The ramp in front of the firing stand will be extended to support the aspirator diffuser used for altitude tests and the exhaust gas scrubber. The exhaust gas scrubber is planned because of the extended duration of operation required. Helium and nitrogen gas storage, propane tanks and liquid nitrogen storage will be located outside of and adjacent to the surrounding barrier. A waste fluorine burn-off stack, aspirator cooling water pumps and an effluent catch-basin will also be installed.

3. Propulsion System Test Facilities

The development of the propulsion system outlined by the test plan includes, in addition to the engine, the tankage and propellant system (fill, vent, and drain), pressurization system and reaction control system, as subsystems. The facilities plan paralleling the test plan provides for component level development of the subsystems at the Bell Aerosystems Company, Wheatfield, and Air Force Plant No. 38. In the plan, propulsion system test is provided for with a new facility, to be located at a remote site. These typical facilities requirements are summarized in Table II-2. The major items in the table, namely, the vertical test stands for propulsion system testing are described broadly below.

The fluorine-hydrogen test facility requirements do not differ significantly from those for more common propellants. Each test stand would be a steel structure, approximately 20 feet square at the base, equipped with a cantilevered beam and crane at the top, to lift the propulsion package from its transport equipment and lower it in place within the fixture. A retractable hoist would be required at the engine level.

Work platforms located at convenient levels, connected by open stairways and rapid-exit poles would be provided. Weather protection would be provided as required as well as vertical hot air ducts, dampered, supply fresh, conditioned air at any level. The steel work at the tower base may be coated with flame-resistant insulation. Areas which may be exposed to high concentrations of oxidizers are better left bare of any coating whatsoever.

Instrumentation cables would be contained in conduits and covered with fire-retardant insulation in the test stand area. Instrument and control cabling outside of the fire area can either be entrenched or above ground in steel raceways, depending on topography. Grounded electrical outlets are made available at each working level.

The flame deflector or bucket at the tower base would be provided with cooling water over an area designed to permit gimballing operation. Water would be pump-fed from a separate test stand reservoir tank. High pressure water to neutralize oxidizer spillage would likewise be supplied to a battery of fog nozzles located around the tower base and ramp area.

The propellants would be supplied from trailers and propellant loading operations remotely controlled. Any fluorine boil-off at the test stand would be recondensed by means of a liquid F_2/liquid N_2 heat exchanger, thereby eliminating the need for vent gas disposal.

With the control room located 500 feet from the test stand, a propellant load of twice the projected size could be safely handled within restrictions from the point-of-blast damage standpoint.

A pond for containment of test stand flame-deflector cooling and wash-down water should be located at least 100 feet from the stands. The pond may be provided with a pump for mixing and for draining or re-use after neutralization. Instruments for pH and redox indications would provide for the determination of neutralizing hypochlorite.

The control room would be of reinforced concrete construction, earth barricaded on the test stand side in conventional fashion. It would be equipped with observation windows to permit visual contact with stands and propellant areas. Individual firing panels would be provided for each stand and a master facilities panel for remote fueling, service, and emergency operations. Required instrumentation would be located in the control room.

A fresh-air supply can be generally assured by having two selective ducts conducting air from a considerable distance on either side of the control room. Stored air must also be available. All ventilation and air conditioning ducts would be designed for rapid closure in case of unfavorable wind conditions.

Closed circuit television would be used to monitor the test stands and other critical areas. Appropriate lighting, vapor proof and shock resistant, would be required for around the clock coverage. Meteorology equipment would monitor wind directions and velocity constantly. Effective and reliable communications with all inhabited areas on the base is essential and all equipment should be provided with an emergency power source. The control room, the center of test operations, would monitor and direct base operations during critical periods of the test procedures.

TABLE II-1

ENGINE FACILITIES SUMMARY

Plant Location	Planned Cell Utilization	Status
AF No. 38	Injector Tests	Existing cell designed and used for fluorine work. Reinstallation of tanks and plumbing required.
"	Limited Duration thrust chamber tests	Existing cell similar to above. Increased duration tankage and plumbing required.
"	Full duration thrust chamber tests	Blockhouse available, modifications and new installation required.
"	Gas generator testing and venturi calibration	Existing cell with some additional equipment required.
"	Pump water calibrations	Existing cell completely equipped.
"	Oxidizer pump and turbine rotor testing	Blockhouse available, modifications and new installation required.
"	Fuel pump testing	Blockhouse available, modifications and new installation required.
"	Turbine Pump Assembly	Blockhouse available, similar to above.
"	Engine testing	Blockhouse available, modifications and new installation required.
"	Engine testing	Blockhouse available, modifications and new installation required.
"	Oxidizer Component flow testing	Blockhouse available, modification and new installation required.
"	Fuel Component flow testing	Blockhouse available, modification and new installation required.

TABLE II-1 (Cont'd)

ENGINE FACILITIES SUMMARY

Plant Location	Planned Cell Utilization	Status
AF No. 38	Fuel Component flow testing	Blockhouse available, modification and new installation required.
"	Data acquisition and recording	Existing center connected with existing cells.
"	Data acquisition and recording	New installation of digital data system to accommodate requirements of new cells.
"	Oxidizer storage	Area preparation, new installation of triple walled storage vessels, distribution system and attendant equipment required.
"	Fuel storage	Parking area preparation for extra trailers required.
Wheatfield	Valve and Control testing	Existing cells and areas requiring minor modification.
"	Component and engine environmental testing	Existing areas and equipment, test fixtures required.

TABLE II-2

PROPULSION SYSTEM FACILITIES SUMMARY

Plant Location	Planned Cell Utilization	Status
AF No. 38	Tank slosh, vibration testing	Blockhouse available, new installation of main fixture, hydraulic power, actuators, pressurization and instrumentation required.
"	Tank expulsion testing, pressurization and propellant system testing	As above. Secondary test fixture for pressurization system and flow testing required.
Wheatfield	Tank model testing. Gas bottle structural testing	Existing area, minor modifications to slosh test fixture required.
Wheatfield	Component level valve and control testing	Existing cells and areas, minor modifications required.
Wheatfield	Component level valve and control environment testing	Existing area and equipment, test fixtures required.
AF No. 38	Attitude control thrust chamber testing	Existing cell, minor modifications required.
"	Attitude control system testing	Existing cell, minor modifications required.
"	Data acquisition and recording	Applicable from engine facility plan.
EAFFTC	Propulsion system testing	New installation required.
EAFFTC	Propulsion system testing	New installation required.

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