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LIQUID ROCKET BOOSTER STUDY
FINAL REPORT

GENERAL DYNAMICS
Space Systems Division

APPENDIX 6

LRB FINAL REPORT FROM ROCKETDYNE

Rockwell International

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RI/RD88-180

LIQUID ROCKET BOOSTER

PHASE II STUDY REPORT

8 June 1988

PREPARED BY

ROCKETDYNE DIVISION
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LIQUID ROCKET BOOSTER PHASE II STUDY REPORT

FOREWORD

This Phase II report, containing results of the Liquid Rocket Booster Study is submitted to General Dynamics Space Systems Division (GDSS) in accordance with General Dynamics contract 08-01290. This program was conducted under the direction of GDSS program manager Paul Bialla and Propulsion Project Manager Gopal Mehta. This document describes the results of a Liquid Rocket Booster engine study conducted in two parts; (1) Pressure fed engine design and analysis carried forward in more detail using the results of the Phase I studies, and (2) Pump-fed engine parametric and design point data. Technology program elements for the booster engines are also presented in this report.

Specific costs are not included in this report due to their proprietary nature; however, they have been submitted to General Dynamics under separate cover.

ABSTRACT

Phase II of the Liquid Rocket Booster Study was conducted over a four month period by Rocketdyne. For the pressure-fed engines, detailed trade studies were conducted defining engine features such as thrust vector control method, thrust chamber construction, etc. This was followed by engine design layouts and booster propulsion configuration layouts.

For the Pump-fed engines parametric performance and weight data was generated for both O_2/H_2 and O_2/RP-1 engines. Subsequent studies by GDSS and NASA resulted in the selection of both LOX/RP-1 and O_2/H_2 propellants for the pump-fed engines. More detailed analysis of the selected LOX/RP-1 and O_2/H_2 engines was conducted during the final phase of the study.

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The following individuals have made significant contributions to the work shown in this report:

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LIQUID ROCKET BOOSTER PHASE II REPORT

1.0 INTRODUCTION

The use of liquid rocket boosters (LRB) for the Space Shuttle is being investigated in detail by General Dynamics Space Systems Division. Rocketdyne, under subcontract to GDSS, is studying pressure fed and pump fed propulsion systems which may be applied to Space Shuttle booster propulsion. The initial effort covered parametric performance, weight and cost data covering a range of propellant combinations and engine thrust levels and chamber pressures. Parametric data from the Rocketdyne 1972 Phase A/B Pressure Fed Space Shuttle Study and the 1986 AFRPL Low Cost Expendable Propulsion Study (LCEPS) were examined for applicability.

Following the initial parametric studies, trade studies were conducted to define the basic elements and features of the engines. These studies covered selection of cooling method, injector type, thrust vector control system, ignition method, and basic engine control method. Engine candidates for both pressure and pump fed applications were formulated based upon the results of the trade studies. Emphasis was placed on expendable engines based on the results of GDSS studies.

Conceptual engine design layouts for both ablative type and regeneratively cooled pressure fed thrust chambers were prepared showing general constructive details for the major engine elements. Emphasis was placed on LOX/RP-1 propellants since they were the propellants of choice based on General Dynamics trade studies. A safety and reliability analysis was conducted to compare ablative and regenerative type chambers with the conclusion that both types of chambers could be developed to have a high degree of safety and reliability. Regeneratively cooled pressure fed engines were chosen based on a careful evaluation of factors such as experience base, overall safety, etc., and on the results of the General Dynamics trade study.

Pump fed engine concepts were defined based on the ongoing STBE (Space Transportation Booster Engine) studies except that RP-1 was used as the fuel rather than methane. RP-1 was selected by GDSS rather than methane or hydrogen based on overall trade studies of size, cost, experience base, etc. For the pressure fed engines a complete list of engine feature options was developed and trade studies were conducted in order to define the most desirable expendable engine features.

1.1 APPROACH

Each alternative characteristic was then briefly considered and compared with alternates , and during the Phase I study the most desirable chosen for combination into five candidate engine configurations. These were then qualitatively evaluated and the most desirable chosen, one for the reusable and one for the expendable category. In Phase II emphasis was placed on expendable engines and detailed evaluations were conducted of cooling method, injector type and gimbal systems. Pump fed engine designs were developed based on the ongoing STBE and STME studies using LOX/RP-1 and LOX/H_2 as the propellants of choice.

This report briefly describes the decisions that were made and the basis for these decisions, and describes the work accomplished.

1.2 REQUIREMENTS

The LRB is intended to replace the solid rocket boosters (SRB) on the STS. As a point of reference, some general characteristics of the SRB are shown in Table 1-1.

Table 1-1. SRB Characteristics

OVERALL WEIGHT	1.25×10^6 lb
PROPELLANT WEIGHT	1.11×10^6 lb
INERT WEIGHT	146,000 lb
NOZZLE WEIGHT ONLY	23,000 lb
CHAMBER PRESSURE AT START	860 PSIA
THRUST AT LAUNCH	3.24×10^6 lbf
EXPANSION RATIO	7.16
MAX GIMBAL ANGLE, FLEX NOZZLE,	8°
LENGTH	149 ft
DIAMETER	12.2 ft

1.2.1 Ground Rules and Baseline System

The Liquid rocket Booster engines will have the following characteristics as a baseline for this study:

1. There shall be four engines in a group feeding from one pair of propellant tanks, with each of these two assemblies replacing the present SRBs on the Space Shuttle assembly and meeting the increased STD requirements defined in the NASA SOW to GDSS.
2. The exit diameter of each engine shall not exceed 108 inches.
3. Mission safety and reliability have top priority considerations ahead of and above all other factors. Volume is to be minimized in preference to weight where they conflict; otherwise, weight and cost are to be minimized.
4. Inlet pressures shall be as low as possible so as to permit lower propellant tank pressures, thus reducing tank weight and pressurant storage volume.
5. Emphasis was placed on expendable type, but both it and reusable designs are to be studied and compared.
6. Chamber pressures and thrust levels for the pressure fed and pump fed applications are determined by GDSS trade studies.
7. The mixture ratio shall be 2.5 for LOX/RP-1 and 6.0/6.9 for LOX/H_2.
8. Thrust vector control shall provide for a six degree excursion each direction at a maximum angular slewing rate of 10 degrees per second and an angular acceleration of one radian per second squared.

Injector Requirements. The prime injector requirements are to provide high combustion performance with dynamic stability at all system operating levels. Dynamic stability means that the system will recover within a prescribed time from a range of chamber overpressures resulting, for example, from the detonation of high explosives within the combustion zone. In addition, chamber pressure oscillation levels shall be less than 10% of the steady-state pressure over the intended operating range. The elimination of pops or self-triggers, however fast they may damp, is a design goal. No damage may result from a self-induced disturbance.

Thrust Chamber Requirements. The thrust chamber maximum exit diameter shall be approximately 108 inches. Safety and reliability are highest priority followed by minimum length, weight and cost. The required specifications are generally those listed above under the heading "Ground Rules and Base Line System"

Throttling Requirements. The thrust level shall be capable of being throttled down to 65% of nominal for the pressure fed engines and +10% to 35% for the pump fed engine. Safety and reliability, and thus stability, will have first priority.

Thrust Vector Control Requirements. The thrust vector control requirements are given in item 8, above, in the section titled "Ground Rules and Base Line System". Safety and reliability are again first priority followed by size, weight and cost.

Controls and Ignition Requirements. Since safety and reliability are first priority items, the control system must utilize relatively simple measures to insure that false signals or system noise do not compromise proper system operation. A careful study of redundancy and automatic supervisory methods as they apply to measurements, controls and health monitoring will be required. A safe shut down (abort) in flight is a prime consideration in the selection of the control system.

2.0 PRESSURE FED LOX/RP-1 ENGINE

This section presents the selected LRB pressure fed booster rocket engine configuration and characteristics resulting from the technical analyses and trades studies.

A baseline engine concept was selected based on previous studies and experience along with trade studies for the STS application. An engine performance and pressure balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, and nozzle characteristics and leading to the present configuration and physical design.

2.1 RECOMMENDED CONFIGURATION AND CHARACTERISTICS

The expendable, pressure fed, engine selected during this study uses the RP-1 fuel as a coolant in a single up-pass, tube wall combustion chamber and nozzle, as illustrated in the simplified schematic shown in Figure 2-1. The oxidizer, LOX, is fed into the engine through a flexible propellant line, through a combined shut-off and throttle valve and into a propellant distribution manifold located above the injector; it then flows into the combustion chamber through the injector orifices. The fuel, RP-1, enters the engine through a similar flexible propellant line and valve before entering a distribution manifold at the nozzle exit. After the fuel passes upward through the tubular wall, it passes into the combustion chamber through the fuel injector orifices. A POGO suppression system located near the oxidizer valve is provided to preclude very low frequency oscillations due to coupling of the engine thrust with the propellant supply system. Key operating parameters developed during this study are shown in Table 2-1.

2.1.1 Regenerative Cooling

Full regenerative fuel cooling of the thrust chamber was selected over an ablative type thrust chamber based upon trade studies conducted jointly by Rocketdyne and GDSS. The major considerations in this evaluation are summarized below.

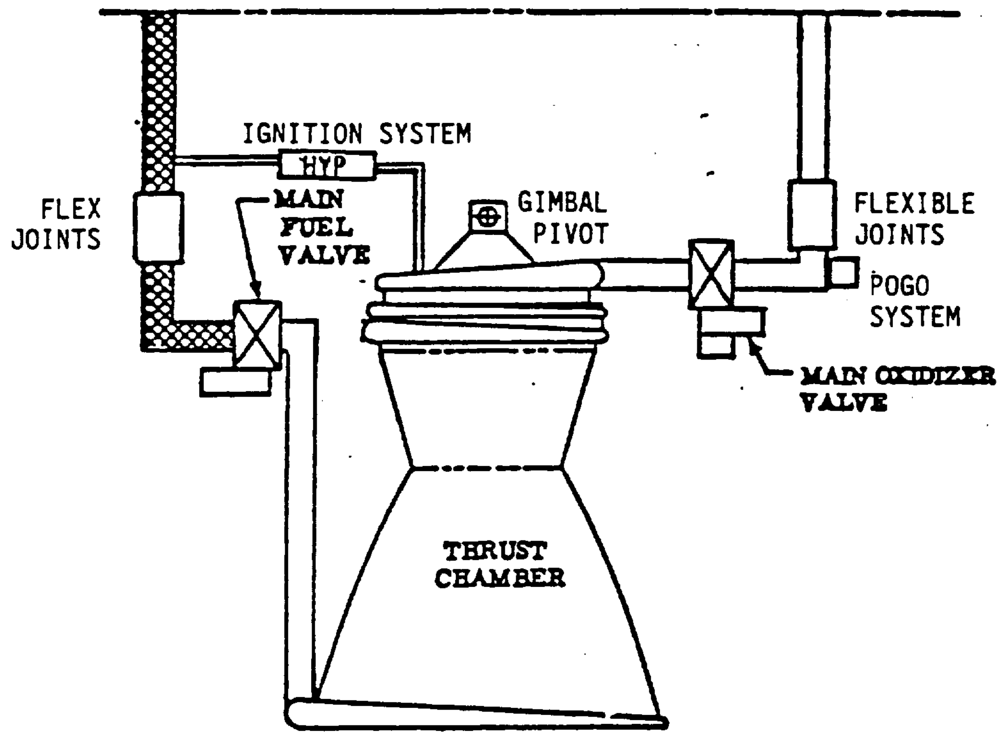


Figure 2-1. Simplified Engine Schematic

Table 2-1. Pressure-Fed Booster Engine

PARAMETERS		FEATURES
• PROPELLANT	LOX/RP-1	
• THRUST (SEA LEVEL)	800 Klbf	
• CHAMBER PRESSURE	330 psia	
• MIXTURE RATIO	2.5	• HYPERGOL IGNITION SYSTEM
• EXPANSION RATIO	5.3:1	• SINGLE BALL TYPE VALVES FOR LOW PRESSURE DROP
• Is, SEA LEVEL	238.7 sec	
• Is, VACUUM	279.0 sec	
• NOZZLE		
• EXIT DIAMETER	108.1 in	• ELECTRO-MECHANICAL ACTUATORS FOR TVC AND VALVES
• C_F, SEA LEVEL	1.40	
• C_F, VACUUM	1.635	• SAME VALVES SERVE AS
• PERCENT LENGTH	80%	• SHUT-OFF
• COMBUSTION CHAMBER		• THROTTLING
• CONTRACTION RATIO	1.676	
• CHAMBER LENGTH	48.35 in	
• THROAT DIAMETER	46.97 in	
• L-STAR	77.67 in	
• INJECTOR		
• DIAMETER	60.81 in	
• MODULE DIAMETER	11 in	
• THRUST PER MODULE	42 K lb	

1. Safety considerations favor the regeneratively cooled engine over the ablative since a burn through on the ablative engine could propagate to adjacent engines or even to the propellant tanks so that unless engine shutdown is quickly initiated, a catastrophic failure may result. On the other hand, if there is a tube leak in a fuel cooled chamber, it generally does no propagate since the leaking fuel tends to cool down the area around the leak preventing spreading of the failed area.

2. Quality control - The detection of debond areas in an ablative chamber or other defects may not be possible through normal non-destructive techniques such as x-raying. On the other hand, regeneratively cooled thrust chambers are pressure tested for possible leaks and then hot fired to check out performance and durability.

3. Performance - The overall booster performance for regenerative and ablative type thrust chambers is nearly equal. Although the regenerative system requires a higher tank pressure (resulting in an increase in booster weight) the performance of the ablative type engine is somewhat lower (~ 1%) than the regenerative system since a fuel rich bias is used at the ablative wall to prevent excessive erosion of the wall. Trade studies have shown that these 2 factors offset each other.

2.1.2 Injector Selection

The selected injector incorporates design techniques based on lessons learned in a number of previous designs built and tested by Rocketdyne over the past years. Although simplification of design and cost reduction have been on-going goals, the major concern in this effort has been reliability and safety and the goal of providing the lowest possible pressure drop requirements consistant with adequate stability margin and performance under nominal as well as off-design conditions such as occur during engine throttling. A cross section drawing of the selected injector is shown in Figure 2-2a, and the pertinent injector characteristics are summarized in Table 2-2.

These are based on and developed from the engine balance for the selected engine. Key injector considerations influencing the design were the

INJECTOR ASSEMBLY

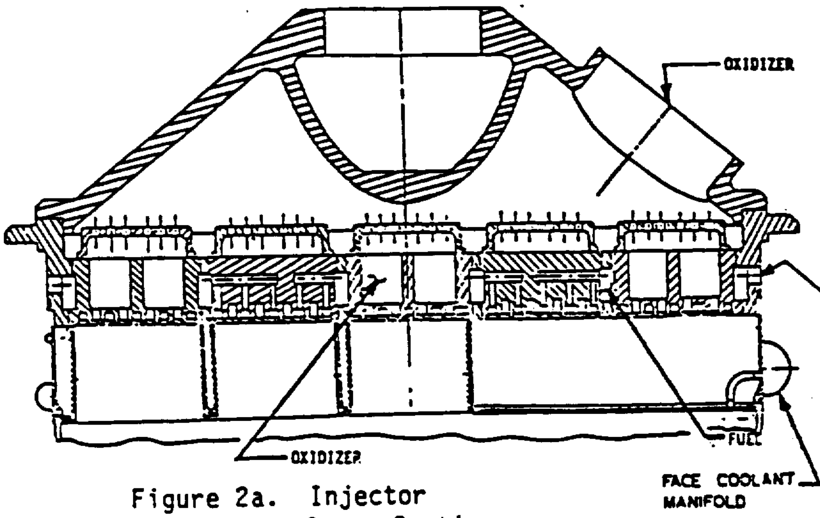


Figure 2a. Injector Cross Section

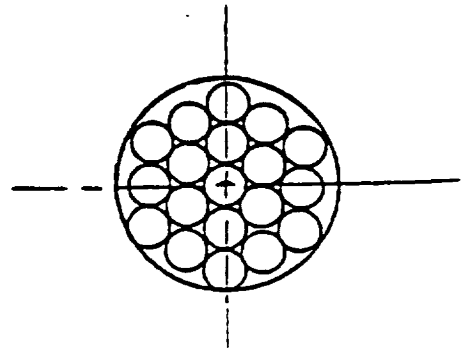


Figure 2b. Distribution of 19 Modules

NOT TO SCALE

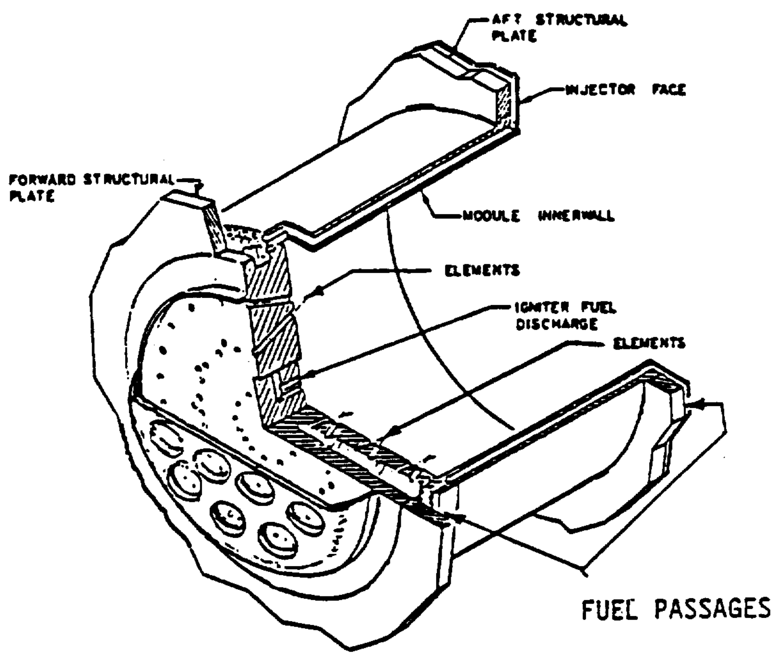


Figure 2c. Cross Section of a Typical Module

Table 2-2. Injector Characteristics

Number of modules	19
Diameter	60.8 in.
Element Pattern Type	like on like in alternate rings

Construction:

- Constructed of CRES, or aluminum, TBD.
- Flat Face with 19 modules sunk into the face.
- Injector face is fuel cooled.
- Cannul walls are fuel cooled.
- Cannular floor containing injector elements are fuel and oxidizer cooled.

propellants (LOX/RP-1), combustion efficiency, chamber pressure, flowrates, propellant injection temperatures, injection pressure drop, and, most important, safety and reliability considerations.

The injector shown in Figure 2-2a is divided into 19 separate segments as shown in Figure 2-2b. Each segment is round in shape and is a miniature injector in the shape of a can (see Figure 2-2c), thus the name "cannular injector". Each can is sunk into the injector face plate. The sides of each can serves the function of a baffle. Each can thus provides 1/19 of the total nominal engine thrust, or about 42,100 lb of thrust per can. Each can contains alternate rings of like on like oxidizer elements, and like on like fuel elements. The outer ring in each segment is a fuel ring to provide a fuel rich environment for the surrounding walls. The injector face and walls of each can are cooled by fuel flowing through cooling passages behind the surface.

A like on like injector orifice element pattern was selected to give a well known, conservative injector orifice layout having the highest performance consistent with an adequate stability margin.

Each can has a central igniter tube which carries the hypergol ignition fluid from the hypergol reservoir and manifold through the injector and into each injector can. The hypergol fluid is in the form of a slug of liquid which is fed through the tube by the pressurized fuel in back of it. For increased reliability, a dual ignition system is used: one system serves to ignite every other can, while the other ignites those in between. If one system should fail, the other will serve to ignite the engine by itself.

2.1.3 Main Combustion Chamber and Nozzle

A conceptual design of the combustion chamber and nozzle is shown in Figure 2-3. A simple tube design was selected for the nozzle, constructed from 347 CRES tubes of constant diameter and wall thickness, available as stock tube material and formed with simplified tooling. Nozzle reinforcing structure will also be low cost and from 347 CRES or composite materials. The tubes are layed next to each other and brazed together along their entire length. This is conventional construction proven to be very reliable. New manufacturing techniques having cost advantages are being considered and are described further in the section on New Technology.

The manifolding will be designed to provide a minimum of pressure drop and will eliminate complicated manifold closing concepts, weld overlays, and weld joints close to the exit diameter. Liberal tolerances will be all allowed where appropriate.

Rocketdyne is conducting a separate study (funded by the Air Force Astronautics Lab) to reduce the construction cost of this type of combustion chamber and nozzle. Improved methods will be used where applicable, and where the end product in no way compromises the safety and reliability of the engine.

2.1.4 Gimbal System Selection

A trade study was made of 3 candidate nozzle/gimbal types after elimination of the liquid injection thrust vector control based on Phase I study results. Table 2-3 shows the result of the gimbal system trade for: (1) regenerating cooled tube wall nozzle with a head end gimbal, (2) an ablative type nozzle

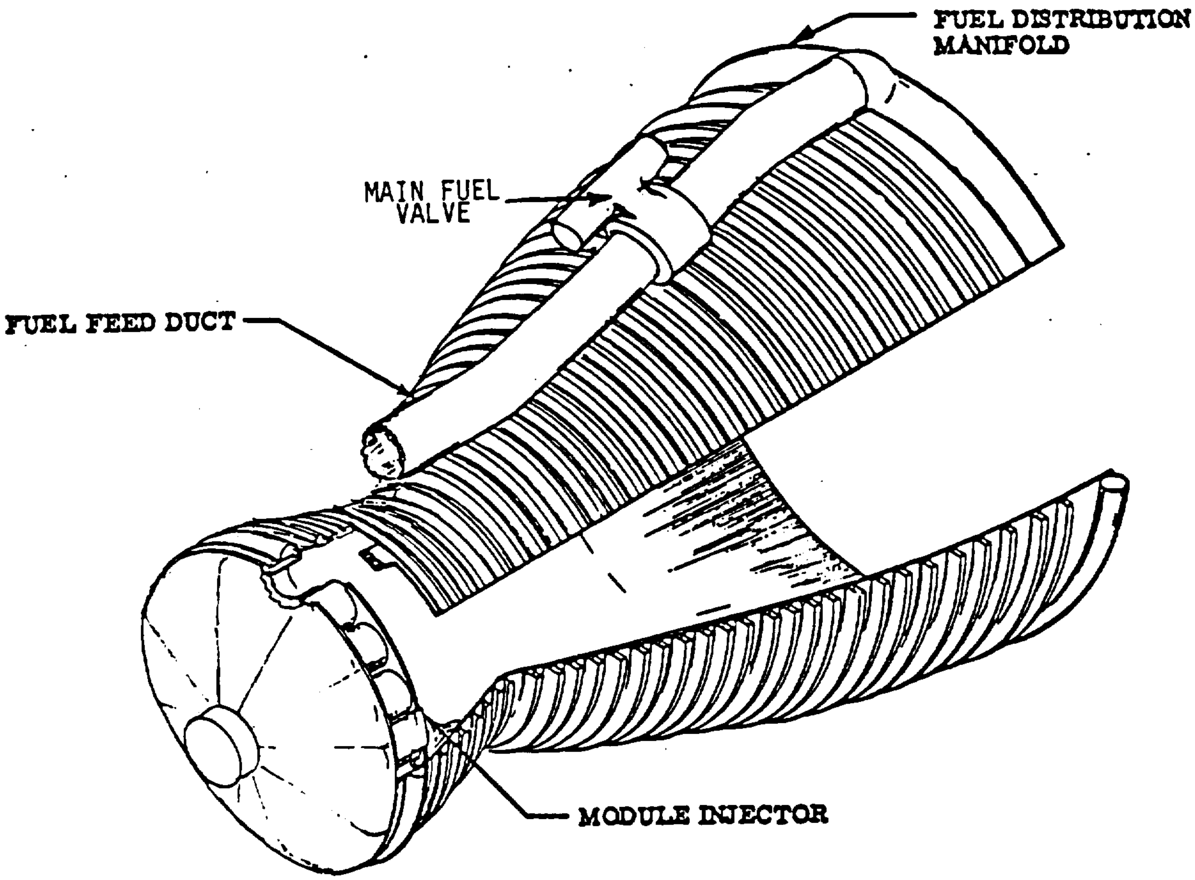


Figure 2-3. Thrust Chamber Assembly

with a head end gimbal, and (3) an ablative nozzle using a flexible nozzle based on solid rocket motor technology. Included in Table 2-3 are all important weight factors which need to be accounted for in the trade study: engine weight, flexible line (bellows) weight, flex nozzle ring weight, actuator weight, and total booster weight increase due to the increase in fuel tank pressure required by the fuel regenerative cooling of the thrust chamber. In addition, there is a slightly lower performance projected for the ablative chamber due to the use of some additional boundary layer fuel cooling at the chamber wall to protect it from possible oxidizer streaking. This effect translates into a booster weight increase of approximately 9000 lbs based on the projected 1% lower I_{sp} for the ablative design.

Table 2-3. Weight Comparison For Pressure Fed Engine

(Thrust = 750,000 LB, Chamber Pressure 400 psia)

	REGENERATIVELY COOLED TUBE WALL HEAD END GIMBAL	ABLATIVE HEAD END GIMBAL	ABLATIVE FLEX NOZZLE	
CONSTRUCTION	TUBES ARE 1/2 IN DIAM.AT NOZZLE EXIT	ABLATIVE THICKNESS CHAMBER--1.6 IN THROAT---2.0 IN EXIT-----1.0 IN	DITTO DITTO DITTO	
ENGINE WEIGHT	4,237 LB	5,760 LB	5,700 LB 1,150 6,850	EXTRA WEIGHT FOR FLEX NOZZLE RING, ETC.
Δ EXTRA WEIGHT FOR FLEX LINES	1,160 5,397	1,160 6,920		
ACTUATORS	332 5,729	332 7,252	332 7,182	
4 ENGINES PER BOOSTER X 4 =	22,916	29,008	28,728	
BOOSTER WEIGHT INCREASE DUE TO DELTA P OF COOLING JACKET	16,800 39,716	29,008	28,728	
Δ I_{sp} IMPACT IN LBS TO PREVENT OXIDIZER STREAKING	0 ~39K	~9000 ~38K	~9000 ~38K	

2.2 PRELIMINARY DESIGN AND ANALYSIS

2.2.1 Engine Layout and Description

The regeneratively cooled LRB engine is shown in Figure 2-4. Fuel is conducted through an 11 inch inside diameter tube to a fuel manifold at the nozzle exit after passing through the main propellant valve. This conduit contains three flexible joints and is bent in a so called "wrap-around" configuration visible

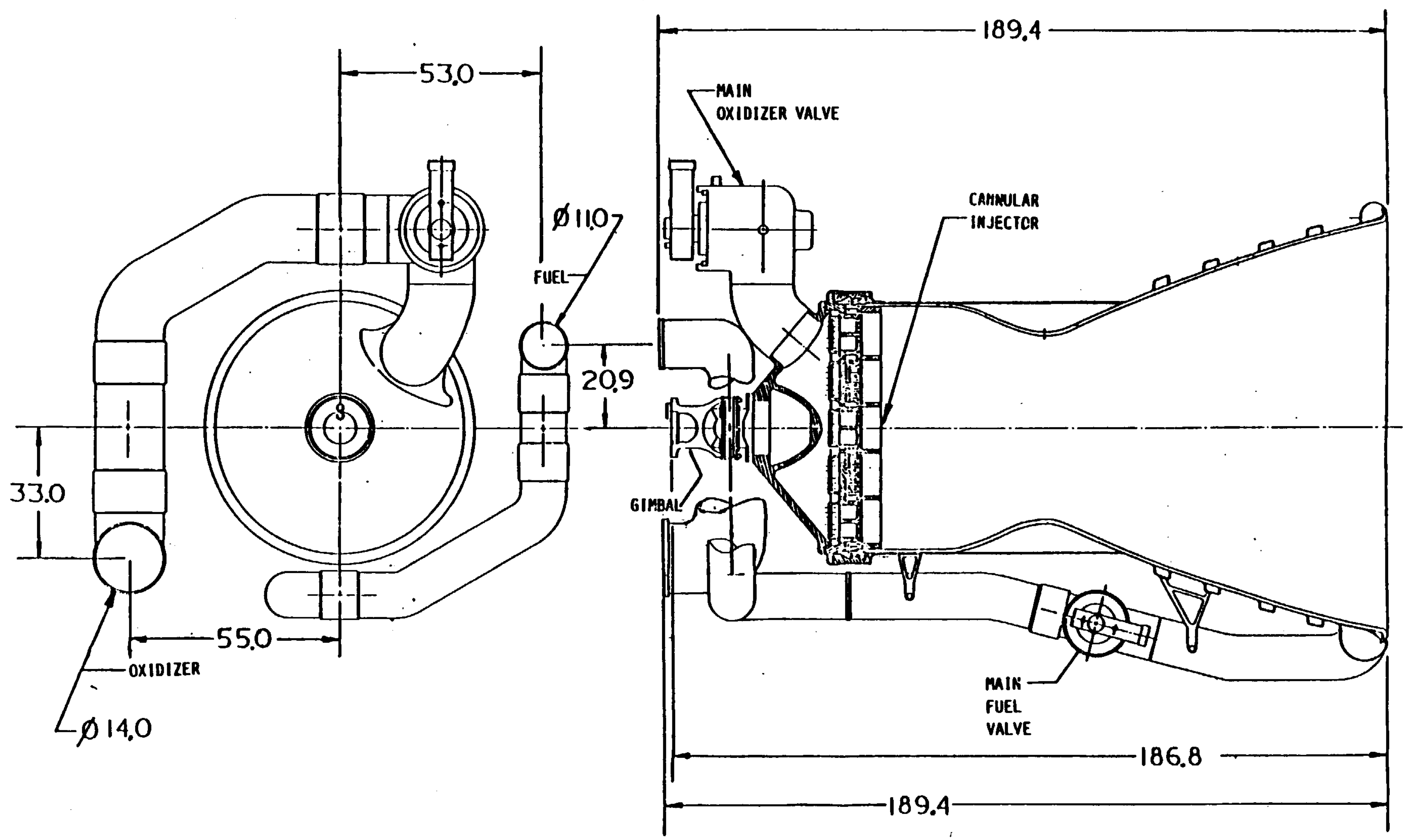


Figure 3-4. LRB Pressure Fed Engine System
Thrust = 800,000. lb.
Chamber Pressure = 330. psia

in the top view shown in Figure 2-4. The fuel then passes from the nozzle exit to the injector through the tubes which constitute the nozzle and combustion chamber walls. There are 680 tubes, each nominally 0.50 in. in diameter which are brazed side by side along their length and brazed into the distribution manifold at the nozzle exit and brazed into the injector fuel manifold at the other end. The tubes are constructed of 347 CRES tubing of uniform wall thickness (0.012 in. thick), but they have each been "booked" and bent identically resulting in a gradually changing crossectional flow area from the nozzle to the throat and from the throat to the injector with the narrowest passage at the throat. The entire fuel flow of 1015. lb/sec of RP-1 passes through these tubes, serving to cool the chamber walls and utilize the collected energy which is returned to the combustion chamber.

The liquid oxygen enters the engine through a wrap-around duct 14 in. inside diameter and containing three flex joints. It then passes through the main oxidizer valve and into the injector manifold at the top of the injector. Just upstream of this valve is a port through which a small flow of gaseous nitrogen is forced during the pre-ignition period of engine startup. This is to prevent the phenomenon of geysering which might otherwise occur in a boiling liquid. The fuel and oxidizer both pass into the combustion chamber through orifices in the injector. The injector is divided into 19 separate sections to increase engine stability. Further details regarding the injector are discussed in a previous section. (See Sec. 2.1.2.)

The engine a has hypergol ignition system similar to that utilized on the F-1, except that it is duplicated. One system feeds a hypergol propellant slug into every other one of the 19 injector cans while the second system feeds a similar slug into the remaining alternate cans. One system will ignite the engine, so the duality also furnishes the engine with a built in backup ignition system. The hypergol slugs are pushed into their respective cans by fuel under propellant tank pressure. The hypergol flow is thus immediately followed by a very small continuous fuel flow. (This is also discussed in section entitled Operations.)

In order to accept the active end of the gimballing actuators, the engine is furnished with outrigger struts spaced 90 degrees and projecting outward from the engine body. The thrust of the engine is carried through a ball pivot located at the top of the LOX manifold above the injector. This contains the engine gimbal pivot point. The ball turns within the socket when the engine is gimballed back and forth by the gimbal thrusters. The ball and socket are lubricated with a dry lubricant having a very low coefficient of friction. (This is further covered in section 2.2.5.) An active POGO suppression circuit is connected to a branch of the oxidizer line just upstream of the main oxidizer valve. It has an associated control system which is automatically activated if the vehicle and booster stage should pass into a POGO type oscillating mode.

The engine is provided with instrumentation for 1) health monitoring, 2) automatic control, and 3) special operations. (This is discussed in the following Section 2.2.2.)

The two main propellant valves are furnished with closed loop valve positioners so that precise position settings can be accurately repeated allowing accurate calibration and open loop control of engine thrust and mixture ratio. All valve actuation is by electric power, obviating the need for a hydraulic control system. Since this type of actuation is relatively new, the engine is designed to allow a change to hydraulic actuation such as has been used in the F-1 rocket engine if the reliability of the all-electric system is deemed too low. Alternatively an electrically actuated valve with pneumatic override can be used of which an example is shown in Figure 2-5.

The system will be monitored by an electronic health monitoring system and will be controlled by an electronic sequencer having a number of different modes consistant with different operating stages such as pre-ignition stage, shutdown, emergency shutdown, etc. This system will receive electronic commands from the vehicle system controller and/or from ground control. Electronic interlocks are provided to ensure the maximum operational safety, especially during startup and emergency shut down. (These are outlined in the sections below.)

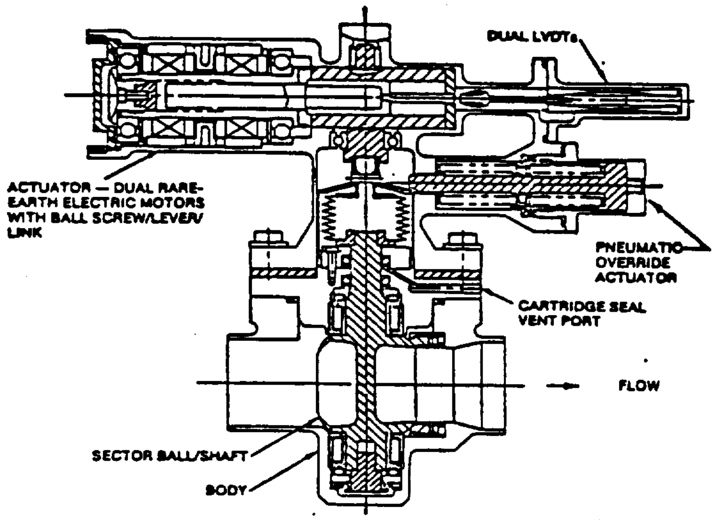


Figure 2-5. Conceptual Valve Design

2.2.2 Schematic and Flight Instrumentation

A schematic of the engine is shown in Figure 2-6. The prevalves shown in each of the propellant lines are not strictly considered to be part of the engine. They are included to permit a clearer description of the engine operation and to better define the engine interface. A list of the engine components shown in the schematic and their symbols is given in Table 2-1. The gimbal actuators are not shown since it is customary that these be considered outside the engine package. On the other hand, an analysis of the torques and power required of these actuators has been calculated and is furnished in Section 2.2.5. The function and use of each of the instruments is discussed in the section entitled Engine Operation, Section 2.2.6.

2.2.3 Performance (Full Thrust and Throttled)

Theoretical parametric performance data was generated to permit optimization of the vehicle. The nozzle exit diameter was limited to 108 inches maximum. The mixture ratio is held constant at 2.5 and the C-star combustion efficiency is held constant at 0.94. This efficiency is considered to be attainable, and to be a reasonable compromise between stability margin and performance. A

Figure 2-6. Pressure Fed LRB Engine Schematic

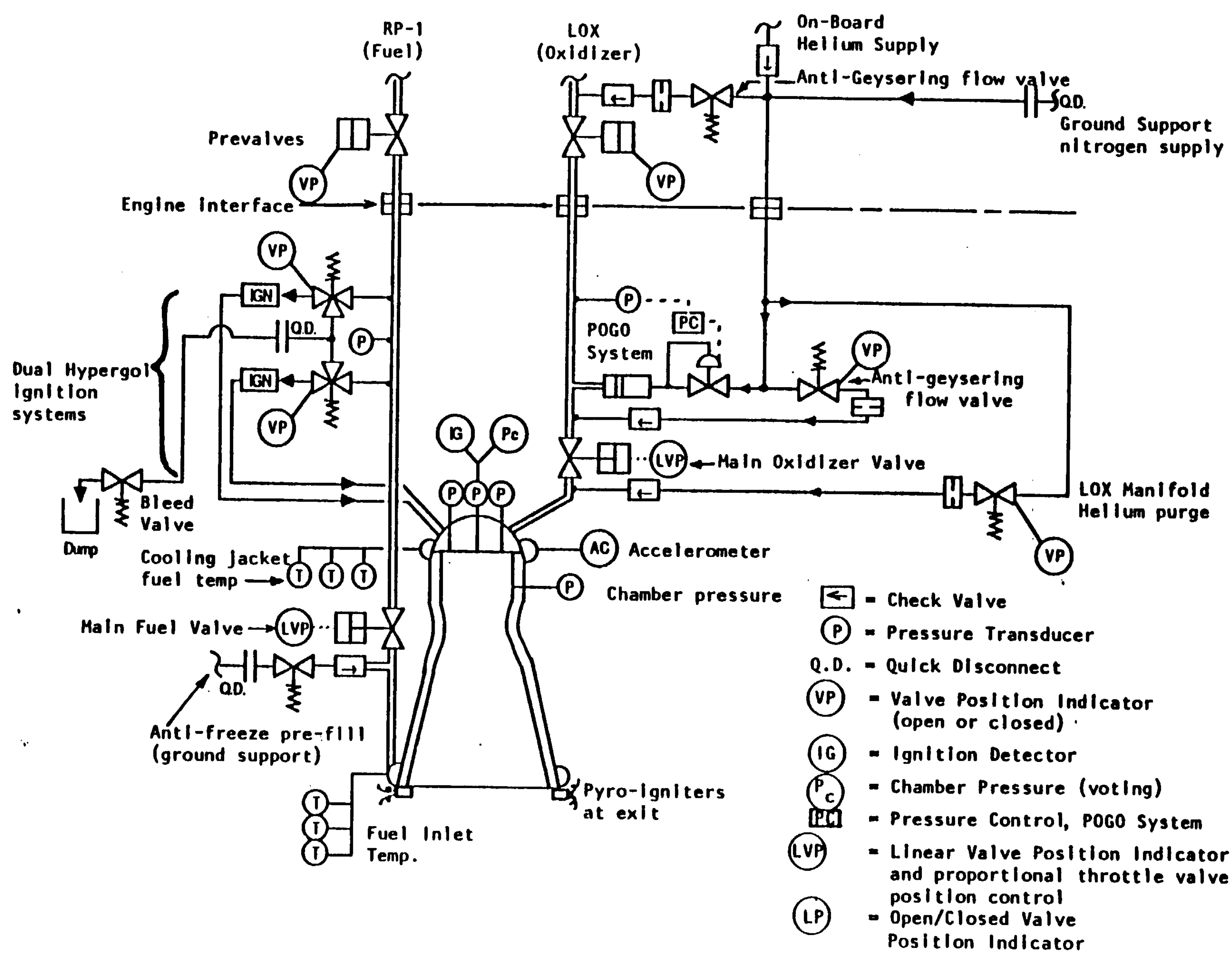


diagram showing the meaning of the dimension terms used in the performance computer printouts, Figures 2-8 and 2-9 is shown in Figure 2-7.

The design point, shown in Figure 2-8, consists of an engine having a sea level thrust of 800,000 pounds and a chamber pressure of 330 psia. The characteristics of this same engine when throttled to 60% of nominal thrust is shown in Figure 2-9. The pressure budget values and their relative changes under throttled conditions and under propellant tank blowdown conditions are shown in Table 2-4. Referring to the table, the engine chamber pressure is first decreased in steps by closing down on both the oxidizer and fuel throttle valves while holding the engine inlet pressure constant (top half of Table 2-4). Secondly, both throttle valves were set so as to give 95 percent of nominal chamber pressure, and this setting was held constant while the engine inlet pressure was decreased in steps. This condition explores the possibility of allowing the propellant tank pressures to decrease in order to

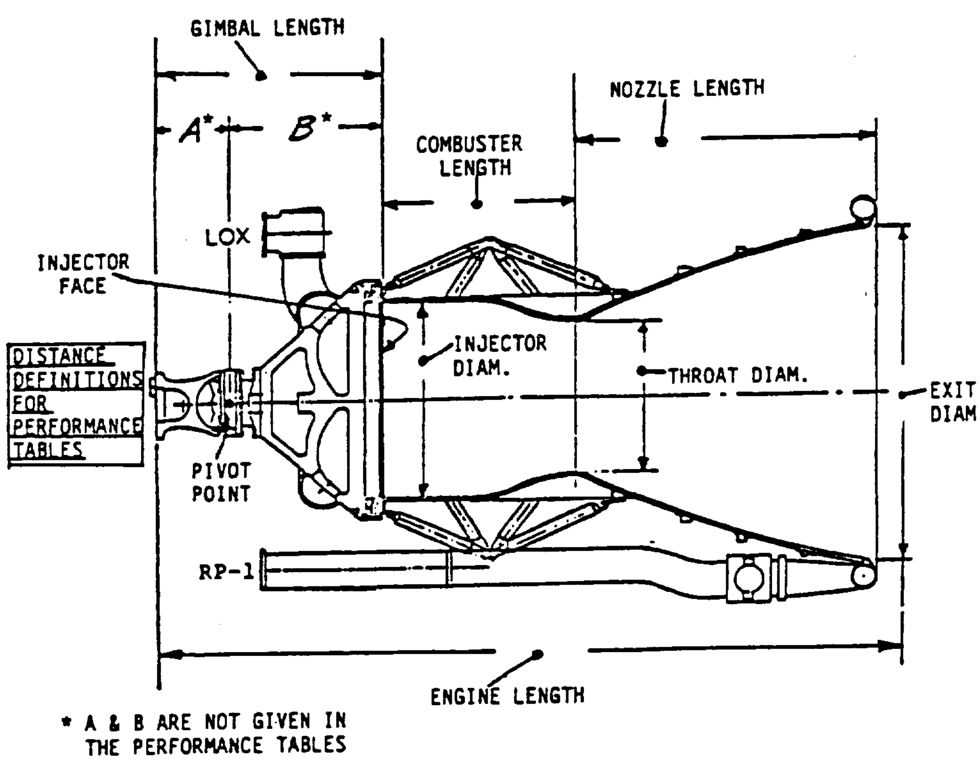


Figure 2-7. Definition of Computer Printout Terms

Case No. 402

Vaccum Thrust (Klb)	= 934.970	Sea Level Thrust (Klb)	= 800.000
Vacuum Isp (sec)	= 279.021	Sea Level Isp (sec)	= 238.742
Chamber Pressure (psia)	= 330.000	T/C Mixture Ratio (O/F)	= 2.500
Nozzle Area Ratio (Ae/At)	= 5.300	Nozzle Percent Length (%)	= 80.000
ODE Isp (sec)	= 305.304	ODE C-star (ft/sec)	=5841.077
Energy Release Eff. (%)	= 94.000	Kinetic Eff. (%)	= 99.843
Divergence Eff. (%)	= 98.161	Boundary Layer Eff. (%)	= 99.265
Throat Area (in**2)	=1732.857	Engine Exit Diameter (in)	= 108.137
Nozzle Length (in)	= 91.389	Engine Length (in)	= 194.389
Contraction Ratio	= 1.676	L-Star (in)	= 77.666
Vacuum Cf	= 1.635	Sea Level Cf	= 1.399
Throat Diameter (in)	= 46.972	Injector Diameter (in)	= 60.813
Gimbal length (in)	= 54.732	Combuster Length (in)	= 48.345

Figure 2-8. Baseline Design Point for Pressure Fed Engine

Case No. 402A

Vaccum Thrust (Klb)	= 614.930	Sea Level Thrust (Klb)	= 480.000
Vacuum Isp (sec)	= 277.559	Sea Level Isp (sec)	= 216.656
Chamber Pressure (psia)	= 217.200	T/C Mixture Ratio (O/F)	= 2.500
Nozzle Area Ratio (Ae/At)	= 5.300	Nozzle Percent Length (%)	= 80.000
ODE Isp (sec)	= 304.154	ODE C-star (ft/sec)	=5813.009
Energy Release Eff. (%)	= 94.000	Kinetic Eff. (%)	= 99.761
Divergence Eff. (%)	= 98.161	Boundary Layer Eff. (%)	= 99.205
Throat Area (in**2)	=1732.342	Engine Exit Diameter (in)	= 108.121
Nozzle Length (in)	= 91.295	Engine Length (in)	= 193.768
Contraction Ratio	= 1.666	L-Star (in)	= 76.552
Vacuum Cf	= 1.634	Sea Level Cf	= 1.276
Throat Diameter (in)	= 46.965	Injector Diameter (in)	= 60.627
Gimbal length (in)	= 54.564	Combuster Length (in)	= 47.92[illegible]

Figure 2-9. Baseline Design Point Throttled to 60% of Nominal Thrust

Table 2-4. Modulation by Throttle Setting Changes in a Pressure Fed Liquid Propellant Booster

Chamb. Press	Percent Nominal Pc	Vacuum Thrust	Percent Nominal Thrust	Fuel Inlet Press	Oxid. Inlet Press	Fuel Injector Del P as Percent Pc	Fuel Valve Opening Percent	Fuel, Valve Press Drop	Cooling Jacket Press Drop	Fuel Lines Press Drop	Fuel Injector Press Drop	Fuel Flow Rate lb/sec	Oxidizer Flow Rate lb/sec
a. Throttling by main propellant valve throttling.													
330	106	985,046	100	471.6	433.6	20.2	100	20.0	40.0	15.0	66.6	1015.4	2538.5
297	90	885,199	89.9	471.6	433.6	18.2	46.14	76.1	32.4	12.1	53.9	913.9	2284.7
264	80	785,651	79.8	471.6	433.6	31.4	31.4	129.8	25.6	9.6	42.6	812.3	2030.0
231	70	686,401	69.7	471.6	433.6	14.1	23.3	181.0	19.6	7.4	32.6	710.8	1777.0
216.3	65.55	642,345	65.2	471.6	433.6	13.2	20.6	203.0	17.2	6.4	28.6	665.6	1664.1
198	60	587,449	59.6	471.6	433.6	12.1	17.7	229.8	14.4	5.4	24.0	609.2	1523.1
b. Throttling by decreasing propellant inlet pressure.													
313.5	95	935,086	94.9	471.6	433.6	19.2	61.1	48.4	36.1	13.5	60.1	964.6	2411.6
297	90	885,199	89.9	438.9	404.8	18.2	61.1	43.4	32.4	12.1	53.9	913.9	2284.7
264	80	785,651	79.8	376.1	349.2	16.1	61.1	34.3	25.6	9.6	42.6	812.3	2030.0
231	70	686,401	69.7	316.8	296.2	14.1	61.1	26.3	19.6	7.4	32.6	710.8	1777.0
216.3	65.55	642,330	65.2	291.6	273.5	13.2	61.1	23.0	17.2	6.4	28.6	665.6	1664.1
198	60	587,449	59.6	261.1	245.9	12.1	61.1	19.3	14.4	5.4	24.0	609.2	1523.1
psia	%	lb force	%	psia	psia	%	%	psia	psia	psia	psia	lb/sec	lb/sec

obtain a programmed reduction in thrust during a typical vehicle boost. This would allow a reduced quantity of pressurant gas to be required. The setting at a constant 95 percent of full open will permit continued control of mixture ratio and/or a vernier adjustment in thrust by small but relatively rapid movement of both throttle valve positions around the nominal 95 percent point.

The design point, shown in Figure 2-8, consists of an engine having a sea level thrust of 800,000 pounds and a chamber pressure of 330 psia. The characteristics of this same engine when throttled to 60% of nominal thrust is shown in Figure 2-9. The pressure budget values and their relative changes under throttled conditions and under propellant tank blowdown conditions are shown in Table 2-4. Referring to the table, the engine chamber pressure is first decreased in steps by closing down on both the oxidizer and fuel throttle valves while holding the engine inlet pressure constant (top half of Table 2-4). Secondly, both throttle valves were set so as to give 95 percent of nominal chamber pressure, and this setting was held constant while the engine inlet pressure was decreased in steps. This condition explores the possibility of allowing the propellant tank pressures to decrease in order to obtain a programmed reduction in thrust during a typical vehicle boost. This would allow a reduced quantity of pressurant gas to be required. The setting at a constant 95 percent of full open will permit continued control of mixture ratio and/or a vernier adjustment in thrust by small but relatively rapid movement of both throttle valve positions around the nominal 95 percent point.

The above pressure budget is not specifically precise to the engines under discussion, but is included to give an indication of typical values to be expected.

2.2.4 Weight Breakdown

An estimate of the engine weight as calculated by estimating the weight of each of the components has been made. The weight breakdown is shown in Table 2-5.

Table 2-5. LRB Pressure-Fed Baseline Engine Weight

COMPONENT	WEIGHT IN LB	TOTALS
INJECTOR AND THRUST CHAMBER	3,690	
OUTRIGGERS	87	
GIMBAL	125	
VALVES	335	
ENGINE	4,237	4,237
ACCESSORIES:		
INLET DUCTS	1,160	
ACTUATORS (HYDRAULIC)	300	
	1,460	1,460
ENGINE SYSTEM TOTAL		5,797

2.2.5 Gimbal System Summary Use and Power

To determine the torque and horse power required to drive the engine through an arc about the gimbal pivot point, a study was made of the various contributory factors. The requirements are:

maximum gimbal excursion $\pm$ 6 degrees
maximum angular velocity 10 degrees/sec
maximum angular acceleration 1 radian/sec^2

The gimbal system weight has been shown in Table 2-5 above. (Note that the following sections on gimbal torque and power have been calculated for a thrust of 750,000 lbs and will require updating to reflect the increased thrust to 800,000 lbs.)

Acceleration Force. The engine moment of inertia about the gimbal pivot joint was estimated by dividing the engine into lumped masses at various distances from the pivot point and adding their moments of inertia. The maximum torque and power were calculated with results shown in Table 2-6.

Table 2-6. LRB TVC Torque Breakdown for Head End Gimbal

Name of Contribution	In-lb of Torque	Percent of Total
Moment of Inertia	66,203 in.lb	8 %
Flex Line Stiffness		
LOX LINE	86,616 in.lb	11 %
FUEL LINE	58,512 in.lb	7 %
Thrust Vector Offset	200,000 in.lb	25 %
Gimbal Friction	264,000 in.lb	33 %
Gravity and Accel. at 3 g	116,144 in.lb	15 %
Total =	791,475 in.lb	100 %
Lever Arm =	70.73 in	
Force Reqd.=	11190.1 lb	

Horse Power at 10 Deg/sec = 20.82 H.P.(input)

Basis:

Engine Thrust = 800000 lb
Engine Mass = 5600 lbm (wet)
Lever Arm = 70.73 in
CG Distance = 62.62 in
Frictn.Coef.= 0.06
Thrust Offset 0.25 in

Requirements:

Angular Excursion = + or - 6 Deg
Angular Slewing Rate = 10 Deg/sec
Angular Acceleration = 1 radian/sec squared
Propellant Line Pres.= 600 psia
Nomin.Fuel Line Diam.= 10 in
Nomin.Oxid.Line Diam.= 13 in

Stiffness of Flexible Lines Using Metal Bellows. A computer program was utilized based on the successful experience with the flexible lines used in past Rocketdyne engines, namely the F-1, the MA-5, the SSME, and others. The present configuration consists of two wrap-around propellant lines each containing three flexible bellows joints. The lines are 10 in. and 13 in. inside diameter for the RP-1 and LOX lines respectively (for an engine of 750,000 lb thrust), while the bellows for these lines are 11 in. and 14 in. inside diameter respectively. The yield stress is based on SSME practice, and is 84,000 psi. The wall thickness for the RP-1 joints is 0.035 in. and for the LOX joints is 0.050 in. The program takes into account internal pressures, angulation, wall thickness, height of the bellows, diameters, system geometry, etc. The results are shown in Table 2-6.

Thrust Vector Offset. The maximum expected offset after adjusting the pivot location subsequent to engine testing is determined to be 0.25 in. (R_0). Development testing may allow a downward adjustment of this value. Note that this torque is one of two major contributors to the total.

Gimbal Friction. The gimbal friction torque shown in Table 6 is based on a coefficient of friction of 0.06. New dry lubricants show promise of reducing this to 0.02. This is the other major contributor to the total torque.

Gravity and Acceleration. Gravity and acceleration both act on the center of gravity of the engine. If the vector sum of these accelerations does not pass through the pivot point and through the center of gravity (C.G.), they will exert a torque on the engine. The value in the table is only approximate since the value depends upon the orientation of the vehicle with respect to the earth and to the acceleration vector of the vehicle.

2.2.6 Engine Operation

Engine Startup and Shutdown. The steps required to start the engine and to perform a shutdown are tabulated and shown in a timeline in Figure 2-10. The steps performed will be initiated by an automatic sequencer and provide complete remote control capability and implementation. The left hand column in Figure 2-10 gives the action taken, while the next column provides a brief explanation for the action. The right hand side shows the estimated time and duration for each of the actions. The times shown are only engineering estimates. Actual times will be obtained during development testing.

2.2.7 Interface Requirements

Utility Requirements. The interface requirements are divided into two, 1) with the vehicle on the ground and utilities furnished by the ground facilities, and 2) during flight, with the vehicle providing the utility requirements.

Figure 2-10. Engine Startup and Shutdown

PRE-PROPELLANT LOADING CHECKOUT ASSUMED COMPLETED

PROPELLANT LOADING

EXPLANATION OF ACTIONS TAKEN

1. Fuel side antifreeze solution fill valve opened to load antifreeze and closed again and disconnected at the quick disconnect.
   Await completion of task signal and enter into logic tree.
2. Close Pre-valves
   Are all prevalves closed?
3. Open oxidizer anti-guysering helium flow valve located above engine interface.
   Check for valve open condition.
4. Fill propellant tanks and Pressurant tank.
   Await propellant tanks full condition signal.

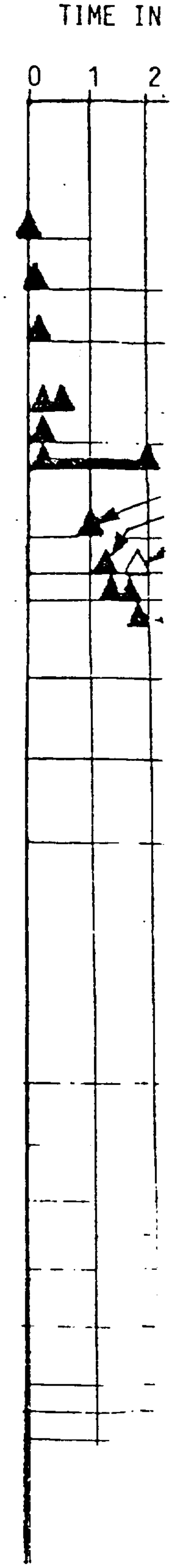
PREPARATION STAGE

1. Carry out instrumentation and control systems checkout and valve position indication checkout.
   a. Confirm availability of on board electrical electrical power.
   b. Confirm igniter circuit checkout.
   c. Exercise valve positions and throttle valve actuation.
   d. Etc.

   Enter confirmation that each component is functional re. a simple pre-launch test and enter data into logic tree of a failure is detected, send warning signal to central control system.
2. Send engine ready signal to central control.

PRE IGNITION ENGINE START SEQUENCE STAGE

1. Receive engine pre-ignition start signal from central control.
2. Activate electronic pre-ignition electronic start sequencer.
3. Open LOX manifold GN2 purge valve.
   This is to prevent moisture from condensing from the outside air.
4. Open LOX prevalve and close anti-geysering valve above it.
5. Open anti-guysering valve just ahead of main LOX valve
   (A very slow flow of gas introduced at the lowest point will prevent geyser.)
6. Open Fuel pre-valve and move hypergol ignition system 3-way bleed valves to the bleed position
7. Open fuel bleed valves for TBD seconds (Valves are part of hypergol ignition system).
   This is vent any trapped gas just ahead of the hypergol slug.
8. Send engine ready signal to central control system indicating that engine is in PRE-FIRE condition and awaits the countdown
9. At a TBD point in the countdown, switch electric power and purge gas supply to the on-board flight configuration systems from the facility system, and send signal to central control confirming this action completed.
   This should be just a few seconds before the commence ignition signal is anticipated to be received.



FOLDOUT FRAME

Figure 2-10. Engine Startup and Shutdown (concluded)

IGNITION STAGE

1. Receive commence-ignition signal from central control.
2. Activate ignition start sequencer and ignition start logic checkout sequencer.
3. Activate engine health monitoring system.
4. Open main LOX valve at programmed rate.
5. Close LOX manifold helium purge valve and close anti-guyser valves.
6. Fire pyro igniters at nozzle exit (Pyro duration is TBD seconds)
7. Activate dual hypergol ignition system.
8. Open main fuel valve at programmed rate.
9. Switch electronic health monitoring system from ignition redline values to engine mainstage values.
10. Switch from ignition sequencer control to main stage logic and contingency sequencer control (if they are different).
11. Activate POGO control system.

Precise opening rate TBD fro development tests. The slow rate is to prevent "water hammer".

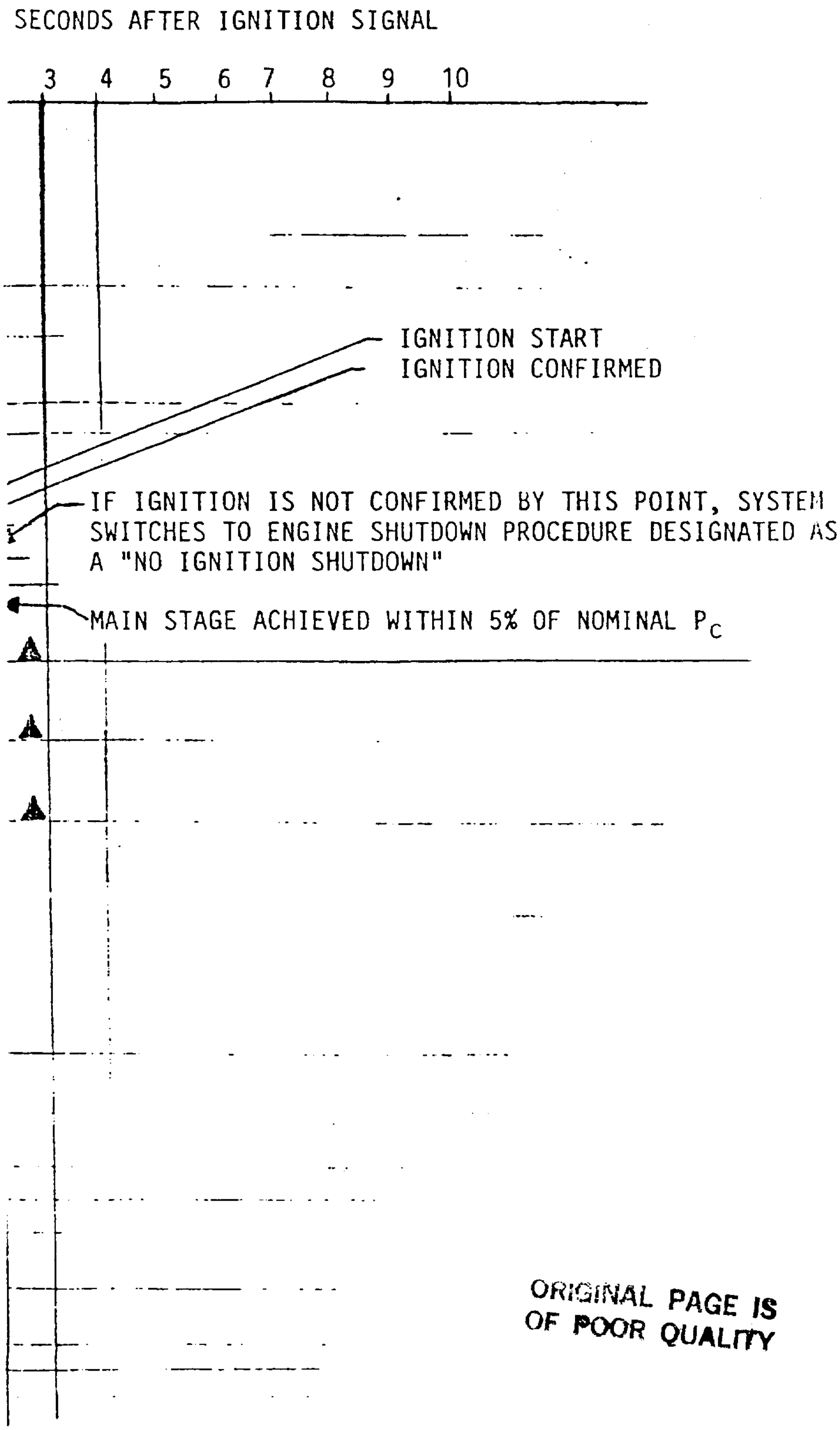
Dual hypergol is to provide backup for ignition system. Fuel valve is opened on sig. from fuel manifold pressure "up" signal.

SHUT DOWN

1. Distinguish between end of burn shutdown emergency shutdown, etc.
2. End of boost time shutdown (normal programmed shutdown) TBD
3. Emergency shutdown (Engine-out shutdown)
   a. Activate shutdown electronic sequencer
   b. Change red-line settings to emergency shut down mode
   c. Ramp LOX valve closed
   d. Ramp fuel valve closed
   e. Deactivate POGO system
   f. Close both prevalves
   g. After TBD seconds (to allow engine cooling) crack main LOX valve open about 1% to allow escape of locked-in oxygen between main LOX valve and its pre-valve
   h. Reset health monitoring red-line values to long term emergency shut down mode
4. Other types of emergency shutdowns---TBD

Estimated Start Transient Propellant Usage

LOX	1500 lb
RP-1	900 lb



FOLDOUT FRAME

A preliminary interface sketch in Figure 2-11 shows the dimensions from a reference point to the centers of the outriggers and propellant lines.

Multi-Engine Ducting. The layout of the bottom of the LRB stage shown in Figure 2-12 is very preliminary and represents a first attempt to combine the engines with propellant manifolds and pre-valves above them. The skirt diameter shown is considered excessively large. Placing the outrigger inboard will allow some reduction in skirt diameter. (Another method of reducing the skirt diameter is considered to be new technology and is covered in that section.)

2.3 POGO & STABILITY ANALYSIS

2.3.1 Introduction & Summary

A critical aspect is that the pressure fed engines not result in vehicle POGO. A study of such an engine was considered in the early 1970's (Rocketdyne report R-8934) and had recommended use of an active POGO suppressor. In subsequent studies of this type of suppressor for application on the SSME, problems were encountered near the upper frequency range for the control. A re-evaluation of the potential POGO problem was conducted and is discussed in this section.

Simulation of the feedline/engine dynamics indicate that the pressurized engine will have a very destabilizing effect on the vehicle, primarily due to the high ratio of thrust/chamber pressure.

Based on a 60 ft LOX feedline, the first resonance will be in the 10 Hz range and have high damping. Net destabilizing thrust feedback will be greater than 8000 lb/G. Estimating a vehicle mode with modal mass of 10^5 lb and 1/2% modal damping the open loop gain was 8.0 with zero loop phase. This would be a highly unstable situation since the maximum gain for a stable system is 1.0.

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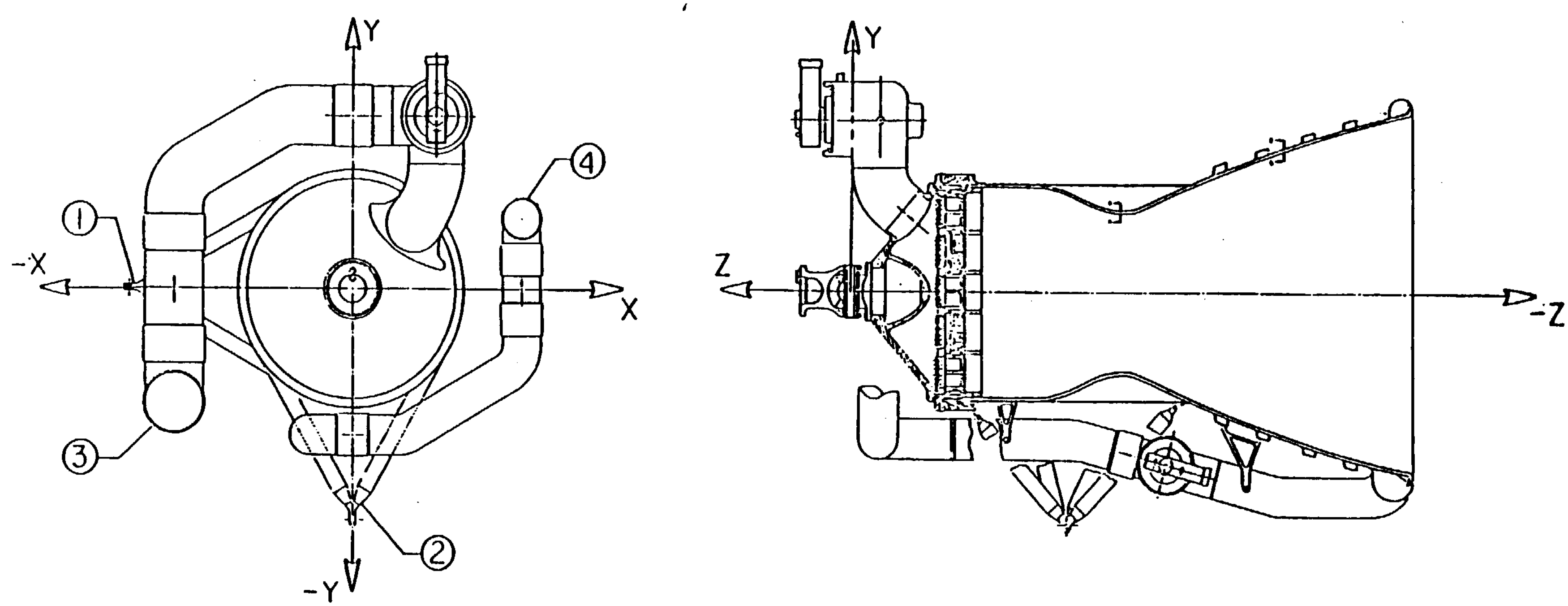

FIND NO.	DESCRIPTION	DIMENSIONS IN INCHES X	Y	Z
1	OUTRIGGER	-57.6	0	-56
2	OUTRIGGER	0	-57.6	-56
3	OXIDIXER INLET	-46	-28	-8
4	FUEL INLET	45	17.6	-8

Figure 2-11. Interface Drawing

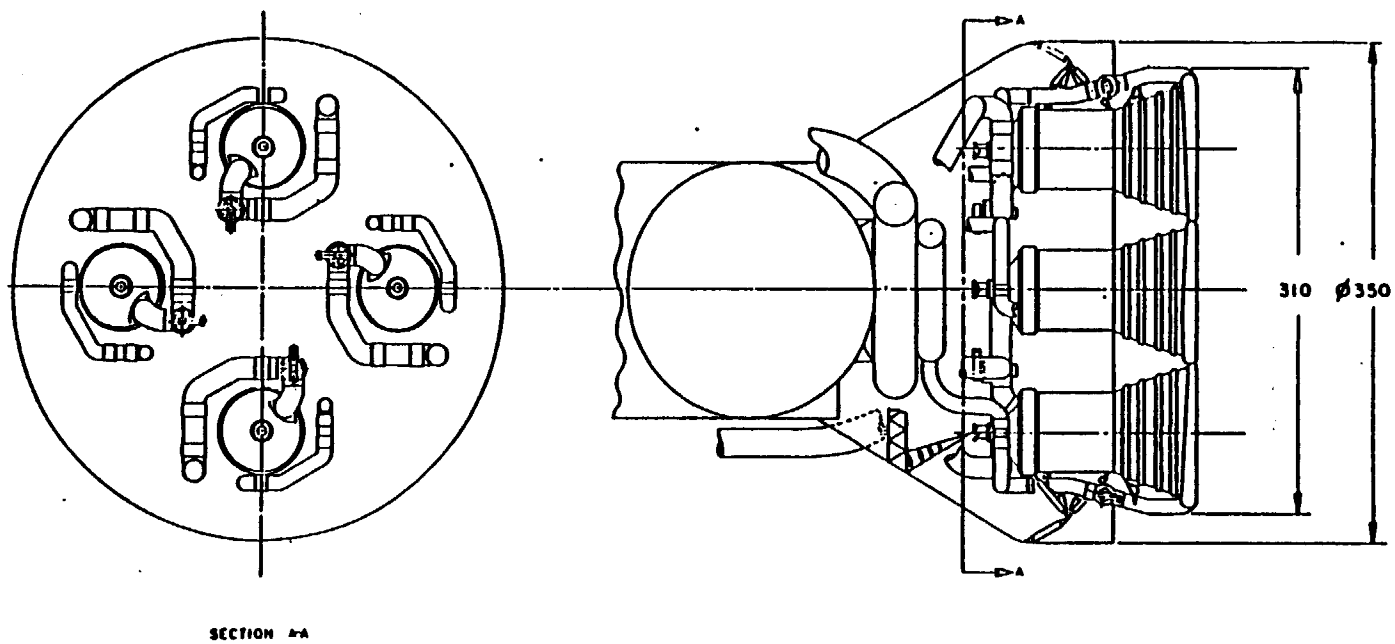

Figure 2-12. Four LRB engines placed at the bottom of the LRB stage showing an example propellant manifolding and skirt arrangement. (Approx. 1/100 scale)

The feedline damping suggests that instability could occur even with considerable frequency mismatch between the feedline and structure. Feedline gain is even larger at lower frequencies so that the engine would decrease stability of the major Orbiter-ET mode in the 4 Hz range, unless the booster is attached near a nodal point of that mode.

A passive suppressor would have an unreasonable size, greater than eleven cubic ft of gas, to move the feedline frequency down to 2 Hz. If additional impedance were added to the feedline above the suppressor connection, a smaller amount of gas would be required. Such impedence would also be required if an active suppression device is considered.

While an active suppressor design might be possible, it is rather high risk. Such a design was tested for the SSME providing nearly 6 db of attenuation in the active band but also providing instability at higher frequencies due to phase roll-off of the servo. A narrow band active suppressor might be possible and some concepts are available which could provide gain roll-off with minimum phase penalty and alleviate the instability situation.

2.3.2 The POGO Phenomenon

POGO is to a space vehicle what flutter is to an airplane; a potentially destructive unstable vibration, which can be a program show stopper. It usually shows up as a low frequency structural vibration occurring during the boost phase, gradually growing out of the background noise, leveling off and decaying back into the noise. An example from the second unmanned Saturn flight is shown in Figure 2-13. It may occur at several times in flight and at different frequencies. During the oscillation growth period, the vehicle is unstable and the maximum amplitude cannot be predicted in advance of flight.

The problem can be very serious on manned vehicles because vibration, which would not cause structural failure on the vehicle, can cause severe pain and severely impare capabilities of the astronauts (Figure 2-14). Based on the Titan-Gemini and Saturn-Apollo programs, NASA decreed that appropriate work would be done to assure that the Space Shuttle is free of POGO. The SSME included a suppressor between the low and high pressure LOX pumps, which

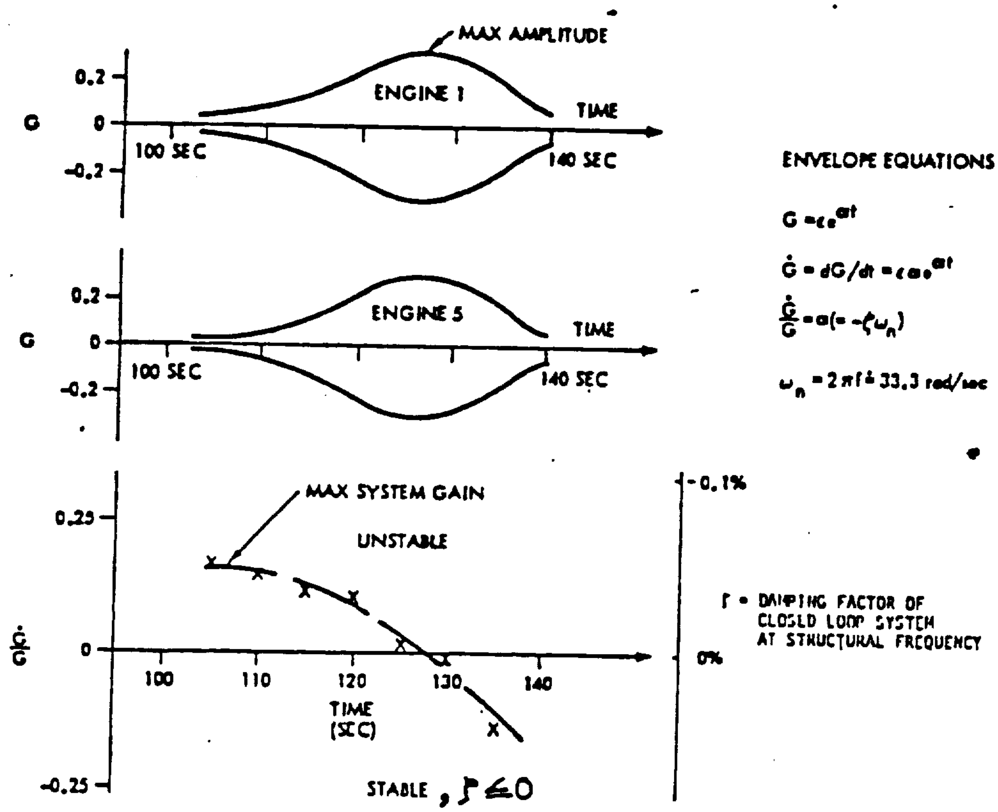

Figure 2-13. Engine Gimbal Black Accelerometer Outputs from Apollo-Saturn 502-S-IC Stage, Showing Envelope of Oscillation and Method of Determining Damping Factor of Closed Loop System at the Structural Mode Frequency (W_n = natural frequency in radius/sec)

absorbs flow fluctuations in the 5-40 Hz range. In addition, the suppressor forces the lowest frequency feedline mode to below the first critical structural frequency. In effect, the entire frequency range below 40 Hz is protected by the suppressor.

The problem involves the vehicle structure, the column of propellant in the feedline and the engine. The structure supports the engine and the engine supports the propellant column. As the engine moves forward, pressure at the engine inlet increases, producing a force acting upward on the propellant and downward component on the engine and structure. This increased pressure causes additional flow into the engine, which is burned in the main thrust chamber, producing an additional upward force component on the structure. If the upward force from the engine is greater than the downward force at the engine inlet, the engine acts like negative structural damping with potential

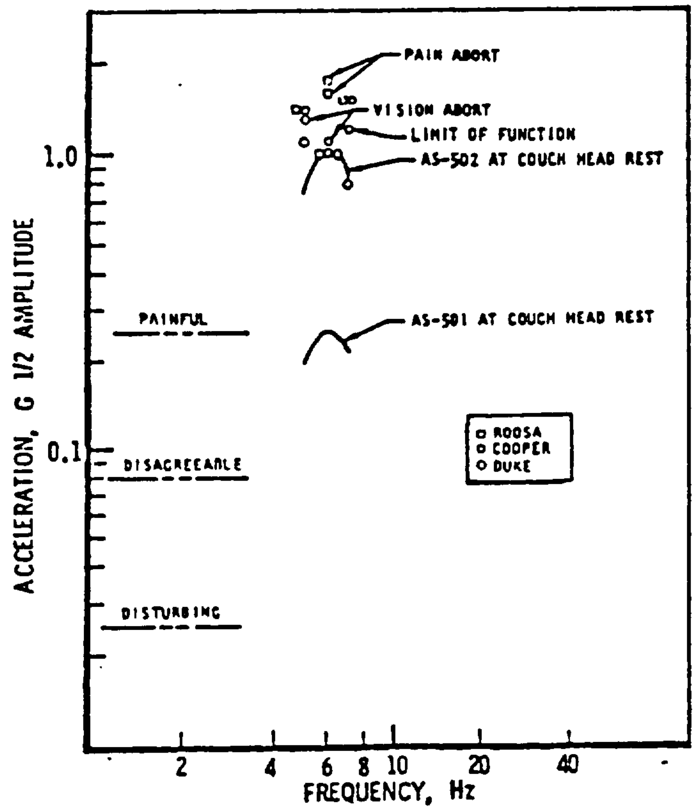

Figure 2-14. Couch Vibration and AS-503 Crew Acceleration Tolerance

for POGO. The instability usually requires tuning of a feed system resonance with a structural resonance. Tuning and detuning occurs naturally during a flight as the propellant in the tank is consumed. Figure 2-15 is a block diagram showing coupling of the significant subsystems. With the structure and feed systems tuned to the same frequencies, the forward loop has maximum gain and zero phase shift. With zero feedback the two resonances are uncoupled. With small negative feedback, damping of the closed loop root associated with the structure is increased resulting in greater stability. With positive feedback, damping is decreased with potential for instability.

The engine has two effects on the POGO loop. It is the lower boundary of the propellant column helping to set the feedline frequencies and damping. In addition, it defines the ratio of upward thrust component to engine inlet

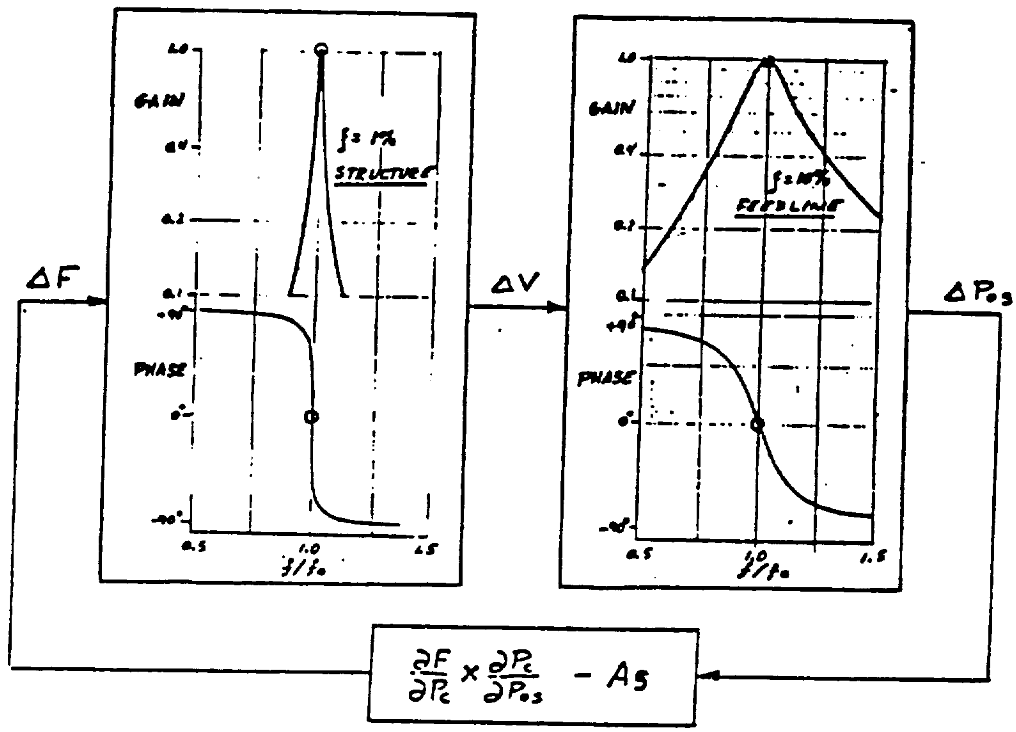

Figure 2-15. POGO Block Diagram

pressure, the so-called engine gain, which is the key to the POGO instability. From the previous discussion, critical engine gain (causing decreased stability of the simple model) occurs when:

$$\frac{\partial F}{\partial P_c} \times \frac{\partial P_c}{\partial P_o} > A_s$$

where:

F = Thrust

P_c = Chamber pressure

P_o = Engine inlet pressure

A_s = Engine inlet area

Values of allowable engine gain for several large rocket engines are shown in Figure 2-16.

VEHICLE	PROPELLANT	$A_s/\frac{\partial F}{\partial P_c}$
THOR	LIQUID OXYGEN	0.126
	RP-1	0.126
TITAN II	NITROGEN TETROXIDE	0.126
	HYDRAZINE/UDMH	0.087
S-IC	LIQUID OXYGEN	0.312
	RP-1	0.241
S-II	LIQUID OXYGEN	0.245
	LIQUID HYDROGEN	0.245
SPACE SHUTTLE	LIQUID OXYGEN	0.722
	LIQUID HYDROGEN	0.722

Figure 2-16. Comparison of $A_s/\frac{\partial F}{\partial P_c}$ For Several Engines Involved with POGO

There is limited experience with POGO on pressurized rocket engines. Normally, this type engine is not used as a major booster engine due to the high tank weight. When used as a minor thruster, the vehicle mass is so great that the closed loop does not have sufficient gain to produce an instability. One exception might be the Lance weapon system, although the frequency was high enough so that it was also considered as a vibration sensitive chug problem.

One of the important features of pressurized systems is that there is no NPSH sensitive pump compliance and pump cavitation gain to consider. Inlet system dynamics are, therefore, much easier to predict. The problem of potential tuning with the widely varying structural resonance, however, remains.

Pressure Fed Engine Gain. The range of critical Space Shuttle structural frequencies is from 2 Hz to about 30 Hz. In the most critical 2 Hz range, the engine itself can be well described by steady state gain values.

Using the fuel and oxidizer engine inlet pressures and flows, the steady state gains can be calculated from the following linear perturbation equations:

1) $P_F - P_c = (2\overline{\Delta P}_F/\bar{\dot{w}}_F)\ \dot{w}_F$

2) $P_o - P_c = (2\overline{\Delta P}_o/\bar{\dot{w}}_o)\ \dot{w}_o$

3) $P_c = \left(\dfrac{\bar{P}_c}{\bar{\dot{w}}_o + \bar{\dot{w}}_F}\right)(\dot{w}_o + \dot{w}_F) + (\bar{P}_c/C^*)\ (\partial C^*/\partial MR)\ \left(\dfrac{1}{\bar{\dot{w}}_F}\right)(\dot{w}_o - \dot{w}_F)$

With operation near the peak of the C* vs MR curve the second factor in equation 3 is small and may be neglected with little error.

For a representative LOX/RP-1 engine, the values are as follows:

$\bar{\dot{w}}_o$ = 3244 lb/sec, oxidizer weight flow rate

$\bar{\dot{w}}_F$ = 1247 lb/sec, fuel weight flow rate

$\bar{P}_c$ = 500 psi , chamber pressure

$\bar{P}_o$ = 750 psi , engine inlet pressure, oxidizer

$\bar{P}_F$ = 750 psi , engine inlet pressure, fuel

$\bar{F}$ = 10^6 lb , thrust

$\overline{M.R.}$ = 2.60 , mixture ratio

This results in the following gains:

$P_F = 0.624\ \dot{w}_F + 0.223\ \dot{w}_o$

$P_o = 0.377\ \dot{w}_o + 0.223\ \dot{w}_F$

$P_c = 0.223\ (\dot{w}_o + \dot{w}_F)$

OR

$$\dot{W}_O = 3.364\ P_O - 1.202\ P_F$$

$$\dot{W}_F = 2.032\ P_F - 1.202\ P_O$$

$$P_c = 0.482\ P_O + 0.185\ P_F$$

Assuming feedline diameters of 15 inches for the LOX and 12 inches for the fuel, the net upward force for a unit inlet pressure variation is:

$$\Delta F/\Delta P_{os}\Big|_{PFS} = \frac{\partial F}{\partial P_c} \times \frac{\partial P_c}{\partial P_{os}} - A_{os} = 964-177 = 787$$ (PFS = Pressure, Fuel, Suction, held constant)

$$\Delta F/\Delta P_{Fs}\Big|_{POS} = \frac{\partial F}{\partial P_c} \times \frac{\partial P_c}{\partial P_{Fs}} - A_{Fs} = 370-113 = 257$$ (POS = Pressure, Oxidizer, Suction, held constant)

In other words, the stabilizing downward force on the engine is only 20-30% of the destabilizing upward thrust force.

On the SSME, a gas type suppressor is used on the engine in the oxidizer system between the LPOTP and HPOTP. Flow fluctuations generated upstream of the engine by vehicle vibration act on the resistance of the LPOTP and provide a stabilizing force. Most of the flow variations are absorbed by the gaseous suppressor minimizing HPOTP inlet pressure variations. The reduced HPOTP inlet pressure variation results in a very small amount of flow being forced into the thrust chamber and a very small amount of destabilizing thrust variation.

Another significant feature of the SSME is the ratio of thrust to chamber pressure. One psi chamber pressure produces only about 160 lb of thrust, while for the pressurized booster, this value is about 2000 lb/psi. Without a suppressor, 1 psi at the engine inlet produces about 1/2 psi in chamber pressure. With a 12" inlet duct, we obtain a downward force of 113 lb and an upward thrust force of only 80 lb. With the pressurefed engine and a 15" LOX inlet duct, 1 psi at the inlet produces 177 lb downward and 964 lb of destabilizing upward force.

Inlet Line Dynamics. The feedline geometry and propellant selection affect the resonant frequency and gain and, therefore, are critical to POGO as indicated in Figure 2-15. RP-1 has a density of 50 lb/ft^3 and an acoustic velocity of 4000 ft/sec while LOX density is about 70 lb/ft^3 and its acoustic velocity is 3000 ft/sec. Typically, the heavier propellant tank is forward to minimize vehicle C.G. motion during boost so that the LOX feed system will have the greatest acceleration head and the lowest resonant frequencies. As previously indicated, the gain of the LOX system (of the engine) is about 3 times the gain of the fuel system, so that from all standpoints, the LOX system is most critical. In this discussion, only the coupling due to the LOX system will be considered, although any actual analysis would consider both propellants. It will be assumed that the feedline is 60 ft long while the propellant height in the tank is a maximum of 80 ft.

Both the tank bottom and engine inlet motion generate flow disturbances, which result in engine inlet pressure variations. Depending on vehicle mode shapes, the tank bottom and engine inlet motion may be in or out of phase. At the lowest (2 Hz) resonance, it can be assumed that the tank and engine are in phase, although they probably have a different amplitude. In this assessment, the tank bottom and engine accelerations were chosen as independent, in order to evaluate their sensitivity. The combined effect could be determined by vector addition of the two components if structural mode shapes were known.

It was assumed that a friction pressure drop of 25 psi exists in the feedline under steady flow and that the entire line from the tank to the oxidizer injector can be described by a 15" diameter line, 60 ft long.

Figures 2-17 and 2-18 show the oxidizer injection pressure response to acceleration of the engine (PSI/G). At the lowest frequencies, the gain is approximately equal to the gravity head. The critical condition is when injection pressure is in phase with engine velocity so that injection pressure logs engine acceleration by 90°. This corresponds to about 10 Hz and a gain

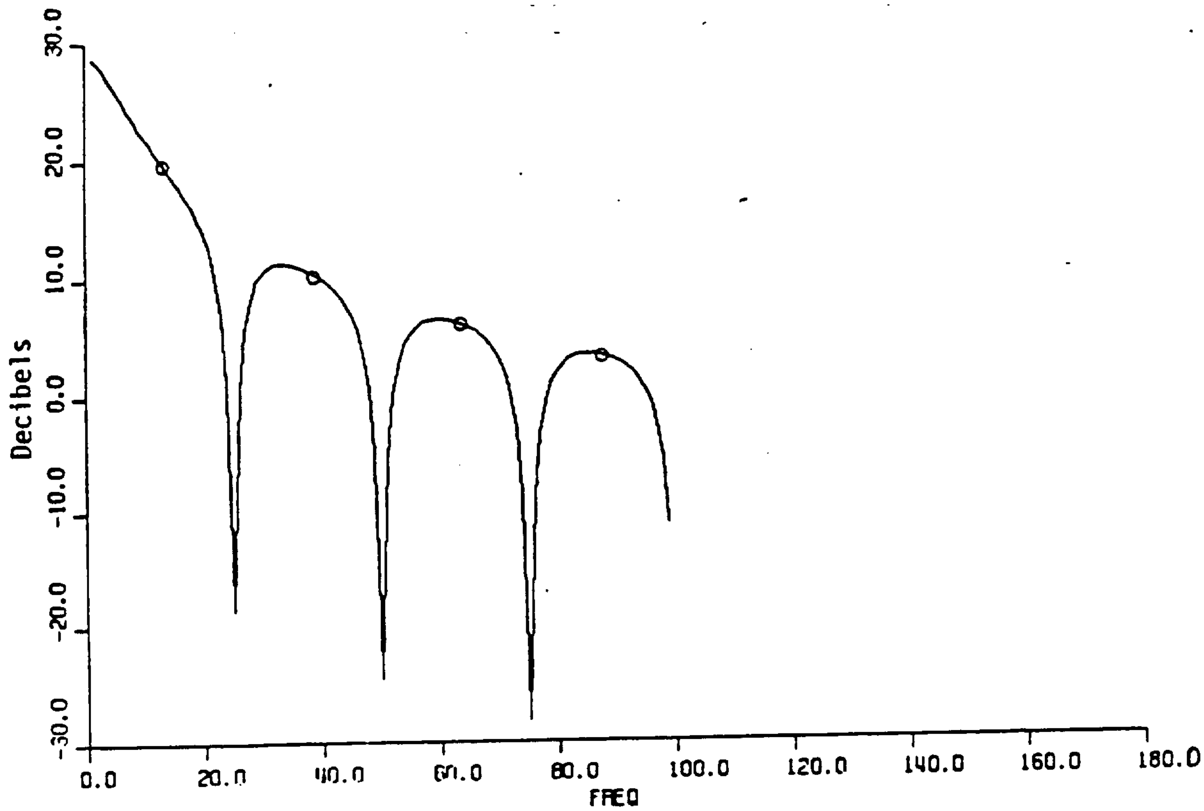

Figure 2-17. Oxidizer Injection Pressure <u>Gain</u> Response to Engine Acceleration

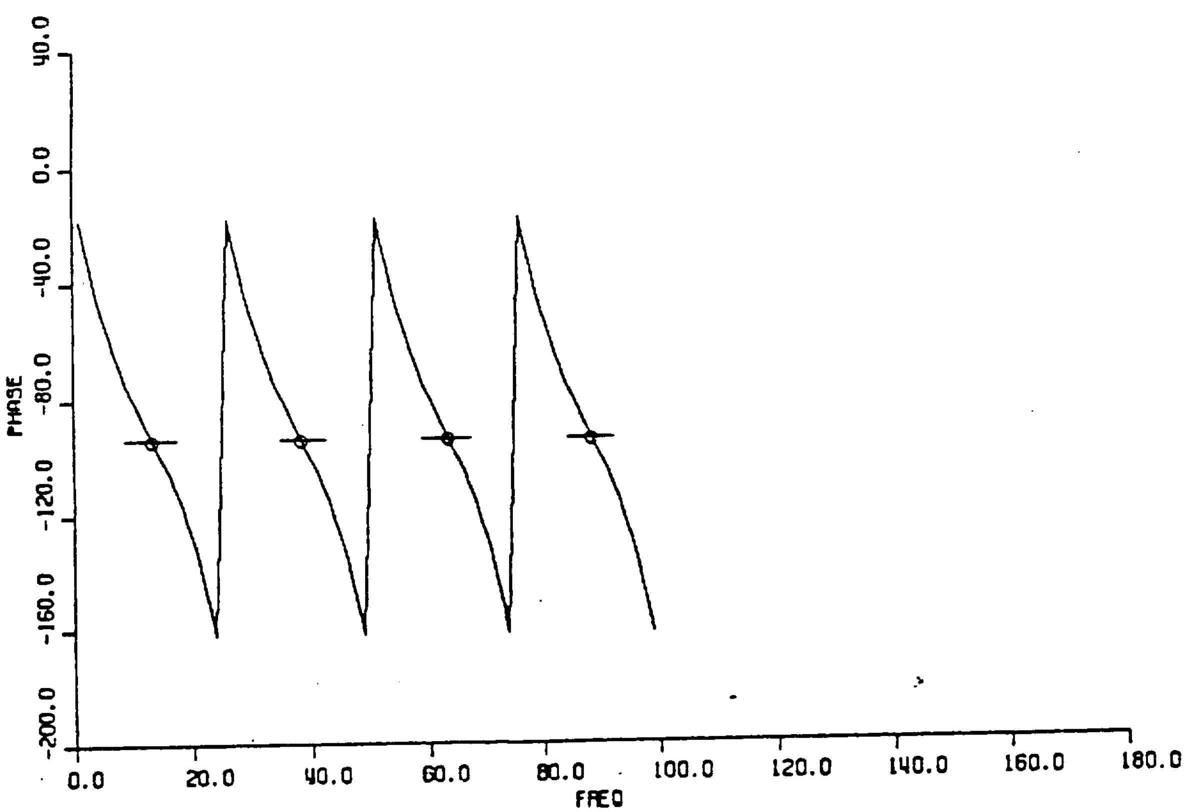

Figure 2-18. Oxidizer Injection Pressure <u>Phase</u> Response to Engine Acceleration

of 20 db (10 psi/G), although several other critical frequencies are also shown. This case also represents the dynamics near end of burn when the tank is empty but the line is full. The anti-resonances (at 25, 50, 75 Hz) result in very low gain at these frequencies and are stable regardless of the critical phase.

Figures 2-19 and 2-20 show the inlet pressure response to tank bottom acceleration for an effective tank height of 10 ft (i.e. 5 psi/G). Acoustics were not used to model the fluid in the tank, only the incompressible gravity head, since much of tank dynamics will be associated with bulkhead motion and tank structure, rather than simply acoustics. Tank motion can result in significant tank bottom pressure, however, because of light damping in the structure. At the first feedline resonance where injection pressure logs acceleration by 90°, the gain is 2.8 psi/G (9.0 db).

Without a significant tank bottom resonance, the acceleration of the engine is the dominant effect, producing about 7870 lbs of net thrust/G in the direction of applied acceleration.

Now assume that the vehicle modal mass associated with a major structural resonance in the 10 Hz range is 10^5 lbs with 1/2% of critical damping. The response at resonance is about $1X10^{-3}$ G/lb. The open loop gain through the structure and back through the engine is 7.87 ignoring tank motion. Since the loop phase shift is zero and the loop gain is greater than 1.0, an instability would result.

At a frequency as low as 2-4 Hz, where they are currently dominant vehicle modes, the phase shift through the feed system is small but the component in phase with the velocity is still appreciable and could lead to an instability at that frequency.

Options. One potential option is the configuration with the LOX tank aft. The effect of the LOX system is decreased while that of the fuel system is increased. The longer fuel line required (~100 ft) would place its resonance (for RP-1) at about 10 Hz. While the injection pressure per G of engine acceleration would be similar to the LOX system, the engine gain ($\Delta P_C/\Delta P_{FI}$)

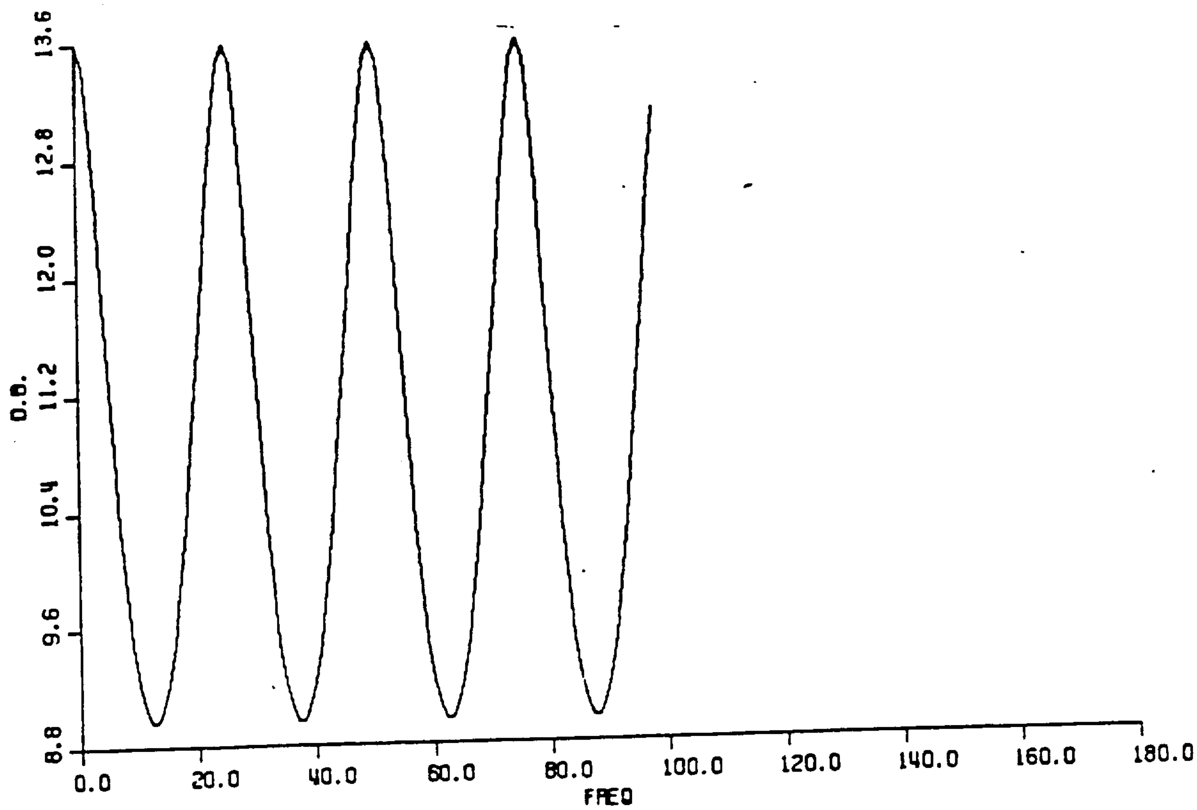

Figure 2-19. Inlet Pressure <u>Gain</u> Response to Tank Bottom Acceleration

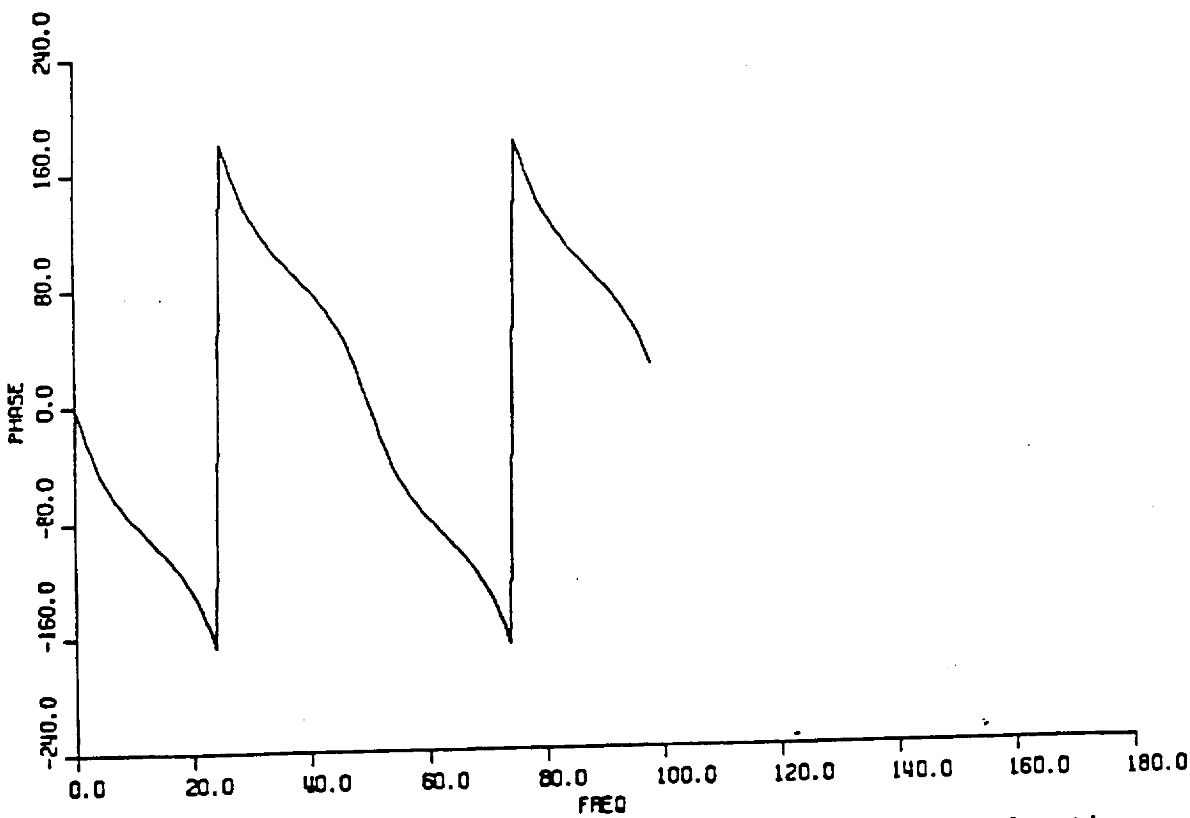

Figure 2-20. Inlet Pressure <u>Phase</u> Response to Tank Bottom Acceleration

is reduced by about a factor of 3. With the LOX tank aft that feedline resonant frequency is much higher but it is nearly assured that tank bottom motion is in phase with engine motion and of similar stroke. In this case, both the fuel and oxidizer play important parts with the likelihood being that either system resonance tuning with the structure would be sufficient for an instability.

Another option is that a passive POGO suppressor might be designed to avoid instabilities. The major problem with a passive suppressor is to obtain a reasonable compliance. Assuming a gas such as helium, the compliance, C (lb/psi), is approximately equal to 42 $V(ft^3)/P(psi)$. For the 60 ft feedline, the gas volume required to reduce the lowest resonant frequency to 2 Hz is about 11 cubic ft. If the feedline damping were smaller and gains were such that the minimum loop gain were only slightly greater than unity, it might be possible to control POGO by only minor detuning so that major feedline and structural resonance were never tuned. The large damping associated with the LOX line as indicated by both the gain and phase response, as well as the high open loop gain, indicate that this is not a feasible approach.

One more option is the design of an active POGO suppressor using a servo driven piston. To be effective, the piston must remove or supply flow to the main duct such that pressure variation at the LOX injector are reduced by an order of magnitude. To augment such a suppressor, a high impedence must be added in the main duct upstream of the suppressor connection. This might take the form of an orifice (~200 psid), a section of smaller diameter ducting (~5' of 8" ducting) or a windmilling inertia wheel built like a pump inducer. The upstream impedence allows the suppressor to have an effect on the local pressure without excessive stroke. A detailed review of impedence vs. suppressor flow requirements should be made.

Based on phase compensation difficulties encountered in attempting such a design for the SSME, the design of a broad band active POGO suppressor is considered to be high risk. It is possible that a narrow band suppressor could be designed but it would require good definition of the structural dynamics over the flight trajectory.

2.3.3 Recommendations

Since it is likely that a large pressurized booster with long feedlines would produce vehicle POGO, it is suggested that further studies of an active suppressor be made. For such a suppression concept to be feasible, additional feedline impedence must be added above the suppressor connection. While an orifice or necked down section of line could be used, a windmilling inducer designed to act as a fly wheel might result in less vehicle weight penalty. Without such impedence, the volumetric flow requirements may be very large.

Any active suppressor study must consider a realistic feedline geometry and an estimate of vehicle modal frequencies and gains. These are required to obtain reasonable estimates of suppressor stroke vs. frequency. Particular attention must be paid to servo system phase errors near the frequency limit of the servo valve. These errors resulted in system instability at moderate frequencies (~40-50 Hz) during concept testing of a similar system for the SSME.

2.4 RELIABILITY ANALYSIS

A preliminary Failure Mode and Effects Analysis (FMEA) is shown in Table 2-8, for a pressure fed LRB. A comprehensive reliability analysis of this engine has not been performed and a reliability history of pressure fed engines of similar size and requirements is not available.

However, based on a cursory item-by-item comparison with pump fed engines of similar requirements that have an established reliability record, the requirement of 0.99 R at 90% confidence appears reasonably attainable. This was done by summing up the known unreliabilities of major components and subassemblies that are not used in this configuration, such as valves, controls and ducting and taking into account the other differences between the engine types and applying them to the known reliability values of the base engine. The result is an indicated positive result on the estimated reliability of this engine at the prescribed confidence level.

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Main Fuel Valve	Fails to open	Prevents M/S operation start. Overboard dump of oxidizer creates a potential fire hazard.	Launch abort	1
	Opens too rapidly	Water hammer causing system vibration and engine damage. Engine shutdown.	Launch abort	2
	Fails to close	Fuel-rich cut-off will occur. After shutdown command, fuel flow will continue and dump overboard.	None	2
	Internal leakage	None	None	3
Main Oxidizer Valve	Fails to open	Prevents engine start	Launch Delay	2
	Fails to close	Delays engine-out shutdown. Prevents normal shutdown	Possible Mission Loss	2
	Internal leakage	Shaft seal leakage allows mixing of propellants within valves, creating a potential fire/explosion hazard	Possible vehicle loss	2

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Thrust Chamber	Fails to attain correct P_c	Engine system switches to engine-out mode of operation.	Single-engine out results in longer systems burn time	2
	External hot-gas leakage	Hot-gas escape causes fire hazard	Launch delay due to fire hazard. Possible vehicle/ mission loss	1
	External major tube	Fuel spill fire hazard. Engine cooling and performance degradation.	Launch delay	1
	Internal tube leaks	Engine cooling efficiency loss. Loss of I_{sp}.	Performance degradation.	3
	Combustion Instability	Structural damage to T/C. Dynamic coupling of loads with propellant systems may cause vehicle damage	Possible vehicle/ mission loss	1
	Dome to T/C leakage	Propellants mixing creates explosion/ fire hazard	Possible vehicle Loss	1

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Fuel PreValve	Fails to close	Cannot load fuel	Launch delay	3
	Fails to open	Cannot start engine	Launch delay	3
Oxidizer Pre-Valve	Fails to close	Cannot load oxidizer	Launch delay	3
	Fails to open	Cannot start engine	Launch delay	3
Oxidizer anti-gysering He flow valve	Fails to open	Cannot load propellants. Can cause surge pressure resulting from oxidizer bubble collapse resulting in damage to oxidizer system ducting and components.	Launch abort	2
	Fails to close	Start sequence inhibited. Helium admitted after the oxidizer.	Launch delay	3
Fuel side anti-freeze solution fill valve	Fails to open.	Cannot procede with preparation stage	Launch delay	3
	Fails to close.	If valve fails to close or if it leaks, the quick disconnect remains as a the single barrier to either fuel or anti-freeze flow.	Launch abort	2
	Leaks.			
	Cannot be disconn-ected.			

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
LOX Manifold Helium purge valve	Fails to open	Pre-ignition engine start sequence inhibited	Launch delay	3
LOX prevalve	Fails to open	Pre-ignition engine start sequence inhibited	Launch delay	3
Fuel prevalve	Fails to open	Pre-igni;tion engine start sequence inhibited.	Launch delay	3
3-Way hypergol ignition system bleed valve	Fails to move bleed	None - Redundant valves	None	3
Nozzle Exit pyrotechnic igniter	Fails to ignite	None - Redundant igniters	None	3

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
POGO system	Fails to operate	Engine system damage	Possible vehicle/ mission loss	1
Check Valve; Hypergol Manifold	Fail to operate and internal leakage	MFV will remain open after engine shutdown command due to pressure backflow against the PA valve	None.	3
	External leakage	Hot-gas escape creates fire hazard	Possible vehicle Loss	1
Hypergol Cartridge	Leaks	Potential fire hazard	Possible vehicle Loss	1
	Diaphragm fails to rupture	None due to redundant cartridges which ignite 1/2 the cannules. The second set of cannules will cross-ignite and may result in a hard start.	None	3

Table 2-8. Preliminary Failure Mode and Effects Analysis Pressure Fed LRB

Specific criteria and groundrules used for criticality rankings are listed below:

Criticality 1

1. Hot gas leakage is assumed to always result in structural/functional damage to at least one engine.
2. Hot gas mixing with LOX is a potential fire/explosion hazard.
3. Oxidizer rich cutoffs always offer the potential for structural damage.
4. Structural failure of rotating machinery or rupture of pressure containment boundaries can both propagate to destruction of one or more engines, followed by loss of life or vehicle.
5. Spark generation in a LOX environment, such as rubbing/fretting of parts in oxidizer pumps or valving, will escalate to a fire/explosion.

Criticality 2

1. Leakage of propellants during start of mainstage is considered as being detectable by hazardous gas monitors or other instrumentation to permit safe engine shutdown. The worst possible scenario of potential mission loss, however, is assigned for conservativeness.
2. Failures precipitating safe engine shutdown. The vehicle is capable of achieving mission success with one engine not operating; however, it is presumed that launch abort, followed by safe shutdown, will be commanded if one engine is not operating prior to liftoff.

Criticality 3

1. External leakage of propellants during preconditioning is assumed to be detected by ambient hazardous gas monitors, which will be cause for launch abort.

2. All others.

2.5 LRB PRESSURE FED ENGINE PROGRAMATICS

The development plan for the LRB pressure fed engine is presented in this section. The overall development schedule is shown in Figure 2-21. The 51 month (4 1/4 years) engine development program is designed to support a first vehicle launch in the third quarter of 1994 and therefore would benefit from a Phase B effort and a technology program directed at defining the best injector configuration. A benefit of the Phase B design effort would be to allow early long lead procurement of casting tooling for some of the major components such as the thrust chamber manifolds. The technology program should be stated in parallel with this Phase B effort and completed in time to provide data for design of the injector. This effort would significantly reduce risk during the hotfire test phase.

As indicated in Figure 2-21, engine test facilities are required by the second quarter of 1992. These facilities are assumed to be provided by the government or the vehicle contractor. Formal Pre-Flight Rating Tests (PFRT) are planned prior to the first flight and Flight Rating Tests (FRT) to certify readiness for production and full operational status are planned after the first flight.

2.5.1 Engine Development Philosophy

The engine test plan has been developed (in terms of numbers of tests and hardware) on the basis that the engine design provides robustness and the design margins are applied to the normal power level (NPL) operating conditions resulting in higher margins at throttled conditions. A design team including engineering, manufacturing, procurement, operations, reliability, producibility, quality and maintainability functions will be fully integrated into the design and procurement process to assure a cost effective low risk engine. Lessons learned from numerous previous large engine development programs will be applied. These include:

1. Component level testing will be conducted in an engine simulating environment to the maximum extent possible.

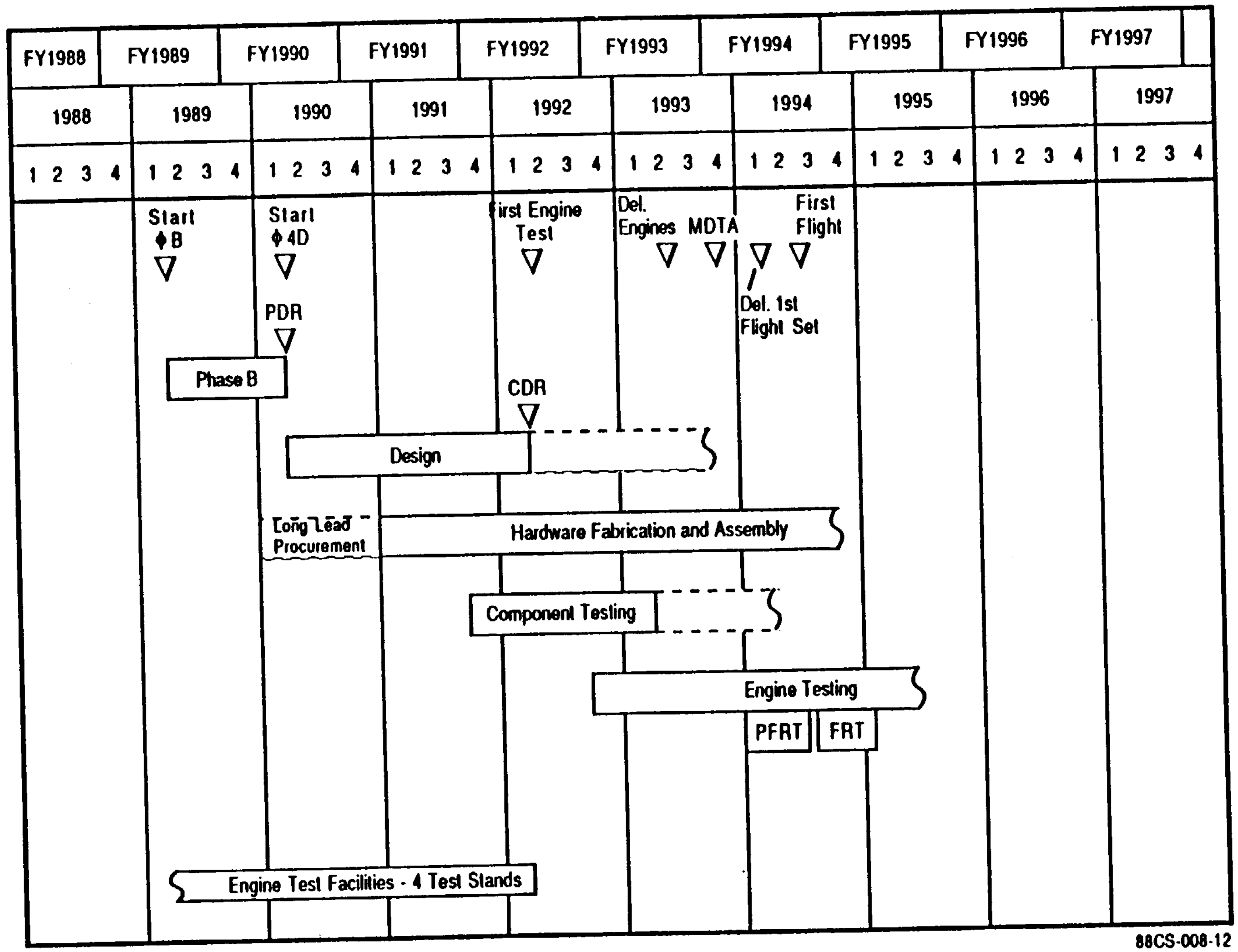

Figure 2-21. LRB Pressure Fed Engine Devlopment Program

2. Extensive limits testing will be conducted at both the component and engine level.

3. Overstress testing will be conducted on a majority of the test units.

2.5.2 Program Approach

Initial effort will consist of analyses and design, making extensive use of Rocketdyne's well anchored analytical tools. Detailed shop drawings will be produced and reviewed during the Critical Design Review (CDR) scheduled 21 months after program start. In parallel with the design effort, procurement of long lead casting tooling will be initiated. It is planned to select the casting supplies early in the program and include them as part of the design team for these parts to be produced by the casting process. Laboratory testing of control system components will be initiated as soon as they are available. The primary objective of the component testing is to drive out design problems and evaluate potential failure modes identified in the Failure Modes and Effects Analysis (FMEA).

The engine test program is designed to drive out random failures and wear-out problems. Engine testing will be initiated as soon as possible. The initial engines will be heavily instrumented to assure that problems can be analyzed and solved in an expeditious manner. Limits and overstress testing will be introduced as soon as possible to verify design margins. Valid component and engine test data will be used to verify the analytical tools used for design and simulation.

2.5.3 Test Plan

The pressure fed booster test plan is presented in Figure 2-22. As indicated, component tests are planned for the control system components and for the injector prior to testing of the complete engine system. The following is a discussion of each of the planned test activities.

Control Components Testing. The control components include the main LOX valve, main fuel valve, control and condition monitoring instrumentation,

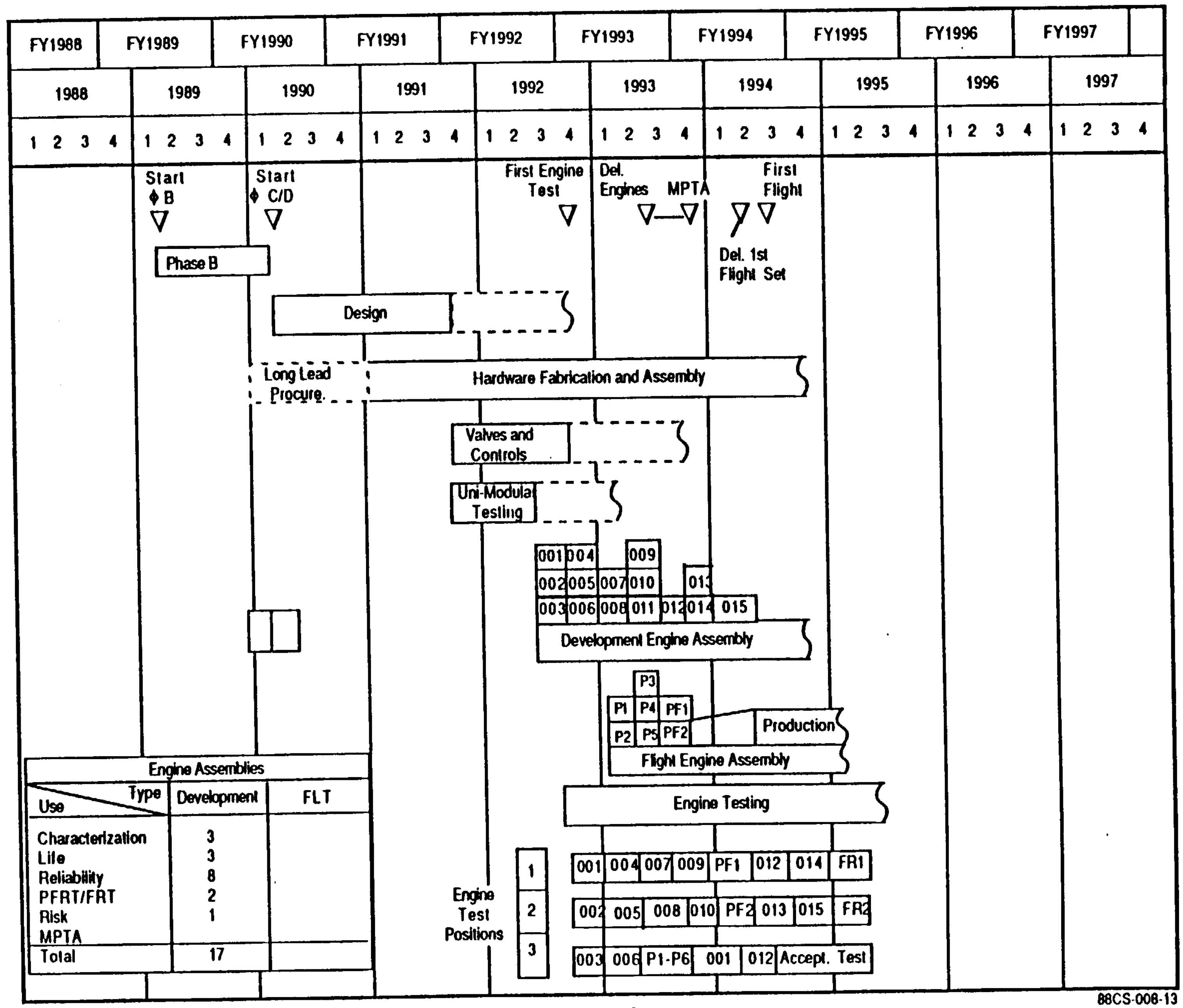

Figure 2-22 LRB Pressure Fed Engine System Test Plan

check valves, igniter fuel valves, pneumatic control console including solenoid valves, electrical burners and an electronic controller package. Control components testing will be conducted at Rocketdyne's existing laboratory test facilities. Three sets of control system components will be procured for laboratory testing. The planned testing is shown in Figure 2-23.

Injector Component Testing. The modular design of the thrust chamber injector allows early testing of a full sized individual module (uni-module) at the component level. Testing of a uni-modules will be accomplished at existing Rocketdyne small engine test facilities at the Santa Susana Field Laboratory. The objectives for this test program are to verify the stability and performance of the uni-module early in the program to allow design changes to be made to the full injector assembly prior to engine test. Three uni-modular test articles are planned. The first test module will be used to evaluate ignition characteristics using a hypergolic fluid and stability and cooling characteristics. The second and third test articles will have the additional objective of demonstrating performance. Each of the 3 test articles will be subjected to bomb tests at limit conditions of propellant inlet pressure and temperature to verify the capability of the stability aids to dampen pressure surges over the full operating range.

Engine Test Program. The first complete engine test is scheduled for the fourth quarter of 1992. The planned development program for the pressure fed booster is divided into 5 phases. These phases are described in Figure 2-24. The first 3 phases are intended to evaluate and demonstrate the maturity and reliability of the engine. The specified demonstrated reliability requirement for the engine is 94 percent at 90 percent confidence. The last 2 phases of the development program are intended to formally demonstrate that the engine is ready for the first flight and subsequent production and operational use. The minimum number of tests and engines required for each of the 5 phases of the development program are defined in Figure 2-25. Also shown in Figure 2-25 is the expected test realization factor, that is, the number of tests that are expected to abort or not produce valid data. This factor is used for planning the total number of tests for the development program. The risk factor for the program is also shown. Since the pressure fed LOX/RP-1 LRB is considered a low risk program, a factor of 5 percent is applied to the number of tests

	Laboratory Development Tests												
Component	Functional	Thermal Vacuum	Thermal Cycling	Vibrational	Acoustic	Pyro Shock	Acceleration	Humidity	Pressure	Leakage	EMF/EMC	Life	Burn-in
Main LOX and Fuel Valves, Igniter Fuel Valves Check Valves	X	X	X	X					X	X	X	X	
Controller Control and Condition Monitoring Instrumentation	X	X	X	X							X		
Pneumatic Control Console and Solenoid Valves	X	X		X				X	X	X	X	X	
Flight Instruments	X	X	X	X		X	X		X	X			X
Harnesses	X		X	X	X			X		X	X		

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Figure 2-23. LRB Controls Component Development Plan

Test Phase Descriptions

Phase	Descriptions
1. Characterization	Testing Designed to Fully Evaluate Engine Operation Includes Tests to Evaluate: • Ignition • Start/Shut • Performance • Stability • Duration • Gimbaling • Limits • Overstress • Failsafe • Heat Exchanger • POGO
2. Life Development	Testing Designed to Evaluate the Life Margin in the Engine Design
3. Reliability Demonstration	Testing Designed to Demonstrate that Engine Reliability is as Specified
4. Pre-Flight Rating (PFRT)	Formal Testing Specifically Designed to Demonstrate Readiness for First Flight
5. Flight Rating (FRT)	Formal Testing Specifically Designed to Demonstrate Readiness for Production and Full Operation

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Figure 2-24. Engine Development Program

Requirements

Phase	Requirements
Characterization	* 100 Tests on 3 Engines
Life Development	* Formal Demonstration Life on 3 of Every Component During Engine Testing
Reliability Demonstration (99% Rel. at 90% Confidence)	* 230 EFDT's on 8 Engines
Pre-Flight Rating (PFRT)	10 Full Duration Tests Each, on 2 Engines
Flight Rating (FRT)	Formal Demonstration Life on 2 Engines
Spares	20 Percent
Factors	
Test Realization	10 Percent
Risk	5 Percent

* Total Number of Tests Can Be Reduced by Combining Objectives

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Figure 2-25. Engine Development Program

and engines. As indicated in Figure 2-24, the characterization phase testing is designed to fully evaluate engine operating characteristics. The test objectives for each engine assigned to this phase are shown in Figure 2-26.

The number of engines required for the development program is based on the design life specified. Since the LRB pressure fed engine will be used in the expendable mode, its mission life requirement is one. However, the demonstration of the maturity and reliability of an expendable engine requiresthat each engine be capable of many tests. By defining the design life requirement at 60 missions, this allows each engine to be tested at least 30 times during the first three phases of the development program with a safety factor of 2 resulting in a significant hardware cost savings. These engines can also be tested 10 times each during the formal PFRT and FRT test phases thus demonstrating a factor of 2 on the number of starts (5) that could be expected for a production flight engine. These 5 starts include: 2 acceptance tests, allowance for 2 on-pad aborts and 1 flight.

The minimum number of tests and engines assigned to each phase of the engine development test plan are shown in Figure 2-27. Additional tests and engines based on the test realization, risk and spares factors are also shown. As indicated a total of 462 tests and 17 engines are required to complete the development and flight certification of the pressure fed LRB engine. Figure 2-22 shows that 3 engine test positions are necessary to complete the planned engine test program. A test frequency of approximately 2 tests per week per test position is planned. Also note in Figure 2-22 that in addition to testing the 17 engines required for the development program, the 5 engines required for the main propulsion test article (MPTA) are acceptance tested prior to delivery.

2.5.4 New Technology Requirements

Large Propellant Valves and Electric Actuators. Recent advancements in the technology of building very high speed, powerful yet small electric motors has made an all electric fighter plane possible; i.e., no hydraulics are required on such a vehicle. This concept may well be applicable to the subject design, especially since smaller versions of such valve actuators have been

Test Objectives	Engine Serial Numbers			Test Allocation*		
	001	002	003	Tests	Engines	Total
Ignition	X			11	1	11
Start and Cutoff	X			10	1	10
Performance	X	X		8	2	16
Stability	X	X	X	5	3	15
Duration		X	X	3	2	6
Gimbaling		X	X	3	2	6
Limits	X	X	X	5	3	15
Overstress		X	X	3	2	6
Fail Safe			X	3	1	3
Heat Exchanger		X	X	2	2	4
Pogo		X	X	4	2	8

* Primary Objective Indicated, Multiple Compatible Objectives Will Be Accomplished on Each Test

Total Tests = 100

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Figure 2-26. LRB Pressure Fed Test Matrix - Characterization Phase

Development										Test Realization	Risk		Spares	Total	
Character-ization		Life		Rel. Demo		PFRT		FRT							
Tests*	Engines	Tests*	Engines	Tests*	Engines	Tests	Engines	Tests	Engines	Tests	Tests	Engines	Equiv. Engines	Tests	Engines
100	3	30	3	230	8	20	2	20	-	40	22	1	4	462	21

* Total Number of Tests Can Be Reduced by Combining Objectives

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Figure 2-27. LRB Pressure Fed Engine Development Program System Test/Hardware Requirements constructed at Rocketdyne specifically for use as main propellant valves in rocket engines. Figure 2-5 in Section 2.2.1 shows a drawing of such a valve. Scaling up to the large diameters required here can be considered to have low technical risk and result in a substantial gain in simplifying the job of valve actuation in very large rocket engines. This concept has also been considered as applicable to implementing gimbal thrusters as discussed with GDSS.

The Use of Tridyne to Furnish Heated Pressurant Gas. Rocketdyne has pioneered the use of Tridyne, which is a mixture of mostly inert gas with a small amount of both oxygen gas and hydrogen gas. The later gasses are present in such small amounts that they cannot burn with a flame or detonate. This gas mixture, however, when passed through a catalyst will cause the oxygen and hydrogen to react almost instantly, creating enough heat to raise the temperature of the gas effluent by as much as 1500 F. By decreasing the percentage of reactants, almost any temperature rise less than this can also be achieved.

Rocketdyne and GDSS have together conducted preliminary analyses to determine the relative merit of such a system when applied to solving the task of minimizing the weight of pressurant gas required in the subject, very large, relatively high pressure propellant tanks. Although considerable progress was made to show that the method has promise, some analytical and experimental work is indicated to determine certain empirical factors as follows:

1. How low in temperature can helium Tridyne be stored and still spontaneously heat itself when passed through a catalyst which is also at a low temperature. It is desirable to store the pressurant at the lowest possible temperature. Use of the catalyst has been successful at temperatures as low as -65 degrees without any evidence to show that the reaction was not normally taking place. However, it should be established experimentally if such a lower limit does exist. If a low catalyst temperature does present a problem, it can be solved by supplying an electric preheat. However, it should be determined where if any such heating might be required, and at what temperature.

2. When gas is allowed to vent from a tank, the remaining gas in the tank undergoes a poytropic expansion which cools it. This causes the pressure to drop and results in an excessive amount of gas to remain in the pressurant storage tank at the end of the burn. The gas in the storage tank should be heated to allow a greater amount of gas to be expelled from it, thereby requiring less mass of gas to be loaded initially. However, heat transfer methods to a stagnant gas inside a high pressure storage tank is difficult and requires an excessively large heat exchanger. A new method of doing this is to provide for two Tridyne storage tanks connected in series, so that when the first one becomes cool, hot gas from a second tank is introduced into the first one utilizing the difference in pressure to perform mixing of the hot gas being introduced with the cold gas already there. This cascading of tanks could be extended to 3 or more tanks. The optimum is not presently known. Further analytical work can be utilized, but only engineering laboratory type tests specifically aimed at evaluating this method will conclusively determine its merits.

3. The amount of cooling which the pressurant gas undergoes when it enters a partly empty LOX tank is not easily calculated and may best be determined experimentally. This is not unique to Tridyne, since it is a problem faced by any method utilizing heated pressurant gas.

4. Tridyne gas contains a certain amount of water after passing through the catalyst bed. It is not known what, if any harm is done to the functioning of the liquid oxygen storage tank and associated feed system by the ice particles which may well form inside the ullage space above the LOX surface. Ice crystals may fall into the LOX where they will float on the surface and should do no harm. However, this action needs to be studied enough to confirm this.

<u>Development of Extended Range Flex Line Joints to Reduce Skirt Diameter</u>. If the flex lines would permit engine pivot angle excursions of plus or minus 12 degrees instead of 6 degrees, then there is the possibility that the four engine cluster could be grouped together much more closely, and might reduce the skirt diameter by as much as 3 feet. This would be achieved by changing the gimballing mode to a single quadrant for each engine, one engine for each

of 4 different quadrants. The same steering moment is achieved by deflecting 2 engines at 12 degrees, as with 4 engines at 6 degrees (approximately). The dynamic response and power required of the actuators is, however, greatly increased. Additional preliminary analysis of this concept should be undertaken to determine its merits.

Development of an Injector Concept Using Ablative Material Between the Cannuls. This concept consists of utilizing a thick ablative layer fastened to the face of the injector which simultaneously serves the function of baffles for engine stabilization and to aid in cooling the injector face. The concept of utilizing ablative material for this purpose is new and will require some development work. However, this design is an extension and combination of known techniques and is considered to present low technical risk. A preliminary design of such an injector has already been made. A development effort on a small scale has promise for advancing large size engine technology.

2.6 LOX/RP-1 PRESSURE FED LIQUID ROCKET BOOSTER PRELIMINARY CONTRACT END ITEM (CEI)

2.6.1 Background

The LOX/RP-1 pressure fed Liquid Rocket Booster engine is being designed to provide booster propulsion for the Space Shuttle. The primary objective of the LRB study was to identify and evaluate a viable LOX/RP-1 pressure fed engine candidates that would meet the requirements for the STS and would have commonality with the Space Transportation Main Engine (STME) currently being studied.

2.6.2 Selected Engine Description

The selected engine configuration utilizes LOX and RP-1 as propellants. RP-1 is used to cool the MCC and nozzle after which it is injected into the injector.

2.6.3 Pressure Fed LOX/RP-1 LRB CEI Requirements

This document presents the preliminary CEI requirements that the LRB must fulfill to satisfy the requirements for the STS. These requirements are as follows:

Performance. All performance values stated herein are nominal values. The minimum and maximum values will be determined during subsequent study efforts.

1) Engine Thrust - The LRB shall be capable of producing 935,000-lb vacuum thrust at the normal power level (NPL) and 614,900-lb vacuum thrust at the minimum power level (MPL). The engine shall be capable of throttled down from NPL to MPL in TBD seconds.

2) Specific Impulse - The specific impulse for the LRB shall be as follows for the two vacuum equivalent thrust operating points:

Thrust Level	Sea Level Is (seconds)	Altitude Is (seconds)
935,000 lb (vac)	239 ± TBD	279 ± TBD
614,900 lb (vac)	217 ± TBD	278 ± TBD

3) Main Combustion Chamber (MCC) Propellants

Propellants	Injected State
Oxidizer - Oxygen (LO_2)	Liquid
Fuel - RP-1 (H_2)	Liquid

4) Engine MR - The engine MR for the pressure fed LRB shall be as follows for the two thrust operating points:

Thrust Level	Mixture Ratio
935,000 lb (vac)	2.5
614,900 lb (vac)	2.5

5) Acceptance Calibration - The acceptance calibration for the LRB shall be as follows:

Thrust (NPL) - 935,000 lb ± 3% (vac)
(MPL) - 614,900 lb ± 3% (vac)
MR (NPL) - 2.5 ± 1%
(MPL) - 2.5 ± 1%

6) Coolants - The coolants for the MCC and nozzle shall be RP-1.

7) Burn Duration - The LRB shall be capable of maximum burn duration of 180 sec at NPL and MPL.

8) Uncoupled Thrust Oscillations - The engine-produced uncoupled oscillatory thrust shall be no greater than the following for the respective specified frequency ranges:

R = 0 to 1.5 Hz	F = ± 6000 lb
R = 0.5 to 1.5 Hz	F = ± 1500 lb
R = 1.5 to 2.5 Hz	F = ± 450 lb
R = 2.5 to 100 Hz	F = ± 1500 lb

For the purpose of performing data analysis to verify engine compliance in the critical frequency range oscillatory shall be defined as the average value of an oscillation over at least 16 cycles.

9) Combustion Stability - The engine-produced main chamber pressure oscillations shall not exceed ± 5% of the mean steady-state pressure.

10) Damping time for artificially induced pressure spires shall be TBD milliseconds maximum.

11) POGO Suppression - The engine shall provide a POGO suppression system in accordance with the following requirements (TBD).

12) Engine Controller - The electrical engine control system shall be capable of continuous operation at ambient temperature for an unlimited period of time during checkout and maintenance.

13) System Checkout and Monitoring Capability - The design shall include onboard checkout capability, redundancy verification, and status monitoring during ground operations. The engine design shall include a limit control system capable of automatically initiating engine shutdown to prevent catastrophic failure.

Operations. The operational requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Engine Start - The engine start system shall have self-contained control within the engine envelope. The start sequence shall be started by a single electrical signal from the vehicle or ground source.

2) The engine shall be capable of one start after each ground servicing.

3) The engine start sequence shall be capable of achieving normal power level (NPL) thrust in less than TBD sec.

4) The thrust buildup rate shall not exceed TBD lb thrust in any 10-msec time period.

5) Starting Impulse - The starting thrust impulse to NPL shall not exceed TBD lb-sec.

6) Throttling Control - The engine shall be equipped with a thrust control system capable of raising the thrust at NPL to the specified thrust at MPL in the event of an engine condition-out during a vehicle launch.

   a) Throttle Rate - The engine thrust control system shall be capable of raising the engine thrust from NPL to MPL at the rate of TBD lb-sec any time after reaching NPL.

   b) The thrust control system shall be capable of a step response of TBD lb thrust increase in less than TBD sec after a step command.

Engine Shutdown. The engine shall be capable of a safe shutdown from any power level including the start sequence.

1) The engine shutdown sequence shall be capable of reducing thrust from NPL to zero in TBD sec.

2) The shutdown impulse shall not exceed TBD lb/sec from NPL.

3) The engine shall be capable of shutdown from any defined thrust level upon receipt of an electrical command at a rate of TBD lb thrust change per any 10-msec time interval.

Environmental Conditions. The engine shall be capable of operating safely under the following conditions:

1) The engine shall be capable of operating safely where exposed to a hat flux of TBD Btu/ft^2-sec and a surface temperature of TBD°F. The heat transfer coefficient that shall be used for design is TBD Btu/sec-ft^2°F.

2) The surface temperature of lines or surface in contact with cryogenic propellants shall be controlled to preclude the formation of liquid air.

3) Acceleration Loads - TBD

4) Shock Loads - TBD

5) Ground Handling and Transportation Loads - TBD

6) Storage Life - The engine shall be capable of being transported and stored over an ambient temperature range of TBD°F to TBD°F, an ambient pressure range of TBD psig to TBD psi,, a relative humidity of 100% at temperatures less than or equal to TBD°F.

   a) The engine shall suffer no degradation of reliability or operating life during the storage period, subject to the inspection and maintenance requirements TBD.

7) Exposure - The engine system and components shall be capable of being transported and stored without deterioration in areas where conditions may be encountered having salt spray and relative humidity as experienced in coastal regions. The engine system and components shall be capable of withstanding exposure to sand and dust when equipped with proper closures.

8) Lightning - The engine controller shall be designed to operate without damage in accordance with TBD lightning protection criteria.

Prelaunch. The engine shall be designed for minimum prelaunch servicing.

1) Ground Service - The engine shall be capable of achieving pre-launch thermal conditioning without ground servicing in less than TBD minutes from the time propellants are supplied to the engine. Recirculation flow rates to achieve thermal conditioning are as follows:

   LOX - TBD lb/sec
   RP-1 - TBD lb/sec

2) The engine shall be capable of servicing and maintenance while in either the horizontal or vertical position.

3) The engine shall not require any servicing from ground equipment within 24 hr after propellants are loaded.

4) External or internal leakage of propellants shall not occur in such a manner as to impair or endanger the engine/vehicle function. Leakage monitoring capability shall be provided with the design objective that separable connections not exceed 1×10^{-4} sec helium at leak check pressure.

5) The engine shall not require any monitored redlines external to the engine prestart and shall provide a continuous engine-ready signal to the vehicle when all critical parameters monitored by the engine control system are within TBD conditions.

Interface. The engine shall require the following conditions at the respective interfaces with the vehicle:

1) Propellant inlet conditions at engine start:

   a) LOX - TBD psia to TBD psia, 163 to 170°R
   b) RP-1 - TBD psia to TBD psia, 510 to 550°R

2) Propellant inlet conditions during mainstage:

   a) LOX - TBD psia to TBD psia, TBD to TBD°R
   b) LH_2 - TBD psia to TBD psia, TBD to TBD°R

3) Electrical

   a) The engine shall be supplied TBD dc V
   b) The engine shall be supplied TBD ac V
   c) The controller shall be engine supplied and mounted.

4) Pressurization Gas - Requirements TBD.

5) Purge Requirements - Nitrogen, in accordance with MIL-P-27401, and helium, in accordance with MIL-P-27407, shall be used for operational and servicing purges and leakage tests.

   a) Operational Purges - TBD
   b) Servicing Purges - TBD

6) Digital Interface

   a) A suitable digital interface shall be provided for vehicle commands to the engine.

Physical Requirements. The physical requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Envelope - the maximum engine width is 108 in. and the engine height is 189 in.

2) Weight - The engine weight is as follows:

	Dry	Wet
Basic engine	4,237 lb	TBD
Accessories	1,460 lb	TBD
Thermal Insulation	TBD lb	TBD

3) Gimbaling - The engine shall be capable gimbaling in a ± 7° square pattern at a gimbal rate of 10°/sec and an acceleration rate of 1.0 rad/sec squared. The engine shall provide attach points for the vehicle-furnished actuators. The gimbal system shall be capable of returning the engine to null position at engine shutdown. The gimbal system control melium is TBD.

4) Engine Alignment - The engine shall be aligned so that the actual thrust vector is within 30 min of an arc to the engine centerline and within 0.25 in. of the gimbal center. The gimbal center shall be within 0.010 in. of the engine centerline.

5) Engine Fluid Interface Ducts and Lines - The engine shall supply all interface ducts and lines with a minimum of TBD in. straight section upstream of the engine interface plane.

6) Engine Electrical Interface - Al engine electrical connections from the vehicle shall be located in a single, engine-mounted panel.

Reliability. The reliability of the configuration upon which the final flight certification is based shall be that which is necessary to ensure functioning within the specified design life.

1) The engine design life is 1.0 missions at NPL.

2) The engine shall be designed for a minimum of 5 main stage ignitions.

3) Fail-Safe Design - The engine shall be capable of shutdown from an internal signal without damage to other systems.

4) Structural Criteria - The engine shall be designed to provide the following minimum factors of safety:

Minimum yield	- 1.1
Minimum ultimate	- 1.4 combined loads
Minimum ultimate	- 1.5 pressure only
Minimum proof	- 1.2 times EPL operating conditions, unless fracture mechanics requires a higher factor
Low cycle fatigue	- 4.0
High cycle fatigue	- 10.0

Note: Components should be designed for 1.25 on endurance limit where feasible

Diagnostic Monitoring. The engine shall be capable of self-diagnostics in real time. Unsafe conditions shall cause an engine-generated shutdown unless inhibited by the vehicle.

1) Diagnostic data will be recorded for postflight analysis.

3.0 PUMP FED LOX/RP-1 ENGINE

This section presents the selected LRB LOX/RP-1 pump fed booster rocket engine configuration and characteristics resulting from the technical analyses and trade studies.

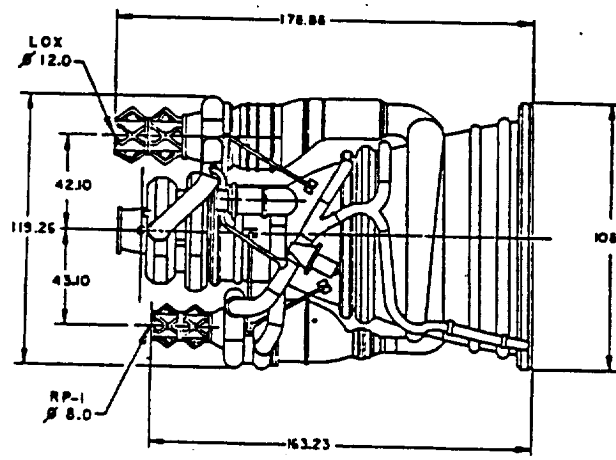
A baseline engine concept was selected based on ongoing Space Transportation Booster Engine (STBE) studies and experience along with trade studies for the STS application. An engine performance and pressure balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the recommended configuration and physical design.

3.1 ENGINE/SUBSYSTEM CONFIGURATION SELECTION

The hydrocarbon engine selected for the pump fed LRB uses LOX/RP-1 propellants at a Emergency Power Level (EPL) chamber pressure of 1400 psia and 2.8 engine mixture ratio. The selected engine cycle is a gas generator cycle producing 1800 R turbine drive gases to drive the RP-1 turbopump and the LOX turbopump which are in series. Series turbines were selected to minimize the secondary flow performance losses of the Gas Generator, (GG gases) which are exhausted into the thrust chamber nozzle at an area ratio of 16:1. The nozzle exit area ratio is 27:1 which represents a nozzle exit pressure of 6 psia at nominal operating design conditions. The nozzle contour is an 80% bell with a 4-degree exit wall angle to accommodate sea level operation at minimum power level without nozzle flow separation. The engine layout is shown in Figure 3-1a and 3-1b. A simplified flow schematic is shown in Figure 3-2.

3.1.1 Thrust Chamber Cooling Selection

The thrust chamber consists of an injector, main combustion chamber (MCC), and a nozzle. The RP-1 fuel is used to cool the surfaces of these components exposed to the 6500 R combustion gas environment. To adequately cool these components while maintaining a minimum component weight, each component will use specific fabrication techniques and materials. It is desirable to use a light weight tubular construction for the nozzle/MCC. This design technique,



Basis:

$N_c^* = 96\%$			
$\epsilon_{noz} = 27.25$		EPL Thrust (VAC)	791.3 Klb
$P_{e\ noz} = 6.24$ psia		P_c	1402 psia
P inlet (oxidizer) = 65 psia		I_s (VAC)	321.9 sec
P inlet (fuel) = 45 psia		I_s (SL)	273.9 sec

Figure 3-1a. LOX/RP-1 LRB Pump-Fed Engine

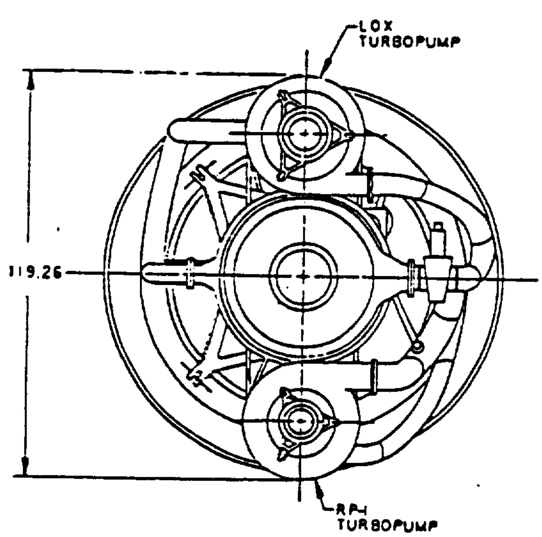
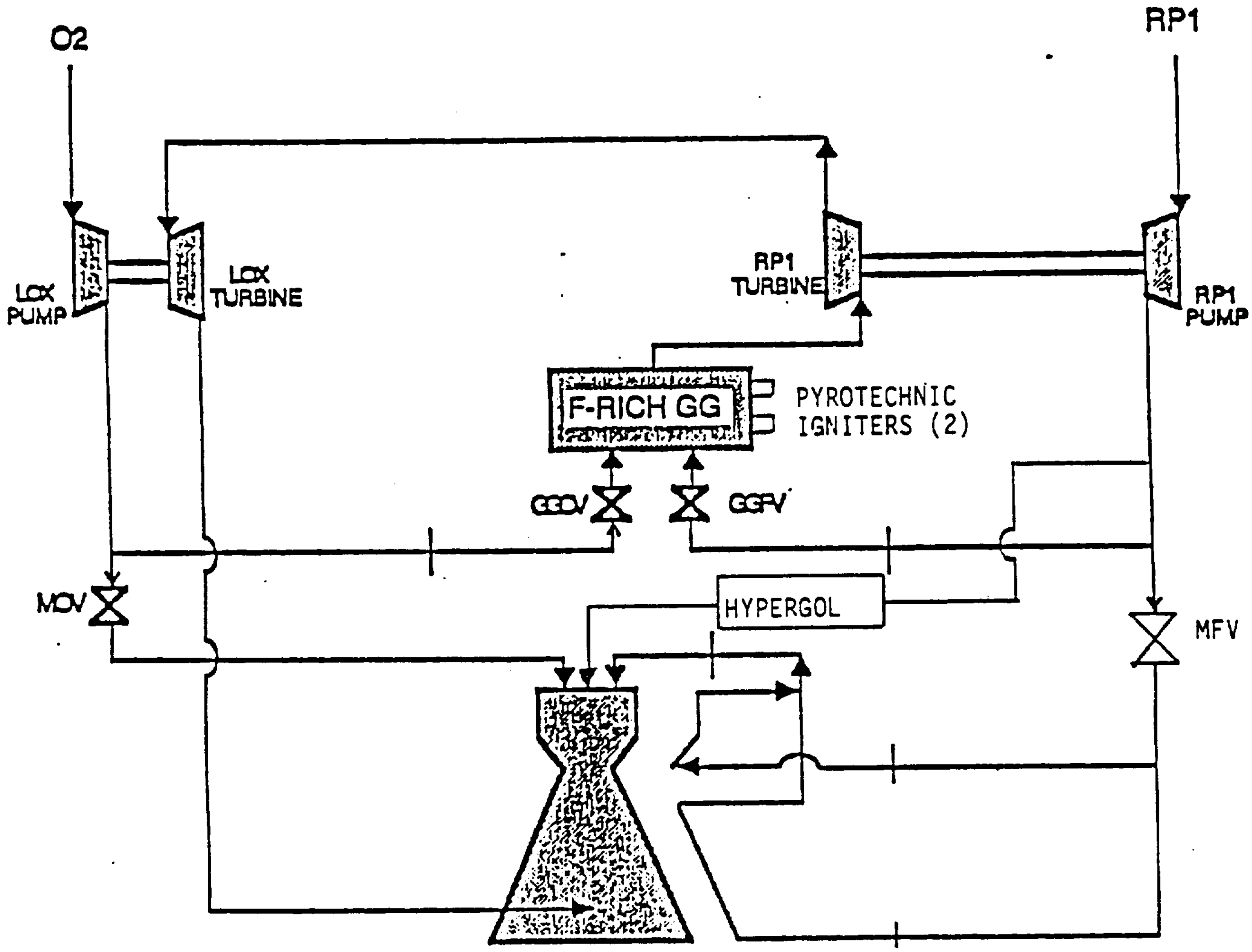


Figure 3-1b. Top View of LOX/RP-1 Engine

Figure 3-2. Simplified LRB LOX/RP-1 Pump-Fed Engine Flow Schematic



using stainless steel tubes, is most satisfactory for a low heat flux nozzle. However, the high heat fluxes of the MCC at 1400 psia requires that it be fabricated of a copper base alloy (NARloy-Z) milled channel configuration, typical of the SSME. Therefore, a one piece construction MCC/Nozzle as used for lower chamber pressure thrust chambers, i.e., the Atlas and the pressure fed LRB is not feasible for the high P_c pump fed engine. As discussed later, the injector will be a ring-type design similar to other LOX/RP-1 injectors and will use OFHC copper rings, as was used in the F-1 injector for adequate injector face cooling.

The nozzle to MCC attachment point is at an area ratio of 5:1 where 50 percent of the RP-1 is used to cool the nozzle and 50 percent is used to cool the MCC. This 50/50 flow split and 5:1 attachment location provides the lightest weight engine with the lowest RP-1 pump discharge pressure. An up-pass cooling circuit is used for both the MCC and nozzle. A fraction of the nozzle coolant is diverted to the gas generator and the remainder is mixed with the MCC coolant and discharged to the main injector. The nozzle coolant ΔP is low compared to the MCC and provides the highest energy level RP-1 to the gas generator. The cooling characteristics and energy levels are depicted in the engine balance tables of section 3.2.3. Fuel cooling was selected over oxidizer cooling from a materials compatibility standpoint.

3.1.2 NPSH Requirements With and Without Boost Pumps

The inlet pressures to the oxidizer and fuel pumps were selected to be 65 psia for the oxidizer and 45 psia for the fuel. With these pressures, boost pumps are not required and a reduced engine weight with fewer components result.

The impact of LOX pump inlet pressure on engine performance, turbine tip diameter, and engine weight are depicted in Figure 3-3. The design point selected was 65 psia without a boost-pump. The design point of 45 psia for RP-1 is based on similar trade factors.

Pump inlet lines from the fuel and oxidizer tanks should have an upstream straight section length five times the inlet diameter. This length may be reduced nearly 50-percent if flow vanes and flow straighteners are

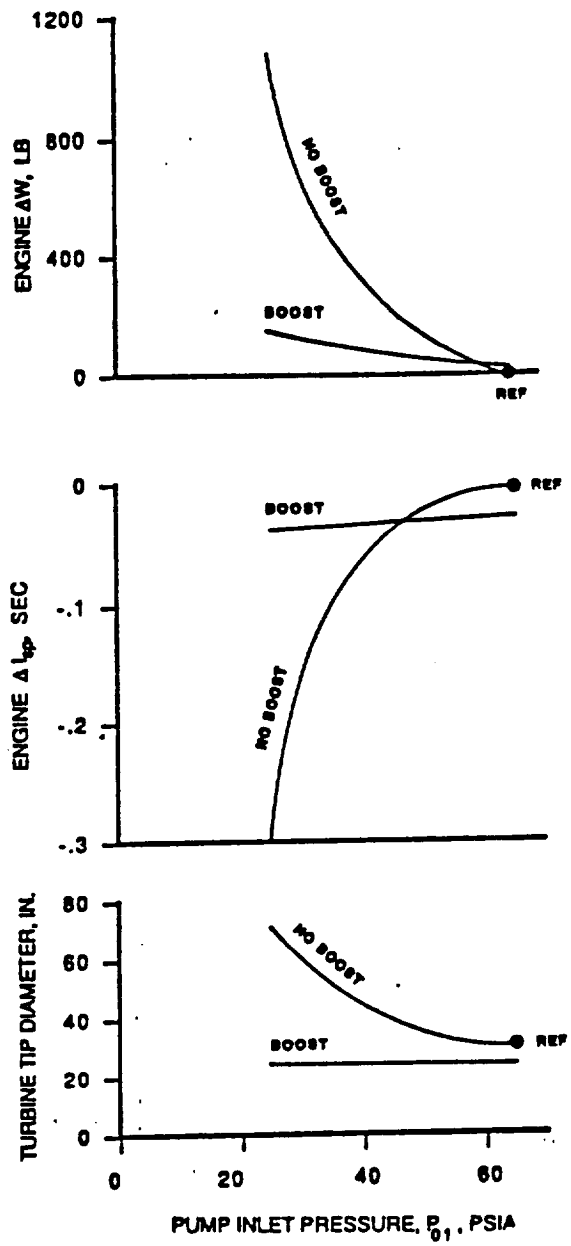


Figure 3-3. LOX Pump Inlet Pressure Effects

properly designed; however, these may require considerable design, analysis and development.

3.1.3 Injector Selection and Rationale

The main injector is a ring type injector with self-impinging oxidizer and fuel doublet orifice pattern similar to past LOX/RP-1 injectors. These rings will be fabricated of OFHC copper, similar to the F-1 injector, to provide adequate injector face cooling at 1400 psia chamber pressure.

The injection pattern will be similar to the high performing RS-27 engine but will be closer packed. Combustion stability aids, in the form of RP-1 cooled baffles and MCC injector-end accoustic absorbers will be employed. Main injector propellant ignition will be attained using multi-element 85/15 TEA/TEB hypergol introduced to the injector at ignition start propelled by the RP-1. Again, this is the well developed ignition system used on previous LOX/RP-1 engines. Start sequencing will also be identical to that developed for previous Rocketdyne engines.

3.1.4 Mixture Ratio Control During Throttling

The pump fed LRB throttling capability is +10% and -25% for a 35% throttling range. This is a fairly large throttling range for incompressible fluids such as LOX/RP-1 and will slightly penalize the pump discharge pressure requirements to provide adequate dynamic and combustion stability of the main injector. A design $\Delta P/P_c$ of the gas generator and main injectors of 20% was used for the LRB engine.

During the throttling excursions the gas generator mixture ratio will be maintained constant to provide a constant combustion gas temperature of 1800 R to drive the hot gas turbines. This approach was taken for three reasons. First, 1800 R is about as high a temperature as one would use without further turbomachinery materials development and elaborate cooling concepts for the gas generator. Secondly, 1800 R has been shown by past experimental testing to be the temperature/mixture ratio that produces the least amount of carbon deposition on the turbomachinery. Thirdly, maintaining a constant gas generator combustion gas temperature requires a minimum secondary flow (GG gases) and results in the highest attainable energy level and maximum engine performance during throttling.

Since the GG gas flowrates are a small percentage of the engine flowrates, the thrust chamber and overall engine mixture ratio is not significantly impacted and the engine operates at maximum efficiency over the mission trajectory.

3.1.5 Gas Generator (GG) Exhaust

The GG gases are symmetrically discharged into the nozzle at an area ratio of 16:1. This concept was selected for packaging purposes and to provide the maximum engine performance by entraining and expanding these exhaust gases with the main propellant gases. There is also a secondary benefit attained through GG gas cooling the nozzle which reduces the RP-1 coolant ΔP of the nozzle.

The selection of the area ratio to discharge the GG gases is based on the minimum area ratio acceptable to maintain the required pump pressure ratio for maximum performance while providing minimum engine weight and minimum engine packaging dimensions.

3.1.6 Control

A closed loop control system will be required for the LRB pump fed engines to accommodate the throttling requirements. For reasons previously discussed, the gas generator mixture ratio will be constant throughout the 35 percent throttling range. To attain this, the gas generator fuel valve will be used to control the GG mixture ratio and turbine inlet hot gas temperature as noted in Figure 3-4.

3.2 ENGINE DESIGN ANALYSIS AND OPERATION

Engine system layouts, design description, engine balances, engine systems weight breakdown, flight instrumentation, and engine systems schematics are presented for a LOX/RP-1 LRB engine which operates at a design baseline

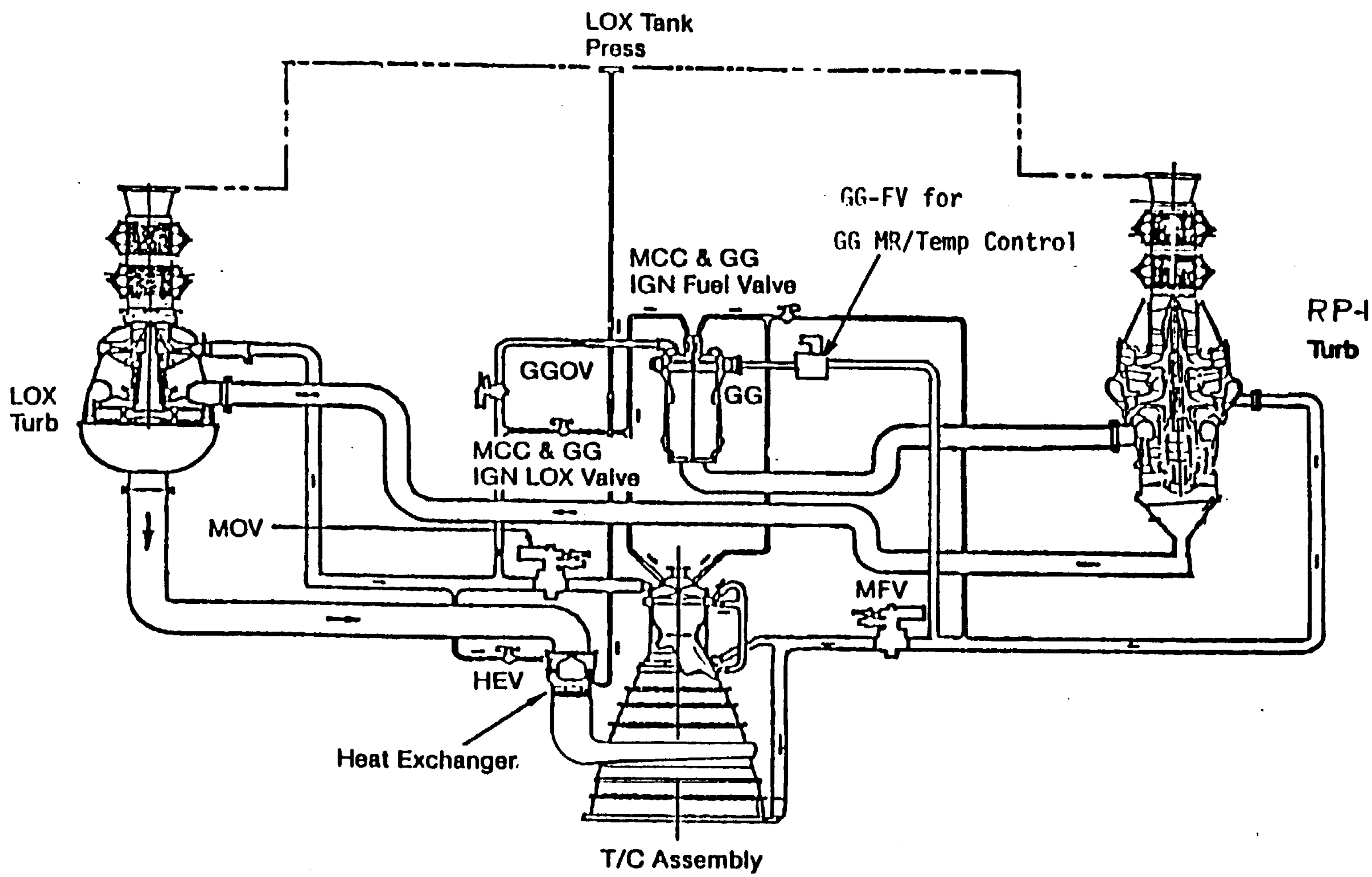


Figure 3-4. LRB Flow Schematic

Emergency Power Level, (EPL) thrust of 791 Klb (Vac) and 1400 psia chamber pressure.

3.2.1 Engine Description

The selected engine configuration was shown in Figure 3-1 with pertinent overall dimensions and interface information. Engine flow schematics were previously presented in Section 3.1. Engine design features are noted in Table 3-1 with engine design and operating characteristics summarized in Figure 3-5, and with combined engine flow schematic and operational characteristics shown in Figure 3-6.

Table 3-1. LOX/RP-1 Pump Fed Engine Design Features

• Thrust	• Throttling	+10%/-25%
• Cycle	• Gas Generator	
• Main Combustion Chamber	• Channel	
• Nozzle	• Tubular	
• Oxygen Turbopump	• 1 Stage Centrifugal	
• Fuel (RP-1) Turbopump	• 1 Stage Centrifugal	
• Boost Pump	• None	
• Control System	• Closed Loop Thrust and Mixture Ratio Control	
• Start Type	• Turbine Spin Start	
• Inlet Ducts	• Scissors	

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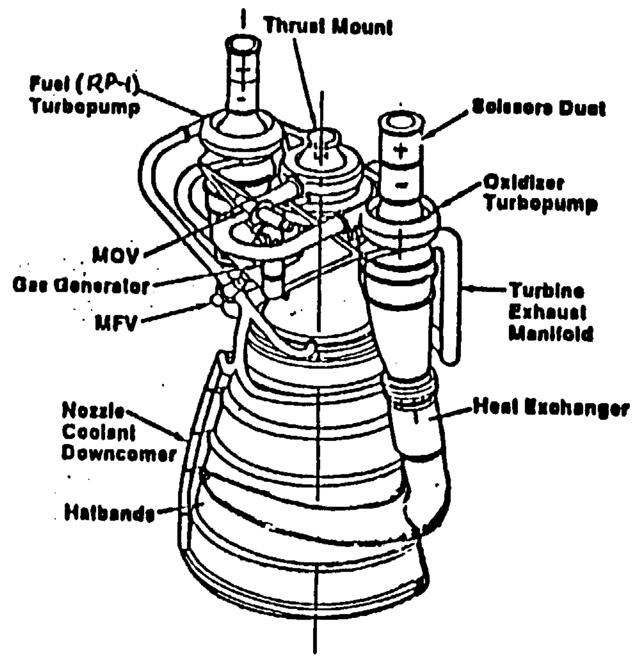
3.2.2 Engine Instrumentation and Control

The LRB control and health monitoring system will utilize both performance and in-situ condition monitoring instrumentation to determine the overall health of the engine system to the extent required for acceptance testing. The health monitoring system will be integrated into the control system functions. The engine condition monitoring sensors are listed in Table 3-2 with a preliminary list of performance and redline instrumentation shown in Table 3-3. The performance instrumentation is used by the controller to modulate the



• Cycle	G.G.
• Thrust, Klb (sl)	673
Klb (vac)	791
• Specific impulse, (sl)	273.9
(vac)	321.9
• Chamber pressure, psia	1402
• LOX pump inlet pressure, psia	65
• Fuel (RP-1) pump inlet pressure, psia	45
• Area ratio, e (nozzle)	27
• Mission life	1
• Delivered life	
Starts	5
Seconds	1500
• Lenght, in	182.5
• Power head diameter, in	129.6
• Nozzle exit diameter, in	108.0
• Weight, lb	8108
• Gimbal capability, deg	±6
• Throttling capability	+10/-25%

Figure 3-5. LRB Pump Fed Engine Characteristics

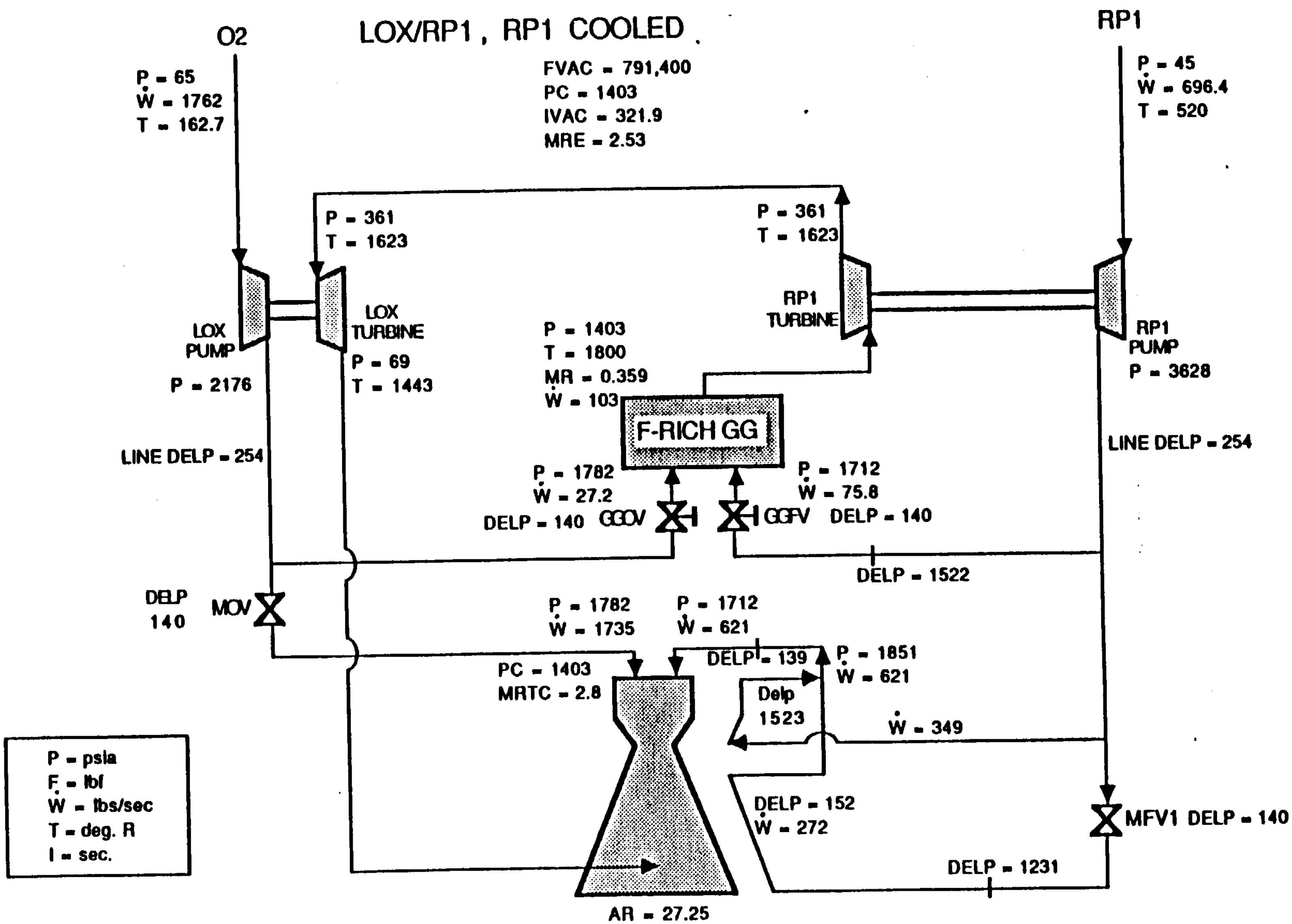


Figure 3-6. LOX/RP-1 Bi-Propellant Engine Flow Schematic

valve actuators to regulate both a constant gas generator mixture ratio and thrust level throttling.

Table 3-2. Engine Acceptance Testing Condition Monitoring Sensors

Bearing Sets (count)	6
Isotope Wear Analyzer Sets	6
Fiberoptic Bearing Deflectometer	12
Shaft Torque Intervals (count)	3
Torquemeter	3
Plume Combustion Monitors	1
Spectormetric Anamalous Combustion Specie Detector System	1
Spectormetric Mixture Ratio Detector	1
Optical Leak Detector System*	1

*Leak detector system is mounted on the facility, 1 per engine.

Table 3-3. Preliminary Performance and Redline Flight Instrumentation List for the STBE

#	MEASUREMENT
1.	Engine LOX Inlet Pressure
2.	Engine LOX Inlet Temperature
3.	LOX Pump Shaft Speed
4.	LOX Pump Acceleration
5.	LOX Pump Discharge Pressure
6.	LOX Pump Discharge temperature
7.	Engine LOX Flowrate
8.	GGOV Inlet Pressure
9.	GGOV Inlet Temperature
10.	GGOV Inlet Flowrate
11.	GGOV Position
12.	GGOV Discharge Pressure
13.	GGOV Discharge Temperature
14.	GG LOX Injector Pressure
15.	GG LOX Injector Temperature
16.	MOV Inlet Pressure
17.	MOV Inlet Temperature
18.	MOV Position
19.	MOV Discharge Pressure
20.	MOV Discharge Temperature
21.	MCC LOX Injector Pressure
22.	MCC LOX Injector Temperature
23.	Engine Fuel Inlet Pressure
24.	Engine Fuel Inlet Temperature
25.	Fuel Pump Shaft Speed
26.	Fuel Pump Acceleration
27.	Fuel Pump Discharge Pressure
28.	Fuel Pump Discharge Temperature

Table 3-3. Preliminary Performance and Redline Flight Instrumentation List for the STBE (Continued)

#	MEASUREMENT
29.	Engine Fuel Flowrate
30.	MFV Inlet Pressure
31.	MFV Inlet Temperature
32.	MFV Position
33.	MFV Discharge Pressure
34.	MFV Discharge Temperature
35.	MCC Fuel Injector Pressure
36.	MCC Fuel Injector Temperature
37.	MCC Coolant Inlet Pressure
38.	MCC Coolant Inlet Temperature
39.	MCC Coolant Discharge Pressure
40.	MCC Coolant Discharge Temperature
41.	Nozzle Coolant Inlet Pressure
42.	Nozzle Coolant Inlet Temperature
43.	Nozzle Coolant Discharge Pressure
44.	Nozzle Coolant Discharge Temperature
45.	GG H_2 Injector Pressure
46.	GG H_2 Injector Temperature
47.	MCC Chamber Pressure
48.	MCC Chamber Temperature
49.	GG Chamber Pressure
50.	GG Chamber Temperature
51.	Fuel Turbine Inlet Pressure
52.	Fuel Turbine Inlet Temperature
53.	Fuel Turbine Discharge Pressure
54.	Fuel Turbine Discharge Temperature
55.	LOX Tubine Inlet Pressure
56.	LOX Turbine Inlet Temperature
57.	LOX Turbine Discharge Pressure
58.	LOX Turbine Discharge Temperature
59.	Nozzle Inlet Turbine Gas Pressure
60.	Nozzle Inlet Turbine Gas Temperature
61.	He GOX Outlet Temperature

3.2.3 Engine Performance and Throttling Characteristics

Engine design and operating parameters at Emergency Power Level (EPL) are presented in the Engine Balance Printout (Table 3-4). Engine performance over the throttling range is presented in Table 3-5.

The nozzle exit area ratio and exit contour was selected for 6 psia exit pressure at the design chamber pressure and without flow separation at minimum power level for sea level testing.

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Table 3-4. Engine Balance Printout

LOX/RP-1 LRB EMERG .POWER LEVEL (FN=GDCRP3) (CODE=GR RF)

ENGINE DESCRIPTION

ENGINE DESCRIPTION	(UNITS)			
TYPE		BELL		
TURBINE DRIVE CYCLE		GAS GENERATOR		
PROPELLANTS		O2/RP-1		
TURBINE ARRANGEMENT		SERIES		
THRUST				
VACUUM	(LBS)	791400.00(ENGINE)	779239.81(T/C)	12160.19(SEC)
SEA LEVEL	(LBS)	673231.50		
MIXTURE RATIO	(NONE)	2.530(ENGINE)	2.785(T/C)	
COOLANT BYPASS - ACTUAL	(PERCENT)	87.79		
COOLANT JACKET BYPASS	(PERCENT)	00.00		
DELIVERED SPECIFIC IMPULSE				
VACUUM	(SEC)	321.92(ENGINE)	330.83(T/C)	118.04(SEC)
SEA LEVEL	(SEC)	273.85		
		OXIDIZER	FUEL	
ENGINE INLET PROPELLANT PRESSURE	(PSIA)	85.00	45.00	
INLET TEMPERATURE	(DEG R)	162.70	520.00	
PUMP INLET NPSH	(FT)	97.04	129.76	
PROPELLANT FLOWRATE	(LBS/SEC)	1701.90503	690.44077(ENGINE)	
		1734.75251	620.63382(T/C)	
PROPELLANT BULK DENSITY	(LB/FT**3)	83.556(ENGINE)		

COMBUSTOR AND NOZZLE DESCRIPTION

COMBUSTOR AND NOZZLE DESCRIPTION	(UNITS)		
CHAMBER PRESSURE	(PSIA)	1402.50	
PRIMARY AREA RATIO	(AE/AT)	26.21	
OVERALL AREA RATIO	(AE/AT)	27.25	
NOZZLE PERCENT LENGTH	(PERCENT)	80.00	
FUEL INLET HEAT OF FORMATION	(KCAL/MOLE)	-4.31	
GIMBAL LENGTH	(IN)	28.67	
COMBUSTOR LENGTH	(IN)	20.45	
NOZZLE LENGTH	(IN)	119.21	
ENGINE LENGTH	(IN)	168.32	
ENGINE WEIGHT	(LB)	8108.44	
PRIMARY ENGINE EXIT DIAMETER	(IN)	99.24	
OVERALL ENGINE EXIT DIAMETER	(IN)	101.18	
CONTRACTION RATIO	(NONE)	2.70	
COMBUSTOR DIAMETER	(IN)	31.85	
THROAT AREA	(IN**2)	295.1055	
THROAT DIAMETER	(IN)	19.3840	
C SUB F	(NONE)	1.8827(T/C)	1.5389(SEC)
C* EFFICIENCY	(NONE)	.9600(T/C)	.9870(SEC)
		NOZZLE	COMBUSTOR
COOLANT FLOWRATE	(LBS/SEC)	272.41	348.22
COOLANT DELTA P	(PSID)	152.28	1522.79
COOLANT EXIT TEMPERATURE	(DEG R)	1082.84	824.47
HEAT INPUT	(BTU/SEC)	88604.55	51632.54
COOLING JACKET OUTLET PRESSURE	(PSIA)	1711.05	1711.05

GG GAS PROPERTIES

GG GAS PROPERTIES		FUEL	OXIDIZER
GAS TEMPERATURE	(DEG R)	1800.00	1622.52
GAS MIXTURE RATIO	(NONE)	.359	.359
GAS MOLECULAR WEIGHT	(GMS/GM-MOLE)	28.720	28.720
GAS PROCESS GAMMA	(NONE)	1.118	1.103
GAS CP	(BTU/LB-DEG R)	.649	.642
GAS FLOWRATE	(LB/SEC)	102.32302	103.02002
FUEL HEAT OF FORMATION	(KCAL/MOLE)	-5.76	
FUEL INLET TEMPERATURE	(DEG R)	520.00	
COMBUSTION PRESSURE	(PSIA)	1402.50	

Table 3-4. Engine Balance Printout (continued)

LOX/RP-1 LRB EMERG .POWER LEVEL (FN=GDCRP3) (CODE=GR RF)

TURBOPUMP DESCRIPTION	(UNITS)	MAIN PUMP OXIDIZER	MAIN PUMP FUEL	KICK PUMP OXIDIZER	KICK PUMP FUEL	BOOST PUMP OXIDIZER	BOOST PUMP FUEL
PUMP							
# OF STAGES	(NONE)	1.00	1.00	.00	.00		
HORSEPOWER	(HP)	16779.311	16664.538	.000	.000	.000	.000
ROTATING SPEED	(RPM)	7867.0	14567.2	.0	.0	.00	.00
EFFICIENCY	(NONE)	.81459	.78504	.00000	.00000	.80000	.00000
INLET PRESSURE	(PSIA)	65.00	45.00	.00	.00	65.00	45.00
OUTLET PRESSURE	(PSIA)	2175.03	3628.29	.00	.00	.00	.00
FLOWRATE	(LB/SEC)	1781.96583	696.44072	.00000	.00000	.00000	.00000
	(GPM)	11104.12	[illegible]	.00	.00	.00	.00
INDUCER							
TIP DIAMETER	(IN)	11.48	7.71		.00		
TIP SPEED	(FT/SEC)	393.74	490.48		.00		
INLET FLOW VELOCITY	(FT/SEC)	39.35	49.01		.00		
FLOW COEFFICIENT	(NONE)	.100	.100	.000	.000	.208	.000
IMPELLER							
TIP DIAMETER	(IN)	15.68	12.63	.00	.00	.000	.000
TIP SPEED	(FT/SEC)	538.80	803.26	.00	.00	.000	.000
TIP WIDTH	(IN)	1.330	.738	.000	.000		
HEAD COEFFICIENT	(NONE)	.473	.515	.000	.000	.207	.200
BLADE ANGLE	(DEG)	25.000	25.000	25.000	25.000		
HEAD RISE (OVERALL)	(FT)	4768.89	10332.27	.00	.00	.00	.00
STAGE SPECIFIC SPEED	(RPM*GPM**.5/FT**.75)	1570.26	1124.80	.00	.00		
BOOST PUMP							
MINIMUM DELTA P	(PSI)	-24.20	-8.16				
HUB/TIP RATIO	(NONE)			.000	.000		

TURBINE		OXIDIZER	FUEL	HYDROGEN
TYPE	(NONE)	PRESSURE	PRESSURE	PRESSURE
# OF STAGES	(NONE)	2.00	2.00	.00
HORSEPOWER	(HP)	16779.31	16664.54	.00
FLOWRATE	(LB/SEC)	103.02002	102.32302	.00000
EFFICIENCY	(NONE)	.77833	.73889	.00000
PRESSURE RATIO	(NONE)	5.201	3.882	.000
ADMISSION	(FRACTION)	1.000	1.000	.000
VELOCITY RATIO	(NONE)	.341	.272	.000
PITCH DIAMETER	(IN)	27.063	11.941	.000
1ST STG BLADE HEIGHT	(IN)	1.498	.818	.000
2ND STG BLADE HEIGHT	(IN)	3.686	1.785	.000
PITCHLINE VELOCITY	(FT/SEC)	929.70	759.68	.00
INLET HUB/TIP RATIO	(NONE)	.895	.872	.000
EXIT HUB/TIP RATIO	(NONE)	.760	.740	.000
TIP SPEED	(FT/SEC)	1056.33	873.12	.00
BEARING DN*E-6	(MM*RPM)	.607	.981	.000
ANNULUS AREA*N**2*E-10	((IN*RPM)**2)	1.940	1.421	.000
INLET PRESSURE	(PSIA)	361.05	1402.32	.00
OUTLET PRESSURE	(PSIA)	69.42	361.23	69.42
INLET TEMPERATURE	(DEG R)	1622.52	1800.00	1800.00
OUTLET TEMPERATURE	(DEG R)	1443.20	1622.52	.00
1ST BLADE TEMPERATURE	(DEG R)	1544.71	1730.96	.00
2ND BLADE TEMPERATURE	(DEG R)	1455.03	1642.21	.00

Table 3-5. Engine Performance vs. Power Level

	THRUST, kLB Vac	SL	P_c, psia	I_{vac}	I_{SL}, sec
EPL	791.4	673.2	1402	321.9	273.9
NPL	719.5	602.3	1286	322.4	269.5
MPL	539.6	421.4	964	323.7	252.8

3.2.4 Engine Weight Summary

A preliminary weight summary is presented in Table 3-6 by component grouping for the LOX/RP-1 LRB engine. The engine design operating conditions and pertinent configuration characteristics are noted. The total engine dry weight is 8108 pounds without the engine accessories noted. The necessity of these accessories should be considered by the vehicle contractor, but will be subject to weight changes depending on vehicle requirements.

3.2.5 Start And Shutdown

The engine start and shutdown for the LOX/RP-1 LRB will be similar to that used for previous LOX/RP-1 engines, such as the ATLAS, RS-27 and F-1. All previous LOX/RP-1 engines used a spin-start with the exception of the F-1 which used a tank head start. Both types of startup were reviewed for the LRB and a final selection will need be made based on cost, reliability, and final vehicle requirements. The two types of start to be considered for the LOX/RP-1 LRB are discussed with advantages/disadvantages noted.

Tank Head Start. The F-1 was a LOX/RP engine using a tank head start, the same kind of start was evaluated for the LRB LOX/RP engine. Control of the GG temperature during the initial part of the start is difficult and will require a modulating GG oxidizer valve. In addition, a tank head start on the LRB may be more difficult than on the F-1 because of the mainstage design pressure levels. The F-1 gas generator pressure at mainstage was about 950 psia. Therefore, if initial GG combustion were started at the minimum pump inlet pressure (about 45 psia), a GG pressure and initial turbine torque of about 5

Table 3-6. LOX/RP-1 Pump Fed Engine Weight Summary

LOX/RP-1 LRB EMERG .POWER LEVEL (FN=GDCRP3)

O2/RP-1 ENGINE WEIGHT SUMMARY

T/C THRUST	(KLB)	779.
CHAMBER PRESSURE	(PSIA)	1402.50
ATTACHED AREA RATIO	(NONE)	5.0
FIXED AREA RATIO	(NONE)	27.2
EXTENDIBLE AREA RATIO	(NONE)	27.2
T/C THRUST COEFFICIENT	(NONE)	1.8827
COMB. CHARACTERISTIC LENGTH	(IN)	39.17
CONTRACTION RATIO	(NONE)	2.7
ENGINE MIXTURE RATIO	(NONE)	2.53
NOZZLE PERCENT LENGTH	(PERCENT)	80.00
GIMBAL ANGLE	(DEG)	11.

TURBOMACHINERY :			
FUEL TURBOPUMP	647.5		
OXID MAIN TURBOPUMP	940.9		
SUB-TOTAL		1588.4	
GAS GENERATOR :		154.9	
EXHAUST GAS MANIFOLD :		105.8	
THRUST CHAMBER :			
GIMBAL BEARING	159.2		
INJECTOR	1377.0		
COMBUSTOR	1301.3		
FIXED NOZZLE	1349.1		
SUB-TOTAL		4186.7	
VALVES AND CONTROLS :			
PROPELLANT VALVES	356.3		
CONTROL VALVES	66.4		
HARNESS AND SENSORS	187.7		
PNEUMATIC CONTROLS	150.5		
HYDRAULIC CONTROLS	60.4		
ATTACH PARTS	227.5		
SUB-TOTAL		1048.7	
ENGINE SYSTEMS :			
PROPELLANT DUCTS	651.2		
ATTACH PARTS	74.6		
DRAIN LINES	61.9		
I.F. OXID. BLEED LINE	9.3		
I.F. FUEL BLEED LINE	25.4		
I.F. HYDRAULIC LINES	14.1		
I.F. GN2/HE LINES	34.4		
IGNITION LINES AND IGNI,RS	47.6		
PRESSURIZATION SYSTEM	105.6		
SUB-TOTAL		1024.0	
ENGINE ACCESSORIES :			
FIXED NOZZLE THERMAL PROTECTION	97.5		
CONTROLLER AND MOUNT	85.0		
POGO SYSTEM	142.9		
SUB-TOTAL		325.4	
TOTAL ENGINE DRY WEIGHT W/O ACCESSORIES :			8108.4
TOTAL ENGINE DRY WEIGHT WITH ACCESSORIES :			8433.9

percent of mainstage would be produced. This would result in only 3 percent of mainstage pressure and torque. This is essentially a deeper throttling of the GG and also a lower effective torque for initial engine bootstrap. Both of these characteristics will cause additional difficulty in system control and a reduced initial turbine speed buildup. Any variations in turbopump drag would cause larger run-to-run changes in pump buildup rates.

The GG LOX and fuel valves open at 0.1 and 0 sec., respectively. Both valves take 0.2 sec. to reach full open. Flow to the gas generator powers the turbopumps. The main LOX valve is set to open at engine start. The valves on the main and kick pump loops on the fuel side start to open at time 0, and both are set to reach full open in 0.3 sec. For model simplification, fuel was not allowed to flow into the chamber until both the chamber and nozzle are primed.

Fuel starts to flow into the gas generator at time 0. The GG LOX valve starts to open at 0.1 seconds. With the LOX side priming volume, LOX does not flow into the GG until 0.4 seconds. At about this time the GG primes and GG chamber pressure begins to rise. The back pressure increase causes a drop in GG fuel flow. This high LOX flow and reduced fuel flow would cause the GG temperature to spike up to 2800 R at 0.9 seconds. Therefore the LOX flow to the GG will be throttled between about 0.7 and 1.1 seconds to eliminate any temperature spike.

The main chamber primes at about 0.8 seconds. With engine start time defined as the time that the engine reaches 90 percent chamber pressure, this engine starts in about 1.3 seconds. Additional throttling of the GG LOX valve to prevent the temperature overshoot will increase the engine start time, but probably will be less than 1.7 seconds.

<u>Turbine Spin Start</u>. A turbine spin system would substantially reduce run-to-run start variations. The spin power is relatively repeatable and is large enough that variations in turbopump drag will have minor effects. A start with a spin system will be less sensitive and probably require less start transient development time.

A spin start for the LRB LOX/RP engine would be similar to an Atlas and RS-27 engine start. The MA-5 and RS-27 system uses pressurized propellant start tanks while the MA-3 engine uses a solid propellant spinner. Either of these methods, in addition to a pressurized helium spin bottle, could be used for the initial turbine power.

The basic sequence used is to open the main oxidizer valve first. At about the same time an igniter fuel valve is opened that allows RP-1 flow from a pressurized tank to a hypergol cartridge. When the hypergol burst diaphrams break, hypergol followed by RP-1 flows through the igniter fuel line to the main chamber causing main chamber ignition. This ignition is confirmed by burnthrough of a wire stretched across the chamber nozzle exit.

Once ignition has been confirmed, the engine goes into the spin start phase. On the MA-5 with pressurized start tanks, the GG valves are opened and combustion at about 300 psi is generated which spins the pumps up to near mainstage speeds. The main fuel valve is signalled open at about the same time as the GG valves. In about one second, the fuel fills the volumes to the main fuel injector, which results in main propellant ignition. The increase in system pressures due to main propellant ignition causes the pump discharge pressures to open check valves in the GG lines, resulting in system bootstrap and check valves shut off the start tank flows. The start time from signal to spin to mainstage is in the order of 1.2 seconds.

Using a solid spin or a helium bottle would be similar except that the GG propellant valves would not be opened until main propellant ignition. With a solid spinner, the grain burning duration has to be matched to terminate flow just after the main propellant ignition. With a helium spin system, a valve would be used to sequence the spin on and off at the proper times. The suggested LOX/RP-1 engine for the LRB uses a helium turbine-spin-start system. A typical start and cutoff sequence of events is shown in Table 3-7 with the propellant consumption during engine start noted. The propellant consumption is from engine start signal to mainstage operating level, and does not include any engine prechill consumption.

Table 3-7. LOX/RP-1 LRB Engine Start and Cutoff Sequence

START TIME (SECS)	EVENT
0.0	Open Main Oxidizer Valve
0.2	Open Igniter Fuel Valve
1.0	Detect Main Chamber Ignition
1.2	1. Signal Spin System Start 2. Ramp Main Fuel Valve Open
2.1	Fuel Primes System to Main Chamber Generating Main Chamber Prime and Engine Boost stage
2.2	Open GG Valves
2.4	Close Spin System Valve
2.6	Engine Reaches Full Thrust

CUTOFF TIME (SECS)	EVENT
0.	Close GG Valves (0.1 to 0.2 sec)
0.	Ramp Main LOX Valve Closed (assume 0.5 sec travel)
0.1	Ramp Main Fuel Valve Closed (assume 0.5 sec travel)

Estimated Propellant Usage During Start/Shutdown

	START	CUTOFF
LOX	2100 lb	500 lb
RP-1	470 lb	300 lb

3.2.6 Thrust Vector Control Actuation Torque and Power Requirements

The various elements of the total torque and power requirements are listed in Table 3-8 along with the major assumptions and conditions. It may be possible to reduce the required torque and power by reducing the allowed thrust vector offset, R_o, which has a strong influence. In addition, the gimbal friction may be lowered utilizing special advanced low friction dry lubricants. The combined effect may substantially reduce the power required.

3.2.7 Nozzle Exit Gas Condition Analysis

The gas condition near the wall at the edge of the boundary layer was determined analytically to aid others in calculating the heat transfer to the base of the vehicle due to radiation and convection. The parameters calculated and the corresponding resulting values are given in Table 3-9.

Table 3-8. LRB TVC Torque Breakdown for Head End Gimbal, LOX/RP-1 Pump Fed

Name of Contribution	In-lb of Torque	Percent of Total
Moment of Inertia	77,914 in.lb	12 %
Flex Line Stiffness		
LOX LINE	40,904 in.lb	6 %
FUEL LINE	26,954 in.lb	4 %
Thrust Vector Offset	168,308 in.lb	26 %
Gimbal Friction	222,166 in.lb	34 %
Gravity and Accel. at 3 g	121,810 in.lb	19 %
Total =	658,056 in.lb	100 %
Lever Arm =	32 in	
Force Reqd.=	20564.2 lb	

Horse Power at 10 Deg/sec = 17.31 H.P.(input)

(0.25 in)
Basis:

Engine Thrust =	673231 lb
Engine Mass =	8553 lbm
Lever Arm =	32 in
CG Distance =	43 in
Frictn.Coef.=	0.06
Thrust Offset	0.25 in

Requirements:

Angular Excursion =	+ or - 6 Deg
Angular Slewing Rate =	10 Deg/sec
Angular Acceleration =	1 radian/sec.squared
Propellant Line Pres.=	65 & 45 psia
Nomin.Fuel Line Diam.=	10 in
Nomin.Oxid.Line Diam.=	13 in

Table 3-9. Analysis Results for Nozzle of RAO Optimum Contour (80%)

Parameter at Nozzle Exit (LIP)	$\epsilon = 27$ LOX/RP-1
Chamber Pressure (psia)	1286
Wall Static Pressure (psia)	8.853
Wall Angle (deg)	8.396
Mach Number	3.41088
Gas Specific Heat Ratio (γ)	1.16743
Static Temperature (°R)	4135.4
Displacement Thickness (in.)	0.0911964
Momentum Thickness (in.)	0.309353
Boundary Layer Thickness (in.)	2.5111
Enthalpy Thickness (in.)	0.4919
Mass Flow in Boundary Layer (Lbm/sec)	278.8
Subsonic Mass Flow in B.L. (Lbm/sec)	0.0675

3.3 POGO & STABILITY ANALYSIS

Pogo is to a launch vehicle what flutter is to an airplane; a potentially destructive unstable vibration. It is a low frequency vibration occurring sometime during the boost phase, gradually growing out of the background noise (atmospheric buffeting etc.), leveling off and then gradually decaying back into the noise. Because it is a transient instability, its maximum amplitude is not predictable. Figure 3-7 shows the envelopes of two accelermotors located on the aft end of the Saturn V during the second unmanned flight. During the Pogo "football" the degree of instability can be inferred from the divergence rate in terms of damping factor. The maximum instability in this case was about -0.05% of critical damping.

3.3.1 POGO Suppressor Design Philosophy

High amplitude vibration can cause structural failure. The problem involves the vehicle structure, the column of propellant in the feedline and the engine. The structure supports the engine and the engine supports the propellant. As the engine moves forward, pressure at the engine inlet

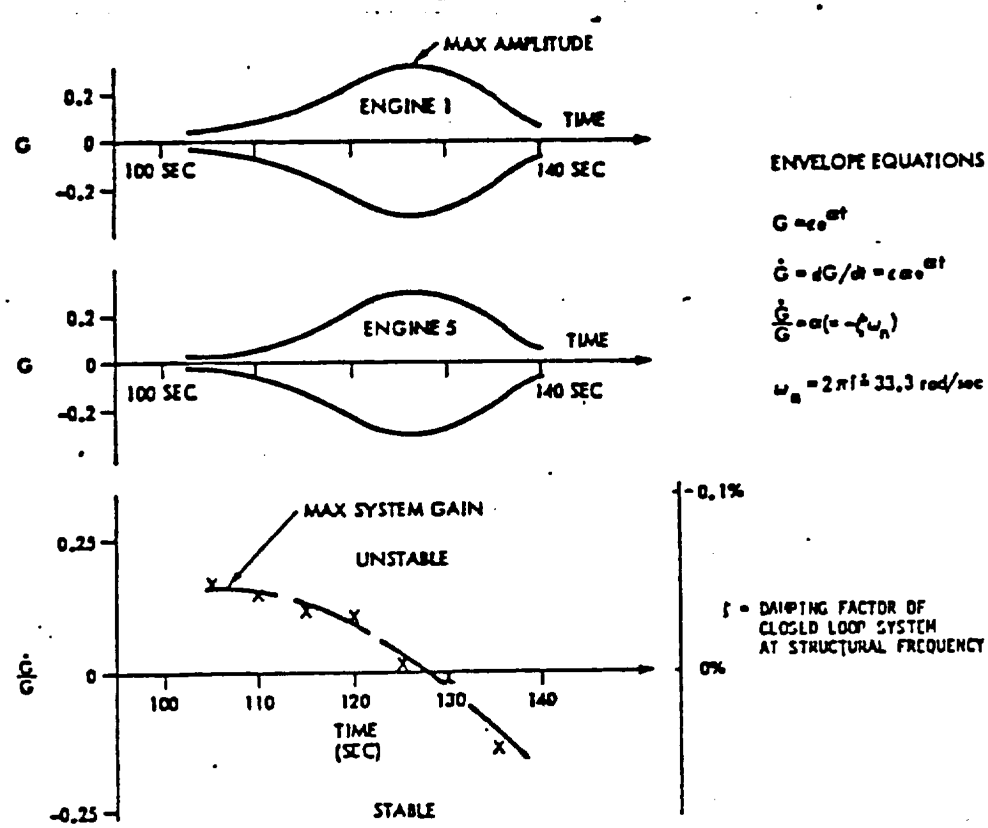

Figure 3-7. Envelope Equations of Two Accelerometers

increases producing a force acting upward on the propellant and downward on the engine and structure. This increased pressure causes additional flow into the engine which is burned in the main thrust chamber producing an additional upward force on the structure. If the upward force from thrust is greater than the downward force at the engine inlet the engine acts like negative structural damping with potential for Pogo. The instability usually also involves tuning of a feed system resonance with a structural resonance. Tuning and detuning occurs naturally during a flight as propellant in the tanks is consumed. Figure 3-8 is a block diagram showing coupling of the significant subsystems. With the structure and feed systems tuned to the same frequencies, the forward loop has maximum gain and zero phase shift. With damping associated with the structural resonance is increased resulting in greater stability. With positive feedback, damping is decreased with potential for instability.

Specific requirements for suppressor are dependent on the vehicle structure and feed system. The suppressor should be located downstream of a reasonably

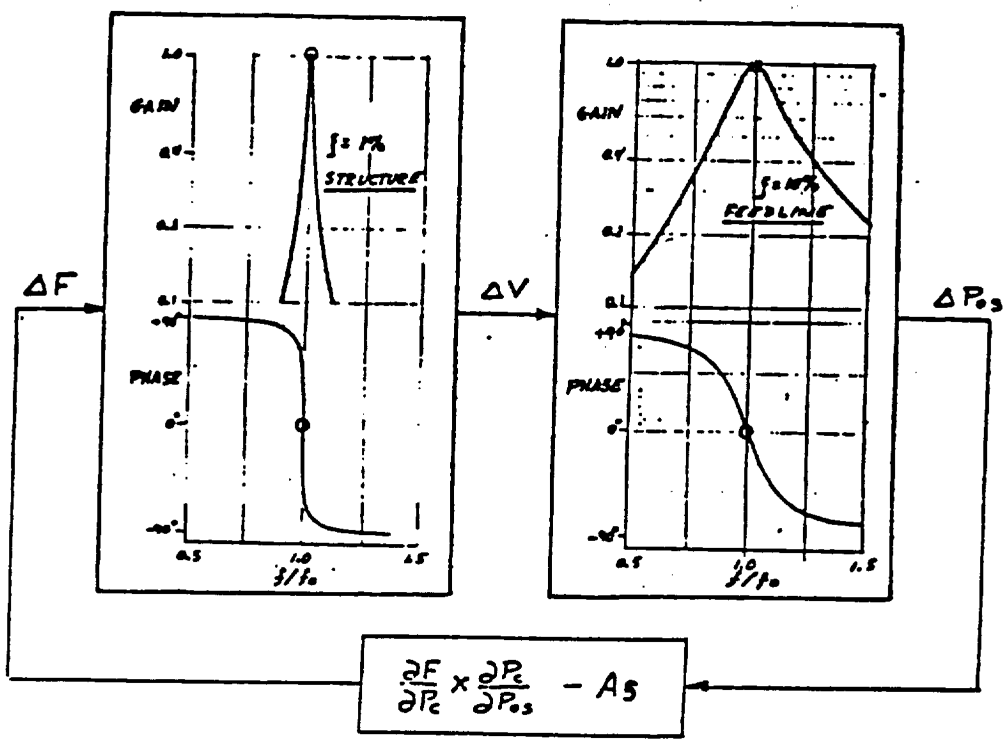

Figure 3-8. POGO Block Diagram

high resistance, in this case at the pump inducer while in the SSME it is located between the low pressure oxidizer pump and the high pressure oxidizer pump. Location in this region provides low enough gas pressure for acceptable compliance with reasonable suppressor volume.

3.3.2 Suppressor Configuration

The recommended suppressor schematic is similar to the SSME system in function. The proposed configuration for the LRB is shown in Figure 3-9. The configuration is shown as an annular volume surrounding the pump inlet. About 1-1/4 cubic ft. of gas is shown with sufficient liquid to allow some interface motion without gas injection into the impeller. An initial helium precharge allows the suppressor to be active at lift off. As mainstage operation is reached, a valve is activated and the ullage is supplied with GOX through a choked orifice from the heat exchanger. Liquid level is controlled by holes drilled in an overflow pipe. The overflow (gaseous and liquid oxygen) is ducted 10-15 ft. upstream of the engine interface and recirculated back

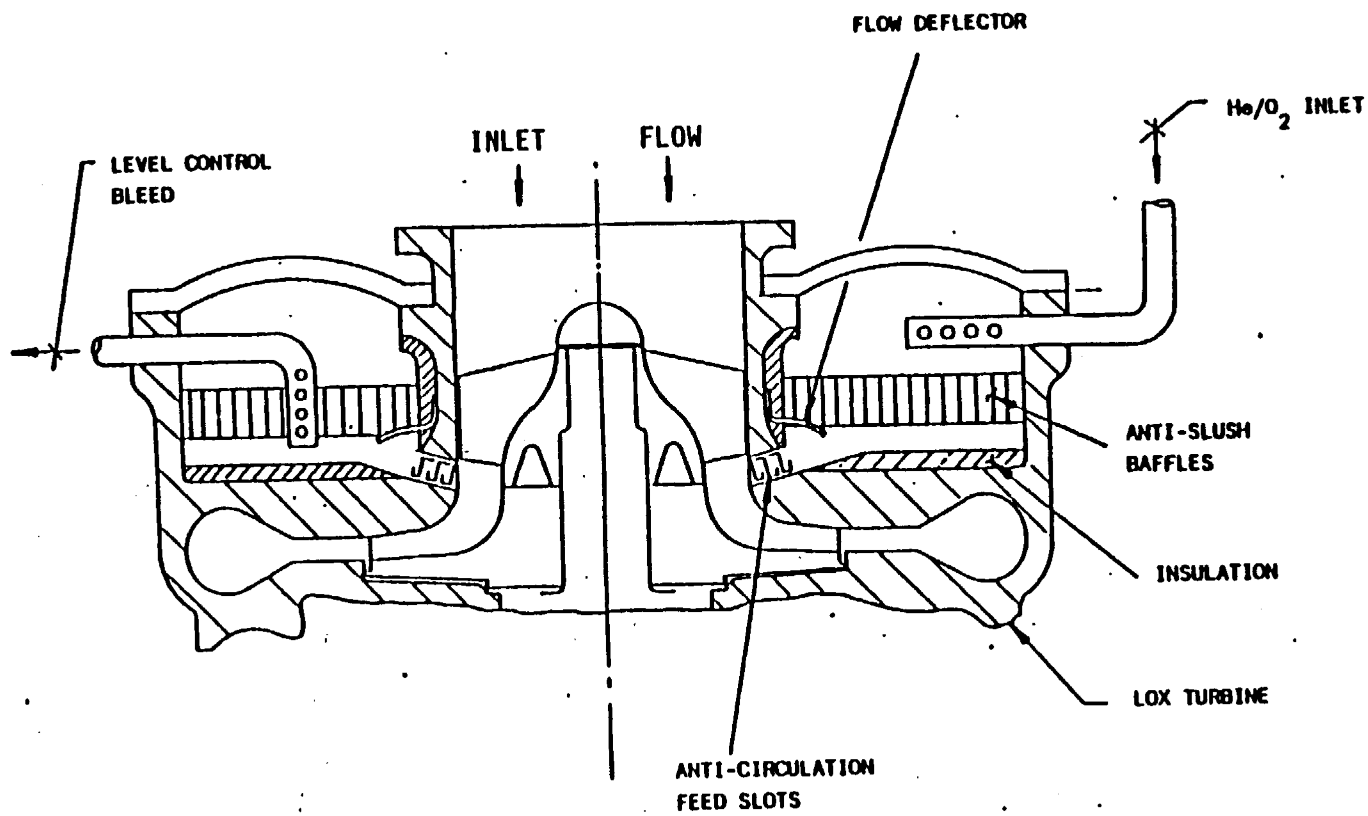

Figure 3-9. POGO Suppression System

into the main flow stream. Baffles in the annular chamber inhibit lateral sloshing as well as circulation thereby precluding collapse of the ullage.

A critical design consideration is the connection between the suppressor and the main flow stream. Testing of the SSME suppressor led to shaped slits. The normal operation vorticies are generated in the slots which minimize circulatory flow with suppressor and adds some resistance to through flow. The total flow area is large enough (about 1/2 the flow area of the main duct) to maintain small fluid inertia so that the suppressor is effective over a wide frequency range (2-40 Hz). Preliminary analysis shows that in the LRB, a good location for the communicating slots will be slightly upstream of the inducer trailing edge. The design of the inducer must be evaluated in conjunction with the suppressor to avoid significant penalties to the turbomachinery.

While the SSME type suppressor is quite effective in POGO suppression, the level control system requires its use at a point in the system where the pressure is very low. Placement downstream of the first pumping stage provides this condition and the desirable feature of significant resistance in the main flow stream between engine inlet and the suppressor tap off point.

The design of the suppressor must be considered early in the pump design to avoid significant impacts to either component. A fluid interface designed upstream of the inducer trailing edge will have least impact on the critical shroud-casing recirculation flow and will allow evaluation of the suppressor-pump interface during early testing without jeopardizing the fuel pump tests.

Gaseous oxygen will be supplied from the engine heat exchanger through a choked orifice. A heat exchanger discharge pressure of about 1500 psi will therefore be required. A helium source pressure of 250 psi or higher for engine start and cutoff is required. This provides Pogo suppression at lift off and it prevents large surges at engine cutoff.

3.4 FAILURE MODE ANALYSIS AND RELIABILITY ESTIMATE

A quantitative reliability analysis of this engine has not been performed, but reliability histories of pump fed engines of similar size and requirements are available. Therefore based on a cursory comparison with those engines that have an established reliability record, the requirement of 0.99 % Reliability at 90% Confidence Level appears attainable.

A preliminary Failure Mode and Effects Analysis (FMEA) is presented. Because of the preliminary nature of this study, only major components and assemblies have been addressed. Criticality codes, as defined at the end of the FMEA, have been assigned to each failure mode.

3.4.1 Preliminary Failure Mode and Effects Analysis - Pump Fed LRB

Specific criteria and groundrules are listed below with criticality rankings listed in Table 3-10.

3.4.2 Criticality Definitions

Criticality 1

1. Hot gas leakage is assumed to always result in structural/functional damage to at least one engine.
2. Hot gas mixing with LOX is a potential fire/explosion hazard.
3. Oxidizer rich cutoffs always offer the potential for structural damage.
4. Structural failure of rotating machinery or rupture of pressure containment boundaries can both propagate to destruction of one or more engines, followed by loss of engine or vehicle life.
5. Spark generation in a LOX environment, such as rubbing/fretting of parts in oxidizer pumps or valving, will escalate to a fire/explosion.

Criticality 2

1. Leakage of propellants during start of mainstage is considered as being detectable by hazardous gas monitors or other instrumentation to permit safe engine shutdown. The worst possible scenario of potential mission loss, however, is assigned for conservativeness.
2. Failures precipitating safe engine shutdown. The vehicle is capable of achieving mission success with one engine not operating; however, it is presumed that launch abort, followed by safe shutdown, will be commanded if one engine is not operating prior to liftoff.

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Turbine Spinner	Fails to spin start turbines	Delays engine start	Launch delay. Possible engine-out launch	2
LOX Pump	Oxidizer seal leakage, Internal	Excess leakage will back up into gear box where intermixing with fuel will cause detonation	Mission Loss	1
	Overspeed, cavitation	High vibration and possible piece failure	Engine shutdown	2
	Piece part failure	Pump imbalance with rubbing causing possible fire	Engine loss	2
	Performance outside specified limits. Overpressure	Engine performance variation. Structural damage or failure resulting in LOX leakage.	Possible mission loss	1
LOX Turbine	Hot gas leakage	LOX/hot gas mix or hot gas impingement. Engine loss	Possible mission loss	1
	Pump imbalance	Possible piece part failure	Engine loss	2

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Fuel Pump	Fuel output press. drops below limits	Engine shut down and system switches to engine-out mode of operation.	Single-engine out results in longer systems burn time	2
	Fuel seal leakage	Gross leakage may result in gear case flooding, performance decay, low P_c and automatic shutdown.	Single-engine out results in longer systems burn time	2
Fuel Turbine	External hot gas leakage	Turbine performance loss results in fuel discharge, pressure decay, auto-cutoff. Fire hazard created. Prob-engine damage.	Launch abort.	1
	Turbine seal leak	None. Redundant seals protect against hot-gas leakage into gearbox	None	3
Main Oxidizer Valve	Fails to open	Engine fails to start. Safe shutdown	Launch delay	2
	Fails to close	None. Shutdown redundancy provided by stage oxidizer prevalve.	None.	3
	Closes prematurely	Automatic shutdown. System switches to engine-out mode of operation.	Single engine-out results in longer systems burn time	2

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Main Fuel Valve	Fails to open	Prevents engine start.	Launch delay	3
	Opens prematurely or internal leak	Excess fuel accumulates in combustion chamber. Possible hard start.	None	3
	Opens too rapidly	Water hammer causing system vibration and engine damage. Engine shutdown.	Launch abort	2
	Opens partially or restricted fuel flow.	Reduced fuel flow to MCC and nozzle. Mixture ratio upset and reduced MCC and nozzle coolant flow may cause burn through. Automatic cutoff.	Launch abort at start. Switch to engine-out mode of operation.	2
	Fails to close or leaks internally	Fuel flow terminated by prevalve. Fuel rich cutoff occurs. Oxidizer valve closing terminates combustion	None.	3
	External leak	Will require correction if detected	Launch delay at start	2
		Fire hazard during M/S partially mitigated by inert atmosphere in boattail .	Switch to engine-out mode if detected.	2
			Mission loss if undetected	1

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Gas Generator	Hot gas overtemp.	Automatic engine cutoff	Engine-out operation	2
Gas Generator Oxidizer Valve	Fails to open	Delayed start	Launch delay or engine-out launch	2
	Erratic response to modulation commands	Erratic LOX flow to GG. Thrust control and/or engine throttling capability lost. Engine shutdown.	Single engine-out results in longer burn time	2
Gas Generator Fuel Valve	Fails to open	Delayed start	Launch delay or engine-out launch	2
	Erratic response to modulation commands	Erratic fuel flow to GG. Loss of Mixture ration control, engine performance decay. Engine shutdown	Single engine-out results in longer systems burn time	2
Gas Generator Igniters	Fail to ignite	None due to redundancy	None	3

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Heat Exchanger Valve (HEV)	Fails to open	No LOX flow to Heat Exchanger. Controller initiates engine shutdown.	Launch delay	1
	Fails to close or leaks internally	LOX flow continues to Hex and POGO accumulator	None	3
	Restricted LOX flow or shutoff	Pressurant loss to POGO accumulator and to vehicle LOX tank	Engine shutdown	2
Heat Exchanger	Coil leakage	Cold LOX and fuel rich hot turbine exhaust gas mix. Hot spot on Hex. Engine shutdown	Possible vehicle / mission loss.	1
	Plugging or flow restriction of LOX	Over heating or reduced LOX flow may cause rupture. No pressurant flow to LOX tank & POGO accumulator	Engine-out operation	2
TC hypergol igniter	Diaphragm fails to rupture	Thrust chamber ignition will not be established and automatic shutdown will follow	Launch delay Possible single engine-out launch	2

0240m

Table 3-10. Preliminary Failure Mode and Effects Analysis Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Main Combustion Chamber	External fuel leak MCC to nozzle joint	Possible fire and loss of engine or adjacent equipment damage	Possible loss of vehicle or mission. Engine-out capability may save mission	1
	Internal fuel leakage to (hot side)	Reduced MCC cooling. Possible local MCC overheating	Possible loss of vehicle or mission. Engine-out capability possible	1
Nozzle	External fuel leak	Possible fire and loss of engine or adjacent equipment damage	Possible loss of vehicle or mission. Engine-out capability may save mission	1
	Internal fuel leakage to (hot side)	Reduced nozzle cooling. Possible local overheating	None	3
POGO system	Fails to operate or maintain operability	Automatic shutdown will occur	Launch delay at start	2
			Engine-out operation after launch	2

Criticality 3

1. External leakage of propellants during preconditioning is assumed to be detected by ambient hazardous gas monitors, which will be cause for launch abort.

2. All others.

3.5 PROGRAMMATICS (PUMP FED ENGINES)

Engine development plans are presented for two pump fed LRB engine configurations. These are (1) pump fed with Lox/RP-1 propellants; and (2) pump fed with LOX/H_2 propellants. Since the development schedules for these two pump fed configurations are the same their program descriptions are combined in the following discussion. Hardware and cost estimates are presented in separate transmittals for each of the programs.

3.5.1 Development Schedule

Both LOX/RP-1 and LOX/H_2 were considered as propellants for the LRB pump fed engines. THe overall development program schedule is generally the same for these engines, and is shown in Figure 3-10. The 63 months (5 1/4 years) development program is designed to support a first vehicle launch in the third quarter of 1995 and therefore would benefit from a Phase B effort and a modest technology program in terms of reduced risk.

First, a benefit of the Phase B design effort would be to allow early long lead procurement of casting tooling for some of the major components such as the pump housings. Secondly, significant benefits in terms of reduced risk would be derived from a technology program that is started in parallel with the Phase B design effort and completed in time to provide data for the development program design phase. The specific technology that would provide the most benefit is in the area of injector design for stability and turbo pump bearings and seals and rotating elements. The details of this technology program are described in a latter section. Thirdly, as indicated in Figure 3-10, engine test facilities are required by the fourth quarter of 1992. These test facilities are assumed to be provided by the government or the vehicle contractor. Formal Pre-Flight Rating Test (PFRT) are planned prior to

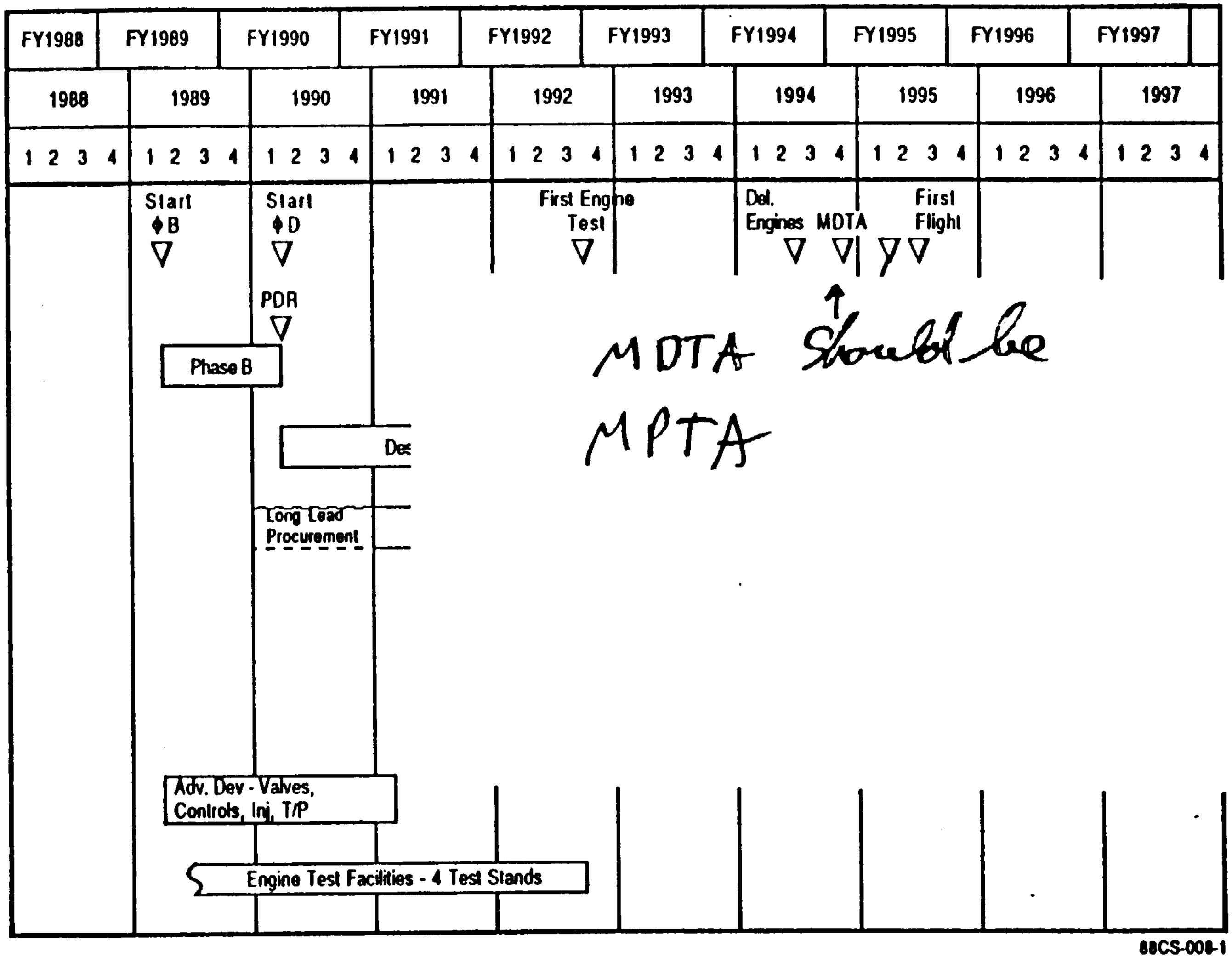

Figure 3-10. LRB Pump Fed Engine Development Program
(LOX/RP-1 or LOX/H_2 Propellants)

the first flight and Flight Rating Tests (FRT) to certify readiness for production and full operational status which are planned after the first flight.

3.5.2 Development Plan

The engine test plan has been developed, ie, in terms of number of tests and hardware, on the basis that the engine design provides robustness and design margins were applied to the emergency power level (EPL) operating conditions resulting in higher margins at the nominal power level (NPL). A design team including engineering, manufacturing, procurement, operations, reliability, producibility, quality and maintainability functions will be fully integrated into the design and procurement process to assure a cost effective, low risk engine. Lessons learned from numerous previous large engine development programs will be applied. These include:

1. Component level testing will be conducted in an engine simulating environment to the maximum extent possible.

2. Extensive limits testing will be conducted at both the component and engine level.

3. Overstress testing will be conducted on a majority of the test units.

3.5.3 Program Approach

Initial effort will consist of analyses and design, making extensive use of Rocketdyne's well anchored analytical tools. Detail shop drawings will be produced and reviewed during the Critical Design Review (CDR) scheduled 24 months after program start. In parallel with the design effort procurement of long lead casting today will be initiated. It is planned to select the casting supplies early in the program and include them, as part of the design team for those parts to be produced by the casting process. Component testing will be initiated as soon as components are available. The primary objective of component testing is to drive out design problems and evaluate potential failure modes identified in the Failure Modes and Effects Analysis (FMEA).

Component hotfire testing of the thrust chamber assembly, gas generator and turbopumps will include limits and overstress testing.

The engine test program is designed to drive out random failures and wearout problems. Engine testing will be initiated as soon as possible. Experience has shown that the actual engine operating environment is the best medium in which to drive out problems. The initial engines will be heavily instrumented to assure that problems can be analyzed and solved in an expeditions manner. Limits and overstress testing will be introduced at the engine level as soon as possible to verify the design margins. Valid component and engine test data will be used to verify the analytical tools used for design and simulation.

3.5.4 Component Test Program

The component test program in terms of schedule, hardware and number of tests is the same for the pump fed, LOX/RP-1 and LOX/H_2 configurations. The component test plan is presented in Figure 3-11. The following is a discussion of each of the component test programs.

Control Components. The control components include the main LOX valve, main fuel valve, gas generator LOX and fuel valves, control and condition monitoring instrumentation; check valves, puenumatic console including solenoid valves, electrical harnesses, a controller package and spark exciter boxes for the LOX/H_2 engine gas generator and main chamber igniters. Controls component testing will be conducted at Rocketdyne's existing laboratory test facilities. As indicated in Figure 3-11, three sets of each of the items described will be procured for laboratory testing. The planned testing is shown in Table 3-11.

Gas Generator Assembly. The gas generator assembly consisting of a combustor body, injector/dome assembly, propellant valves and ignition system is planned to be hot fire tested at a government or vehicle contractor facility. Ignition for the LOX/H_2 engine gas generators will be provided by an augmented spark ignition (ASI) system. The gas generator for the LOX/RP-1 engine will utilize dual 28 volt pyrotechnic igniters. Figure 3-11 shows the

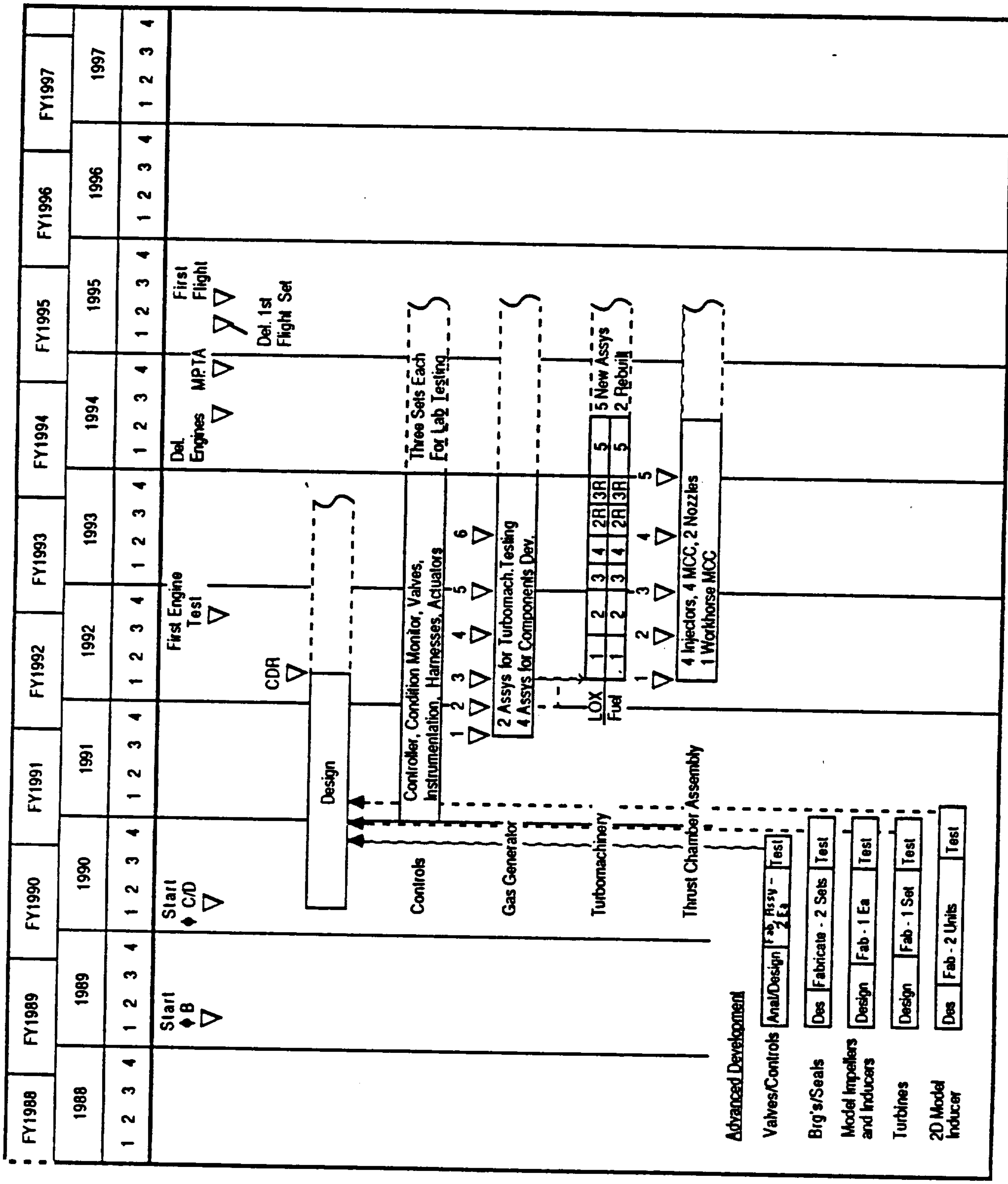

Figure 3-11. LRB Pump Fed Engine Component Development Program (LOX/RP-1 and LOX/H_2)

Table 3-11. LRB Controls Component Development Plan

	Laboratory Development Tests												
Component	Functional	Thermal Vacuum	Thermal Cycling	Vibrational	Acoustic	Pyro Shock	Acceleration	Humidity	Pressure	Leakage	EMF/EMC	Life	Burn-in
Main LOX and Fuel Valves, Igniter Fuel Valves Check Valves	X	X	X	X					X	X	X	X	
Controller Control and Condition Monitoring Instrumentation	X	X	X	X							X		
Pneumatic Control Console and Solenoid Valves	X	X		X				X	X	X	X	X	
Flight Instruments	X	X	X	X		X	X		X	X			X
Harnesses	X		X	X	X			X		X	X		
Spark Exciter Boxes (for LOX/H_2 Engine Only)	X		X	X	X			X		X	X		

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schedule and planned hardware for this test program. As indicated, 6 complete assemblies will be procured, assembled and tested during the development program. After sufficient testing has been accomplished on unit number 1 to understand the operations of the gas generator assembly, the next 2 units will be hot fire tested to verify their operation and then delivered to the turbopump test facilities where they will be used to drive the pump turbines during the turbopump component hotfire test program. The gas generator assembly hot fire test program will then continue with units 4, 5, and 6. The gas generator assembly hotfire test plan is shown in Table 3-12.

Table 3-12. LRB Gas Generator Component Hotfire Test Plan
($LOX/RP-1$ and LOX/H_2)

Units	Test Objectives						Number of Tests
	Ignition	Performance	Dynamic Stability	Gas Temperature Uniformity	Limits	Overstress	
Unit No. 1	X	X	X	X			75
Unit No. 2	X	X	X	X			50
Unit No. 3		X		X	X	X	60
Unit No. 4		X			X	X	80

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Total = 265 tests

Turbomachinery. The LRB has separate LOX and fuel turbopumps. Hotfire testing of these turbopumps is planned at a government or vehicle contractor test facility. Table 3-10 shows the schedule and planned hardware for this test program. As indicated 5 new and 2 rebuilds each for a total of 7 LOX turbopumps and 7 fuel turbopumps will be procured, assembled and hot fire tested. As stated previously, the 2 gas generator assemblies required to support the turbopumps testing are planned to be provided by the gas generator assembly test program. The turbopump hot fire test plan is shown in Table 3-13. The turbomachinery design and development risk would be significantly reduced if the technology programs described in a latter section are started during the Phase B program and completed in sufficient time to provide data during the development program design phase.

Table 3-13. LRB Turbomachinery Component Hotfire Test Plan (LOX/RP-1 and LOX/H_2)

Units		Test Objectives										Number of Tests	
LOX	Fuel	Head-Flow vs. Speed	Head-Flow Full Speed	Suction Specific Speed	Pressure Oscillations	Rotor Dynamics	Critical Speed	Duration	Life	Limits	Overstress	LOX	Fuel
001	001	X	X	X								40	40
002	002	X	X	X								40	40
003	003		X	X	X	X	X	X				50	50
004	004		X	X	X	X	X	X	X	X	X	50	50
002R	002R		X	X	X	X	X	X				50	50
003R	003R			X	X	X	X	X				50	50
005	005			X	X	X	X	X	X	X	X	50	50

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Total Tests = 330 + 330 = 660

Thrust Chamber Assembly. The thrust chamber assembly consists of an injector, main combustion chamber (MCC) nozzle, ignition system and LOX dome inlet manifold. Hotfire testing of the thrust chamber assembly is planned at a government or vehicle contractor test facility. Ignition for the LOX/H_2 engine MCC is provided by an augumented spark ignition (ASI) system. A hypergolic fluid ignition system will be used for the LOX/RP-1 MCC.

Table 3-10 shows the schedule and planned hardware for this program. As indicated, 5 assemblies will be tested. Assembly number 1 consists of prototype injector, LOX dome and a solid (workhorse) MCC. This unit will be used to develop the ignition sequence and demonstrate performance and combustion dynamic stability. Subsequent units will be utilized for testing regeneratively cooled MCC's and nozzles to demonstrate cooling. The thrust chamber assembly hot fire test plan is shown in Table 3-14. The thrust chamber assembly is the other major engine subsystems that could benefit from a technology plan that is started in Phase B and compiled in sufficient time to provide data during the development program phase. This technology program is described in a latter section.

Table 3-14. LRB Thrust Chamber Assembly Component Hot Fire Test Plan

Units	Configuration	Test Objectives						Number of Tests
		Ignition	Performance	Dynamic Stability	Gas Temperature Uniformity	Limits	Overstress	
Assembly No. 1	Injector LOX Dome, Solid Wall Mcc	X	X	X				20
Assembly No. 2	Injector, LOX Dome, Regen. Cooled MCC and Nozzle	X	X	X	X			30
Assembly No. 3	Injector, LOX Dome Regen. Cooled MCC	X	X	X	X			40
Assembly No. 4	Injector, LOX Dom, Regen. Cooled MCC and Nozzle		X		X	X	X	50
Assembly No. 5	Injector, LOX Dome, Regen., Cooled MCC		X		X	X	X	60

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Total = 200 Tests

3.5.5 Engine Test Program

The engine development program schedule Table 3-15, is the same for the pump fed LOX/RP-1 and LOX/H_2 configurations. However the amount of hardware and number of tests required for the development program are different. Since the experience base and the potential problem areas are different for LOX/H_2 and LOX/RP-1. The throttleable LOX/H_2 pump fed engine is judged to require less testing and hardware because of the considerable existing experience with this propellant combination in terms of combustion stability and cooling at the required operating conditions. The LOX/RP-1 pump fed engine is judged to require a slightly higher number of development tests and hardware because the chamber pressure is slightly higher than previously experienced and cooling at the higher chamber pressure level has not been demonstrated.

The planned development program for the LRB pump fed engines is divided into 5 phases. These phases are described in Figure 3-12. The first 3 phases are

TEST PHASE DESCRIPTIONS

PHASE	DESCRIPTION
1. Characterization	Testing designed to fully evaluate engine operation, Includes tests to evaluate: • Ignition • Gimbaling • Start/shut • Limits • Performance • Overstress • Stability • Failsafe • Duration • Heat Exchanger • Throttling • POGO
2. Life Development	Testing designed to evaluate the life margin in the engine design
3. Reliability demonstration	Testing designed to demonstrate that engine reliability is as specified
4. Pre-flight rating (PFRT)	Formal testing specifically designed to demonstrate readiness for first flight
5. Flight rating (FRT)	Formal testing specifically designed to demonstrate readiness for production and full operation

Figure 3-12. Engine Development Programs

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Table 3-15. Engine Development Program

Requirements

Phase	Requirements	
Characterization	*320 Tests on 8 engines	
Life Development	*Formal life demonstration on 3 of every component during engine testing	
Reliability Demonstration (99% Rel. at 90% confidence)	*230 Equivalent Full Duration tests on 8 engines	
Pre-flight Rating (PFRT)	10 full duration tests each, on 2 engines	
Flight Rating	Formal life demonstration on 2 engines	
Spares	20%	
Factors	LOX/RP-1	LOX/H_2
Test Realization	10 Percent	10 Percent
Risk	15 Percent	5 Percent

*Total number of tests can be reduced by combining objectives

intended to evaluate and demonstrate the maturity and reliability of the engine. The specified demonstrated reliability requirement for the LRB is 99 percent at 90 percent confidence. The last 2 phases of the development program are intended to formally demonstrate the engine for first flight and subsequent operational and production readiness. The test requirements for each of the 5 phases (see Figure 3-12) of the development program are defined in Table 3-15. Also shown in Table 3-15 is the expected test realization factor; that is, the number of tests that are expected to abort or not produce valid data. This factor is used for planning the number of tests required. The risk factors for the LOX/RP-1 and LOX/H_2 configurations are also shown.

The number of engines required for the development program is based on the design life specified. since the LRB's are expendable engines their mission life is one. However, the hardware cost in the development program can be substantially reduced by being able to conduct many tests on each engine. In order to determine the number of engines required for the development program the life definitions given in Figure 3-13 were used. By defining the design life requirement at 60, the engines can be tested at least 30 times each during the first 3 phases with a safety factor of 2 and 10 times each during the PFRT and FRT test phases thus demonstrating a factor of 2 on the potential test life of a production engine which could require 5 starts. There are 2 acceptance tests, potential for 2 on-pad aborts and 1 flight.

Life Definitions

	MISSION LIFE*	DEVELOPMENT LIFE (EFDTS)	DESIGN LIFE	FORMAL DEMONSTRATION LIFE
EXPENDABLE	1	30	60	10

*Plus: 2 acceptance tests
1 Full duration
1 Start
2 on-pad

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Figure 3-13. Engine Life Development Program

The number of tests and engines assigned for development of both the LOX/RP-1 and LOX/H_2 pump fed configurations are shown in Table 3-16. Note that the total includes the minimum requirements for each phase, the test realization factor, plus tests and hardware to account for risk differences and spare hardware based on a 20 percent factor. The engine development plan for the LOX/RP-1 engine is shown in Figure 3-14. As indicated 4 test positions are necessary to complete the 784 tests. A test frequency of approximately 2 tests per week is planned. Also note that in addition to development testing of the 24 engines required for the development program, the 5 engines required for the main propulsion test article (MPTA) are acceptance tested prior to delivery. The engine development plan for the LOX/H_2 LRB is not shown but would be the same as the LOX/RP-1 plan, Figure 3-14, but with 2 less engines.

Table 3-16. LRB Pump Fed Engine Development Program System Test/Hardware Requirements

	DEVELOPMENT										TEST REALI-ZATION	RISK		SPARES	TOTAL	
	CHARACTER-IZATION		LIFE		RELIABILITY DEMON-STRATION		PFRT		FRT							
	TESTS	ENGINES	TESTS	ENGINES	TESTS	ENGINES	TESTS	ENGINES	TESTS	ENGINES	TESTS	TESTS	ENGINES	EQUIV. ENGINES	TESTS	ENGINES
LOX/RP-1	320	8	30	3	230	8	20	2	20	Use PRFT Engines	62	102	3	5	784	29
LOX/H_2	320	8	30	3	230	8	20	2	20	Use PRFT Engines	62	34	1	4	716	26

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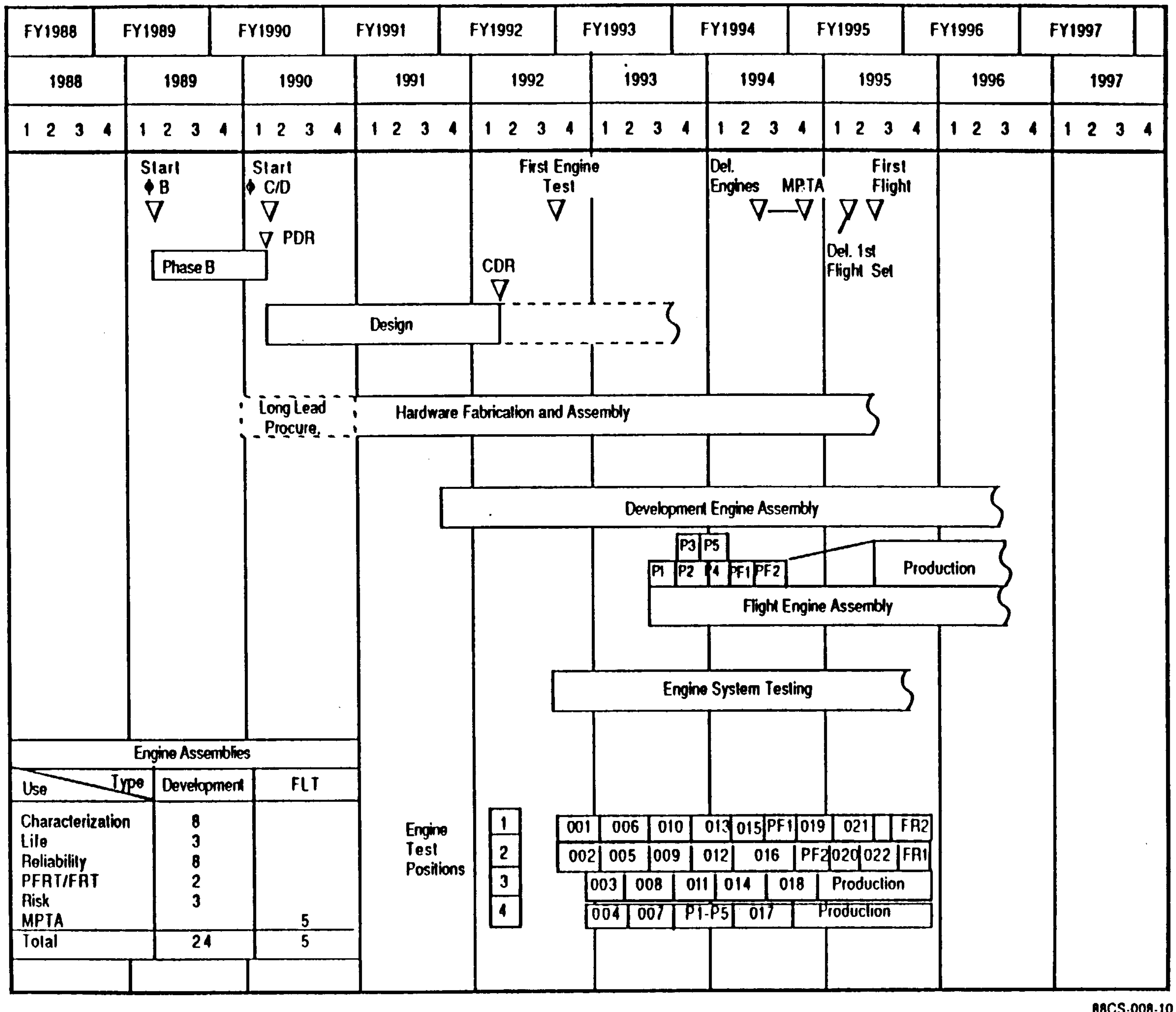


Figure 3-14. LRB Pump Fed Engine System Devlopment Program (LOX/RP-1 and LOX/H_2)

3.5.6 Technology Plan For LRB Pump FED Engine

The development plan for the LRB pump fed engine is designed to support the first vehicle launch in the third quarter of 1995. With a planned start date in the second quarter of 1990 this 63 month (5 1/4 year) program must use a low risk approach in the design and fabrication of the test hardware. A significant risk reduction would be achieved by starting a Phase B design effort and several technology programs one year prior to start of the phase C/D effort. The schedule for this Phase B effort and the technology programs and the relationships to the overall development program are shown in Figure 3-10. The following is a discussion of the technology programs proposed to support the LRB pump fed development program.

Valves and Controls. The LRB engine will be equipped with a closed loop thrust and mixture ratio control system. The major components include modulating main oxidizer and main fuel valves, modulating gas generator oxidizer and fuel valves, an electronic controller; control and conditioning monitoring instrumentation, a pneumatic control console, harnesses, check valves and flight instrumentation and data recording equipment. A system with the same capabilities is used on the Space Shuttle Main Engine. This system is based on 1970's technology and is very costly and complex. Recent advances in control/computer/modulating actuator technology promises to greatly simplify the required closed loop control system resulting in a 60 to 70 percent cost reduction. By initiating the control system architective study during Phase B and completing a preliminary design, advanced technology low cost features could be evaluated in time to provide data for the Phase C/D design phase, Figure 3-10. Some of the features to be evaluated include; fail safe and redundancy features, copper vs fiber optic interconnect harnesses, hydraulic vs pneumatic vs electric motor actuators for the main and gas generator valves and low production cost features. This technology program is scheduled to be completed in 18 months.

Pump Bearings. Bearing technology for high speed rocket engine turbopumps has progressed significantly in recent years. However the majority of this experience has been with propellants other the hydrocarbons. The LRB LOX and fuel pumps will operate at relatively high speeds at normal power level and operate off design as the engine thrust is throttled plus 10 percent, minus 25 percent. Ideally the bearings will use the propellants, LOX and RP-1, as the lubricating/cooling fluid. A technology program in parallel with a Phase B design effort would evaluate if current bearing technology can be applied to the LRB pumps. The bearing evaluation program could be completed in 22 months.

Pump Seals. Large diameter, high pressure, high speed, liquid oxygen and RP-1 rotating shaft seals are subject to severe distortions caused by thermal and mechanical loading. A design and analysis effort during the Phase B effort would allow early procurement of the best candidate seals for test evaluation. Some of the configurations to be evaluated include:

1. Face type metal bellow seals
2. Face type plastic lip seals
3. Face type elastomeric seals
4. Hydrostatic seals
5. Floating ring seals

The seal evaluation could be combined with the bearing tester setup and completed in the same schedule.

Pump Inducers and Impellers. The oxidizer and fuel pumps for the LRB engine will operate at relatively high speeds and pressures. As a result structural limits are being pushed requiring that the thickness of the parts be increased. Recent experience has shown the compromises in inducer and impeller performance must be made because of the structural requirements. An advanced technology program during the Phase B design effort would allow time to achieve the best design for these rotating pump elements and build and test sub-scale parts for testing. This sub-scale hardware would be tested in Rocketdyne's existing water test facility. The resulting data would be used to design the full size hardware during the Phase C/D design effort. This program could be completed in 22 months.

Thrust Chamber Injector. Design of a stable high performance LOX/RP-1 main thrust chamber injector would significantly reduce the risk of a design iteration which could adversely impact the Phase C/D schedule. A technology program to design, fabricate and test a 2 Dimensional (2D) model of the full size main injector in conjunction with a Phase B design effort would provide data input into the Phase C/D design effort. This test data would greatly increase the confidence in the stability and performance of the full sized injector tested in Phase C/D. Data expected from the 2D technology test model includes baffle compartment size, baffle length, acoustic cavity arrangement, verification of injection element performance and chamber pressure dampening characteristics following a pressure disturbance caused by a bomb. This technology program is scheduled to be completed in 24 months.

Turbine. The turbines for the LRB oxidizer and fuel pumps will operate at high speed and pressure ratios resulting in supersonic flow velocities in the nozzle and rotor blade passages. These operating conditions make turbine performance very sensitive to nozzle and blade geometry. The emphasis on low cost fabrication including the potential use of castings to net dimensions will produce a lowered cost product with attendant increased potential for part to part dimensional variation. An advanced technology program during the Phase B design effort will allow time to evaluate the sensitivity of turbine performance to the variation in part to part geometry resulting from low cost fabrication processes. this data will be used in the Phase C/D design resulting in reduced risk in the development program. The testing will be accomplished in an air flow test facility with each element of the turbine added to the test fixture in series. Sufficient model size parts of each element (nozzles and blades) will be procured and tested. This program is scheduled for 24 months.

3.6 LOX/RP-1 LIQUID ROCKET BOOSTER PRELIMINARY CONTRACT END ITEM (CEI)

3.6.1 Background

The LOX/RP-1 Liquid Rocket Booster engine is being designed to provide booster propulsion for the Space Shuttle. The primary objective of the study was to identify and evaluate viable LOX/RP-1 pump fed engine candidates that would meet the requirements for the STS and select the best candidate.

3.6.2 Selected Engine Description

The selected engine configuration utilizes the GG cycle with LOX and RP-1 as propellants. RP-1 is used to cool the MCC and nozzle after which it is injected into the injector except for a small amount that is diverted to the GG where it combines with LOX for the combustion process that produces the turbine drive gas. After passing through the turbines, this gas is dumped into the nozzle.

The bulk of the RP-1 is first used to cool the thrust chamber and is then injected into the MCC as a gas where it combines with LOX for the MMC process, after which it is expanded through the nozzle to produce the engine thrust.

3.6.3 LRB CEI Requirements

This document presents the preliminary CEI requirements that the LRB must fulfill to satisfy the requirements for the OSTS. These requirements are as follows:

Performance. All performance values stated herein are nominal values. The minimum and maximum values will be determined during subsequent study efforts.

1) Engine Thrust - The LRB shall be capable of producing 719,500-lb vacuum thrust at the normal power level (NPL) and 791,400-lb vacuum thrust at the emergency power level (EPL). The engine shall be capable of throttling up from NPL to EPL in TBD seconds. The engine shall be capable of being throttled down to a minimum power level (MPL) of 539,600 LBS vacuum thrust in TBD sec.

2) Specific Impulse - The specific impulse for the LRB shall be as follows for the 3 vacuum equivalent thrust operating points.

	Thrust Level	Sea Level Is (seconds)	Altitude Is (seconds)
EPL	791,400 lb (vac)	273.9 ± TBD	321.9 ± TBD
NPL	719,500 lb (vac)	269.5 ± TBD	322.4 ± TBD
MPL	539,600 lb (vac)	252.8 ± TBD	323.7 ± TBD

3) Main Combustion Chamber (MCC) Propellants

Propellants		Injected State
Oxidizer	- Oxygen (LO_2)	Liquid
Fuel	- RP-1	Liquid
MCC MR	- O/F NPL	O/F EPL

4) GG Propellants

Propellants		Injected State
Oxidizer	- Oxygen (LO_2)	Liquid
Fuel	- RP-1	Liquid
GG MR	- O/F NPL	O/F EPL

5) Engine MR - The engine MR for the LOX/RP-1 LRB shall be as follows for the two thrust operating points:

Thrust Level	Mixture Ratio
719,500 (vac)	2.8
791,400 (vac)	2.8
539,600 (vac)	2.8

The engine shall equipped with a closed loop engine MR control system capable of controlling MR within ± 1.0% of the nominal value.

6) Acceptance Calibration - The acceptance calibration for the LRB shall be as follows:

Thrust (NPL) - 719,500 lb ± 3% (vac)
(EPL) - 791,400 lb ± 3% (vac)
MR (NPL) - 2.8 ± 1%
(EPL) - 2.8 ± 1%

7) Coolants - The coolants for the MCC and nozzle shall be RP-1.

8) Burn Duration - The LRB shall be capable of maximum burn duration of 180 sec at NPL nad EPL.

9) Uncoupled Thrust Oscillations - The engine-produced uncoupled oscillatory thrust shall be no greater than the following for the respective specified frequency ranges:

R = 0 to 1.5 H_z	F = ± 6000 lb
R = 0.5 to 1.5 H_z	F = ± 1500 lb
R = 1.5 to 2.5 H_z	F = ± 450 lb
R = 2.5 to 100 H_z	F = ± 1500 lb

For the purpose of performing data analysis to verify engine compliance in the critical frequency range oscillatory shall be defined as the average value of an oscillation over at least 16 cycles.

10) Combustion Stability - The engine-produced main chamber pressure oscillations shall not exceed ± 5% of the mean steady-state pressure.

11) Damping time for artificially induced pressure spires shall be TBD milliseconds maximum.

12) POGO Suppression - The engine shall provide a POGO suppression system in accordance with the following requirements (TBD).

13) Engine Controller - The electrical closed loop engine control system shall be capable of continuous operation at ambient temperature for an unlimited period of time during checkout and maintenance.

14) System Checkout and Monitoring Capability - The design shall include onboard checkout capability, redundancy verification, and status monitoring during ground operations. The engine design shall include a limit control system capable of automatically initiating engine shutdown to prevent catastrophic failure.

Operations. The operational requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Engine Start - The engine start system shall have self-contained control within the engine envelope. The start sequence shall be started by a single electrical signal from the vehicle or ground source.

2) The engine shall be capable of one start after each ground servicing.

3) The engine start sequence shall be capable of achieving normal power level (NPL) thrust in less than 5 sec.

4) The thrust buildup rate shall not exceed TBD lb thrust in any 10-msec time period.

5) Starting Impulse - The starting thrust impulse to NPL shall not exceed TBD lb-sec.

6) Throttling Control - The engine thrust control system shall be capable of raising the engine thrust from NPL to EPL at the rate ot TBD lb-sec any time after reaching NPL.

   a) Throttle Rate - The engine thrust control system shall be capable of raising the engine thrust from NPL to EPL at the rate of TBD lb-sec any time after reaching NPL.

b) The thrust control system shall be capable of a step response of TBD lb thrust increase in less than TBD sec after a step command.

Engine Shutdown. The engine shall be capable of a safe shutdown from any power level including the start sequence.

1) The engine shutdown sequence shall be capable of reducing thrust from NPL to zero in TBD sec.

2) The shutdown impulse shall not exceed TBD lb/sec from NPL.

3) The engine shall be capable of shutdown from any defined thrust level upon receipt of an electrical command at a rate of TBD lb thrust change per any 10-msec time interval.

Environmental Conditions. The engine shall be capable of operating safely under the following conditions:

1) The engine shall be capable of operating safely where exposed to a hat flux of TBD Btu/ft^2-sec and a surface temperature of TBD°F. The heat transfer coefficient that shall be used for design is TBD Btu/sec-ft^2°F.

2) The surface temperature of lines or surface in contact with cryogenic propellants shall be controlled to preclude the formation of liquid air.

3) Acceleration Loads - TBD

4) Shock Loads - TBD

5) Ground Handling and Transportation Loads - TBD

6) Storage Life - The engine shall be capable of being transported and stored over an ambient temperature range of TBD°F to TBD°F, an ambient pressure range of TBD psig to TBD psi,, a relative humidity of 100% at temperatures less than or equal to TBD°F.

   a) The engine shall suffer no degradation of reliability or operating life during the storage period, subject to the inspection and maintenance requirements TBD.

7) Exposure - The engine system and components shall be capable of being transported and stored without deterioration in areas where conditions may be encountered having salt spray and relative humidity as experienced in coastal regions. The engine system and components shall be capable of withstanding exposure to sand and dust when equipped with proper closures.

8) Lightning - The engine controller shall be designed to operate without damage in accordance with TBD lightning protection criteria.

Prelaunch. The engine shall be designed for minimum prelaunch servicing.

1) Ground Service - The engine shall be capable of achieving pre-launch thermal conditioning without ground servicing in less than TBD minutes from the time propellants are supplied to the engine. Recirculation flow rates to achieve thermal conditioning are as follows:

   LOX - TBD lb/sec
   RP-1 - TBD lb/sec

2) The engine shall be capable of servicing and maintenance while in either the horizontal or vertical position.

3) The engine shall not require any servicing from ground equipment within 24 hr after propellants are loaded.

4) External or internal leakage of propellants shall not occur in such a manner as to impair or endanger the engine/vehicle function. Leadage monitoring capability shall be provided with the design objective that separable connections not exceed 1×10^{-4} sec helium at leak check pressure.

5) The engine shall not require any monitored redlines external to the engine prestart and shall provide a continuous engine-ready signal to the vehicle when all critical parameters monitored by the engine control system are within TBD conditions.

Interface. The engine shall require the following conditions at the respective interfaces with the vehicle:

1) Propellant inlet conditions at engine start:

   a) LOX - 65 psia to TBD psia, 163 to 170°R
   b) RP-1 - 45 psia to TBD psia, 38 to 40°R

2) Propellant inlet conditions during mainstage:

   a) LOX - 65 psia to TBD psia, TBD to TBD°R
   b) RP-1 - 45 psia to TBD psia, TBD to TBD°R

3) Electrical

   a) The engine shall be supplied TBD dc V
   b) The engine shall be supplied TBD ac V
   c) The controller shall be engine supplied and mounted.

4) Pressurization Gas - The engine shall provide GOX to pressurize the vehicle oxygen tank and helium to pressurize the RP-1 tank.

   a) Oxygen Tank Pressurant - The engine shall be capable of supplying GOX pressurant as indicated in Table TBD.

5) Purge Requirements - Nitrogen, in accordance with MIL-P-27401, and helium, in accordance with MIL-P-27407, shall be used for operational and servicing purges and leakage tests.

   a) Operational Purges - TBD
   b) Servicing Purges - TBD

6) Digital Interface

   a) A suitable digital interface shall be provided for vehicle commands to the engine.

Physical Requirements. The physical requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Envelope - the maximum engine width is 119 in. and the engine height is 178 in.

2) Weight - The engine weight is as follows:

	Dry	Wet
Basic engine	8108 lb	TBD
Accessories	TBD lb	TBD
Thermal Insulation	TBD lb	TBD

3) Gimbaling - The engine shall be capable gimbaling in a ±6° square pattern at a gimbal rate of 10°/sec and an acceleration rate of 10 rad/sec squared. The engine shall provide attach points for the vehicle-furnished actuators. The gimbal system shall be capable of returning the engine to null position at engine shutdown.

4) Engine Alignment - The engine shall be aligned so that the actual thrust vector is within 39 min of an arc to the engine centerline and within 0.25 in. of the gimbal center. The gimbal center shall be within 0.010 in. of the engine centerline.

5) Engine Fluid Interface Ducts and Lines - The engine shall supply all interface ducts and lines with a minimum of TBD in. straight section upstream of the engine interface plane.

6) Engine Electrical Interface - All engine electrical connections from the vehicle shall be located in a single, engine-mounted panel.

Reliability. The reliability of the configuration upon which the final flight certification is based shall be that which is necessary to ensure functioning within the specified design life.

1) The engine design life is 1.0 mission at EPL.

2) The engine shall be designed for a minimum of TBD missions at EPL.

3) Fail-Safe Design - The engine shall be capable of shutdown from an internal signal without damage to other systems.

4) Structural Criteria - The engine shall be designed to provide the following minimum factors of safety:

Minimum yield	- 1.1
Minimum ultimate	- 1.4 combined loads
Minimum ultimate	- 1.5 pressure only
Minimum proof	- 1.2 times EPL operating conditions, unless fracture mechanics requires a higher factor
Low cycle fatigue	- 4.0
High cycle fatigue	- 10.0

Note: Components should be designed for 1.25 on endurance limit where feasible

Diagnostic Monitoring. The engine shall be capable of self-diagnostics in real time. Unsafe conditions shall cause an engine-generated shutdown unless inhibited by the vehicle.

1) Diagnostic data will be recorded for postflight analysis.

4.0 LOX/HYDROGEN PUMP FED ENGINE

4.1 INTRODUCTION

The third propulsion concept selected for the LRB employs LO_2/LH_2 propellants with a gas generator cycle engine. The basic reasons for the selection of LO_2/LH_2 system are low technical risk, no environmental concerns, and commonality with the current shuttle ET propellants.

4.2 MAIN PROPULSION SYSTEM

A baseline engine concept was selected based on previous studies and experience along with trade studies for the STS application. An engine performance and pressure balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the recommended configuration and physical design.

4.2.1 Engine System.

The engine selected is of the expendable type with continuous variable thrust capability of 75% to 110% of the normal power level. The rationale for the engine thrust and engine throttling range were set by GDSS. The propulsion system described here is based on a mixture ratio of 6.0/6.9 and expansion ratio of 41.4. This gives close to maximum mean I_{sp} since nozzle exit pressure is approximately equal to the mean flight ambient pressure.

A side view and top view of the selected LOX/H_2 LRB engine preliminary design are shown in Figure 4-1.

Engine Feature Selection. The engine features selected here are based on data generated by Rocketdyne for the STME studies. It is interesting to note that the LRB LO_2/LH_2 vacuum EPL thrust (619 Klb) is close to the STME vacuum EPL thrust (570 Klb). The various features selected are summarized below.

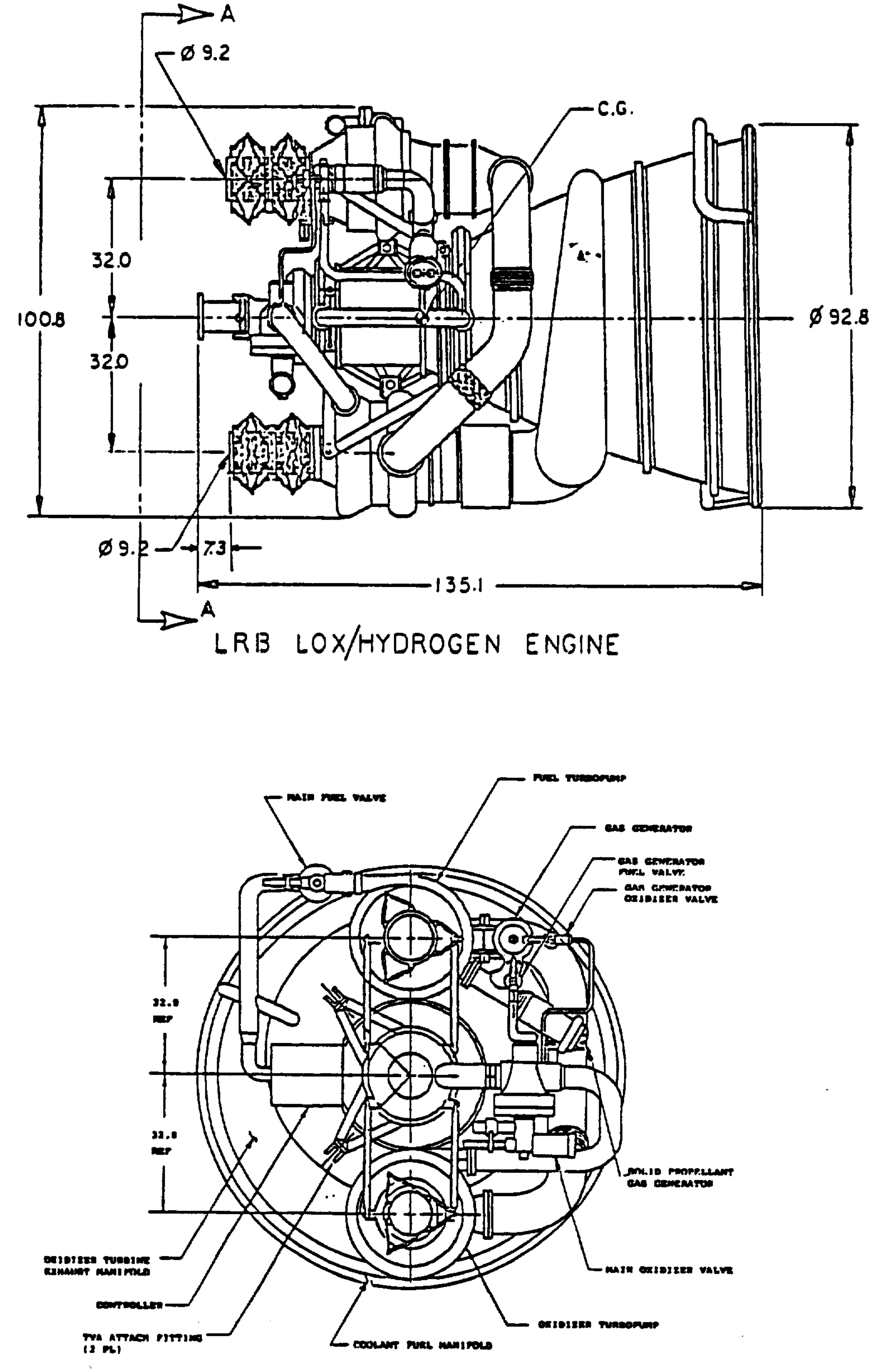


Figure 4-1. LRB LOX/Hydrogen Engine (Dimensions in Inches)

The current baseline engine assumes no boost pumps, and the engine inlet pressure of 65 psia for LOX and 25 psia for LH_2 are baselined. Boost pumps in the STME studies have been identified as increasing engine weight, cost, and complexity. Although no overall vehicle trade studies have been done by Rocketdyne for the above stated engine inlet conditions, it is felt that these baseline conditions are appropriate for the STS boosters. The rationale for the LOX pump inlet engine pressure was described in Section 3.1.2. The rationale for the LH_2 case is shown in Figure 4-2. This figure shows that pump size starts to level off at about 25 psia inlet pressure.

A closed loop control is baselined because of the continuous throttling requirement. The basic overall control system features a P/U system. The main propellant valves function as throttle valves, providing precisely repeatable valve area settings.

Various options for disposing of the engine exhaust discussed in the STME studies are shown in concepts A, B, and C in Figures 4-3, 4-4, and 4-5. In concept A, the exhaust gases are uniformly injected into the main nozzle through the nozzle wall at an expansion ratio of 16:1 with sonic flow through gaps between tubes. In concept B, the exhaust gases are uniformly injected through supersonic flow nozzle at the same expansion ratio with supersonic flow parallel to the mainflow. In concept C, the gases are injected through a nozzle at the main nozzle exit plane. Although the overboard exhaust concept C, is the lowest cost (by about $0.2 M), concept A has been baselined by GDSS because of concerns regarding the impact on gimbal actuator forces, base heating, and engine layout.

The turbine spin start using GSE helium is selected over the tank head start because it provides more repetitive starts. In addition, as shown in Figures 4-6 and 4-7, the tank head start is comparatively slow compared to other types of start, and this may complicate optimization of ignition sequencing for the vehicle.

Various features of the selected engine are shown in Table 4-1.

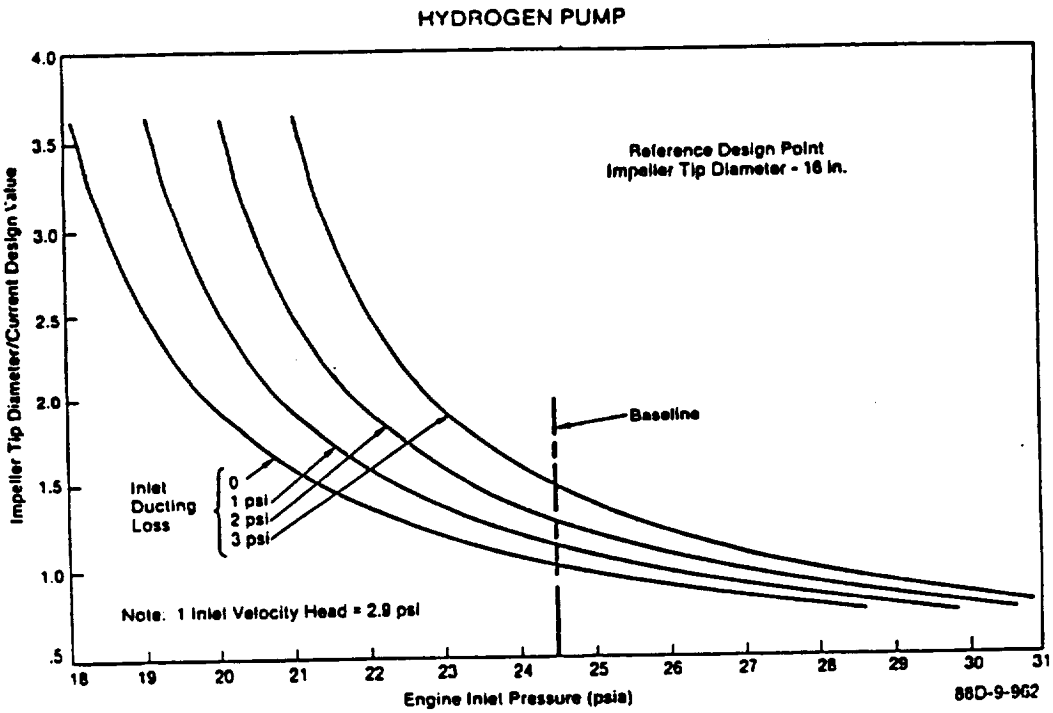


Figure 4-2. Inlet Pressure Impact on Pump Size

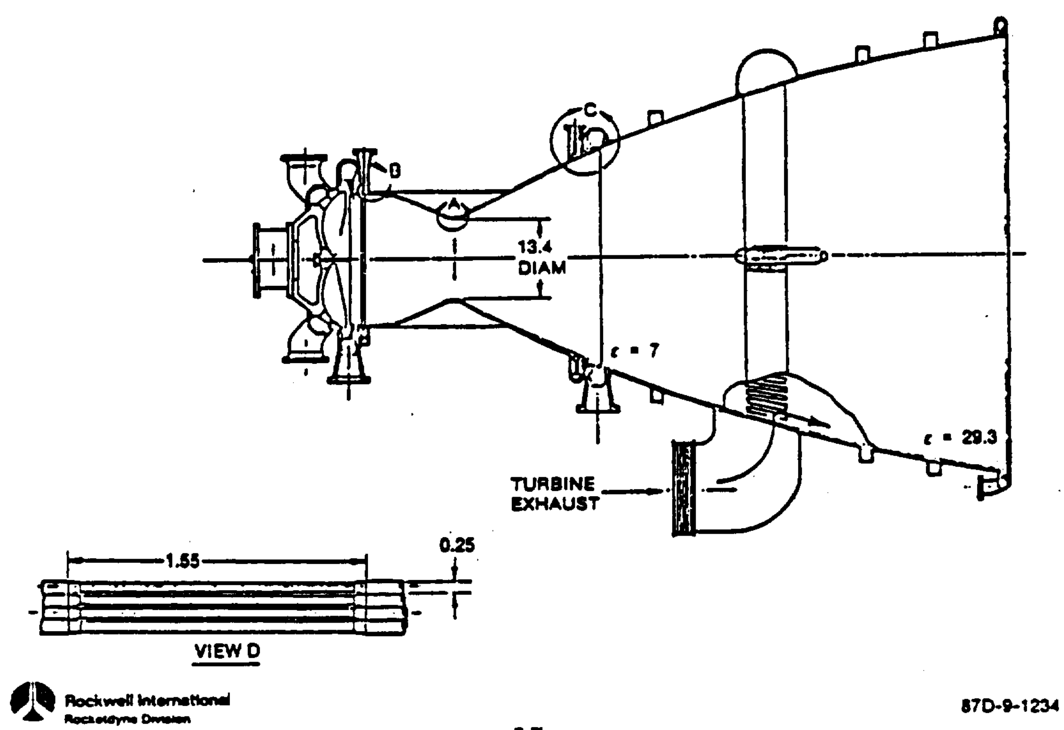
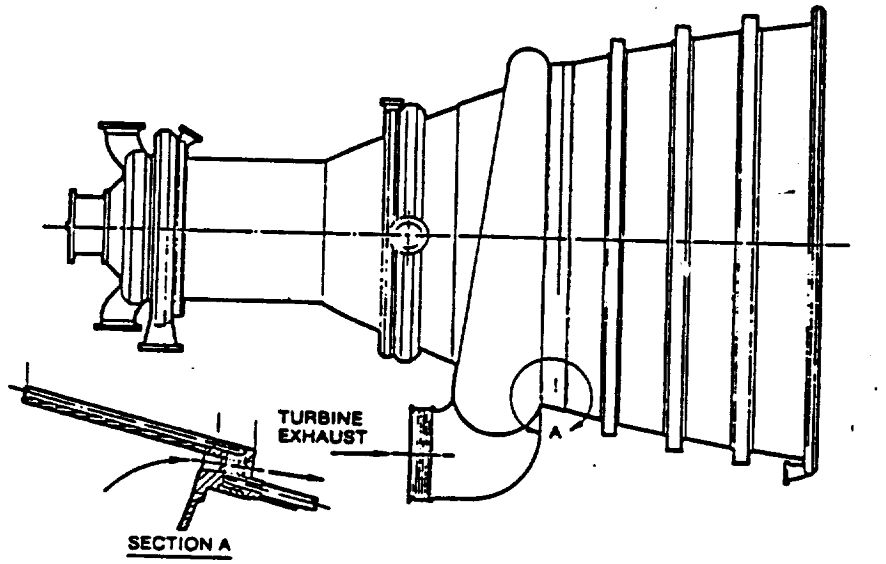


Figure 4-3. Concept A - Turbine Exhaust Through Nozzle Wall Sonic Flow



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Rocketdyne Division

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Figure 4-4. Concept B - Turbine Exhaust Through Nozzle Wall Supersonic Flow

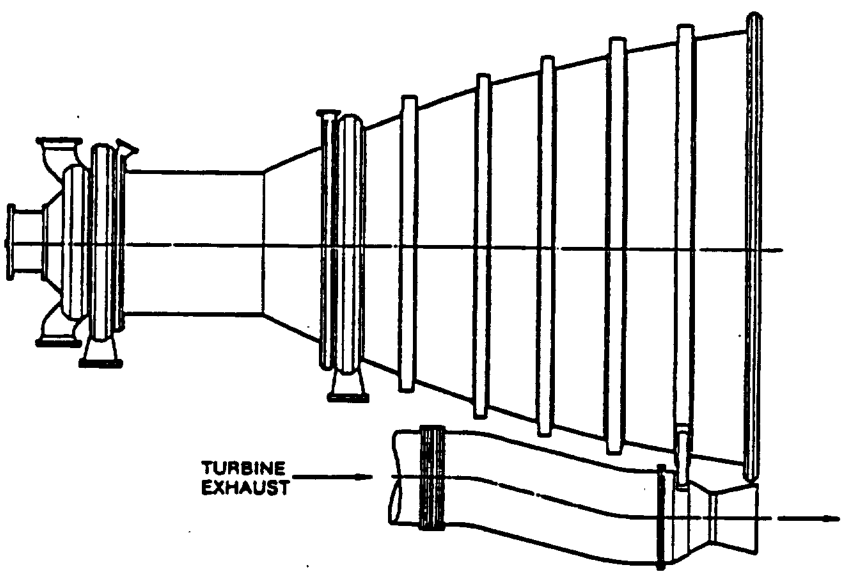

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Figure 4-5. Concept C - Turbine Exhaust at Exit Plane Single Round Nozzle

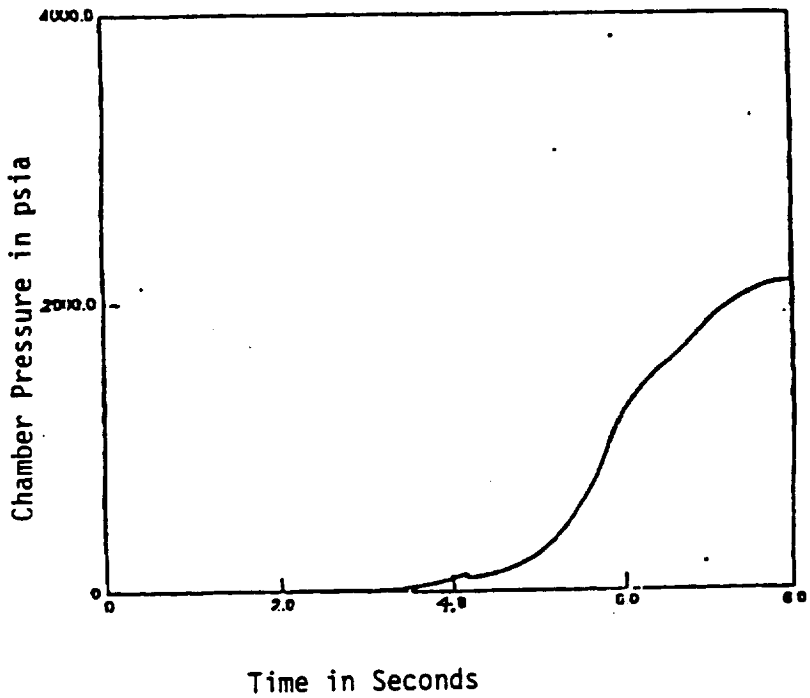


Figure 4-6. STME Tank Head Start

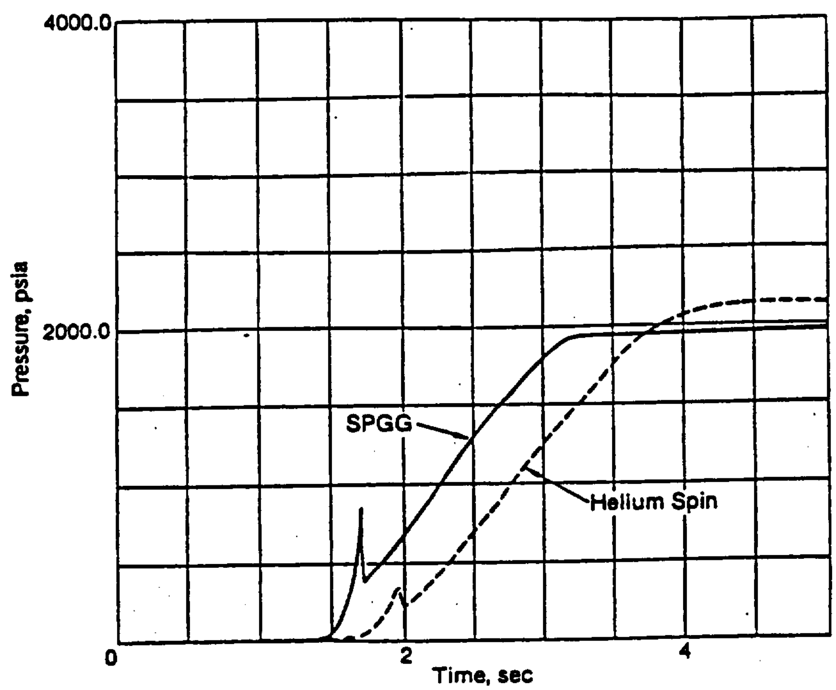


Figure 4-7. Chamber Pressure During Turbine Spin Start

Table 4-1. Main Features of LO_2/LH_2 Engine

Cycle	Gas generator
Boost pumps	None
Throttling capability	Continuous, 75 to 110%
Control System	Closed loop using P/U system
Turbine exhaust disposal	Injected in nozzle at $\epsilon = 16:1$
Turbine start	GSE Helium
Inlet ducts	Scissors
Ignition	Spark ignition
Nozzle	80% Bell nozzle
Gimbal	Head end gimbal; 6-square pattern
Delivered life	5
Burn qual duration	150 secs.
Engine inlet requirement: LO_2	65 psia
LH_2	25 psia

Selected Engine Characteristics. An engine cycle balance was done on the point design arrived at using the sizing program. The main engine characteristics are shown Table 4-2a and 4-2b. A more detailed computer generated tabulation of engine charactertistics is given in Table 4-2c. It should be noted in Table 4-2b that the vacuum I_{sp} changes with throttling; however for simplicity it was assumed constant in the sizing runs. The turbopump characteristics of the LO_2/LH_2 engine are shown in Table 4-3.

A typical cutoff sequence of events is shown in Table 4.4 with the propellant consumption noted during engine start and shutdown. The propellant consumption is from engine start signal to mainstage operating level and does not include any engine prechill consumption.

4.2.2 Engine Schematic and Operation.

Figure 4-8a shows the LRB engine schematic for the LOX/LH_2 engine, while the LRB schematic of Figure 4-8b shows the propellant flow rates and conditions at various parts of the engine at EPL (110% NPL). LH_2 is used for cooling both the main combustion chamber (MCC) and the nozzle. A portion of the H_2 from the nozzle is used as the fuel for the GG, which provides the turbine drive gas. The rest of the hydrogen from the nozzle is mixed with the flow from the MCC and the nozzle. A small amount of GH_2 is tapped from this mixture for the tank pressurization and the rest of the hydrogen in gaseous form (above critical temperature) is injected into the combustion chamber.

Table 4-2a. LOX/LH_2 Pump-fed Engine Performance Summary

Parameter	EPL (110%)
Thrust Vac (k lbs)	619.9
Thrust SL (k lbs)	542.9
Chamber pressure (psia)	2538
C star efficiency	99%
Expansion ratio	41.4
Mixture ratio, Eng/TC	6.0/6.0
Isp vac (sec)	427.03
Isp SL (sec)	373.99
Total flow TC (lb/sec)	1399.11
Oxidizer flow TC (lb/sec)	1222.0
Fuel flow TC (lb/sec)	177.11
Total GG flow (lb/sec)	52.54
GG mixture ratio	0.736
GG oxidizer flow (lb/sec)	22.27
GG fuel flow (lb/sec)	30.27
Length (inch)	135.08
Exit dia (inch)	81.67
Dry weight (lbs)	6671.6

Table 4-2b. Engine Performance vs Power Level

Operating Condition	F_{vac} lb	F_{sl} lb	P_c psia	I_{vac} sec	I_{sl} sec
EPL*	619900	542908	2538	427.03	373.99
NPL*	563545	486553	2366	427.90	396.44
MPL*	422659	345667	1775	430.08	351.74

* EPL stands for Emergency Power Level
NPL stands for Nominal Power Level
MPL stands for Minimum Power Level

Table 4-2c. Pump Fed LOX/LH_2 Engine Characteristics

LOX/H2 619.0K VAC PC OPTIMIZED FN=LR814

ENGINE PERFORMANCE SUMMARY

SEA LEVEL THRUST	(LBS)	542908.
VACUUM THRUST	(LBS)	619900.
AVERAGE THRUST	(LBS)	619000.
ENGINE MIXTURE RATIO	(NONE)	6.000
CHAMBER PRESSURE	(PSIA)	2538.00
OVERALL AREA RATIO	(AE/AT)	41.43
NOZZLE PERCENT LENGTH	(PERCENT)	80.00
NOZZLE WALL EXIT PRESSURE	(PSIA)	8.76
THROAT DIAMETER	(IN)	12.689
ENGINE LENGTH	(IN)	135.08
OVERALL ENGINE EXIT DIAMETER	(IN)	81.67
COMBUSTOR LENGTH	(IN)	14.42
CONTRACTION RATIO	(NONE)	2.98
ENGINE VACUUM C-SUB-F	(NONE)	1.862
ENGINE VACUUM ISP	(SEC)	427.03
ENGINE SEA LEVEL ISP	(SEC)	373.99
ENGINE AVERAGE ISP	(SEC)	427.03
CONSTANT USED IN AVE. CAL.	(NONE)	1.000

TURBOPUMP DESCRIPTION	(UNITS)	MAIN. OXIDIZER	PUMP FUEL
PUMP			
# OF STAGES	(NONE)	1.00	3.00
HORSEPOWER	(HP)	22385.370	70000.582
ROTATING SPEED	(RPM)	9798.0	23147.4
EFFICIENCY	(NONE)	.75920	.75682
INLET PRESSURE	(PSIA)	85.00	24.80
OUTLET PRESSURE	(PSIA)	3781.62	4570.67
FLOWRATE	(LB/SEC)	1244.27804	207.38278
	(GPM)	7841.59	21062.09
INDUCER			
TIP DIAMETER	(IN)	9.49	9.90
TIP SPEED	(FT/SEC)	405.88	1000.86
INLET FLOW VELOCITY	(FT/SEC)	40.56	100.01
FLOW COEFFICIENT	(NONE)	.100	.100
IMPELLER			
TIP DIAMETER	(IN)	15.80	16.36
TIP SPEED	(FT/SEC)	662.97	1655.27
TIP WIDTH	(IN)	.934	.980
HEAD COEFFICIENT	(NONE)	.550	.550
BLADE ANGLE	(DEG)	80.000	80.000
HEAD RISE (OVERALL)	(FT)	7513.89	140512.81
STAGE SPECIFIC SPEED	(RPM*GPM**.5/FT**.75)	1075.11	1055.26
BOOST PUMP			
MINIMUN DELTA P	(PSI)	-22.71	-2.73
HUB/TIP RATIO	(NONE)		
TURBINE		OXIDIZER	FUEL
TYPE	(NONE)	PRESSURE	VELOCITY
# OF STAGES	(NONE)	2.00	2.00
HORSEPOWER	(HP)	22385.37	70000.66
FLOWRATE	(LB/SEC)	82.54862	62.38859
EFFICIENCY	(NONE)	.76000	.60000
PRESSURE RATIO	(NONE)	1.921	10.800
ADMISSION	(FRACTION)	1.000	1.000
VELOCITY RATIO	(NONE)	.311	.178
PITCH DIAMETER	(IN)	32.220	15.818
1ST STG BLADE HEIGHT	(IN)	2.070	.811
2ND STG BLADE HEIGHT	(IN)	3.026	1.705
PITCHLINE VELOCITY	(FT/SEC)	1378.55	1698.93
INLET HUB/TIP RATIO	(NONE)	.879	.891
EXIT HUB/TIP RATIO	(NONE)	.828	.805
TIP SPEED	(FT/SEC)	1508.02	1771.29
BEARING DN*E-6	(MM*RPM)	.769	1.992
ANNULUS AREA*N**2*E-10	((IN*RPM)**2)	2.940	4.540
INLET PRESSURE	(PSIA)	241.36	2537.68
OUTLET PRESSURE	(PSIA)	125.63	241.68
INLET TEMPERATURE	(DEG R)	1147.51	1600.00
OUTLET TEMPERATURE	(DEG R)	1003.26	1147.51
1ST BLADE TEMPERATURE	(DEG R)	1088.22	1370.74
2ND BLADE TEMPERATURE	(DEG R)	1016.03	1158.55

Table 4-3. Turbo-Pump Characteristics

Component	LOX	LH_2
<u>Turbine</u>		
Stages	2	2
Efficiency	0.769	0.590
Horsepower	22385	70000
Tip Speed (ft/sec)	1508.02	1771.29
Inlet Temperature (deg R)	1147	1600
Outlet Temperature (deg R)	1003	1147
Inlet pressure (psia)	241.36	2537.68
Outlet pressure (psia)	125.63	241.68
<u>Pumps</u>		
Stages	1	3
Efficiency	0.759	0.757
Inlet pressure (psia)	65.0	24.5
Outlet pressure (psia)	3782	4571
<u>Inducer</u>		
Tip dia (in)	9.49	9.90
Tip speed (ft/sec)	405.06	1000.86
<u>Impeller</u>		
Tip dia (in)	15.50	16.38
Tip speed (ft/sec)	662.97	1655.27
Stage specific speed (RPM*GPM**0.5/Ft**0.75)	1075	1055

Table 4-4. Estimated Shutdown Times and Propellant Usage During Start/Shutdown

CUTOFF TIME (SECS)	EVENT
0.	Close GG Valves (0.1 to 0.2 sec)
0.	Ramp Main LOX Valve Closed (assume 0.5 sec travel)
0.1	Ramp Main Fuel Valve Closed (assume 0.5 sec travel)

Estimated Propellant Usage During Start/Shutdown

	START	CUTOFF
LOX	2100 lb	500 lb
LH_2	TBD lb	TBD lb

Liquid oxygen enters the engine system through a POGO suppression system located at the entrance of the LOX single stage turbopump which has a dual-discharge volute to minimize radial loads. A small amount of LOX is bled for the tank pressurization; this flow passes through the heat exchanger mounted on the turbine exhaust system to convert LOX to GOX. The rest of the LOX enters the combustion chamber through the injector.

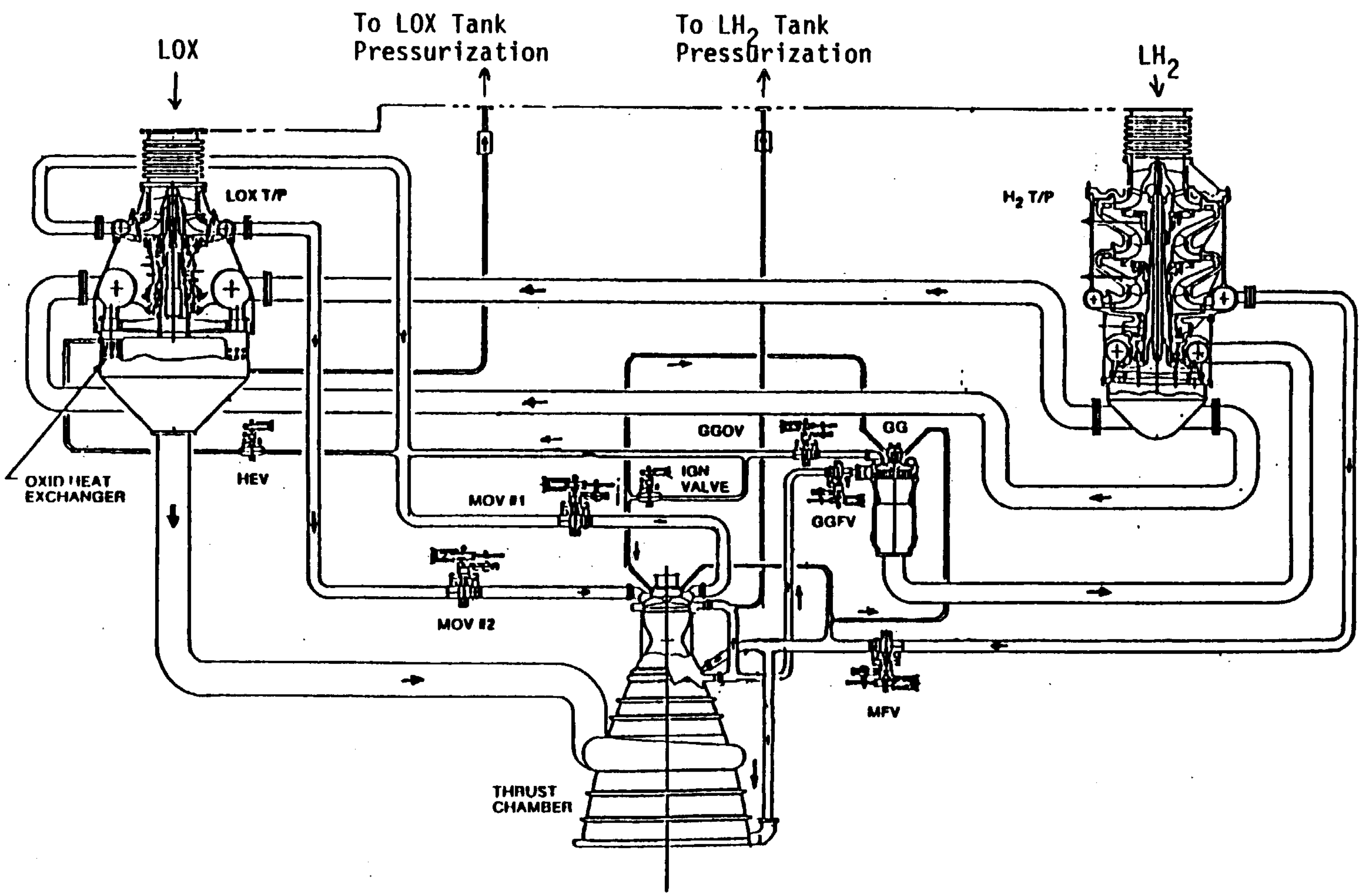


Figure 4-8a. LRG Engine Schematic

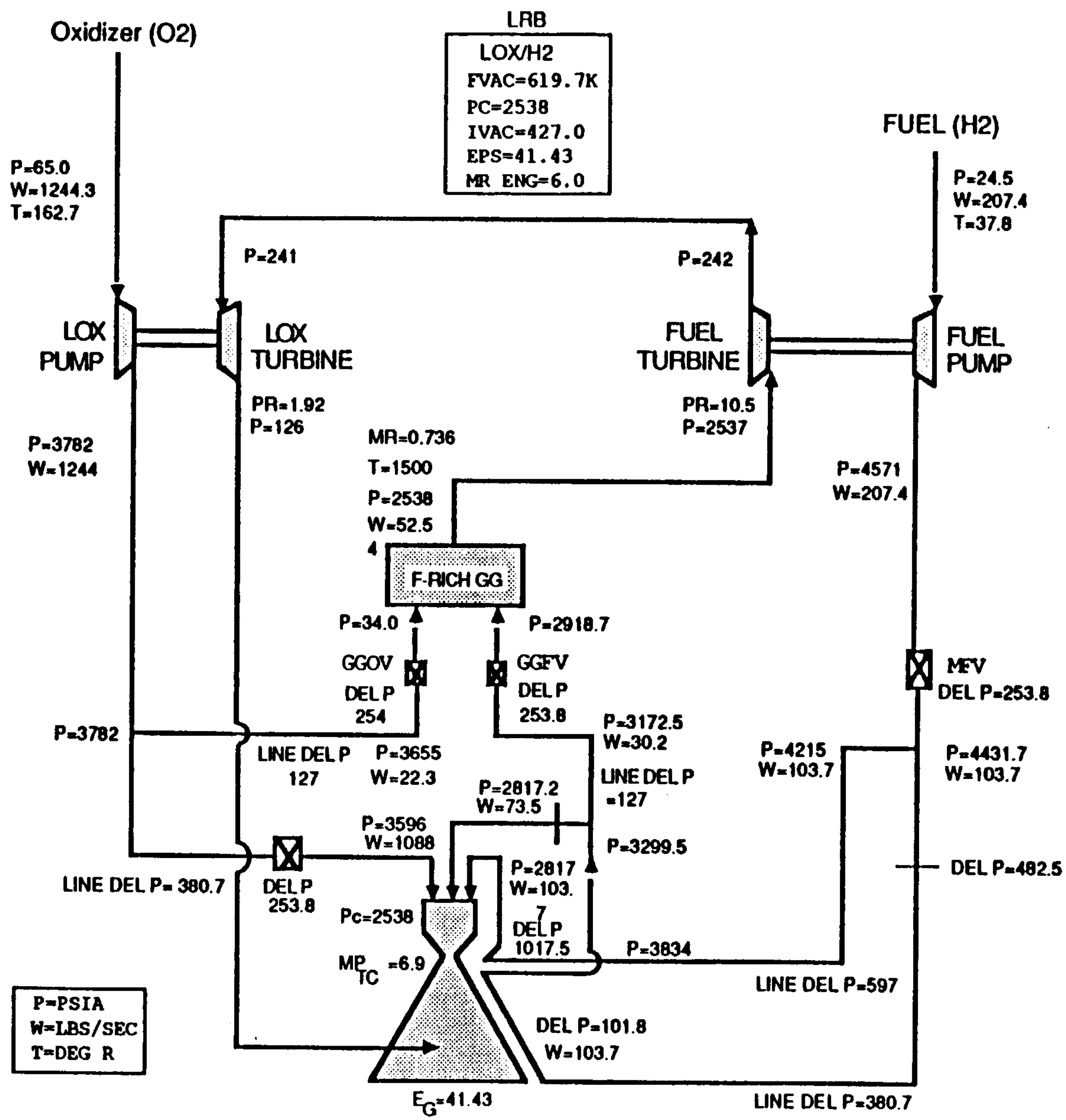


Figure 4-8b. LOX/H_2 LRB Engine Schematic Showing Flowrates, Pressures, and Temperatures at EPL

Purge. A helium purge must be provided downstream of the main hydrogen propellant valve continuously starting just before the hydrogen prevalve is opened to exclude air. A similar bleed is required through the associated turbine and housing for the same reason.

A similar purge, as described above, only using dry nitrogen gas instead of helium is required on the LOX side of the engine. The quantity required of these purge gases is best determined during development testing.

Chilldown. The vehicle must be provided with small helium bleeds just upstream of the propellant pre-valves to prevent geysering during tank loading. Similarily, the engine is provided with bleed lines and valves just upstream of the pump inlets for use during engine chilldown for the same reason. Estimated flowrates, time and total propellant consumption will be determined during engine development testing.

Start. A start transient analysis was performed utilizing a computer program developed for the STME. At engine start, the main fuel valve (MFV), the main oxidizer valve (MOV), the gas generator fuel valve (GGFV), and gas generator oxidizer valve (GGOV), were opened at the times and ramp rates illustrated in Figure 4-9. The resulting chamber pressure and mixture ratio as functions of time are shown in Figure 4-10 and Figure 4-11 respectively. It is expected that LRB LOX/H_2 engine will show very similar analytical characteristics.

The selected method for starting the engine is to utilize a solid propellant gas generator (SPGG). The helium spin start data are included in Figures 4-9, 4-10, and 4-11 for comparison. The SPGG gives a faster and a more stable start, and removes the operational complication of providing a large helium flow from ground support immediately before liftoff with a quick disconnect

Shutdown. The estimated time required and propellants utilized during engine cutoff were given in Table 4-4.

Abort Considerations. The two main propellant valves are provided with pneumatic overrides which permit slamming these valves to the closed

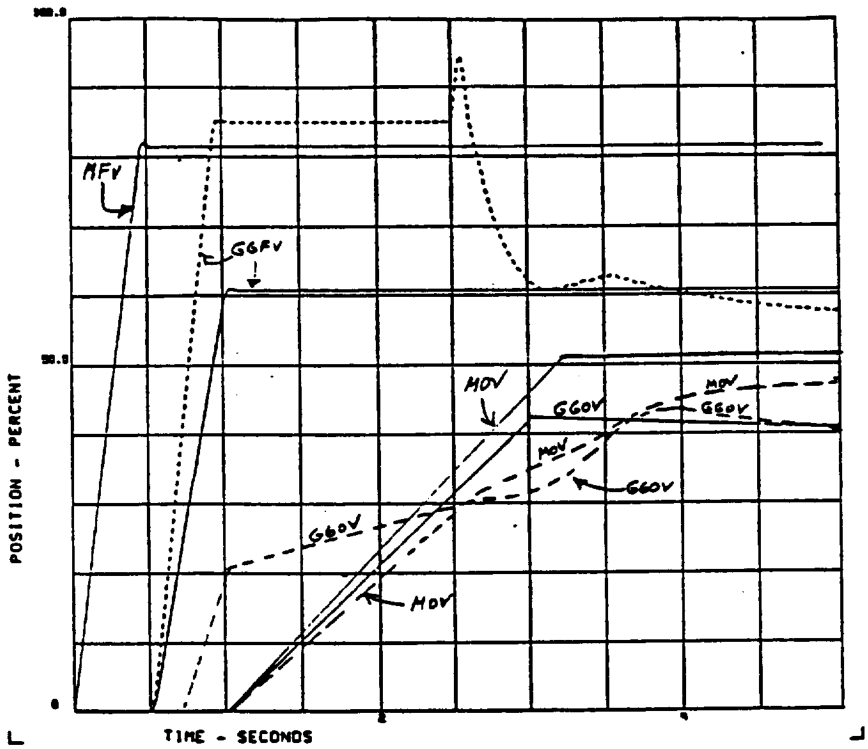


Figure 4-9. EngineStart Valve Position Ramp Rates

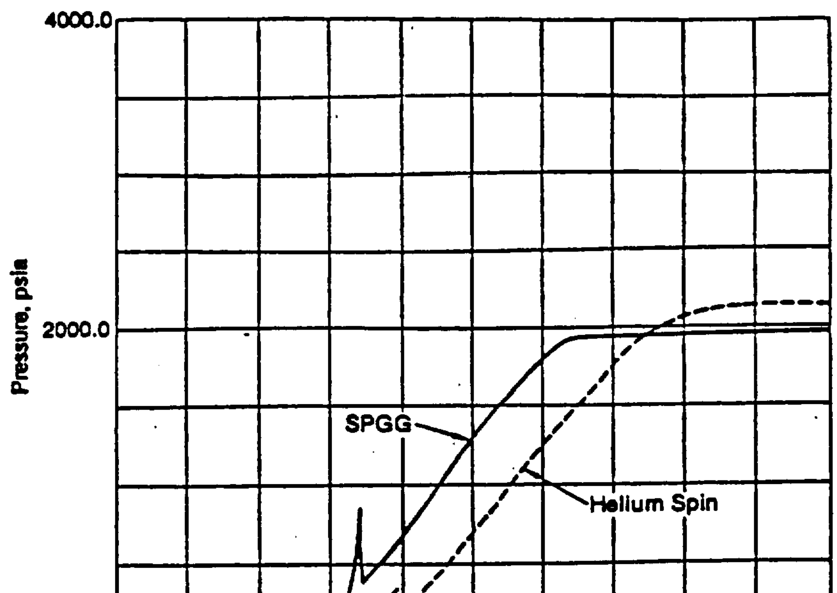


Figure 4-10. Chamber Pressure as a Percent of Nominal During the First Five Seconds of the Start Transient

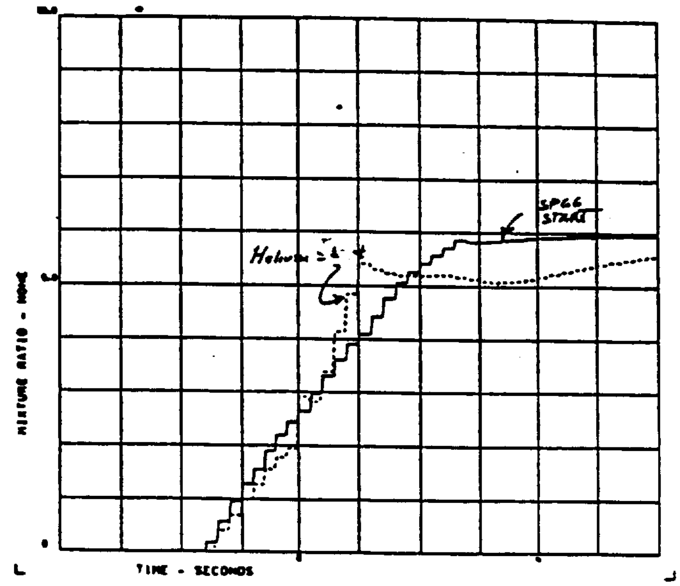


Figure 4-11. LOX/H_2 LRB Engine Mixture Ratio During Start Transient

position. The rate of closure for a quick emergency shutdown will be limited to a valve which will prevent catastrophic failure of the piping system (due to water hammer effects), but will otherwise be as rapid as possible. Simultaneous closure of the prevalves at a similar rate will also ensure that no damage occurs to the upstream equipment should be provided.

The relative timing of the closure of the two main propellant valves will be determined based on minimizing the quantity of unburned propellant which may exit the engine nozzle during shutdown.

4.2.3 Engine Control and Condition-Monitoring System.

The O_2/H_2 LRB control and condition-monitoring system will utilize both derived and measured condition-monitoring instrumentation to determine the overall health of the engine system. The function of the health-monitoring system is to assess the operational capability of the engine. The basic STME engine control system is shown in Figure 4-12 except that the closed loop flow control has been deleted.

Therefore, the LRB control system will operate as follows. The thrust is controlled by changing the flow of LOX to the gas generator which in turn has a temperature control loop which will maintain the GG exhaust gas temperature constant by modulating the GG fuel flow; the GG fuel flow will thus follow the GG LOX flow. This will change the turbine inlet conditions and slow or speed up the pumps producing the required change in chamber pressure and thrust.

Independent of the above, the mixture ratio will be changed to accomodate propellant utilization by slowly opening or closing the main LOX valve slightly utilizing a precise closed loop valve position controller. This valve will be accurately calibrated and its required position calculated from known engine characteristics and from the small change in mixture ratio required as determined by the vehicle central controller in order to accomplish the propellant utilization function.

Ground data processing consists of diagnostics, prognostics, conclusions, and decisions pertaining to engine operational capability. The engine condition monitoring sensors are listed in Table 4-5 with a preliminary list of performance and redline instrumentation are shown in Section 4.2.5. The performance instrumentation is used by the controller to modulate the valve actuators to correspond to a command mixture ratio and thrust level. The LRB control system diagram in Figure 4-12 depicts the basic control concept.

Rocket engine control valves have traditionally been fluid power actuated because of requirements for high speed, coupled with high delta-P forces. Recent studies have shown that electric actuation can provide significant advantages in cost, maintainability, and reliability. A major contributor to these advantages is the high-energy samarium cobalt dc motor.

The overall actuator and valve design goal is minimum weight and simplicity, which complements high reliability. Further, the valve element must be capable of accurate modulation control with minimum force to meet throttling requirements.

The actuator and valve concept illustrated in Figure 2-5, Section 2.2.1 represents a great simplification in complexity and number of detail parts

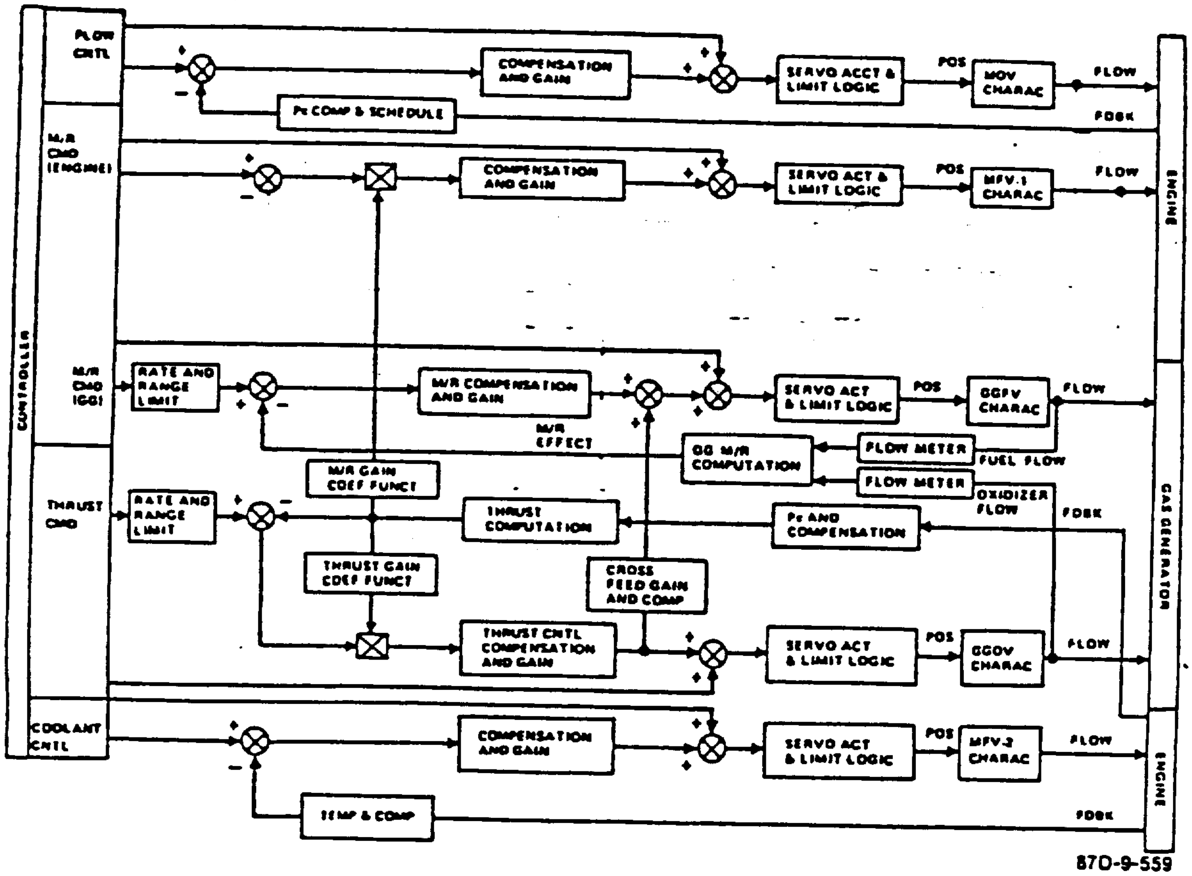


Figure 4-12. STME Closed Loop Control Diagram Modified For Use With the LRB

Table 4-5. Engine Acceptance Testing Condition Monitoring Sensors

Bearing Sets (count)	6
Isotope Wear Analyzer Sets	6
Fiberoptic Bearing Deflectometer	12
Shaft Torque Intervals (count)	3
Torquemeter	3
Plume Combustion Monitors	1
Spectormetric Anamalous Combustion Specie Detector System	1
Spectormetric Mixture Ratio Detector	1
Optical Leak Detector System*	1

*Leak detector system is mounted on the facility, 1 per engine.

compared with hydraulically actuated valves. The valve shown is typical of both main propellant valves. The mechanical drive features all rolling bearings via ball screw and a needle bearing-supported lever/link piece. For fail-safe operation, a pneumatic override actuator decouples the valve shaft from the electric actuator and closes the valve.

All elements of the valve are easily disassembled with primary access through a single flange. With reduced complexity and fewer parts the proposed concept will provide excellent reliability, maintainability, ease of fabrication, and long life with lower cost than other comparably sized valves.

4.2.4 Engine Description

The gas generator cycle engine is designed based upon demonstrated technologies for all of the major components of the engine.

Injector Design. Key injector parameters influencing the design are the propellants (LOX/LH_2), combustion efficiency, chamber pressure, flow rates, propellant injection temperatures, and injection pressures. A coaxial pattern was selected for the injector primarily because this type of element has been successfully used with similar gas/liquid propellant combinations giving high performance, stability, and compatibility. Figure 4-13 shows the main injector conceptual design with the coaxial elements. One thousand and twenty-six elements were incorporated into the design, sized with a LOX post inner diameter (ID) of 0.170 in., an outer diameter (OD) of 0.190 in., and a fuel annulus gap of 0.016 in. The element design selected is shown in Figure 4-13. This element is furnace brazed into the injector body. Coaxial elements projecting beyond the face of the injector are provided to form six baffle compartments. These compartments may be eliminated, if warranted, depending on final stability characteristics determination. A centrally located, dual-spark torch igniter was selected for the ignition source. Inconel 625 materials were selected for the injector elements, injector body, and the manifolding because of its high-strength, brazeability, and weldability characteristics. Regimesh (porous) material is used for the injector face to provide for transpiration cooling between elements.

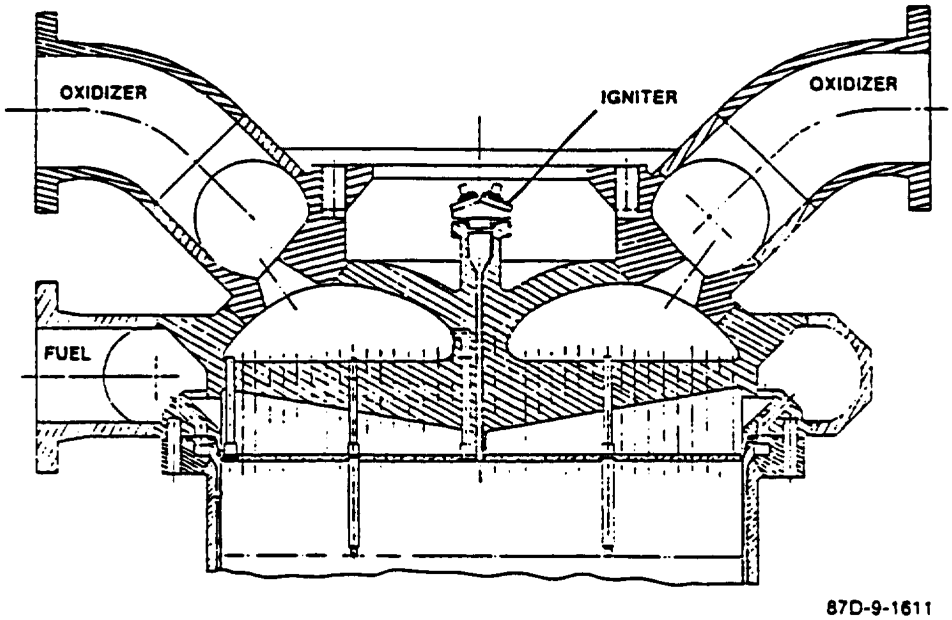
Simplified fuel and oxidizer manifold designs are used for ease of fabrication and minimization of cost. These designs employ constant diameter cross-sectional area flow passages and are sized to minimize flow maldistribution. Two inlets are used to feed the oxidizer manifold and one for the fuel. Ignition will be accomplished with a single, centrally located spark torch igniter with dual spark plugs.

Main Combustion Chamber (MCC) and Nozzle. The basic geometry and operating requirements for these components evolved from the engine balance parameters. The combustion chamber utilizes a channel wall coolant passage approach with a Narloy-Z high-conductivity copper alloy liner. In the design, the channels are machined into the Narloy-Z liner, followed by an electrodeposited (ED) copper closeout (seal) on the back side with support structure of either graphite epoxy or ED nickel cobalt. Again, simplified manifold designs are used for ease of fabrication, low cost, and inspectability. Inconel 625 material was selected for the manifolds.

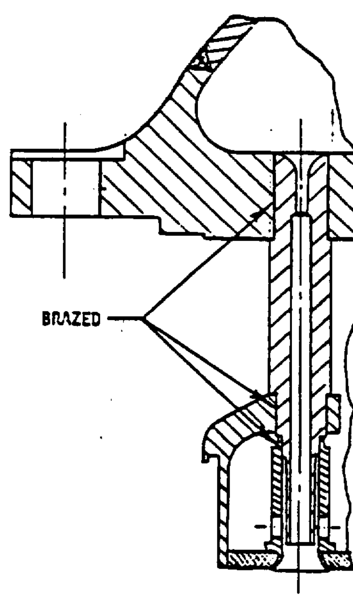
The channel wall combustion chamber structure extends aft to an area ratio of 7:1 where the tube wall coolant passage primary nozzle begins. Selection of this 7:1 transition provides more room for the combustion chamber and nozzle manifolding (compared to the SSME at 5:1) and relaxes the nozzle cooling requirements at the forward end. An up pass coolant circuit was selected for the H_2-cooled combustion chamber and also for the H_2-cooled primary nozzle to simplify the design and minimize the size of the feed line. A simple tube design was selected for the primary nozzle, made from 347 CRES tubes of constant diameter and wall thickness, and formed with simplified tooling. Nozzle reinforcing structure will also be low cost and from 347 CRES or composite materials. The turbine exhaust gas passages through the primary nozzle wall is accomplished by reducing the tube cross section at the appropriate location to form flow passages between tubes.

Gas Generator Design

Some of the key operating requirements that control GG design are type of propellants, chamber pressure, condition of the propellants, and hot gas temperature requirements. A coaxial injector element has been successfully



Injector Cross Section



Coaxial Element

Figure 4-13. LOX/H_2 LRB Injection Assembly and Coaxial Element Conceptual Design used under similar GG operating requirements and, consequently, it was selected for the O_2/H_2 booster engine. High performance, stability, and compatibility are primary attributes of this element concept. The design incorporates coaxial elements, each with a LOX post ID of 0.065 in., an OD of 0.085 in., and a fuel annulus gap of 0.030 in. A brazed element design was selected with all of the elements identical.

Inconel 625 materials were selected for the injector elements, body, and manifolding. This body/manifolding will be fabricated from a casting and hipped to eliminate porosity. A single inlet will be used to supply the fuel and oxidizer manifolds. These inlets will be part of the casting. This single-piece casting will minimize potential high-cost machining and welding operations. Constant cross-sectional area manifolding will be used to reduce fabrication costs. Ignition will be accomplished with a single, centrally located spark torch igniter with dual spark plugs.

The GG combustor has been designed with an Inconel 625 outer-body structure and an H_2-cooled Haynes 188 liner. A choke ring is used to enhance hot gas mixing and provide a uniform temperature gas to the turbine. Other low-cost and design simplification features, such as the manifold simplification techniques and standardized injector elements, have been incorporated into this design.

<u>Turbopump Design Features</u>. The LOX turbopump is a single-stage centrifugal pump with an inducer to provide good suction performance and a vaned diffuser and double-discharge volute to minimize radial loads. The pump is driven by an impulse turbine with a double inlet manifold to minimize duct and torus size. The shaft is supported by two hybrid bearings, each of which combine a hydro- static and two ball bearings. Startup and shutdown transient axial and radial loads are reacted by the ball bearings. At speeds near operating speed, the radial loads are reacted by the hydrostatic bearings and the axial loads are reacted by a balance piston located on the back of the pump impeller. The pump-end hybrid bearing is pressurized with LOX while the turbine-end bearing is pressurized with LH_2. This allows the placement of the LOX seal package between bearings and provides a larger bearing span and minimizes turbine overhang. LOX bearing pressurant is returned to the inlet of the

impeller along with balance piston flow through holes in the impeller. The seal package consists of a face-riding LOX primary seal with a drain between it and a helium-purged floating ring intermediate seal. The intermediate seal is separated from the H_2 floating rings seal by a slinger and drain. A drain is also included between the floating ring seals of the H_2 seal. The H_2 bearing pressurant is routed into the turbine through a labyrinth seal.

The H_2 turbopump consists of three centrifugal stages preceded by an inducer for good suction performance. The stages are connected with radial diffusers and diffusing crossovers. The third-stage impeller discharges into a vaned diffuser and volute to minimize radial loads. The pump is powered by a two-stage, velocity-compounded turbine. The shaft is supported by two externally pressurized hydrostatic bearings. Steady-state axial loads are reacted by a balance piston on the back of the third-stage impeller while transient axial loads are reacted by a single ball bearing. The first two stages are equipped with rear wear rings to aid in balancing thrust. Rear wear ring balance piston flow and bearing coolant/pressurant are recirculated to the inlet of each impeller through holes in the impellers. Part of the turbine-end hydrostatic bearing pressurant discharges past the lift-off seal (open during operation) and then into the turbine through a pair of floating ring seals. The entire H_2 pump assembly is installed in a pressure- containing barrel, thus simplifying assembly and construction.

<u>Main Propellant Valve Design Features</u>.

The electrically activated sector valve concept proposed for the LRB is a significant improvement over prior main propellant valves and were discussed in Section 2.2.

<u>Weight</u>. A breakdown of the engine component weights is given in Table 4-5. These are derived from a computer program based on actual weights of similar components and adjusted for size.

Table 4-6. LRB Engine Component Weight Breakdown Computer Generated (All Values are in Pounds)

TURBOMACHINERY :		
FUEL TURBOPUMP	1160.2	
OXID MAIN TURBOPUMP	980.5	
SUB-TOTAL		2140.7
GAS GENERATOR :		178.0
EXHAUST GAS MANIFOLD :		136.4
THRUST CHAMBER :		
GIMBAL BEARING	124.0	
INJECTOR	670.3	
COMBUSTOR	615.8	
FIXED NOZZLE	944.0	
SUB-TOTAL		2354.2
VALVES AND CONTROLS :		
PROPELLANT VALVES	264.5	
CONTROL VALVES	46.3	
HARNESS AND SENSORS	136.5	
PNEUMATIC CONTROLS	96.6	
HYDRAULIC CONTROLS	33.3	
ATTACH PARTS	153.4	
SUB-TOTAL		730.6
ENGINE SYSTEMS :		
PROPELLANT DUCTS	804.3	
ATTACH PARTS	100.8	
DRAIN LINES	33.1	
I.F. OXID. BLEED LINE	10.3	
I.F. FUEL BLEED LINE	21.8	
I.F. HYDRAULIC LINES	8.7	
I.F. GN2/HE LINES	21.9	
IGNITION LINES AND IGNI,RS	34.0	
PRESSURIZATION SYSTEM	96.9	
SUB-TOTAL		1131.8
ENGINE ACCESSORIES :		
FIXED NOZZLE THERMAL PROTECTION	66.8	
CONTROLLER AND MOUNT	85.0	
POGO SYSTEM	111.5	
SUB-TOTAL		263.3
TOTAL ENGINE DRY WEIGHT W/O ACCESSORIES :		6671.6
TOTAL ENGINE DRY WEIGHT WITH ACCESSORIES :		6934.9

4.2.5 Engine Instrumentation

A preliminary list of engine flight instrumentation is given in Table 4-7. Outputs will be utilized by the health monitoring system to evaluate engine operation based on any instrument indications which may exceed given "red line" values. These red-line values will be established during development testing.

Table 4-7. Preliminary Flight Instrumentation List for the O_2/H_2 LRB Engine

No.	Measurement
1	Engine LOX inlet pressure
2	Engine LOX inlet temperature
3	LOX pump shaft speed
4	LOX pump acceleration
5	LOX pump discharge pressure
6	LOX pump discharge temperature
7	Engine LOX flow rate
8	GGOV inlet pressure
9	GGOV inlet temperature
10	GGOV inlet flow rate
11	GGOV position
12	GG LOX injector pressure
13	GG LOX injector temperature
14	MOV inlet pressure No. 1
15	MOV inlet pressure No. 2
16	MOV position No. 1
17	MOV position No. 2
18	MCC LOX injector pressure
19	MCC LOX injector temperature
20	Engine fuel inlet pressure
21	Engine fuel inlet temperature
22	Fuel pump shaft speed
23	Fuel pump acceleration
24	Fuel pump discharge pressure
25	Fuel pump discharge temperature
26	Engine fuel flow rate
27	MFV position
28	MFV discharge pressure
29	MFV discharge temperature
30	MCC fuel injector pressure
31	MCC fuel injector temperature
32	GGFV inlet flow rate
33	Nozzle coolant discharge pressure
34	Nozzle coolant discharge temperature
35	GG fuel injector pressure
36	GG fuel injector temperature
37	MCC chamber pressure
38	GG chamber pressure
39	GG discharge temperature
40	Fuel turbine inlet pressure
41	Fuel turbine inlet temperature
42	LOX turbine inlet pressure
43	LOX turbine inlet temperature
44	LOX turbine discharge pressure
45	LOX turbine discharge temperature

Table 4-7. Preliminary Flight Instrumentation List for the O_2/H_2 LRB Engine (continued)

No.	Measurement
46	Nozzle turbine gas inlet pressure
47	Nozzle turbine gas inlet temperature
48	Hex GOX outlet temperature
49	Hex GOX outlet pressure
50	Bearing deflection--fuel pump
51	Bearing deflection--LOX pump
52	Fuel pump torque
53	LOX pump torque
54	Fuel turbine blade temperature
55	LOX turbine blade temperature
56	LOX pump intermediate seal purge pressure
57	Fuel system purge pressure
58	LOX dome purge pressure
59	LOX ASI valve position
60	Hex inlet valve position

*ASI stands for Augmented Spark Ignition

LRB TVC Control System Actuator Requirements. The torque and power requirements for the TVC actuators are shown in Table 4-8.

4.2.6 POGO and Stability Analysis

The simplified analysis permitted for this preliminary effort resulted in virtually identical results for the LOX/H_2 engine as that for the LOX/RP-1 engine. For a discussion regarding POGO, see the section with the above-title covering the LOX/RP-1 pump fed engine in this report, Section 3.3.

4.2.7 Failure Mode and Effects Analysis and Reliability Estimate

A quantitative reliability analysis of this engine has not been performed, but reliability histories of Lox/Hydrogen pump fed engines of similar size and requirements are available. Therefore, based on a cursory comparison with those engines that have an established reliability record, the requirement of 0.99 reliability at 90% confidence appears reasonably attainable.

Table 4-8. LRB TVC Torque Breakdown for Gimbaled LOX/H_2 Pump Fed Engine

Name of Contribution	In-lb of Torque	Percent of Total
Moment of Inertia	33,846 in.lb	6 %
Flex Line Stiffness		
LOX LINE	40,904 in.lb	8 %
FUEL LINE	26,954 in.lb	5 %
Thrust Vector Offset	135,727 in.lb	25 %
Gimbal Friction	179,160 in.lb	33 %
Gravity and Accel. at 3 g	126,330 in.lb	23 %
Total =	542,920 in.lb	100 %
Lever Arm =	32 in	
Force Reqd.=	16966.3 lb	

Horse Power at 10 Deg/sec = 14.36 H.P.(input)

Basis:

Engine Thrust = 673231 lb
Engine Mass = 6935 lbm
Ro = Lever Arm = 32 in
CG Distance = 55 in
Frictn.Coef.= 0.06
Thrust Offset 0.25 in

Requirements:

Angular Excursion = + or - 6 Deg
Angular Slewing Rate = 10 Deg/sec
Angular Acceleration = 1 Radian/sec.squared
Propellant Line Pres.= 65 & 25 psia
Nomin.Fuel Line Diam.= 9 in
Nomin.Oxid.Line Diam.= 9 in

A preliminary Failure Mode and Effects Analysis (FMEA) is presented. Because of time constraints and lack of more design details, only major components and assemblies have been addressed. Criticality codes as defined at the end of the FMEA have been assigned to each failure mode.

Specific criteria and ground rules are noted below with component criticality listed in Table 4-9.

4.2.8 Programmatics (Pump Fed Engines)

The engine development plans of the LOX/H_2 engine are essentially identical to the pump fed LOX/RP-1 engine and are presented in Section 3.5 which provides the schedules.

4.3 LOX/LH_2 LIQUID ROCKET BOOSTER PRELIMINARY CONTRACT END ITEM (CEI)

4.3.1 Background

The LOX/H_2 Liquid Rocket Booster (LRB) is being designed to provide booster propulsion for the Space Shuttle. The primary objective of the LRB study was to identify and evaluate a viable LOX/H_2 engine candidate that would meet the requirements for the STS and would have commonality with the Space Transportation Main Engine (STME) currently being studied.

4.3.2 Selected Engine Description

The selected engine configuration utilizes the GG cycle with LOX and LH_2 propellants. LH_2 is used to cool the MCC and nozzle and nozzle after which it is injected into the injector for a small amount that is diverted to the GG where it combines with LOX for the combustion process that produces the turbine drive gas. After passing through the turbines, this gas is dumped into the nozzle.

The bulk of the LH_2 is first used to cool the thrust chamber and is then injected into the MCC as a gas where it combines with LOX for the MMC process, after which it is expanded through the nozzle to produce the engine thrust.

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Helium Spin Start Assembly	Fails to start turbines	Delays engine start	Launch delay. Possible engine-out launch	2
Main Oxidizer Valve	Fails to open at start or only partially opens	Engine cutoff occurs at timer expiration. Prior to cutoff GG overtemp-damage could occur.	Launch abort due to possible engine damage/loss	1
	Closes to intermediate position only.	Oxidizer rich mixture ratio and continued Ox. flow at cutoff could damage thrust chamber.	None due to engine-out capability	2
	Closes prematurely	Automatic shutdown. System switches to engine-out mode of operation.	Single engine-out operation results in longer cluster burn time.	2
	Fails to close at cutoff	None. Shutdown redundancy provided by stage oxidizer prevalve.	None	3

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Main Fuel Valve	Fails to open at start or partial opening only	Engine cutoff occurs at timer expiration and by low P_c pickup switch. Possible ASI erosion.	Possible engine -out start	2
	Internal leakage	None.	None	3
	Fails to close at cutoff	Fuel flow terminated by stage pre - valve. Fuel rich cutoff. Oxidizer valve closing terminates combustion	None	3
	External leak	Will require correction if detected Fire hazard during M/S partially mitigated by inert atmosphere in boattail .	Launch delay. Switch to engine-out mode of operation if detected.	2
			Mission loss if not detected	1

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Fuel Turbopump	Fails to start	Engine cutoff occurs at timer expiration. Prior to cutoff GG overtemp may occur.	Engine out operation	2
	Internal shaft seal leak	Turbopump seals contamination and icing cause binding and delayed start	Launch delay or engine-out operation	2
	Pump stall	Oxidizer rich composition resulting in Gas Generator, Turbine and Thrust chamber erosion. Possible engine shutdown.	Possible engine-out operation.	2
	External hot gas leakage	Degraded engine performance. Possible engine shutdown due to low chamber pressure. Possible fire.	Possible mission loss	1
	Turbine seal leak	None. Redundant seals protect against hot-gas leakage.	None	3

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Fuel Turbopump (Continued)	Pump imbalance	Possible piece part failure and subsequent engine loss	Engine-out operation	2
	Overspeed, cavitation	High vibration and possible piece part failure. Engine shutdown.	Engine-out operation	2
	Failure to remain in operation during mainstage.	Thrust decay, low P_c, automatic engine cutoff.	Engine-out operation	2
Oxidizer Turbopump	Failure to start	Engine cutoff occurs at time expiration. Prior to cutoff GG overtemp could occur.	Engine-out operation	2
	Overspeed, cavitation	High vibration and possible piece part failure	Engine shutdown	2
	Internal shaft seal leak	Turbopump seals contamination and icing cause binding and delayed start	Launch delay or engine out operation	2

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Oxidizer Turbopump (continued)	Failure to remain in operation during mainstage.	Thrust decay, low P_c, automatic engine cutoff	Engine-out operation	2
	External hot gas Leakage	Degraded engine performance. Possible engine shutdown due to low P_c. Possible fire	Possible mission loss	1
	Pump imbalance	Possible piece part failure and engine loss	Engine-out operation	2
	Pump stall	Fuel rich composition. Low performance. Engine shutdown if not corrected quickly.	Possible engine out operation.	2
Gas Generator	Hot gas overtemp.	Automatic engine cutoff	Engine-out operation	2
Gas Generator Oxidizer Valve (GGOV)	Fails to open	Delayed start	Launch delay or engine-out launch	2

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
GGOV (continued)	Erratic response to modulation commands	Erratic LOX flow to GG. Thrust control and/or engine throttling capability lost. Engine shutdown.	Single engine-out results in longer burn time	2
Gas Generator Fuel Valve (GGFV)	Fails to open	Delayed start	Launch delay or engine-out launch	2
	Erratic response to to modulation commands.	Erratic fuel flow to GG. Loss of Mixture ration control, engine performance decay. Engine shutdown	Single engine-out results in longer systems burn time	2
Gas Generator Igniters	Fail to ignite	None due to redundancy	None	3
Heat Exchanger Valve (HEV)	Fails to open	No LOX flow to Heat Exchanger. Controller initiates engine shutdown.	Launch delay	1
	Fails to close or leaks internally	LOX flow continues to Hex and POGO accumulator	None	3

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
HEV (continued)	Restricted LOX flow or shutoff	Pressurant loss to POGO accumulator and to vehicle LOX tank. Engine shutdown	Engine-out operation	2
Heat Exchanger	Coil leakage	Cold LOX and fuel rich hot turbine exhaust gas mix. Hot spot on Hex. Engine shutdown	Possible vehicle / mission loss.	1
	Plugging or LOX flow restriction	Over heating or reduced LOX flow may cause rupture. No pressurant flow to LOX tank & POGO accumulator Engine shutdown	Engine-out operation	2
POGO system	Fails to operate or or maintain operability	Automatic shutdown.	Launch delay at start and engine-out operation after launch	2

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Thrust Chamber Assembly	Injector oxidizer post deformation	Localized oxidizer-rich combustion and thrust chamber damage. Performance loss.	Engine-out operation	2
	External fuel leak MCC to nozzle joint	Possible fire and loss of engine or adjacent equipment damage	Possible loss of vehicle or mission. Engine-out capability may save mission	1
	Internal fuel leakage to (hot side) in MCC	Reduced MCC cooling. Possible local MCC overheating	Possible loss of vehicle or mission. Engine-out capability possible	1
	External fuel leak at the nozzle	Possible fire and loss of engine or adjacent equipment damage	Possible loss of vehicle or mission. Engine-out capability may save mission	1
	Internal fuel leakage to (hot side) at the nozzle	Reduced nozzle cooling. Possible local overheating	None	3
			launch	

Table 4-9. Preliminary Failure Mode and Effects Analysis LOX/Hydrogen Pump Fed LRB

COMPONENT	FAILURE MODE	EFFECT ON ENGINE	EFFECT ON MISSION	CRITICALITY
Augmented Spark Igniter (ASI)	Fails to ignite	MCC pressure gases do not ignite. Ignition confirmed limits not satisfied due to low chamber pressure. Engine shutdown.	Engine-out operation	2
	Loss of fuel to ASI	High mixture ratio erosion of ASI combustion chamber walls, possible manifold invasion, injector burnout and LOX rich operation resulting in uncontained engine damage	Possible vehicle loss (unless engine-out operation can be initiated before spread of damage)	1
	Blockage of ASI LOX passage	Localized LOX-rich operation, erosion of ASI combustion chamber walls, manifold invasion and injector burn-out, fire. Uncontained engine dam - age.	Possible vehicle loss (Unles engine-out operaton can be initiated before spread of damage.	1

Table 4-9. Preiminary Failure Mode and Effects Analysis
LOX/Hydrogen Pump Fed LRB (concluded)

Criticality 1

1. Hot gas leakage is assumed to always result in structural/functional damage to at least one engine.

2. Hot gas mixing with LOX is a potential fire/explosion hazard.

3. Oxidizer rich cutoffs always offer the potential for structural damage.

4. Structural failure of rotating machinery or rupture of pressure containment boundaries can both propagate to destruction of one or more engines, followed by loss of life or vehicle.

5. Spark generation in a LOX environment, such as rubbing/fretting of parts in oxidizer pumps or valving, will escalate to a fire/explosion.

Criticality 2

1. Leakage of propellants during start of mainstage is considered as being detectable by hazardous gas monitors or other instrumentation to permit safe engine shutdown. The worst possible scenario of potential mission loss, however, is assigned for conservativeness.

2. Failures precipitating safe engine shutdown. The vehicle is capable of achieving mission success with one engine not operating; however, it is presumed that launch abort, followed by safe shutdown, will be commanded if one engine is not operating prior to liftoff.

Criticality 3

1. External leakage of propellants during preconditioning is assumed to be detected by ambient hazardous gas monitors, which will be cause for launch abort.

2. All others.

4.3.3 LRB CEI Requirements

This document presents the preliminary CEI requirements that the LRB engine must fulfill to satisfy the requirements for the STS. These requirements are as follows:

Performance. All performance values stated herein are nominal values. The minimum and maximum values will be determined during subsequent study efforts.

1) Engine Thrust - The LRB shall be capable of producing 563,000-lb vacuum thrust at the normal power level (NPL) and 619,000-lb vacuum thrust at the emergency power level (EPL). The engine shall be capable of throttling up from NPL to EPL in TBD seconds. Throttling to the minimum power level (422,600) shall be provided in the engine design.

2) Specific Impulse - The specific impulse for the STME shall be as follows for the two vacuum equivalent thrust operating points:

Thrust Level	Sea Level Is (seconds)	Altitude Is (seconds)
563,000 lb	273.9 ± TBD	427.0 ± TBD
619,000 lb	369.4 ± TBD	427.9 ± TBD
422,600 lb	351.7 ± TBD	430.0 ± TBD

3) Main Combustion Chamber (MCC) Propellants

Propellants	Injected State
Oxidizer - Oxygen (LO_2)	Liquid
Fuel - Hydrogen(H_2)	Gas

4) GG Propellants

Propellants	Injected State
Oxidizer - Oxygen (LO_2)	Liquid
Fuel - Hydrogen(H_2)	Gas
MCC MR - O/F NPL	O/F EPL

5) Engine MR - The engine MR for the LOX/H_2 LRB shall be as follows for the two thrust operating points:

Thrust Level	Mixture Ratio
563,000 lb (vac)	6.0
619,000 lb (vac)	6.0
422,600 lb (vac)	6.0

The engine shall be equipped with a closed loop engine MR control system capable of controlling MR within ± 1.0% of the nominal value.

6) Acceptance Calibration - The acceptance calibration for the LRB shall be as follows:

Thrust	(NPL)	- 563,000 lb ± 3%
	(EPL)	- 619,000 lb ± 3%
MR	(NPL)	- 6.0 ± 1%
	(EPL)	- 6.0 ± 1%

7) Coolants - The coolants for the MCC and nozzle shall be LH_2.

8) Burn Duration - The LRB shall be capable of maximum burn duration of 180 sec at NPL and EPL.

9) Uncoupled Thrust Oscillations - The engine-produced uncoupled oscillatory thrust shall be no greater than the following for the respective specified frequency ranges:

R = 0 to 0.5 H_z	F = ± 6000 lb
R = 0.5 to 1.5 H_z	F = ± 1500 lb
R = 1.5 to 2.5 H_z	F = ± 450 lb
R = 2.5 to 100 H_z	F = ± 1500 lb

For the purpose of performing data analysis to verify engine compliance in the critical frequency range oscillatory shall be defined as the average value of an oscillation over at least 16 cycles.

10) Combustion Stability - The engine-produced main chamber pressure oscillations shall not exceed ± 5% of the mean steady-state pressure.

11) Damping time for artificially induced pressure spires shall be TBD milliseconds maximum.

12) POGO Suppression - The engine shall provide a POGO suppression system in accordance with the following requirements (TBD).

13) Engine Controller - The electrical closed loop engine control system shall be capable of continuous operation at ambient temperature for an unlimited period of time during checkout and maintenance.

14) System Checkout and Monitoring Capability - The design shall include onboard checkout capability, redundancy verification, and status monitoring during ground operations. The engine design shall include a limit control system capable of automatically initiating engine shutdown to prevent catastrophic failure.

Operations. The operational requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Engine Start - The engine start system shall have self-contained control within the engine envelope. The start sequence shall be started by a single electrical signal from the vehicle or ground source.

2) The engine shall be capable of one start after each ground servicing.

3) The engine start sequence shall be capable of achieving normal power level (NPL) thrust in less than 5 sec.

4) The thrust buildup rate shall not exceed TBD lb thrust in any 10-msec time period.

5) Starting Impulse - The starting thrust impulse to NPL shall not exceed TBD lb-sec.

6) Throttling Control - The engine shall be equipped with a closed loop thrust control system capable of raising the thrust at NPL to the specified thrust at EPL in the event of an engine condition-out during a vehicle launch.

    a) Throttle Rate - The engine thrust control system shall be capable of raising the engine thrust from NPL to EPL at the rate of TBD lb-sec any time after reaching NPL. Also throttling to the minimum power level shall be provided at a rate TBD.

    b) The thrust control system shall be capable of a step response of TBD lb thrust increase in less than TBD sec after a step command.

Engine Shutdown. The engine shall be capable of a safe shutdown from any power level including the start sequence.

1) The engine shutdown sequence shall be capable of reducing thrust from NPL to zero in TBD sec.

2) The shutdown impulse shall not exceed TBD lb/sec from NPL.

3) The engine shall be capable of shutdown from any defined thrust level upon receipt of an electrical command at a rate of TBD lb thrust change per any 10-msec time interval.

Environmental Conditions. The engine shall be capable of operating safely under the following conditions:

1) The engine shall be capable of operating safely where exposed to a hat flux of TBD Btu/ft^2-sec and a surface temperature of TBD°F. The heat transfer coefficient that shall be used for design is TBD Btu/sec-ft^2°F.

2) The surface temperature of lines or surface in contact with cryogenic propellants shall be controlled to preclude the formation of liquid air.

3) Acceleration Loads - TBD

4) Shock Loads - TBD

5) Ground Handling and Transportation Loads - TBD

6) Storage Life - The engine shall be capable of being transported and stored over an ambient temperature range of TBD°F to TBD°F, an ambient pressure range of TBD psig to TBD psi, a relative humidity of 100% at temperatures less than or equal to TBD°F.

   a) The engine shall suffer no degradation of reliability or operating life during the storage period, subject to the inspection and maintenance requirements TBD.

7) Exposure - The engine system and components shall be capable of being transported and stored without deterioration in areas where conditions may be encountered having salt spray and relative humidity as experienced in coastal regions. The engine system and components shall be capable of withstanding exposure to sand and dust when equipped with proper closures.

8) Lightning - The engine controller shall be designed to operate without damage in accordance with TBD lightning protection criteria.

Prelaunch. The engine shall be designed for minimum prelaunch servicing.

1) Ground Service - The engine shall be capable of achieving pre-launch thermal conditioning without ground servicing in less than TBD minutes from the time propellants are supplied to the engine. Recirculation flow rates to achieve thermal conditioning are as follows:

   LOX - TBD lb/sec
   LH_2 - TBD lb/sec

2) The engine shall be capable of servicing and maintenance while in either the horizontal or vertical position.

3) The engine shall not require any servicing from ground equipment within 24 hr after propellants are loaded.

4) External or internal leakage of propellants shall not occur in such a manner as to impair or endanger the engine/vehicle function. Leadage monitoring capability shall be provided with the design objective that separable connections not exceed 1 x 10^{-4} sec helium at leak check pressure.

5) The engine shall not require any monitored redlines external to the engine prestart and shall provide a continuous engine-ready signal to the vehicle when all critical parameters monitored by the engine control system are within TBD conditions.

Interface. The engine shall require the following conditions at the respective interfaces with the vehicle:

1) Propellant inlet conditions at engine start:

   a) LOX - 65 psia to TBD psia, 163 to 170°R
   b) LH_2 - 25 psia to TBD psia, 38 to 40°R

2) Propellant inlet conditions during mainstage:

   a) LOX - 65 psia to TBD psia, TBD to TBD°R
   b) LH_2 - 25 psia to TBD psia, TBD to TBD°R

3) Electrical

   a) The engine shall be supplied TBD dc V
   b) The engine shall be supplied TBD ac V
   c) The controller shall be engine supplied and mounted.

4) Pressurization Gas - The engine shall provide GOX to pressurize the vehicle oxygen tank and GH_2 to pressurize the H_2 tank.

5) Purge Requirements - Nitrogen, in accordance with MIL-P-27401, and helium, in accordance with MIL-P-27407, shall be used for operational and servicing purges and leakage tests.

   a) Operational Purges - TBD
   b) Servicing Purges - TBD

6) Digital Interface

   a) A suitable digital interface shall be provided for vehicle commands to the engine.

Physical Requirements. The physical requirements presented herein are preliminary and represent nominal values. The maximum and minimum values will be determined during subsequent study efforts.

1) Envelope - the maximum engine width is 108 in. and the engine height is 135 in.

2) Weight - The engine weight is as follows:

	Dry	Wet
Basic engine	6,671 lb	TBD
Accessories	263 lb	TBD
Thermal Insulation	6,934 lb	TBD

3) Gimbaling - The engine shall be capable gimbaling in a ± 6° square pattern at a gimbal rate of 10°/sec and an acceleration rate of 1.0 rad/sec squared. The engine shall provide attach points for the vehicle-furnished actuators. The gimbal system shall be capable of returning the engine to null position at engine shutdown.

4) Engine Alignment - The engine shall be aligned so that the actual thrust vector is within 30 min of an arc to the engine centerline and within 0.25 in. of the gimbal center. The gimbal center shall be within 0.010 in. of the engine centerline.

5) Engine Fluid Interface Ducts and Lines - The engine shall supply all interface ducts and lines with a minimum of TBD in. straight section upstream of the engine interface plane.

6) Engine Electrical Interface - All engine electrical connections from the vehicle shall be located in a single, engine-mounted panel.

Reliability. The reliability of the configuration upon which the final flight certification is based shall be that which is necessary to ensure functioning within the specified design life.

1) The engine design life is 1 mission at EPL.

2) Fail-Safe Design - The engine shall be capable of shutdown from an internal signal without damage to other systems.

3) Structural Criteria - The engine shall be designed to provide the following minimum factors of safety:

Minimum yield	-	1.1
Minimum ultimate	-	1.4 combined loads
Minimum ultimate	-	1.5 pressure only
Minimum proof	-	1.2 times EPL operating conditions, unless fracture mechanics requires a higher factor

Low cycle fatigue - 4.0
High cycle fatigue - 10.0

Note: Components should be designed for 1.25 on endurance limit where feasible

<u>Diagnostic Monitoring</u>. The engine shall be capable of self-diagnostics in real time. Unsafe conditions shall cause an engine-generated shutdown unless inhibited by the vehicle.

1) Diagnostic data will be recorded for postflight analysis.

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RI/RD89-112

LIQUID ROCKET BOOSTER
PHASE III STUDY REPORT

18 JANUARY 1989

PREPARED BY

ROCKETDYNE DIVISION
ADVANCED LAUNCH SYSTEMS

CONTRACT NUMBER
08-01290

LIQUID ROCKET BOOSTER PHASE III STUDY REPORT

FOREWORD

This Phase III report, containing results of the extension to the Liquid Rocket Booster system Feasibility Study is submitted to General Dynamics Space Systems Division (GDSSD) in accordance with General Dynamics contract 08-01290. This program was conducted under the direction of GDSS program manager Paul Bialla and Propulsion Project Manager Gopal Mehta. This document describes the results of a Liquid Rocket Booster engine study conducted during Phase I and II and extended to carry design analysis studies to a further degree of detail.

Specific engine development and production costs are not included in this report due to their proprietary nature; however, they have been submitted to General Dynamics under separate cover.

ABSTRACT

Phase III of the Liquid Rocket Booster Study was conducted over a five month period by Rocketdyne. Three engine types were compared: 1) LOX/RP-1 pressure fed, 2) LOX/RP-1 pump fed, and 3) LOX/H_2 pump fed. For the pressure fed engine, trade studies were conducted to determine the influence of chamber pressure on engine performance, stability, and cost as well as its influence on the vehicle's tendency toward POGO. Technology items for the pressure fed engine are identified.

The acoustic pressures generated by the above engine systems were compared in a preliminary way with the Solid Rocket Booster (SRB) based on analytical studies. Various engine control system options were also compared.

New engine balances and new design sketches of the above engines were generated based on final engine requirements generated by GDSS.

The engine of choice, at the conclusion of this effort, is the LOX/H_2 pump fed engine of a design almost identical with the STME engine.

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ACKNOWLEDGEMENTS

The following individuals have made significant contributions to the work shown in this report:

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Alex J. Sobin - Project Manager
John J. Vrolyk - Engine System Analysis
Angelo DiBlasi - Engine Programmatics
Chris Erickson - Engine Performance Parameters
Jim Fenwick - POGO System Analysis
Jim Fang - Combustion Stability Analysis
Wally A. Geniec - Engine Performance Parameters
Ben Gutierrez - Acoustic Analysis
Judie Goldstein - Report Coordinator and Production
Irvin Kaith - Engine System Design
Joseph H. Lavender - Cost Parametrics
Rich Nelson - Engine Transient Dynamics
Duc G. Nguyen - Engine Performance Parameters
Hao V. Nguyen - Engine System Design
Paul Novacek - Reliability and Safety
Dipankar Roy - Acoustic Analysis
Roland Szabo - Acoustic Analysis
Wen Hsiung Tu - Engine Parametrics

LIQUID ROCKET BOOSTER PHASE III REPORT

1.0 INTRODUCTION

The use of liquid rocket boosters (LRB) for the Space Shuttle is being investigated in detail by General Dynamics Space Systems Division. Rocketdyne, under subcontract to GDSS, is studying pressure fed and pump fed propulsion systems which may be applied to Space Shuttle booster propulsion. The initial effort covered parametric performance, weight, and cost data for a range of propellant combinations and engine thrust levels and chamber pressures. Parametric data from the Rocketdyne 1972 Phase A/B Pressure Fed Space Shuttle Study and the 1986 AFRPL Low Cost Expendable Propulsion Study (LCEPS) were examined for applicability.

1.1 PHASE I AND PHASE II STUDIES REVIEW

Following the initial parametric studies, trade studies were conducted to define the basic elements and features of the engines. These studies covered selection of cooling method, injector type, thrust vector control system, ignition method, and basic engine control method. Engine candidates for both pressure and pump fed applications were formulated based upon the results of the trade studies. Emphasis was placed on expendable engines based on the results of GDSS studies.

Conceptual engine design layouts for both ablative type and regeneratively cooled pressure fed thrust chambers were prepared showing general constructive details for the major engine elements. Emphasis was placed on LOX/RP-1 propellants since they were the propellants of choice based on General Dynamics trade studies. A safety and reliability analysis was conducted to compare ablative and regenerative type chambers with the conclusion that both types of chambers could be developed to have a high degree of safety and reliability. Regeneratively cooled pressure fed engines were chosen based on a careful evaluation of factors such as experience base, overall safety, etc., and on the results of the General Dynamics trade study.

Hydrocarbon pump fed engine defintions were based on the ongoing STBE (Space Transportation Booster Engine) studies except that RP-1 was used as the fuel rather than methane. GDSS selected RP-1 and hydrogen based on overall trade studies of size, cost, experience base, etc. For the pressure fed engines a complete list of engine feature options was developed and trade studies were conducted in order to define the most desirable expendable engine features.

1.2 PHASE III STUDY APPROACH

Initial work consisted of generating parametric engine data to permit comparison of the following three engines: 1) a pressure fed LOX/RP-1 engine, 2) a pump fed LOX/RP-1 engine (similar to an STBE design), and 3) a pump fed LOX/LH_2 engine (similar to an STME design). Since many of the pump fed engine characterizations were at hand as a result of other program efforts, they were used as background for this study.

Since the reduction of propellant tank pressure for the pressure fed concept was an important goal, the factors to be considered in holding the injector pressure drop and the engine combustion chamber pressure to a minimum were addressed, especially as related to the future program efforts which would be required to stabilize the engine. The relation between engine design parameters, performance, and stability margin were generated for a pressure fed engine.

Similarly, the influence of lowering both the injector pressure drop and the chamber pressure upon the POGO stability predicted for the pressure fed vehicle were studied to determine the approximate size and weight of the stabilization hardware required.

The relative complexity and cost of the control systems associated with the three engines were determined and the type of control suitable for each was recommended. A detailed sound pressure level study (acoustical) was conducted which compared the predicted levels for the LRB engines with that of the existing solid rocket motors.

Once the choice of the LOX/LH_2 pump fed LRB baseline engine was made by GDSS, engine balances were generated for this engine to finalize characterization for the selected engine of choice.

1.3 SCOPE

This study effort focused on the following issues as well as to provide GDSS with assistance in other areas as required.

The minimum Pc and minimum ΔP for combustion stability in LOX/RP-1 pressure-fed engines will be established. Consideration of possible "Pogo" effects will be included. Low P_c and ΔP are desired to keep propellant tank pressures as low as possible. However, combustion stability and very low frequency feed system coupled instability (POGO) can be more easily induced as P_c and ΔP are lowered. Combustion stability analyses will be made to determine P_c and ΔP minimum requirements for stable combustion, and the risk of encountering a POGO induced instability will be determined.

Rocketdyne will advise GDSS on engine requirements for technical demonstration testing at MSFC on an F-1 test stand, both for the pressure-fed F-1 and the new thrust chamber. The pre-designs of 2 selected pump-fed engine concepts for LRB (LOX/RP-1 and LOX/LH_2) will be optimized. These pump-fed engines will be modified to reflect changes in system requirements. The chamber pressure level and other design characteristics will be re-examined and modified to improve overall engine performance and reduce weight and cost.

When improved pre-designs of the selected engines have been completed, their development and production costs will be re-evaluated. These engines are: 1) LOX/RP-1 pressured-fed, 2) LOX/RP-1 pump-fed, and 3) LOX/H2 pump-fed engines, all throttlable to the required level.

2.0 LRB ENGINES USING LOX/RP-1 PROPELLANTS

This section describes the work performed during this Phase II period on: 1) the LOX/RP-1 pressure-fed engine, and 2) the LOX/RP-1 pump-fed engine.

2.1 LOX/RP-1 PRESSURE FED ENGINE

A simplified schematic and engine layout of the LOX/RP-1 pressure fed engine are shown in Figures 2-1 and 2-2 respectively for reference. The characteristics for this engine are shown in Table 2-1.

Table 2-1. LRB LOX/RP-1 Pressure Fed Engine Characteristics

ENGINE PARAMETERS	NOMINAL THRUST	MINIMUM THRUST
Weight (lb)	7017	
Throttle (%) of vacuum thrust	100.0	60.0*
Oxidizer flow rate (lb/sec)	2433.0	1470.3
Fuel flow rate (lb/sec)	973.2	588.1
Vacuum thrust (lb)	971595	582,957*
Sea level thrust (lb)	841482	452542
Chamber Pressure (psia)	334.0	200.4
Vacuum Isp (sec)	285.2	283.2
Sea level Isp (sec)	247.0	220.0
Mixture ratio	2.5	2.5
Nozzle area ratio	4.96	
Area (in2)	8854	
Throat radius (in)	23.84	
Exit diameter (in)	106.2	
Overall length (in)	205.5	
*60% of the Nominal Vacuum Thrust		

2.1.1 Startup and Shutdown Transients

The startup and shutdown transient flow rates were estimated for a pressure fed LOX/RP-1 engine of the above type. The results are shown graphically in Figures 2-3 and 2-4. Since chamber pressure is an accurate indication of engine thrust level, this data permitted the thrust transient for LRB to be calculated and its effect on the STS vehicle dynamics just before lift-off to be estimated by GDSS.

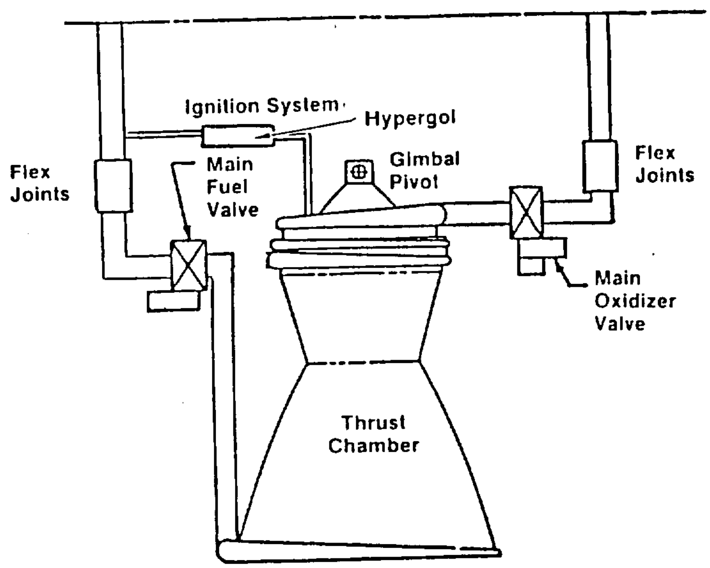

Figure 2-1. Simplified Schematic of the LOX/RP-1 Pressure Fed Engine

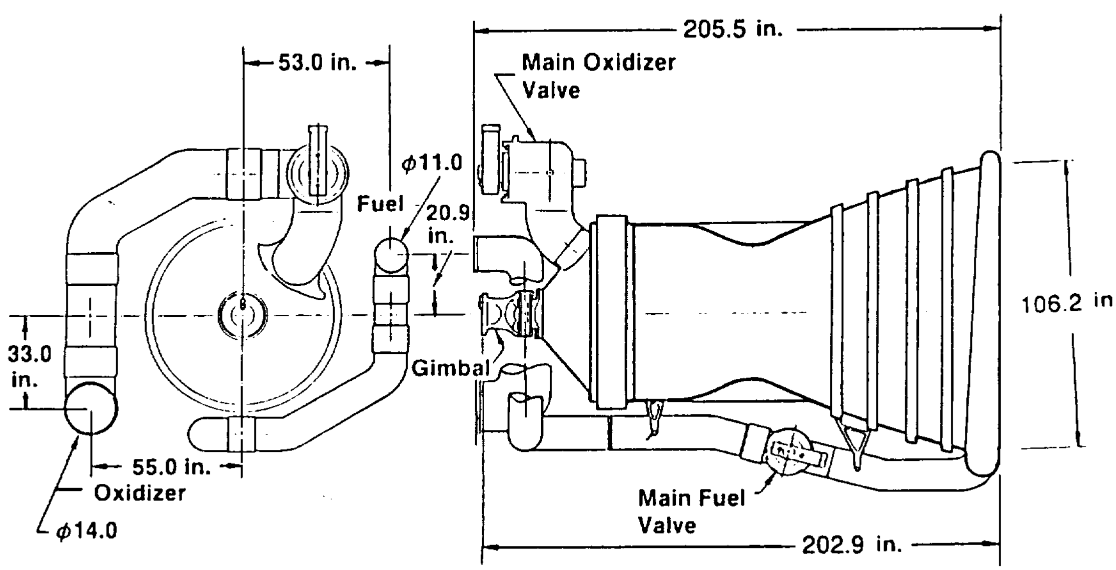

Figure 2-2. Top and Side Views of the LOX/RP-1 Pressure Fed Engine

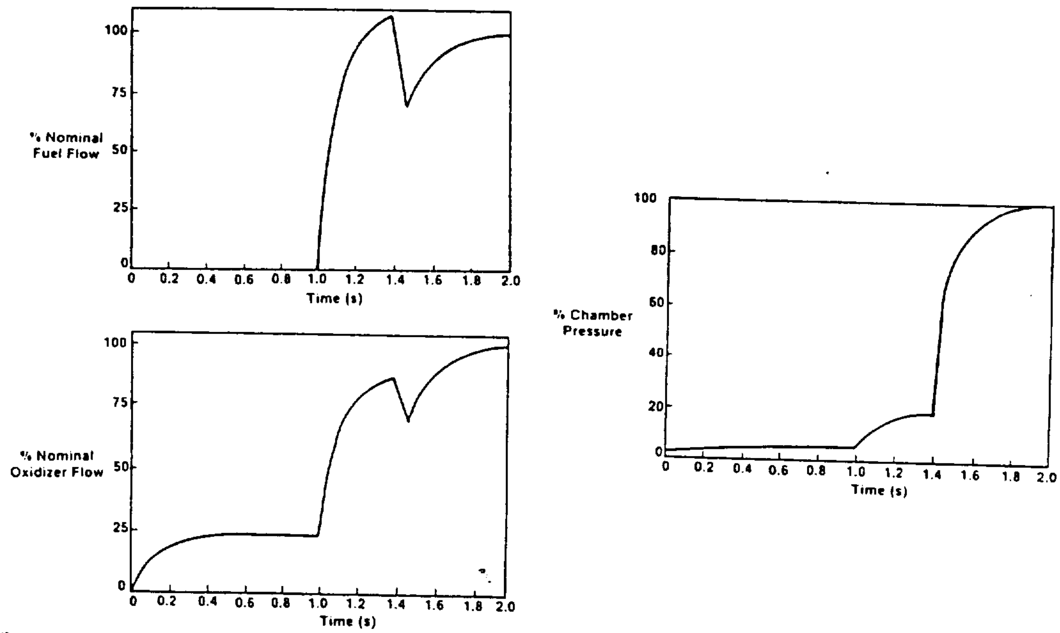

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Figure 2-3. Pressure Fed LOX/RP-1 Start Transients

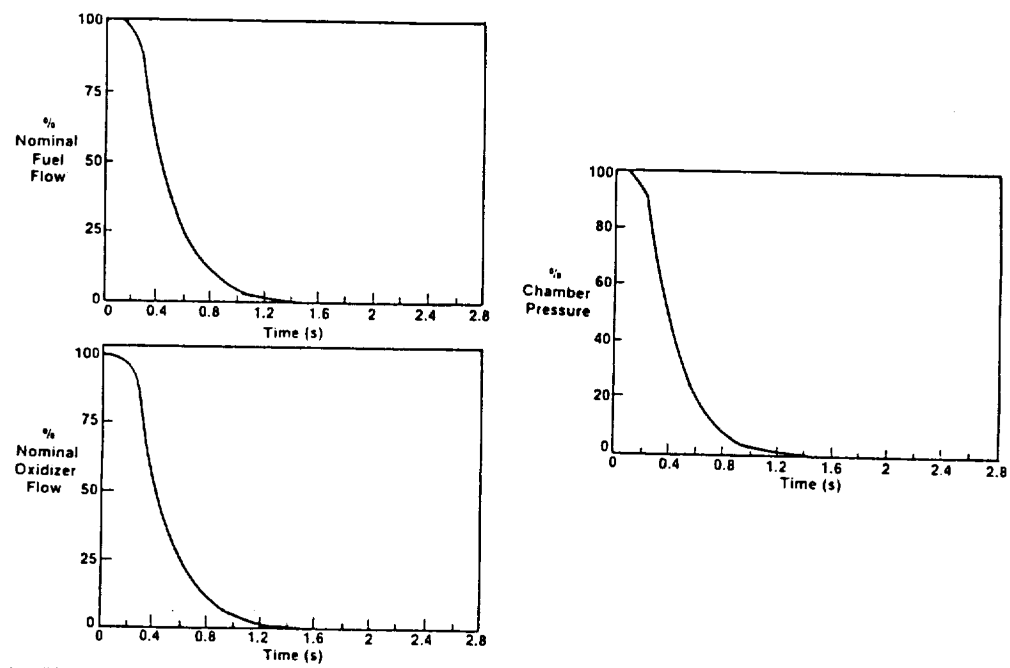

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Figure 2-4. Pressure-Fed LOX/RP-1 Shutdown Transients

The approximate propellant weight used by the engine during startup was estimated from the flow rate curves. The results are shown in Table 2-2 and give an indication of the propellant weights for the baseline engine of 841 Klb thrust.

Table 2-2. Propellant Weight Utilized During Startup of a Pressure Fed Engine

F = 750 K lb Pc = 500 psia

	LOX	RP-1	TOTAL
Time from start, sec	lbs (used)	lbs (used)	
0.231	44.7	2.5	
0.4615	173.7	6.86	
0.692	317.	11.85	
0.923	461.	16.84	
1.154	648.	73.60	
1.385	978.	259.5	
1.615	1400.	492.7	
1.846	1880.	741.0	
2.077	2397.	992.9	
2.308	2932.	1247.4	
2.538	344.2	1477.9	
2.77	3965.	1697.5	
3.00	4531.	1925.1	6456.

2.1.2 Engine Behavior Just Before and After Lift-off

A brief study was made to determine the extent of the engine operating point excursion between a moment just prior to lift-off and just after liftoff. The change is due to the small change in inlet pressure at the engine due to the comparatively sudden change in acceleration from approximately 1 g before liftoff to approximately 1.4 g afterward.

The steps in determining the design point shift and the results are shown in Figure 2-5. It can be seen that the increase in thrust is only 4% and the increase in mixture ratio is from 2.4 to 2.5 when the engine goes from a 1 g environment to an approximately 1.4 g. The engine is of course calibrated to operate at the higher values. The conclusion is that no attempt should be made to require a control system to adjust the flows and mixture ratio to make the thrust and mixture ratio the same before and after liftoff since this requirement is hereby shown to be unnecessary.

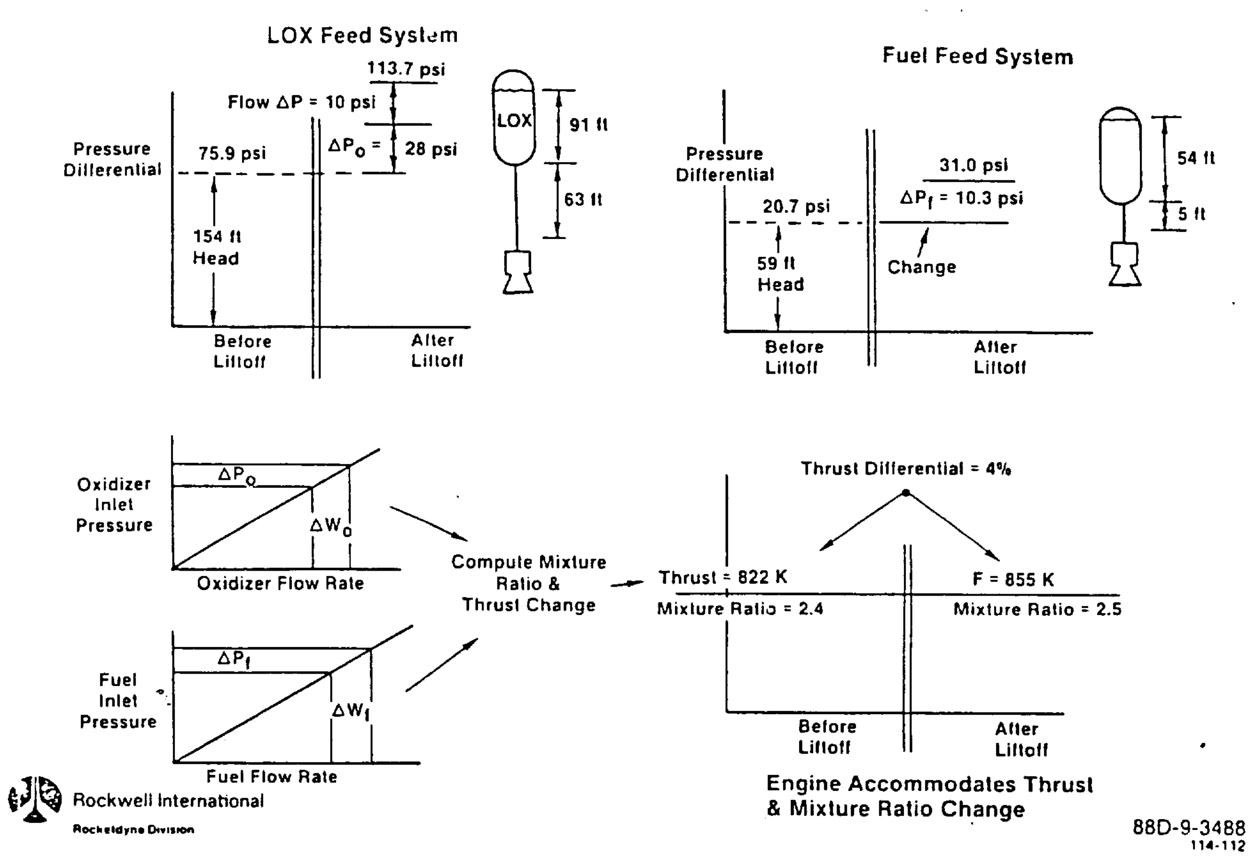


Figure 2-5. LRB Pressure Fed Engine Response During Liftoff

2.1.3 Engine Parametrics

In general, the parametrics generated during Phases I and II were adequate. However, a few more were generated during Phase III as follows. The values of the required inlet pressure as a function of chamber pressures were generated for a typical LRB pressure fed engine. The results are shown in Figure 2-6. The estimated engine weight as a function of chamber pressure and thrust level were determined. The results are shown in Figure 2-7. The engine performance, assuming a realistic ηC^* of 0.96, as a function of chamber pressure is shown in Figure 2-8.

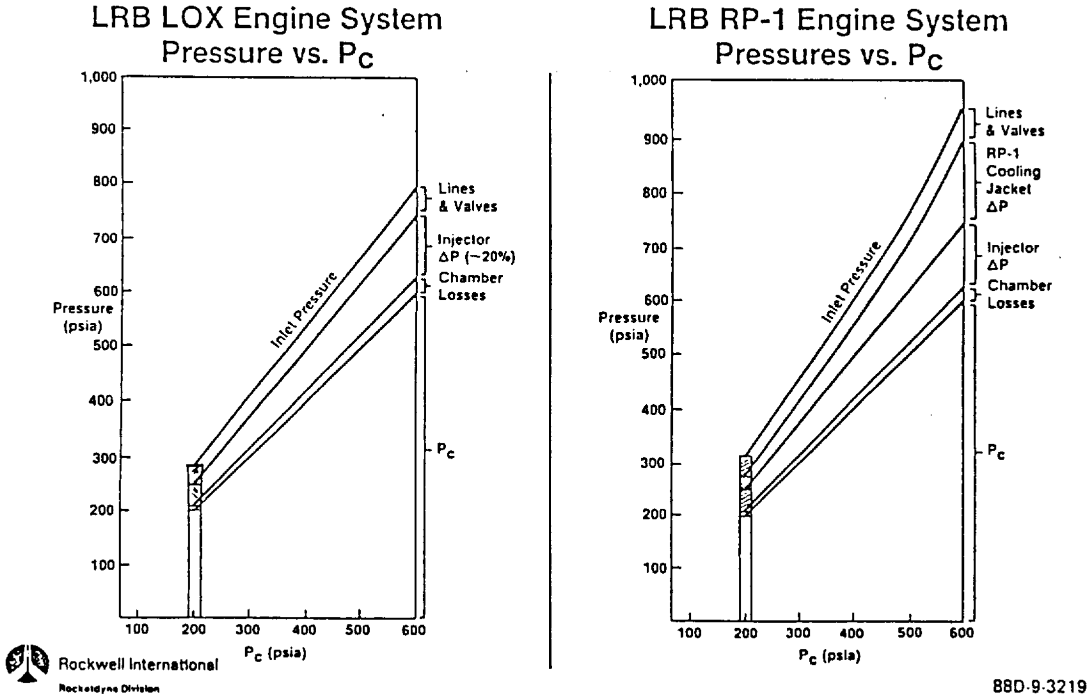
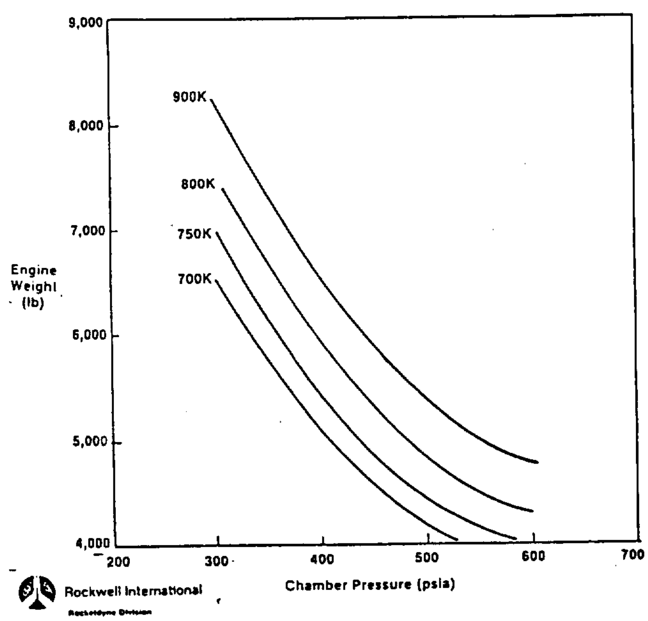


Figure 2-6. Inlet Pressure as a Function of Chamber Pressure for an LRB Pressure Fed Engine



—— Pressure-Fed
Exit Diameter = 108 in. (max.)

Thrust (Sea Level) (lb)	P_c (psia)	Weight (lb)
750K	334	6,400
750K	425	5,100
750K	500	4,460
750K	600	4,000
800K	500	4,800
800K	600	4,300
900K	334	7,680
900K	500	5,350
900K	600	4,800
700K	334	6,030
700K	500	4,200

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Figure 2-7. LRB Pressure Fed Engine Weight as a Function of Chamber Pressure and Thrust

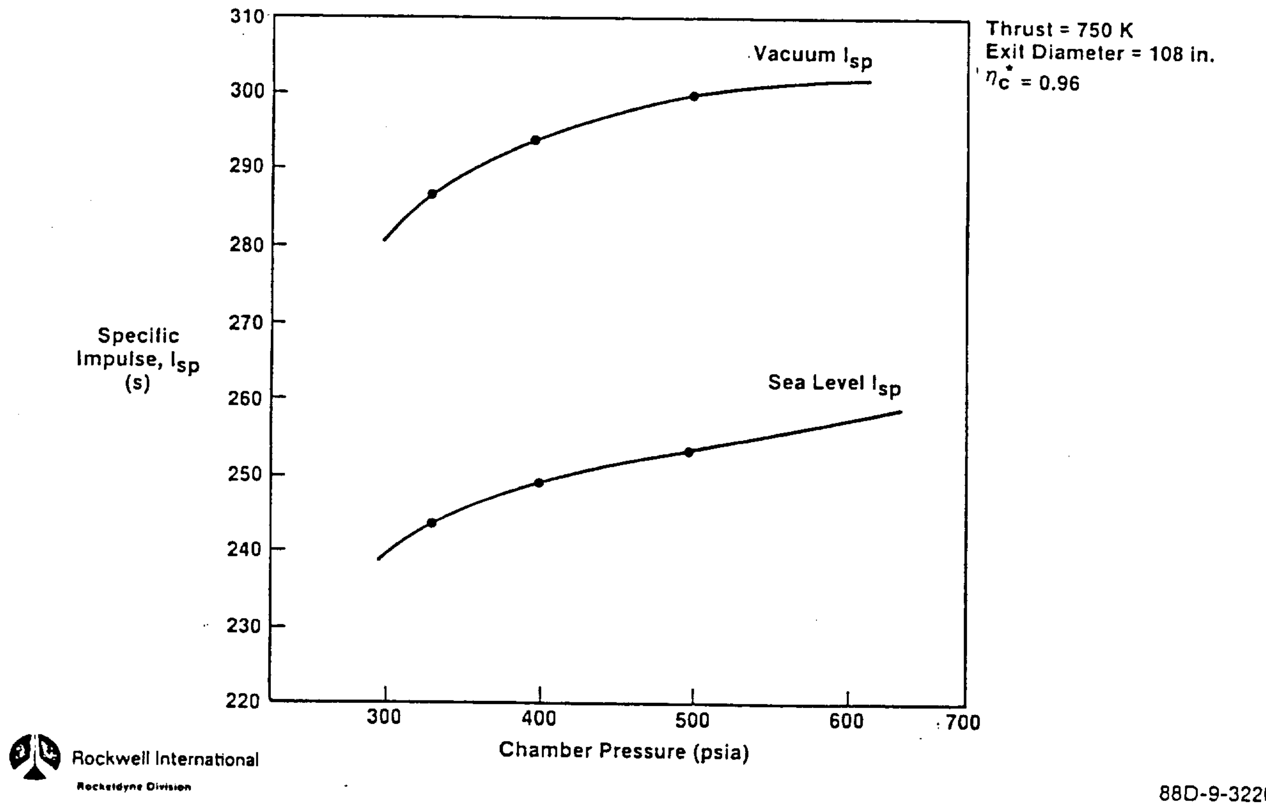

Figure 2-8. LRB Pressure Fed Engine Performance vs. Chamber Pressure

2.1.4 LRB PRESSURE FED ENGINE PROGRAMMATICS

The overall development schedule for the LRB pressure fed engine is shown in Figure 2-9. The 51 month (4 1/4 years) engine development program is designed to support a first vehicle launch in the third quarter of 1994 and therefore would benefit from a Phase B effort and a technology program directed at defining the best injector configuration. A benefit of the Phase B design effort would be to allow early long lead procurement of casting tooling for some of the major components such as the thrust chamber manifolds. The technology program should be started in time to provide data for design of the injector. This effort would significantly reduce risk during the hot fire test phase.

As indicated in Figure 2-9, engine test facilities are required by the second quarter of 1992. These facilities are assumed to be provided by the government or the vehicle contractor. Formal Pre-Flight Rate Tests (PFRT) are planned prior to the first flight and Flight Rating Tests (FRT) to certify readiness for full operational status after the first flight.

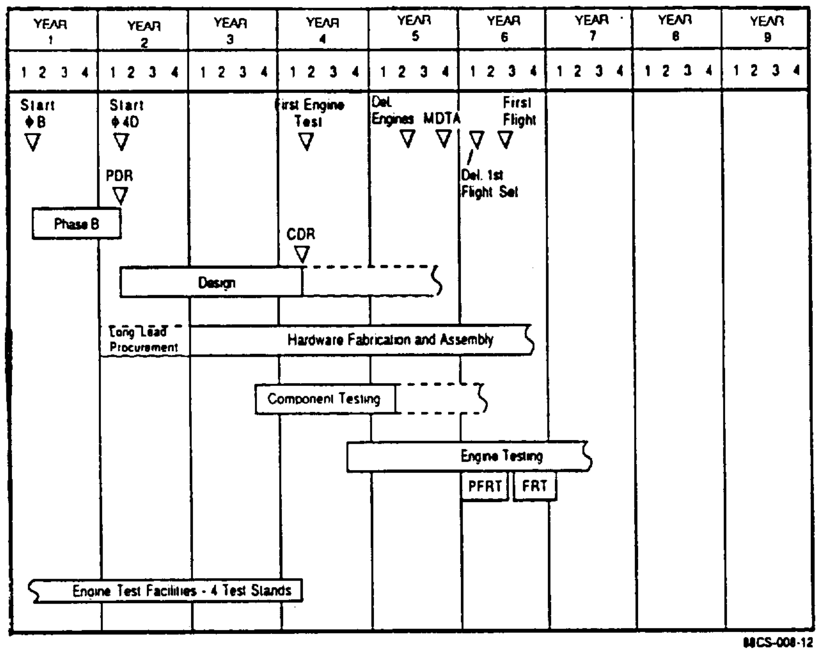

Figure 2-9. LRB Pressure Fed Engine Development Program

The engine test plan (shown in Figure 2-10) has been developed (in terms of numbers of tests and hardware) on the basis that the engine design provides robustness and the design margins are applied to the normal power level (NPL) operating conditions resulting in higher margins at throttled conditions. A design team including engineering, manufacturing, procurement, operations, reliability, producibility, quality and maintainability functions will be fully integrated into the design and procurement process to assure a cost effective low risk engine. Lessons learned from numerous previous large engine development programs will be applied. These include:

1. Component level testing will be conducted in an engine simulating environment to the maximum extent possible.
2. Extensive limits testing will be conducted at both the component and engine level.
3. Overstress testing will be conducted on a majority of the test units. (Further details can be obtained by referring to the Phase II report.)

The development program cost is estimated to total $435M, and is spread over the life of the program as shown in Figure 2-11. (This includes MPTA, PFRT and FRT engines.)

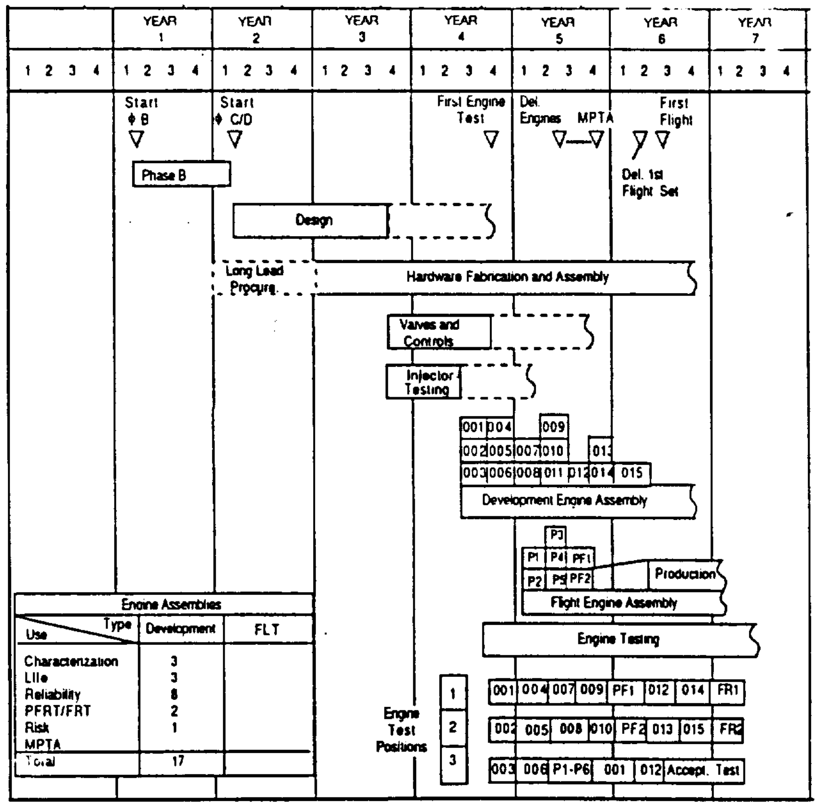

Figure 2-10. LRB Pressure Fed Engine System Test Plan

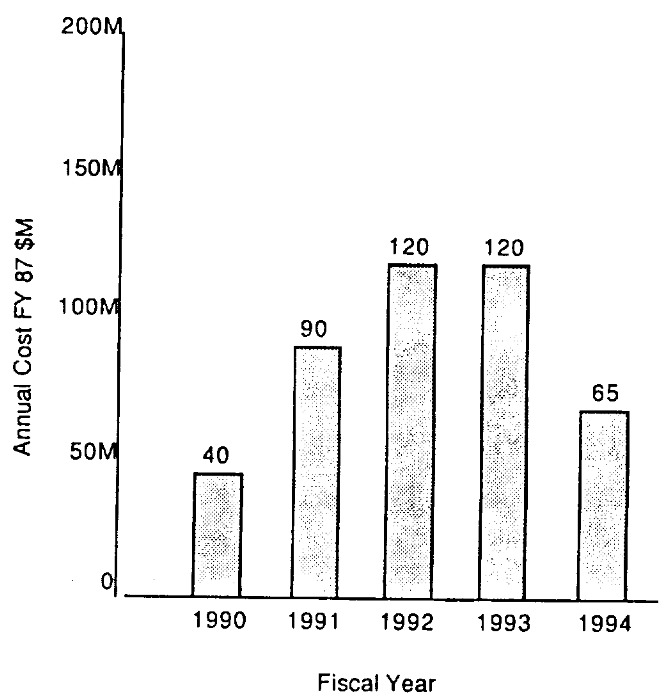

Figure 2-11. Fiscal Year Full Scale Development Cost

This section presents the selected LRB/RP-1 pump fed booster rocket engine configuration and characteristics resulting from the technical analyses and trade studies.

A baseline engine concept was selected based on ongoing Space Transportation Booster Engine (STBE) studies and experience along with trade studies for the STS application. An engine performance and pressure balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the recommended configuration and physical design.

2.2.1 LOX/RP-1 Gas Generator Engine Characteristics

The hydrocarbon engine selected for the pump fed LRB uses LOX/RP-1 propellants at an Emergency Power Level (EPL) chamber pressure of 1400 psia and a 2.53 engine mixture ratio. The selected engine cycle is a gas generator cycle producing 1800 R turbine drive gases to drive the RP-1 turbopump and the LOX turbopump which have turbines connected in series. Series turbines were selected to minimize the secondary flow performance losses of the gas generator, (GG gases) which are exhausted into the thrust chamber nozzle at an area ratio of 14.2. The nozzle contour is an 80% bell with a 4-degree exit wall angle to accommodate sea level operation at minimum power level without nozzle flow separation. The engine layout is shown in Figure 2-12. A simplified flow schematic is shown in Figure 2-13 and the engine characteristics are shown in Table 2-3.

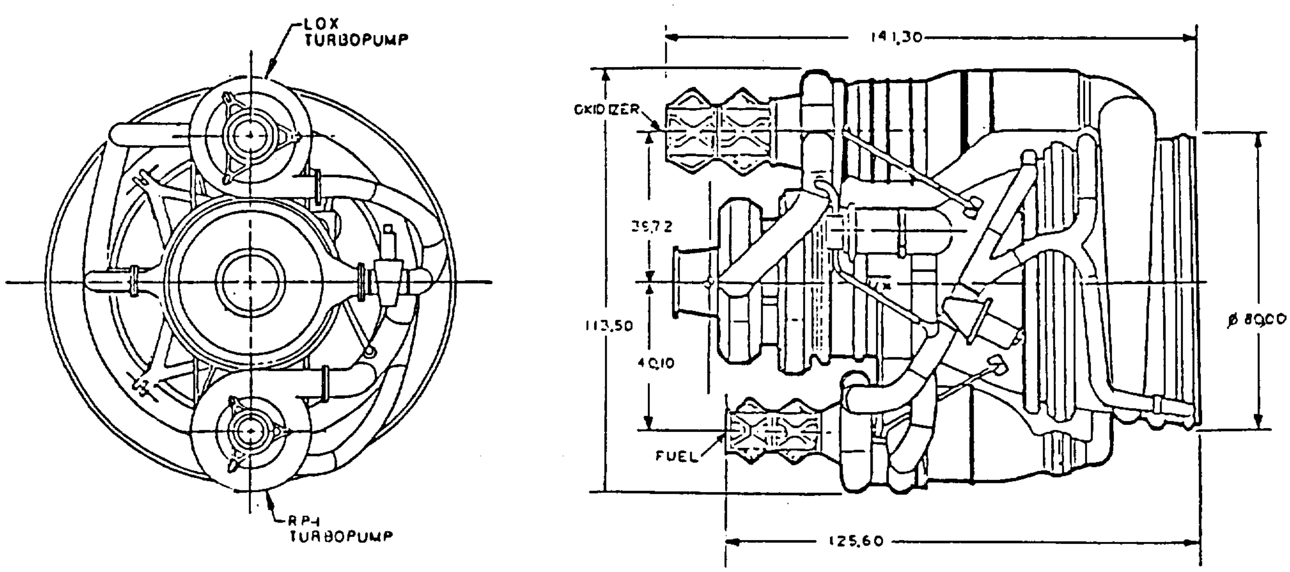

Figure 2-12. Top and Side View of LRB LOX/RP-1 Gas Generator Engine

Table 2-3. LRB LOX/RP-1 Gas Generator Engine Characteristics

Engine Parameters	Nominal Thrust (NPL)	Abort Thrust (EPL)	Minimum Thrust (MPL)
Weight (lb)	6216		
Throttle (percent)	100.0	110.0	75.9
Oxidizer flow rate (lb/sec)	1455.6	1603.7	1087.1
Fuel flow rate (lb/sec)	575.3	633.8	429.7
Vacuum thrust (lb)	629871	692858	472403
Sea level thrust (lb)	564881	627858	407403
Chamber Pressure (psia)	1272.7	1400.0	954.6
Vacuum Isp (sec)	310.1	309.6	311.4
Sea level Isp (sec)	278.1	280.6	268.6
Mixture ratio	2.53	2.53	2.53
Nozzle area ratio	16.5		
Area (in^2)	4423		
Throat radius (in)	9.232		
Exit diameter (in)	75.0		
Overall length (in)	130.3		

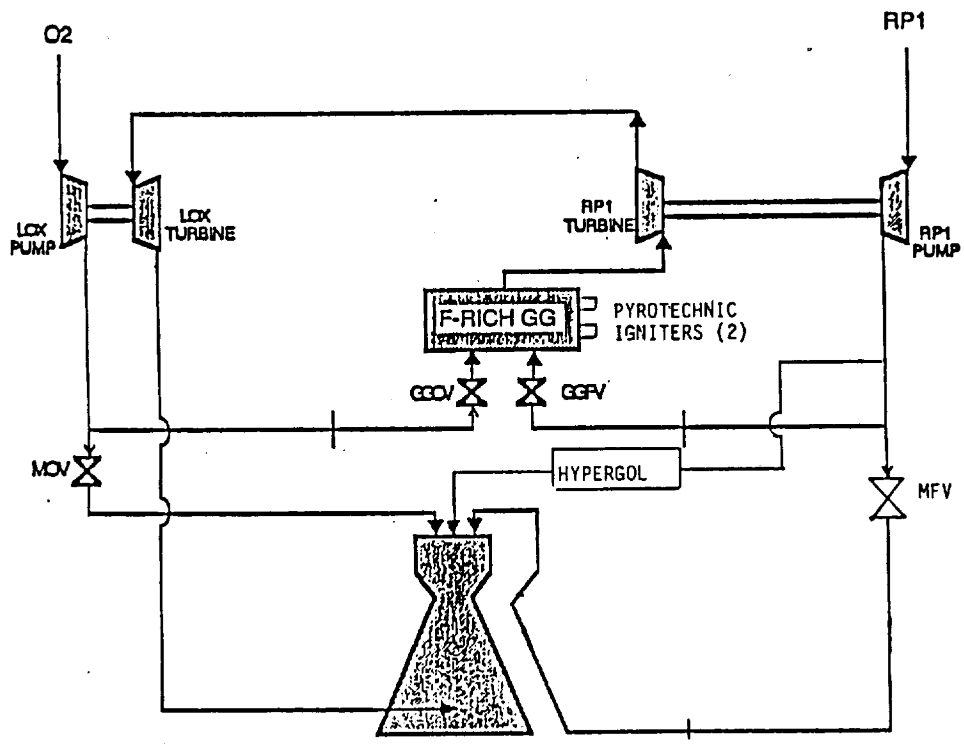

Figure 2-13. Simplified LRB LOX/RP-1 Pump Fed Engine Flow Schematic

The RP-1 fuel is used to cool the thrust chamber. The thrust chamber consists of a one piece construction MCC/nozzle using stainless steel tubes. It is desirable to use a light weight tubular construction for the nozzle/MCC for a low heat flux thrust chamber at chamber pressures up to about 1500 psia. The high heat flux at chamber pressures above that requires fabrication of a copper base alloy (NARloy-Z) milled channel configuration, typical of the SSME. The injector will be a ring-type design similar to other LOX/RP-1 injectors and will use OFHC copper rings, as was used in the F-1 injector for adquate injector face cooling.

The nozzle to MCC attachment point is at an area ratio of 5:1 where 50 percent of the RP-1 is used to cool the nozzle and 50 percent is used to cool the MCC. This 50/50 flow split and 5:1 attachment location provides the lightest weight engine with the lowest RP-1 pump discharge pressure. An up-pass cooling circuit is used for both the MCC and nozzle. A fraction of the nozzle coolant is diverted to the gas generator and the remainder is mixed with the MCC coolant and discharged to the main injector. The nozzle coolant ΔP is low compared to the MCC and provides the highest energy level RP-1 to the gas generator. The cooling characteristics and energy levels are depicted in the engine balance table. Further details regarding the design selection criteria are found in the Phase II report.

3.0 LOX/H_2 GAS GENERATOR ENGINE SYSTEM

This section presents the characteristics of the chosen engine configuration which is a LOX/ H_2 gas generator pump fed engine virtually identical to the present Space Transportation Main Engine (STME) configuration now being studied by Rocketdyne under a separate contract.

The specific baseline engine concept was selected based on previous studies and experience along with trade studies for the STS application. This engine has the following main advantages: 1) low technical risk, 2) no environmental concerns, 3) commonality with current shuttle ET propellants, 4) reduced POGO stability compensation hardware size and complexity, and 5) reduced exit diameter obviating the need to make major launch platform alternations.

3.1 PERFORMANCE AND CHARACTERISTICS

An engine performance and pressure balance was generated for the selected configuration and the resultant parameters were used to establish the pertinent combustion chamber, injector, nozzle, and turbopump characteristics leading to the recommended configuration and physical design. The engine characteristics are tabulated in Table 3-1.

The engine selected is of an expendable type with step throttling capability of 100% to 75% of the nominal thrust level. The rationale for engine thrust, chamber pressure, expansion ratio, and engine throttling range were determined by GDSS. The propulsion system described here is based on an overall mixture ratio of 6.0 and expansion ratio of 20.

The engine is baselined with no boost pumps, and minimum inlet pressures of 65 psia for LOX and 45 psia for LH_2. Boost pump trades conducted in the STME studies showed an increase in engine weight when boost pumps are included, cost and complexity and the STME is baselined without boost pumps. The rationale for the LOX pump inlet pressure is described in the Phase II final report. Various options for disposing of the engine gas genertator (GG) exhaust were studied previously and are given in the Phase II report. A solid propellant gas generator (SPGG) assisted start method is selected over the tank head start method because it provides more repeatable starts. In addition, the tank head start is comparatively slow compared to other types of starts, and this may complicate optimization of ignition sequencing for the STS vehicle.

Table 3-1 Baseline LRB LOX/H_2 Gas Generator Engine Characteristics

ENGINE PARAMETERS	NOMINAL	MINIMUM
Weight (lb)	6100	
Throttle (percent)	100	75.0
Oxidizer Flow Rate (lb/sec)	1162.7	893.3
Fuel Flow Rate (lb/sec)	193.8	148.9
Vacuum Thrust (lb)	558,000	418,500
Sea Level Thrust (lb)	518,574	388,930
Chamber Pressure (psia)	2250	1701
Vacuum Isp (sec delivered)	411.4	412.3
Sea Level Isp (sec)	382.3	373.2
Mixture Ratio	6.0	6.0
Nozzle Area Ratio	20.0	
Throat Radius (in)	6.543	
Exit Diameter (in)	58.44	
Overal Length (in0	100.8	
Inlet Pressure: LOX (psia)	65	
Inlet Pressure: LH_2 (psia)	45	
Throttling Type	Step-Open Loop	
Mission Life	1	
No. of Starts	5	
Boost Pump	None	
Bleed Required	None	
Engine Start	SPGG	
Thrust Vector Control Actuator Type	Electromagnetic	
Valve Actuator Type	Electromagnetic	
Inlet Temperature	Saturation	
Inlet Line Diam. (both oxid. & fuel) (in.)	10	
Reliability	99% @ 90% confidence level	
No. of Pump stages		
LOX	Single Stage	
LH_2	Two Stage	

3.2 SCHEMATIC AND OPERATION

A schematic diagram of the engine is shown in Figure 3-1 and a side view and top view of the engine are shown in Figure 3-2.

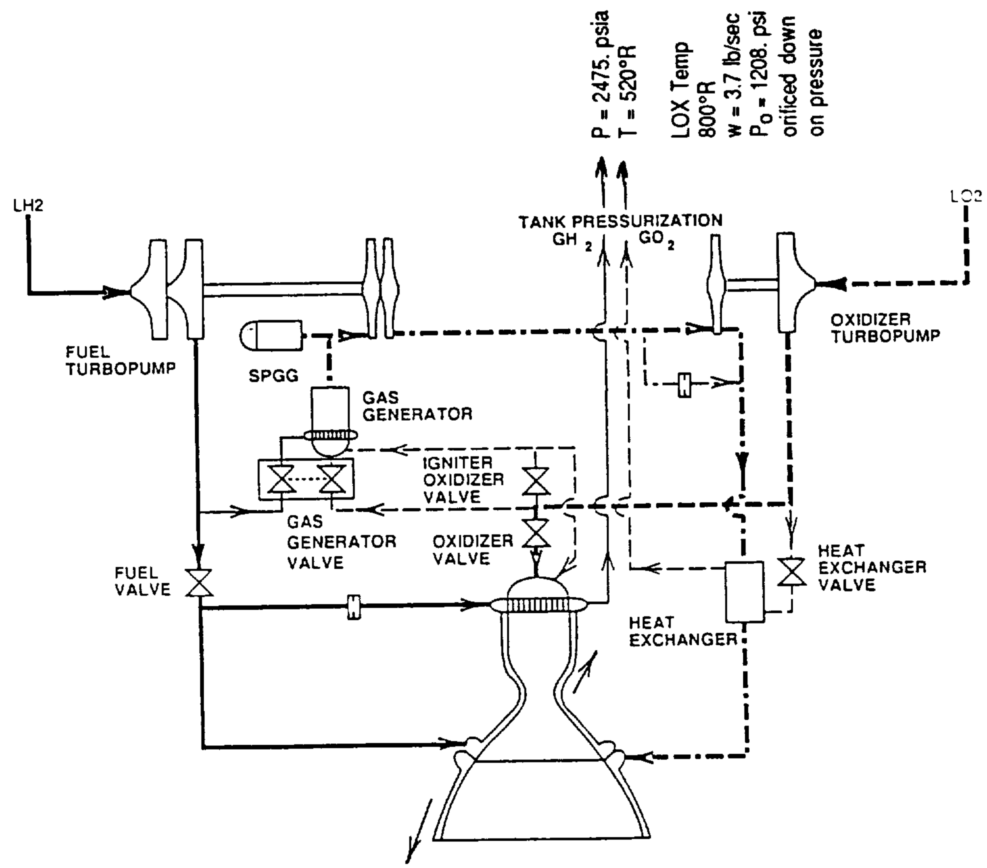

Figure 3-1. LOX/H_2 Gas Generator Engine With Gas Cooled Nozzle

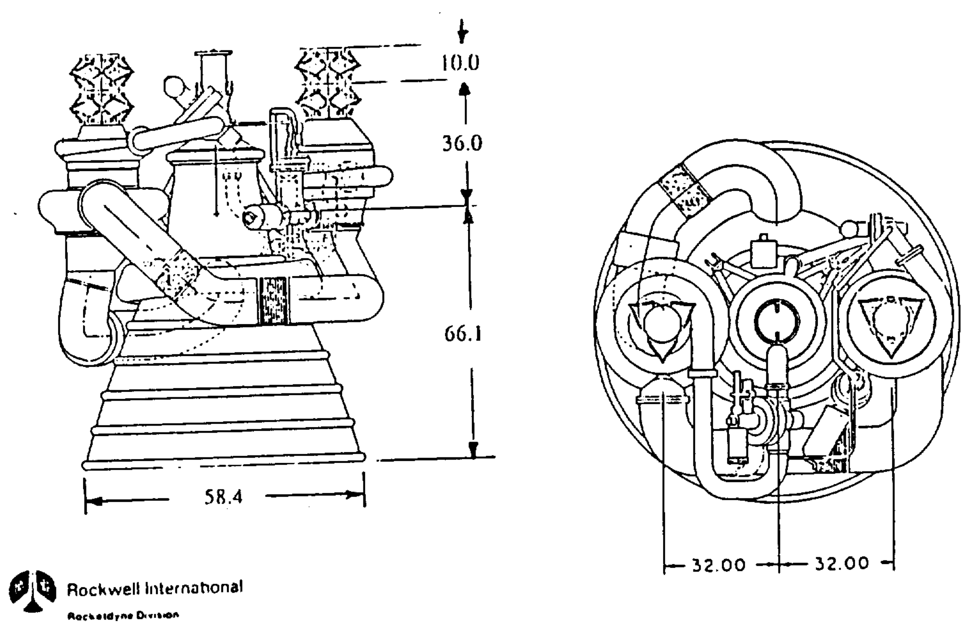

Figure 3-2. LRB LOX/Hydrogen Engine

The engine has separate LOX and liquid hydrogen turbopumps. The two turbines are driven in series by the same gas generator. The GG exhaust gases first drive the fuel turbine and then the LOX turbine, The LOX heat exchanger is located downstream of the LOX turbine and supplies LOX for use in pressurizing the LOX propellant tank. The GG exhaust gas is then utilized to cool the nozzle and is dumped at the nozzle exit around the periphery of the nozzle. Vaporized hydrogen required to pressurize the hydrogen propellant tank is supplied from the combustion chamber coolant.

Steady state operation is reached in approximately 3.5 seconds. The valve start and shutdown sequences and the moment of ignition of the SPGG are shown in Figure 3-3. The transient flows during startup (and during shutdown) are shown in Figure 3-4 and Figure 3-5, resulting in the changes shown in the main chamber pressure and GG chamber pressure shown in Figure 3-6. The LOX heat exchanger valve is then opened allowing a small amount of LOX to be vaporized and utilized to pressurize the LOX tank.

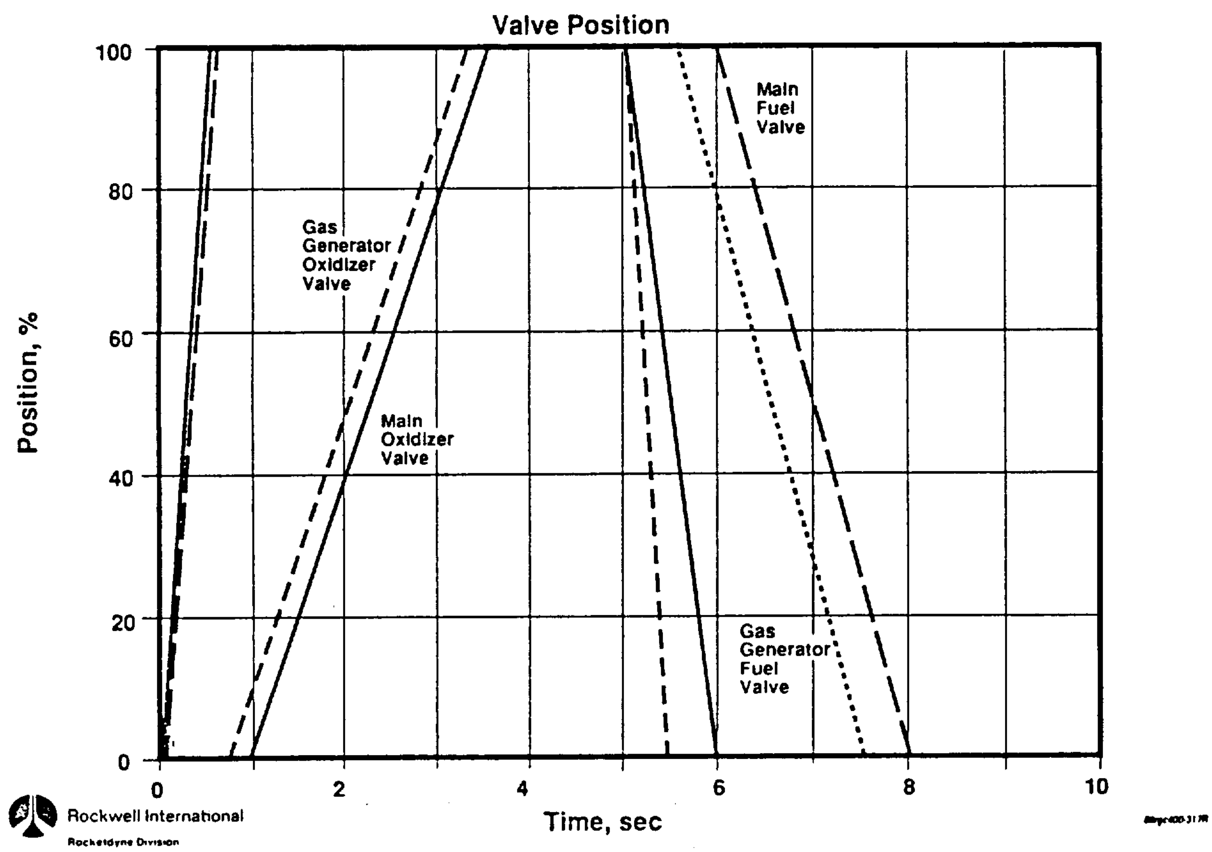

Figure 3-3. Startup and Shutdown Sequence and Valve Movement for LRB LOX/H2 GG Engine.

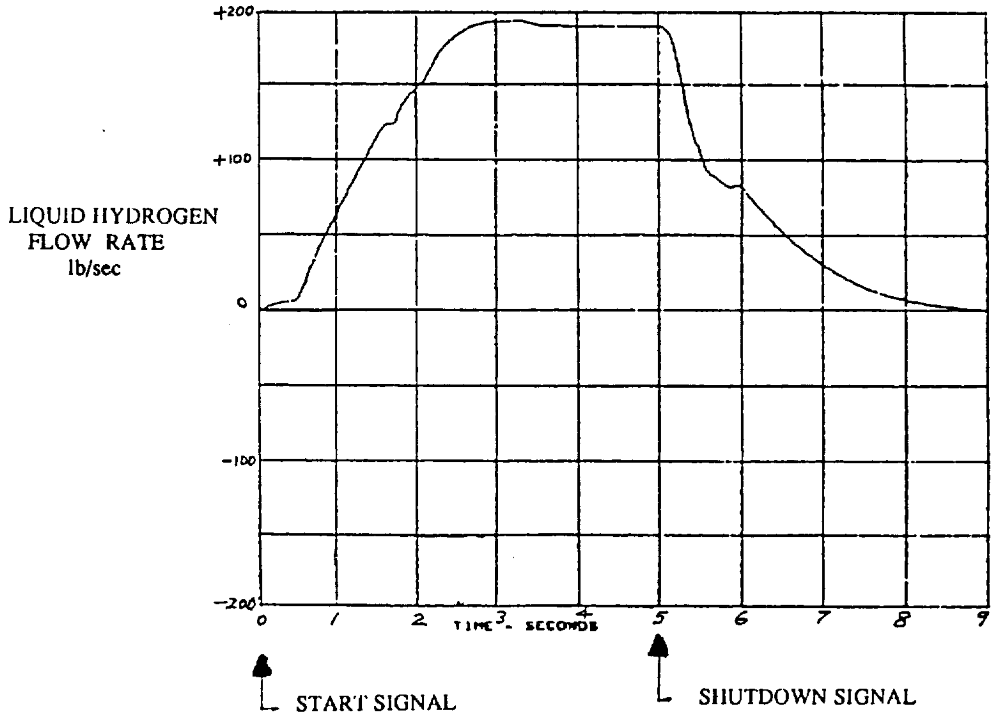

Figure 3-4. Start and Shutdown Transient LH_2 Flow

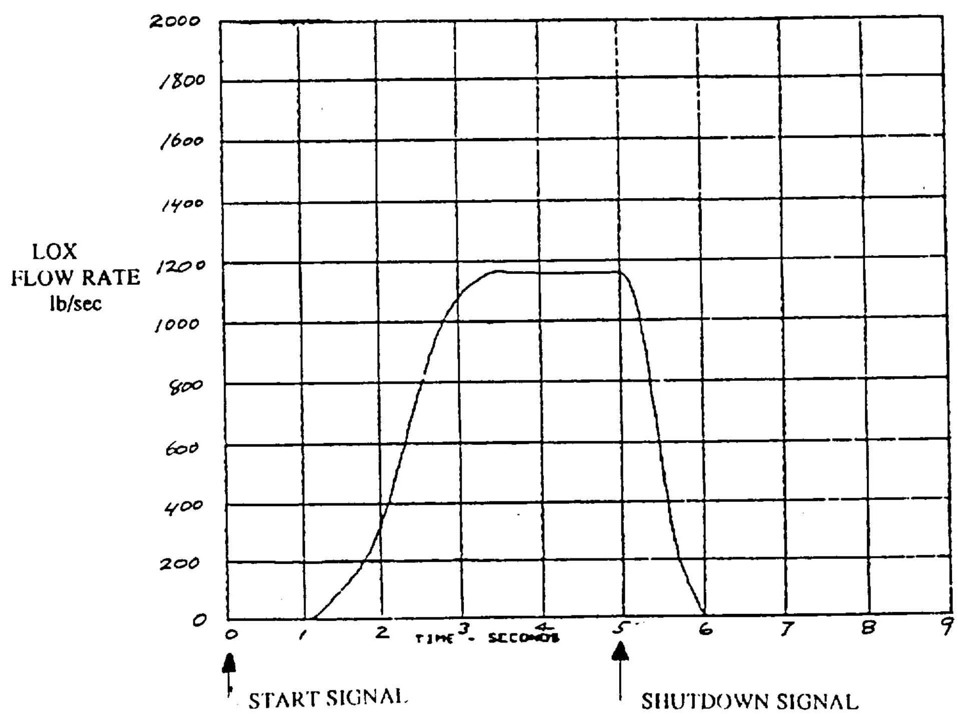

Figure 3-5. Start and Shutdown LOX Transient Flow

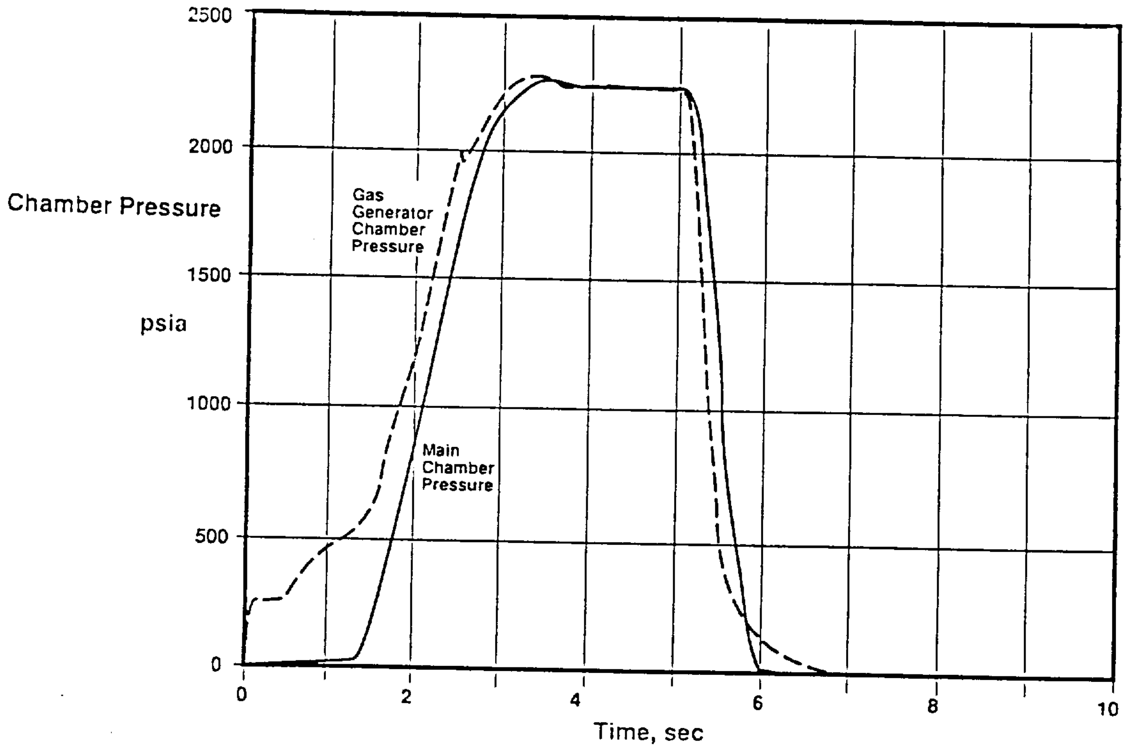

Figure 3-6 GG Cycle Transient Performance - Main Chamber Pressure and GG Chamber Pressure

3.3 DESIGN

A side and top view of the LOX/H_2 engine are shown in Figure 3-2. The selected expansion ratio of 20 has resulted in a relatively short nozzle. The reduced exit diameter and length of the engine are a distinct advantage since the overall plume diameter and gimballing space required are both substantially reduced.

The regeneratively cooled combustion chamber has a 7:1 expansion ratio. A GG exhaust gas cooled nozzle extends the expansion ratio from 7 to 20. The LRB nozzle design will have an optimized 80% bell nozzle from the throat to an expansion ratio of 20 at the nozzle exit.

The nozzle has a tubular wall construction. The nozzle tubes, from an expansion ratio of 7 to 20, are cooled with the exhaust gas coming from the LOX turbine exhaust duct. At the point where the gas enters the tubes, part of the gas flow is diverted into the nozzle and is used as film coolant. The remainder conductively cools the tubes. The coolant gases (GG exhaust gas) flow in the same direction as the primary nozzle flow and are dumped out of the tubes at the nozzle exit plane.

An injector design cross-section is shown in Figure 3-7. It is a typical gas/liquid coaxial injector of conventional design used in LOX/H_2 rocket engines. The detail design has been specially constructed to reduce fabrication costs.

Cross sections of the fuel and LOX turbopumps are shown in Figures 3-8 and 3-9 respectively. The single stage LOX turbopump is driven by a single stage turbine. The fuel pump has two stages driven by a two stage turbine. Again, the designs minimize fabrication costs.

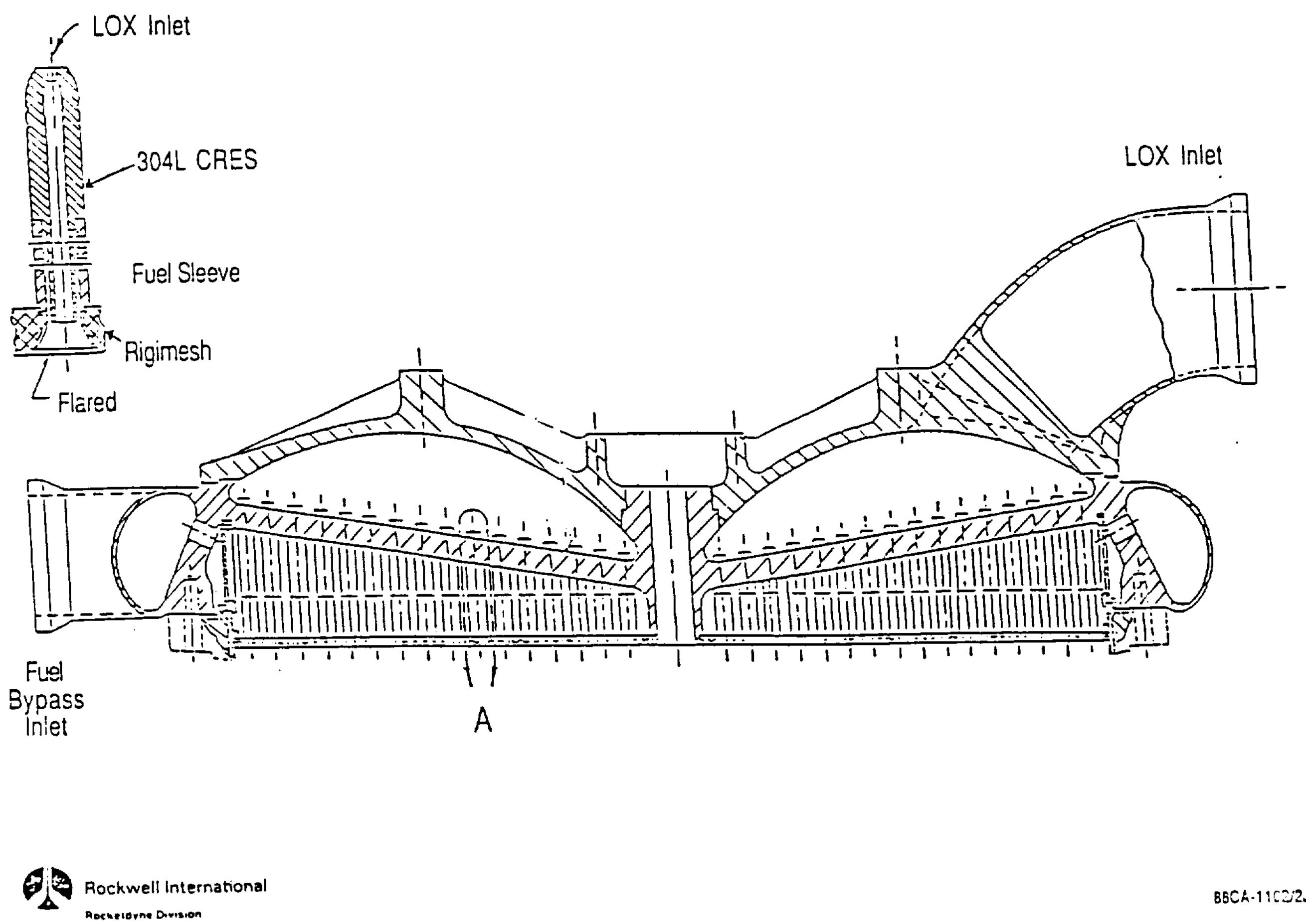

Figure 3-7 Injector Design

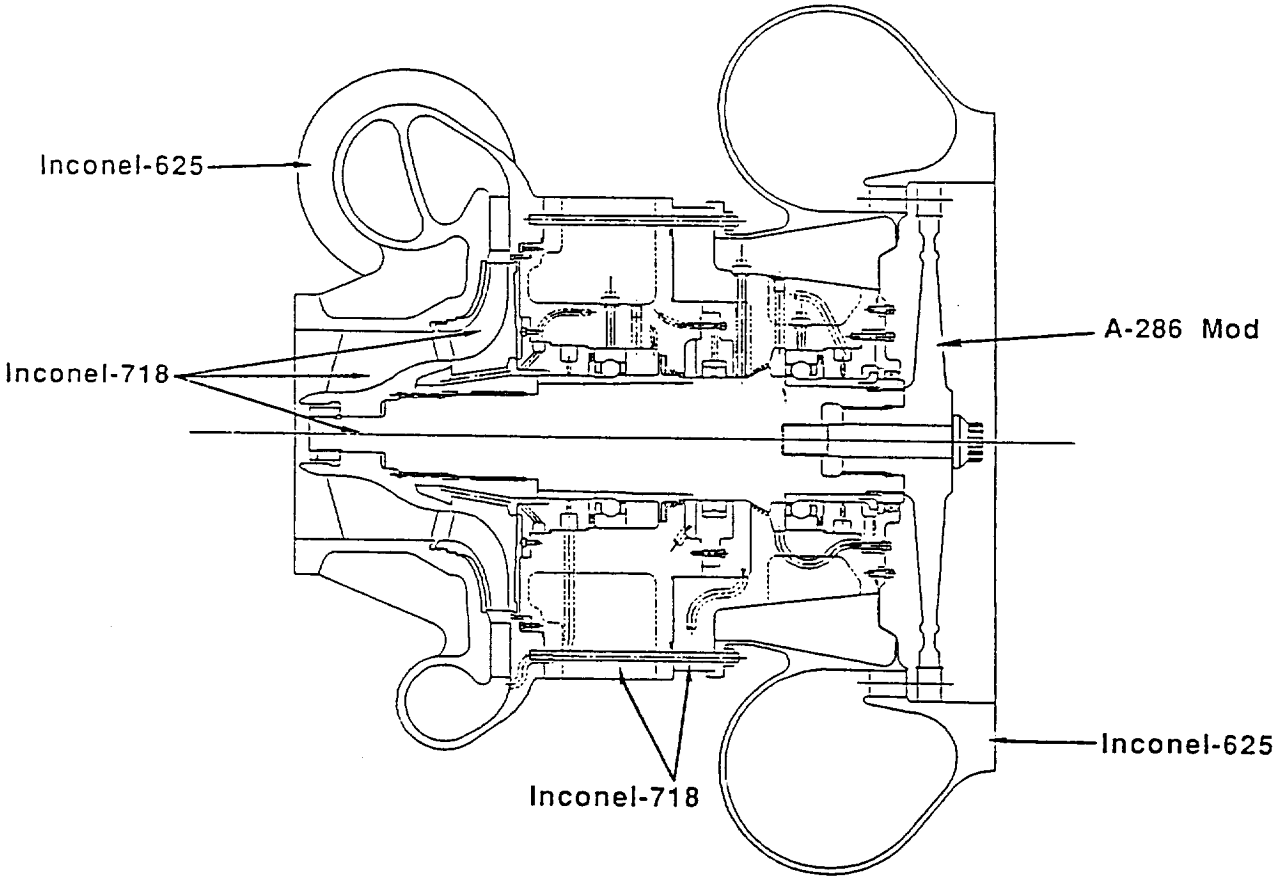

Figure 3-8 Gas Generator Cycle LOX Turbopump

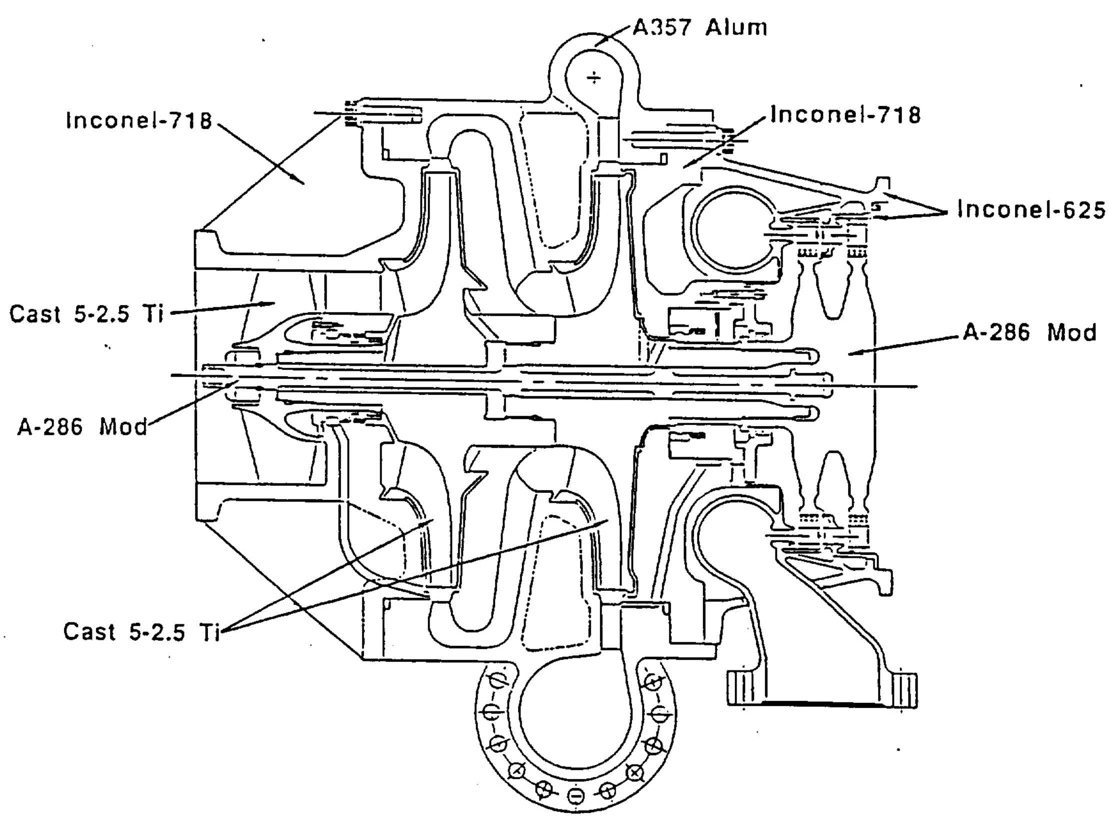

Figure 3-9 Gas Generator Cycle Fuel Turbopump

A heat exchanger design concept for vaporizing LOX and pressurizing the LOX tank is shown in Figure 3-10.

3.4 POGO SYSTEM

A preliminary estimate was made of the size of a POGO compensator for the engine of choice. It was estimated that a unit of about 1 cu. ft. volume placed just above each of the 4 LOX prevalves will be adequate. A single helium supply line branching to each of the four compensators can be provided. Three slightly different POGO suppressor device concepts are shown in Figure 3-11. In each case a spherical or cylindrical volume surrounds the LOX feed line with connecting ports at the bottom to allow rapid propellant flow in and out of the volume, thus suppressing feed line flow oscillations to the engine. The action is similar to that of a piston accumulator. Concept 1 uses very little helium, since it is filled with helium only once just before liftoff. However, as vehicle acceleration is increased, the gas volume will decrease due to an increase in static head pressure. Counteracting this is a gradually decreased static head due to lowering of the level in the propellant tank. Concept 2 maintains the gas volume independent of the static pressure, but requires a small helium or GOX bleed flow throughout the boost period. Concept 3 has an active liquid level control to ensure that the static volume remains relatively constant. Trade-offs can be made during more detailed design efforts. Meanwhile, Concept 2 is considered the suppressor of choice at this time.

3.5 ENGINE CONTROL

The control system is an open loop, step throttled type. The system for controlling the state of the engine and for engine condition monitoring is shown Figure 3-12. Changing the thrust in steps is accomplished by changing the gas generator propellant flow in steps by means of the GG propellant valve. When GG flow output is reduced, the power to the turbopumps is reduced and the main propellant flows are decreased. For example, referring to Figure 3-12, a signal to reduce thrust coming from the Vehicle Command Bus is received by the State Controller which in turn signals the Control Module to energize the appropriate valve actuator. Except for the ignition and shutdown operation, the balance of the operations by the controller are of the condition monitoring type. Signals from the instrumentation shown in Figure 3-13 are compared with preset high/low limits. If these limits are not exceeded, no action is taken. If they are exceeded, warnings to vehicle command and/or automatic engine shutdown are initiated.

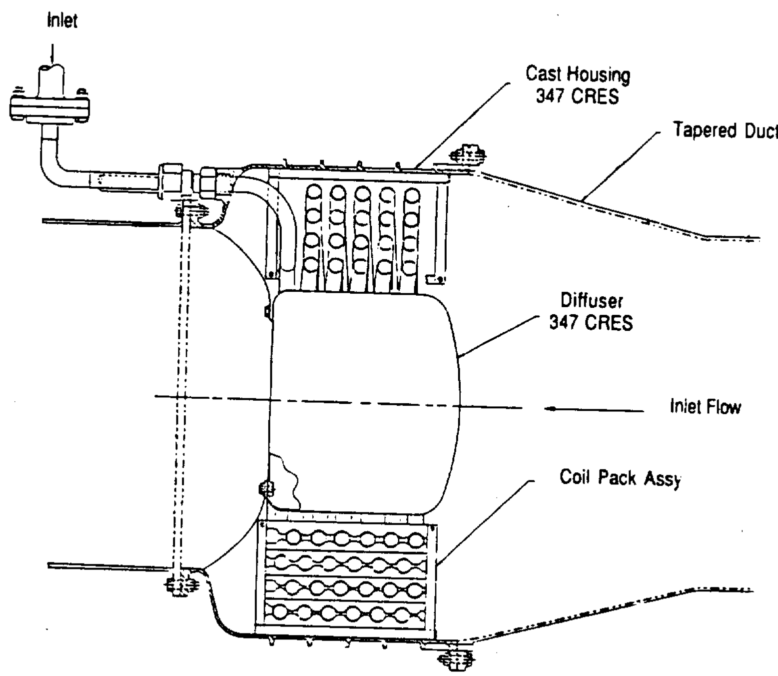

Figure 3-10. Oxygen Heat Exchanger Design

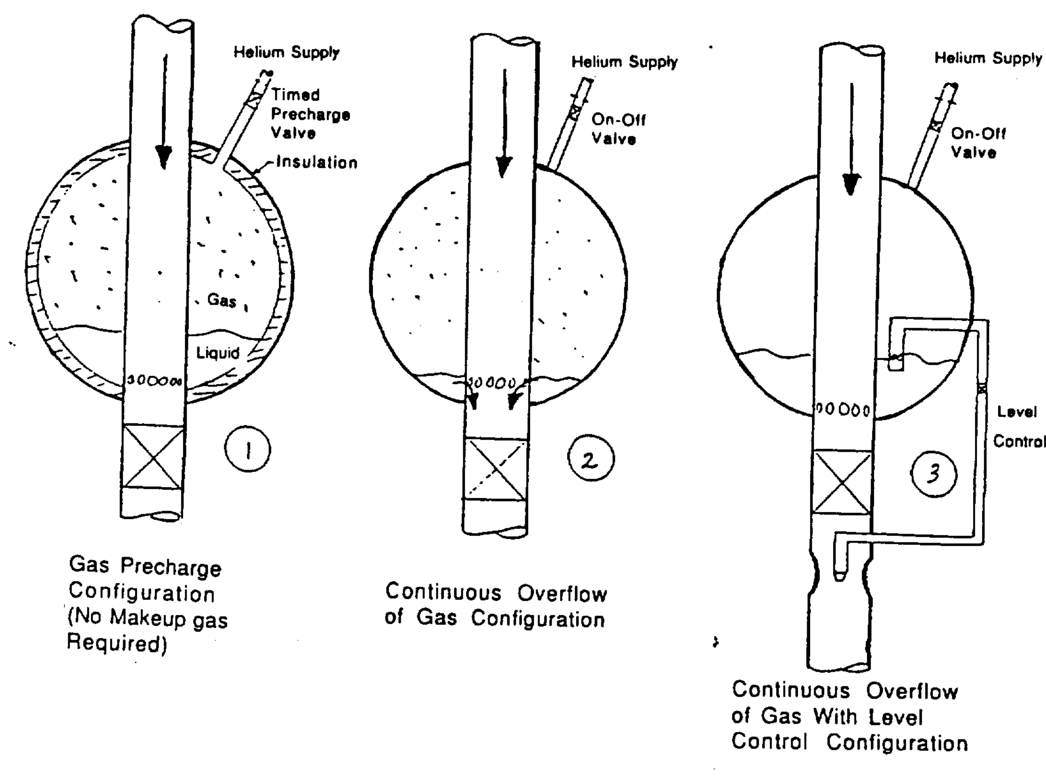

Figure 3-11. Schematics of Concepts for a POGO Supressor

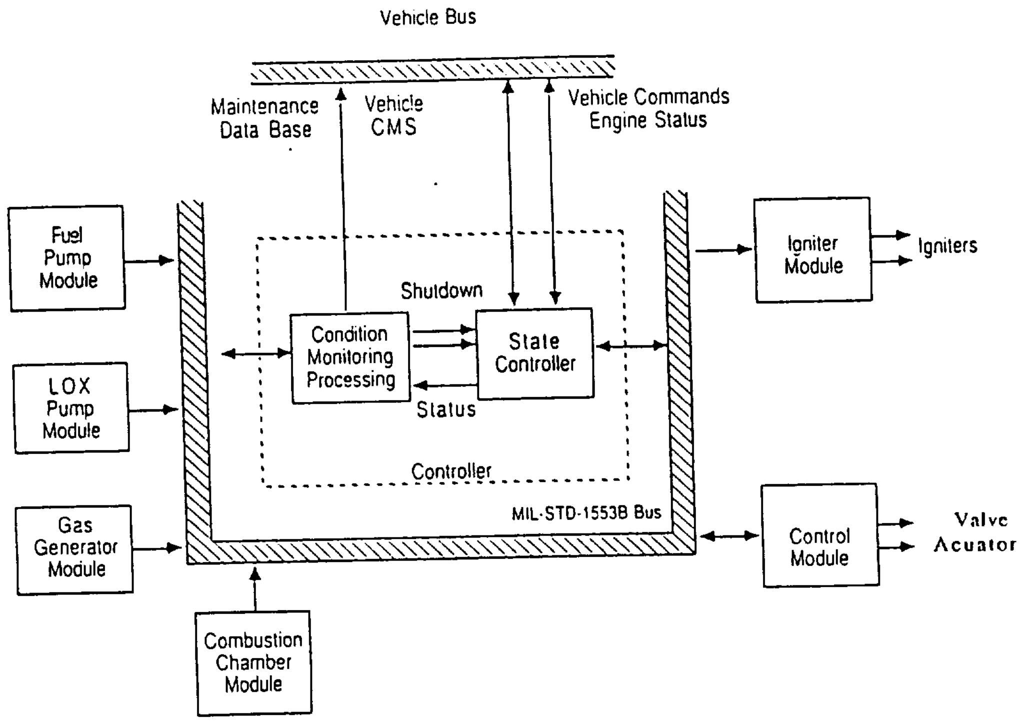

Figure 3-12. Control System Diagram

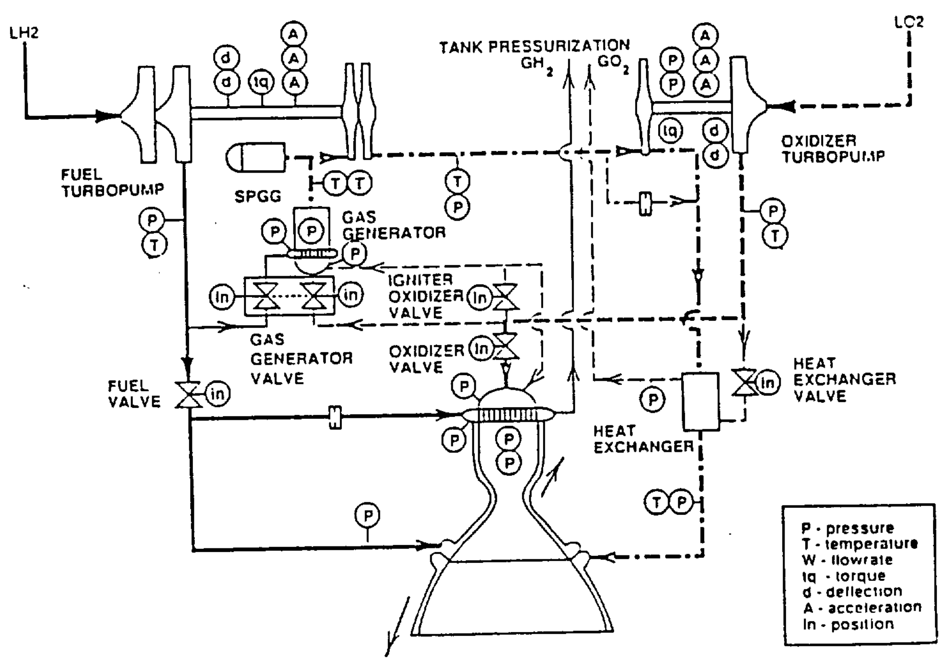

Figure 3-13. Flight Instrumentation Schematic

The number and type of instruments utilized to carry out the engine condition health monitoring function is a trade-off between 1) the cost, weight and reliability of instrumentation hardware, computer hardware, and software, and 2) the engine reliability requirements needed to meet the overall vehicle reliability requirements. Subsequent Engine Phase B studies will define the health monitoring functions and system design.

3.6 ENGINE INTERFACE REQUIREMENTS

The following interface requirements have been defined from overall LRB studies (Table 3-2). More detailed interface requirements will be defined in Phase B studies.

Table 3-2. LRB LOX/H_2 Interface Conditions

Gimbal requirement ±6°
Inlet pressure (min), psia
Hydrogen - 45
Oxygen - 65
Inlet Temperature (min), °R
Hydrogen - 37.5
Oxygen - 164
Mixture ratio tolerance(1) - ±3%
Thrust tolerance(1) - ±3%
(1) at standard propellant inlet conditions

3.7 ENGINE CHECKOUT ON THE PAD

The engine condition monitoring system and its associated measuring system will be used for the engine checkout operation. A fault detection algorithm can then be used to aid in locating the source of any anomalous operating condition.

For in-flight operation, however, only the decision of whether or not to initiate an engine shut down signal and to continue the flight under a one-engine-out condition or not is of importance. The fault diagnosis is only of secondary importance and any hardware and software required is considered ground support equipment. In any case, the detailed analysis to determine the characteristics of abort procedures must be determined from a vehicle standpoint with consideration for engine condition monitoring, shutdown and throttling capabilities and limitations.

3.8 ENGINE SCHEDULE AND PROGRAMMATICS

The overall development program schedule for the LOX/LH2 pump fed engine (and applicable to the LOX/RP-1 pump fed engine), is shown in Figure 3-14. The 63 months (5 1/4 years) development program is designed to support a first vehicle launch in the third quarter of 1995 and therefore would benefit from a Phase B effort and a modest technology program in terms of reduced risk. (For further details see also the Phase II report, RI/RD88-180 of June 1988, page 102, ff.)

First, a benefit of the Phase B design effort would be to allow early long lead procurement of casting tooling for some of the major components such as the pump housings. Secondly, significant benefits in terms of reduced risk would be derived from a technology program that is started in parallel with the Phase B design effort and completed in time to provide data for the development program design phase. The specific technology that would provide the most benefit is in the area of injector design for stability and for turbo pump bearings and seals and rotating elements. Thirdly, as indicated in Figure 3-14, engine test facilities are required by the fourth quarter of 1992. These test facilities are assumed to be provided by the government or the vehicle contractor. Formal Pre-Flight Rating Tests (PFRT) are planned prior to the first flight and Flight Rating Tests (FRT) to certify readiness for production; full operational status which are planned after the first flight.

The development program has been estimated to cost $987M and is spread out in time as shown in Figure 3-15.

3.9 ENGINE INLET PRESSURE

A study was made to determine the influence of propellant inlet pressure on engine weight for the baseline LOX/H_2 gas generator engine. The results are shown in Table 3-3. It can be seen that the engine weight is reduced significantly as the inlet pressure increases for the LOX and for the hydrogen.

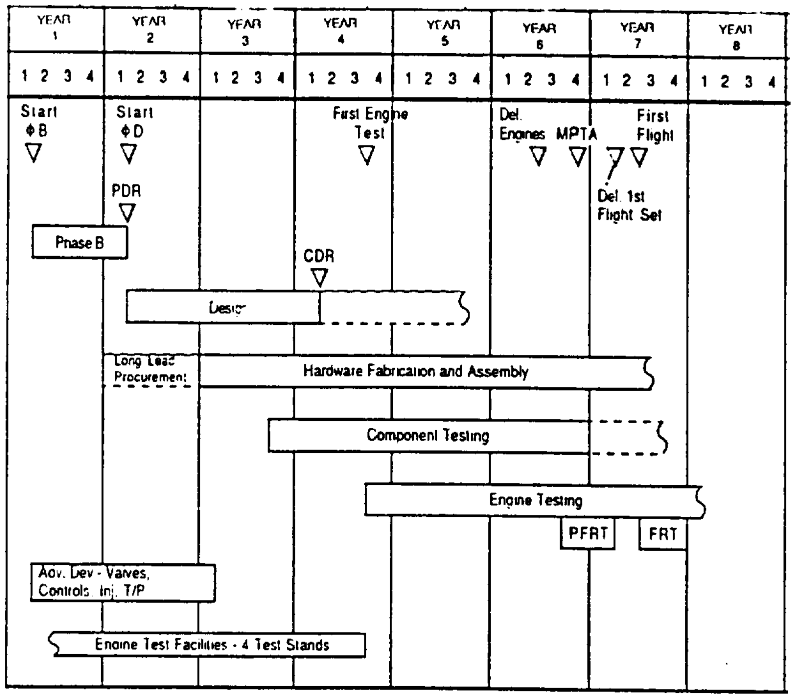

Figure 3-14. LOX/H_2 Pump Fed Engine Development Program

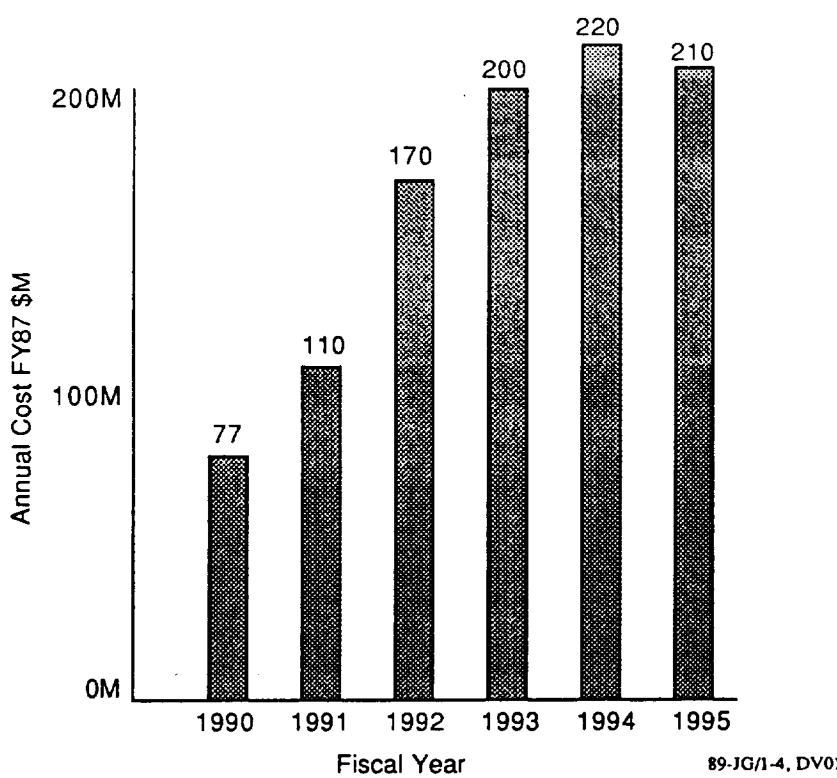

Figure 3-15. LOX/H_2 LRB Pump Fed Full Scale Engine Development Cost

Table 3-3. Performance of the LRB LOX/H_2 Pump Fed Engine as a Function of Propellant Inlet Pressure

INLET PRESSURES					
LH_2 (psia)	24.5	35.0	45.0	65.0	100.0
LOX (psia)	47.0	54.0	65.0	75.0	100.0
CHARACTERISTICS					
Specific Impulse, sl (sec)	431.0	431.7	431.9	431.9	432.0
Specific Impulse,vac (sec)	358.8	359.7	359.9	360.0	360.1
Engine Weight (lb)	6393	6154	6060	6002	5995
Engine Length (in)	153	153	153	153	153
Engine Exit Diameter (in)	94	94	94	94	94
Engine Expansion Ratio	47.9	47.7	47.6	47.6	47.6
Turbo Pump Weight					
Lox Pump (lb)	874	802	749	702	698
LH_2 Pump (lb)	977	805	762	752	747

(Vacuum Thrust Constant at 612,000 lb.)
(Chamber Pressusre Constant at 2250 psia.)

4.0 SUPPORTING STUDIES

4.1 ACOUSTICS

4.1.1 Introduction

This study was performed to provide initial predictions regarding the relative acoustic environment for the STS using LRBs in place of the SRBs as now configured on the STS. The LRB versions considered are those using the three engine types which were the candidates throughout this phase III study.

The method of analysis is based on those found in the literature. A computer model was utilized and the results are described of applying the model to the anticipated LRB STS and to the present SRB configurations. The following is a summary of the results. A detailed treatment is given in Reference 3.

4.1.2 Summary

Comparison between predicted and measured Sound Pressure Levels (SPL) at liftoff for the Space Shuttle at near-field locations demonstrates that the prediction method agrees satisfactorily with the measured data within ±5 dB.

Comparisons were made of acoustic sound pressure levels (SPL) at liftoff for Solid Rocket Boosters on the Space Shuttle and the proposed Liquid Rocket Boosters with (a) a pressure-fed LOX/RP engine system, (b) a pump-fed LOX/RP engine system, and (c) a pump-fed LOX/LH_2 engine system.
Computed results show the following:

a) The LRBs are significantly quieter than the SRBs close to the vehicle surface.

b) The LRBs are louder than the SRBs at 1000 feet from the launch pad along the path of the deflected exhaust. The LRBs are not necessarily louder than the SRBs at 1000 feet along other paths away from the deflected exhaust.

c) The differences between the engine systems become small at distances greater than 5000 feet from the launch pad.

Based on the preceding analysis, the LRB concept looks promising. The LRBs are actually quieter at the vehicle surface than the SRBs. Nothing was found in the analysis results that should discourage further development of the LRB concept.

4.1.3 Exhaust Noise Prediction Model Description

Four methods can be used to estimate acoustic loads due to exhaust noise. They are listed below:

- Teledyne method
- Northrop
- Wyle Labs method
- Noise Control handbook method.

The Teledyne method was found to be the best among the four in terms of test data correlation. A computer program was written which incorporates the Teledyne method. A detailed description is given in reference 3.

4.1.4 Near-Field Results

The major conclusions are:

The SRBs are significantly louder than LRBs up to 100 Hz (Figure 4-1). This is because the modeled point source distribution for SRBs is much closer to the nozzle exit than for LRBs (see Figure 4-2), and the receiver is on the nozzle. Therefore, the low frequency SRB sources are much closer to the receiver and thus sound louder. The SSME's data are presented for both booster configurations; their addition to the rms acoustic power is the same for the configurations.

Different LRB designs have the same loudness because the modeled point source distribution is nearly the same for all three LRB designs. Small differences in Sound Power Level (<3 db) are due to the different mechanical powers of the engines.

The computed SRB curve matches the Space Shuttle test data to within ± 5 db. This was intended. The SRB point source distribution was designed to produce computed SPLs that fit the test data recorded by sensors on the orbiter "belly". However, this process has also succeeded in matching the test data with predictions at other locations satisfactorily. Similarly, the LRB point source distribution was designed to produce computed SPLs that fit Saturn V (LRB) test data.

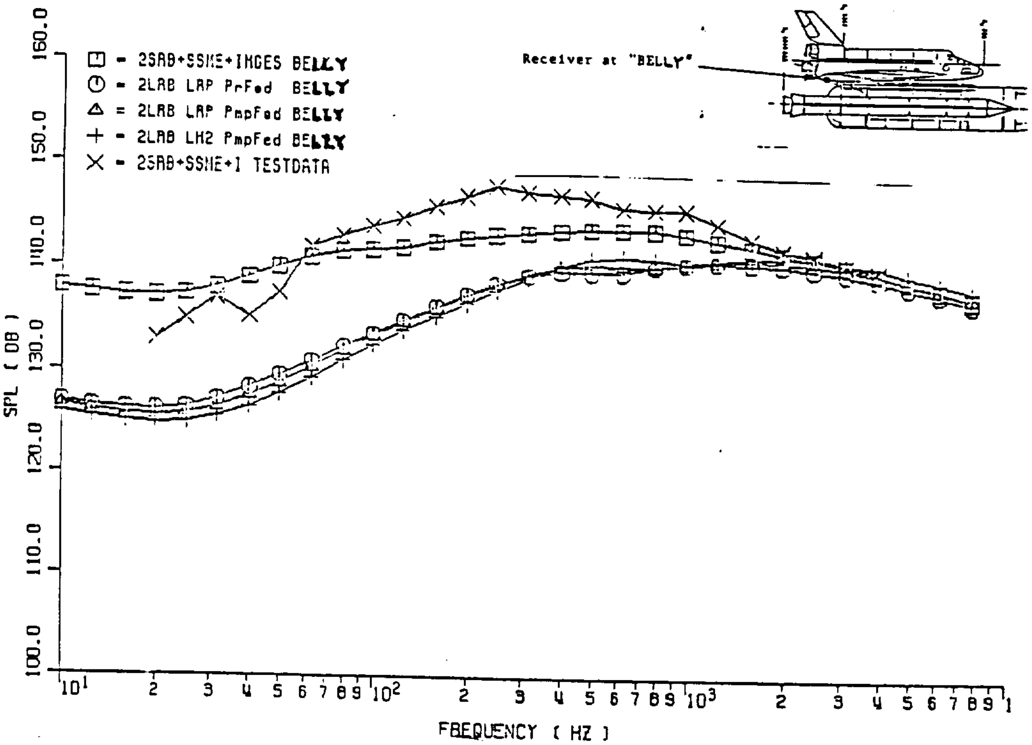

Figure 4-1. Sound Power Level vs Frequency Comparison at the Orbiter "belly".

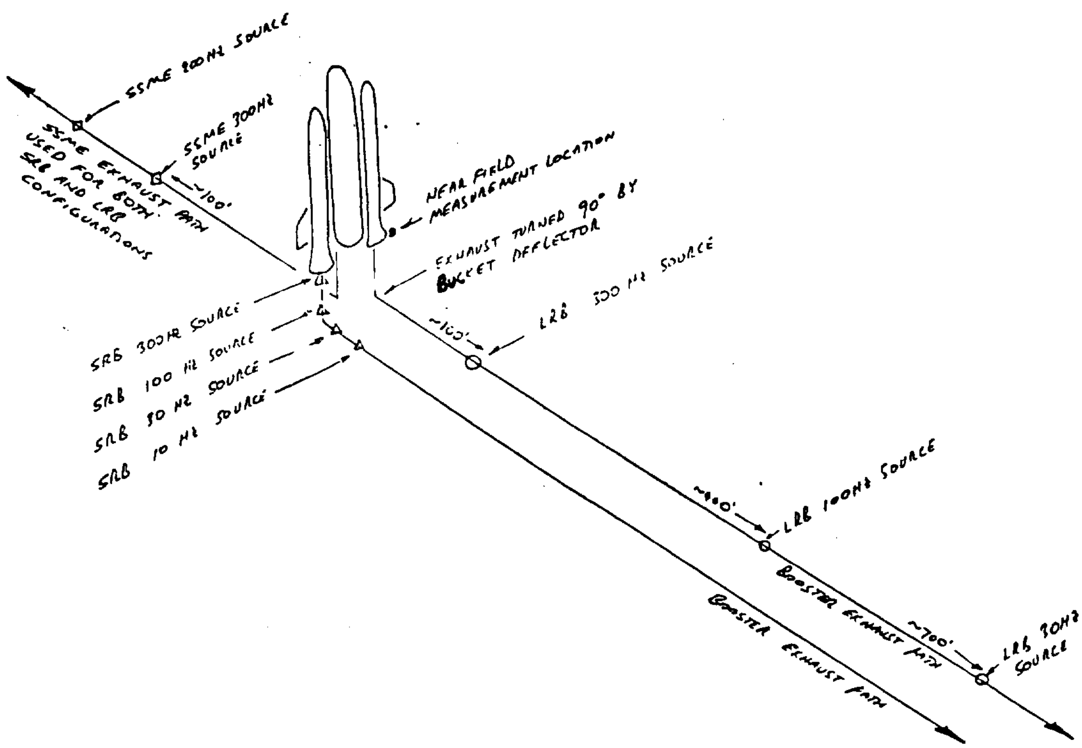

Figure 4-2. Point Source Distributions for the STS with LRB's and SRB's

The sound power level spectrum at the near-field shows the following results. At the belly, the agreement between data and the SRB space shuttle prediction is reasonable (within ±5 dB at most frequencies) as shown in Figure 4-1. At the wing, the agreement between the Space Shuttle test data and the Shuttle (with SRBs) prediction is satisfactory (about ±5 dB), as shown in Figure 4-3. At low frequencies, the mismatch is somewhat larger, showing perhaps that further "fine-tuning" of source distributions is possible. At the tail, the agreement between the Space Shuttle test data and the SRB shuttle prediction is reasonable (within ±5 dB) over a large frequency range), as shown in Figure 4-4. At low frequencies, the prediction is higher than the test data. This effect can be lessened by "fine-tuning" the source distribution curve, as was previously mentioned. At the attach ring, the agreement between test data and the SRB space shuttle prediction is satisfactory (within ±5 dB over most one third octave band center frequencies), as shown in Figure 4-5. The low frequency mismatch can be decreased by further adjustment of the source distributions. At the tank, the agreement between the Space Shuttle test data and the SRB shuttle prediction is reasonable, as shown in Figure 4-6. At mid-range frequencies, the difference between the test data and the prediction is somewhat larger. If significant coherence between the waves off the two reflecting surfaces close to the tank is assumed, the predicted results could be further increased by 0-3 dB; this would narrow the discrepancy between measured and predicted values.

4.1.5 Far Field Results

The major conclusions are:

LRBs are louder than SRBs at 1000 feet (Figure 4-7) because the modeled point source distribution for SRBs is much closer to the nozzle exit than for LRBs (see Figure 4-8). The LRB point sources are far down the exhaust path. They are closer to the far-field receivers, especially the receiver at 1000 ft. on the exhaust path. The latter receiver perceives the biggest difference between the loudness of the LRBs and SRBs. Once again, the SSMEs are present in both con-figurations and add the same amount to the rms acoustic power for both configurations.

The pressure-fed LRB is louder than the pump-fed LRBs at frequencies below 60 Hz (see Figure 4-7). This is because below 60 Hz, acoustic power scales as the product of weight flowrate, exit velocity, and effective diameter: (WVD) this product is almost twice as big for the pressure-fed system. Above 60 Hz, acoustic power scales roughly as WV^3/D. This product is smaller for the pressure-fed system. Therefore, the pressure-fed system is quieter than the pump-fed system above 60 Hz.

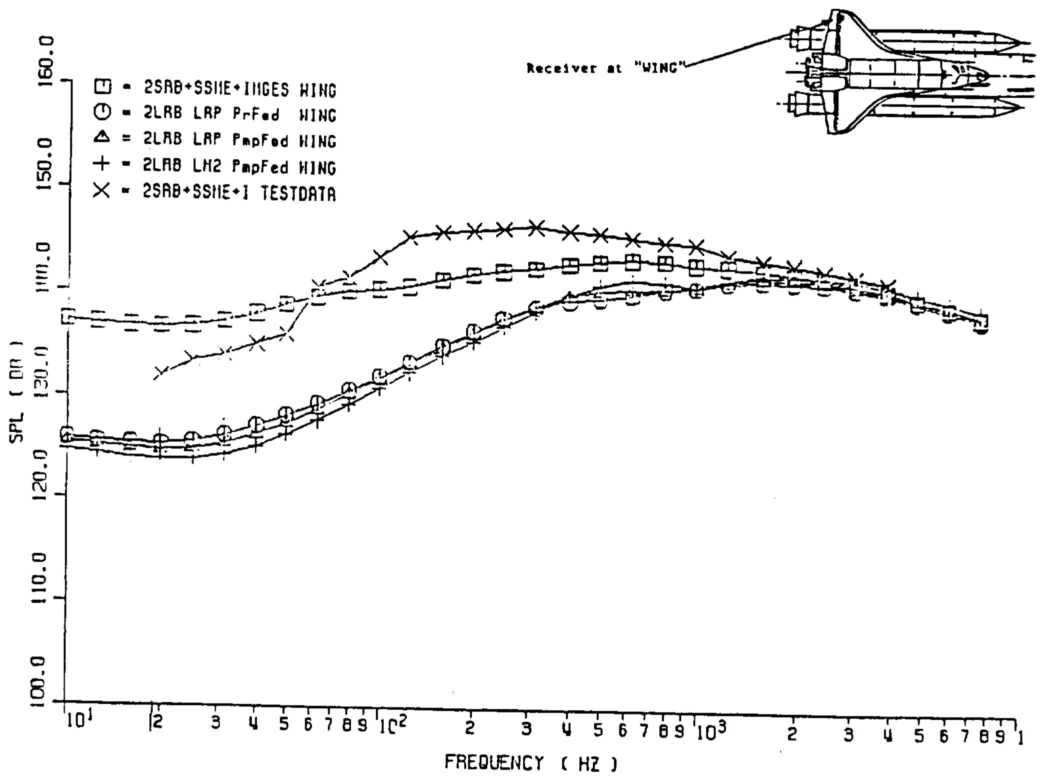

Figure 4-3. Sound Power Level vs. Frequency Comparison at the Orbiter Wing.

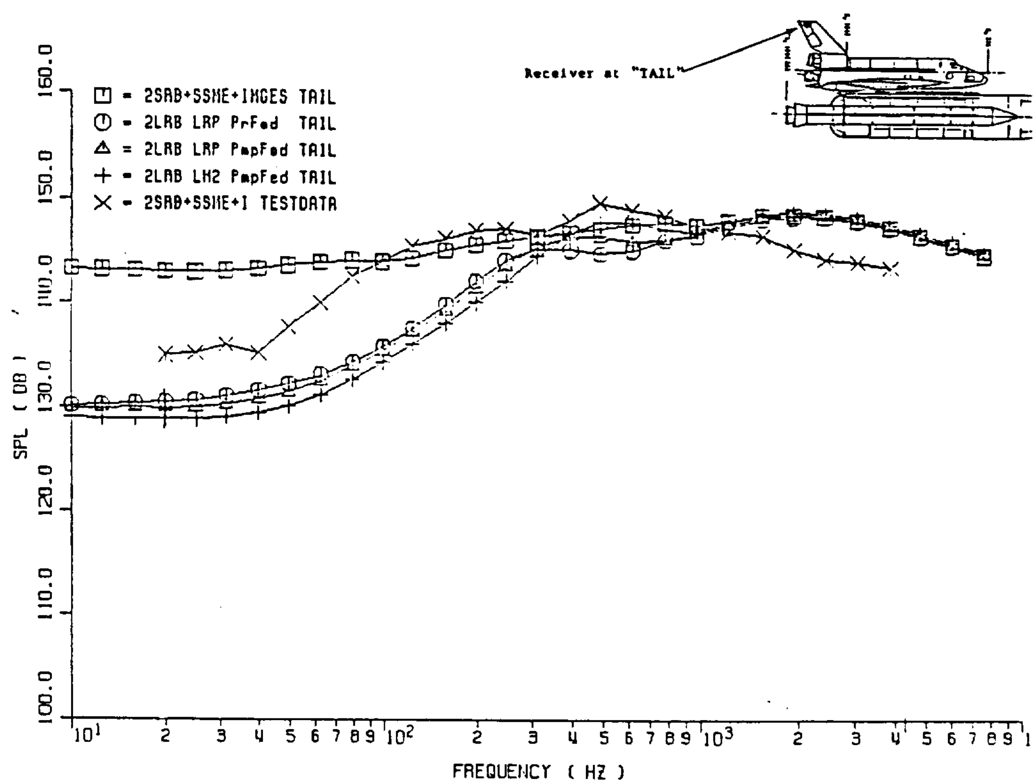

Figure 4-4. Sound Power Level vs. Frequency Comparison at the Orbiter Tail.

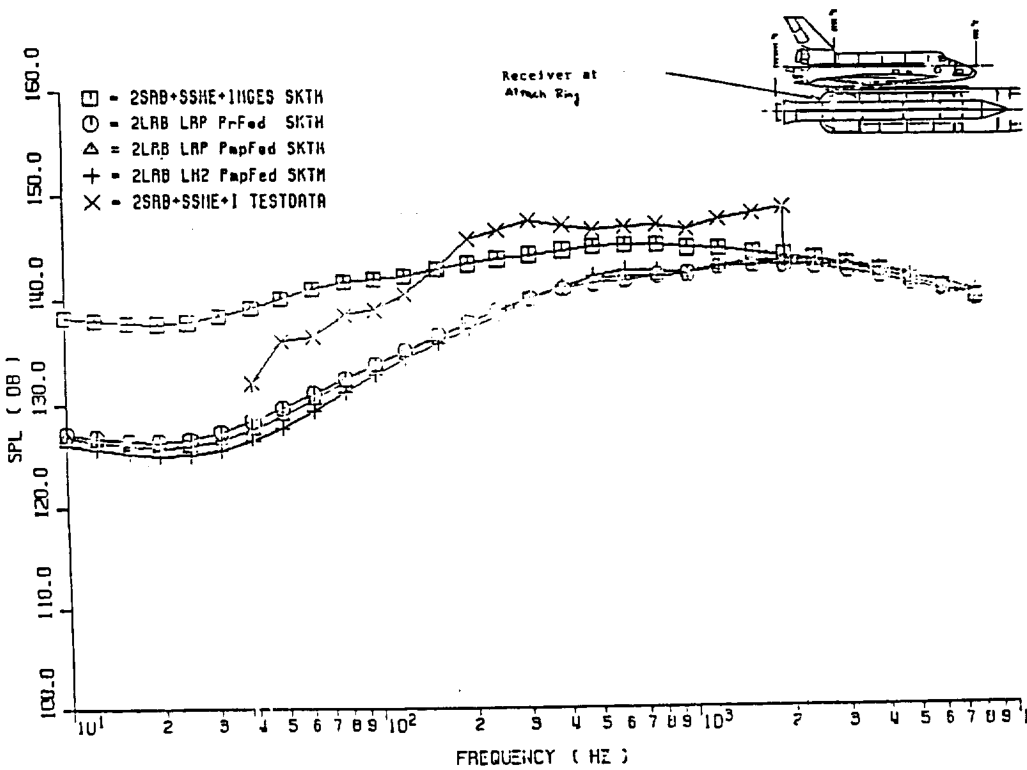

Figure 4-5. Sound Power Level vs. Frequency Comparison at the Orbiter Attach Rin

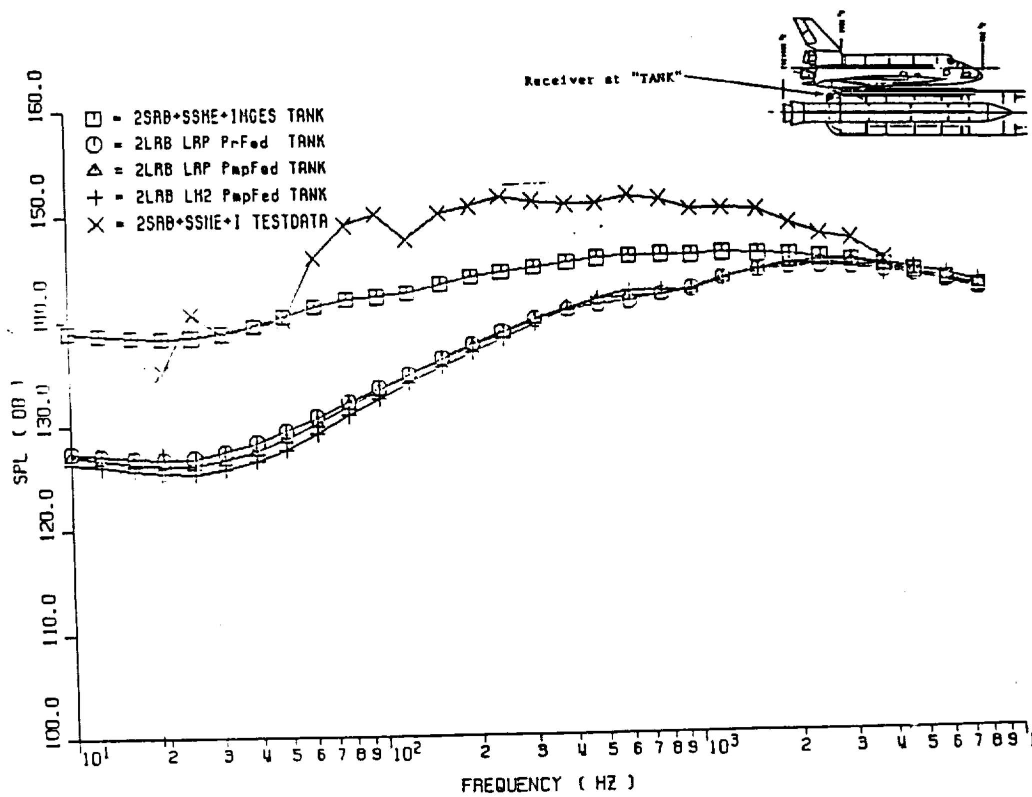

Figure 4-6. Sound Power Level vs. Frequency Comparison at the ET Bottom.

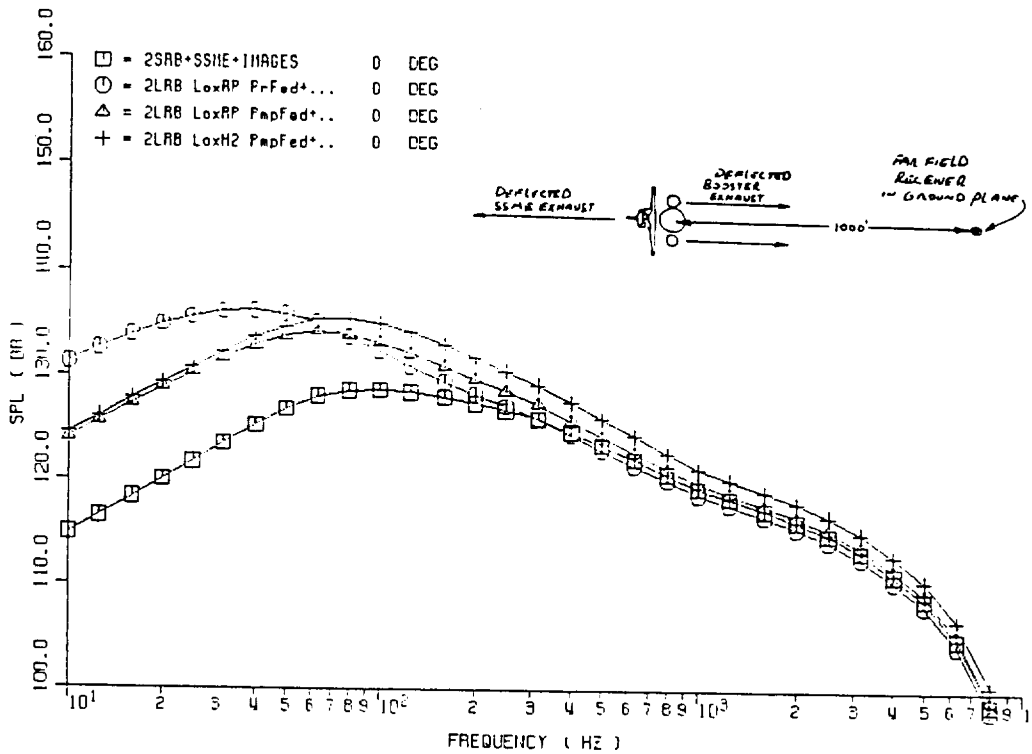

Figure 4-7. Far Field Sound Power Levels vs. Frequency Comparison at 1000 ft. at the Ground Plane.

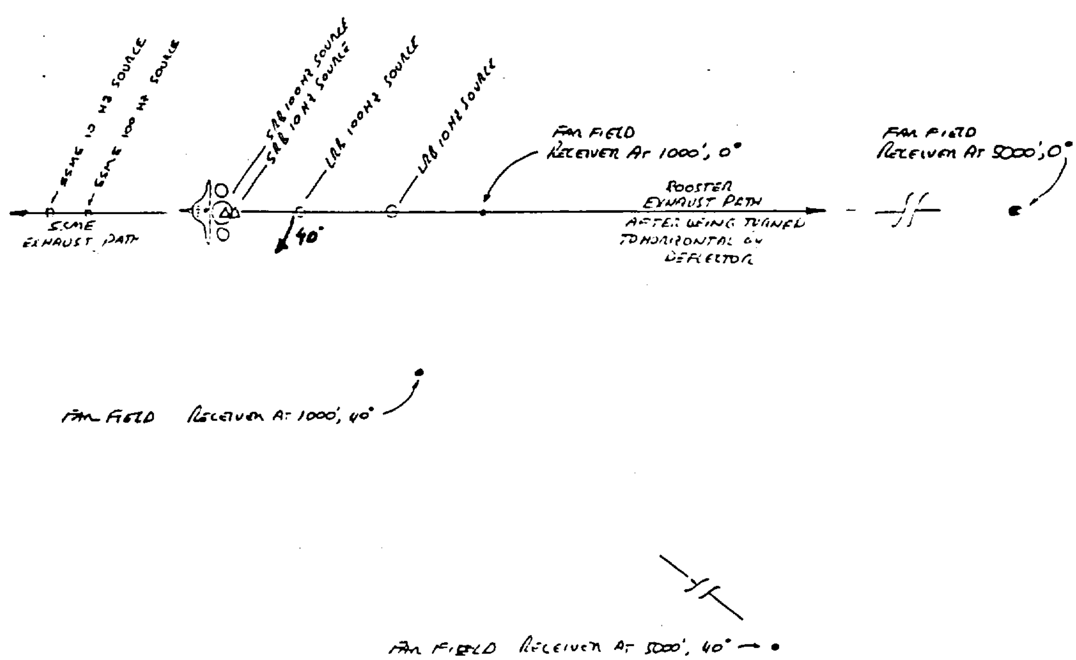

Figure 4-8. Point Source Distributions for LRBs and SRBs, for the Far Field Analysis

All the engine systems are louder in the low frequencies at a 40° angle from the deflected exhaust than in line with the deflected exhaust (see Figure 4-9). This is a well documented effect of jet acoustics and was incorporated into the prediction program using the published empirical far-field directivity data mentioned previously.

All the engine systems sound about the same above 5000 ft (see Figure 4-10), because the point source distributions that has such a strong effect closer-in look about the same at very large distances. The effect of different engine sizes is no more than 3 dB. The dominant effect at large distances is directivity and absorption. These are independent of the actual engine design.

The results at 5000 ft and an angle of 40° to the deflected exhaust (Figure 4-11) show two previously noted effects: All the engine systems are louder at low frequencies at a 40° angle from the deflected exhaust than in line with the deflected exhaust. All the engine systems sound about the same for distances greater than 5000 ft (see Figure 4-11).

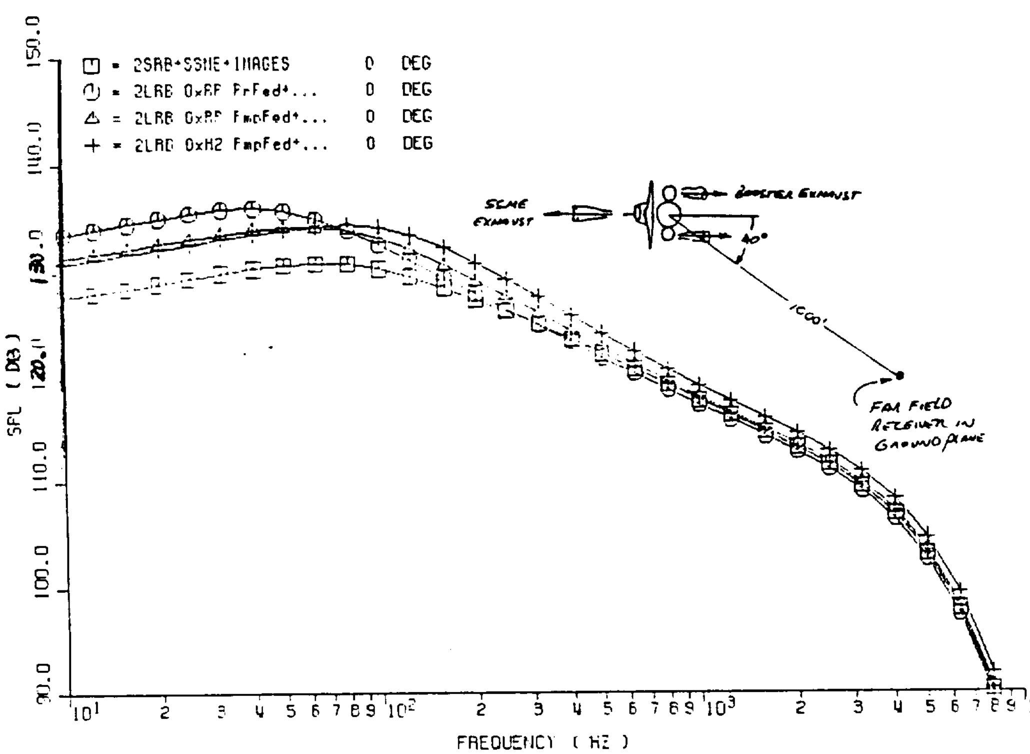

Figure 4-9. Sound Power Levels vs. Frequency Comparison for a Distance of 1000 ft. at an Angle of 40°.

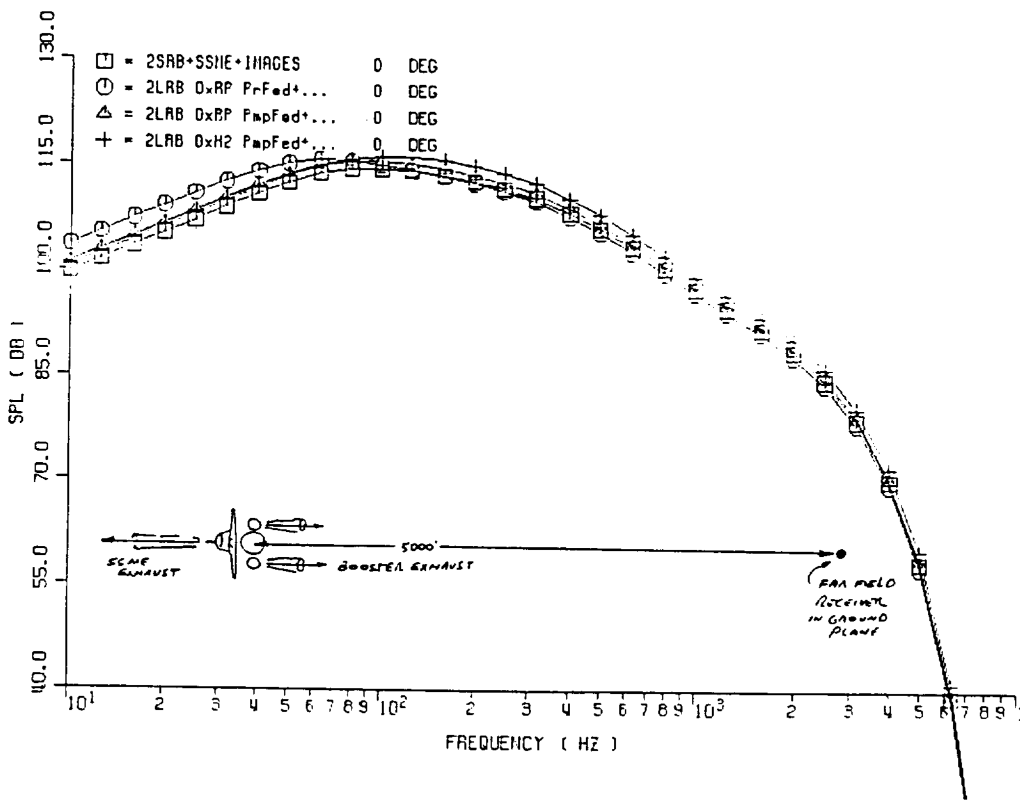

Figure 4-10. Sound Power Levels vs. Frequency Comparison for a Distance of 5000

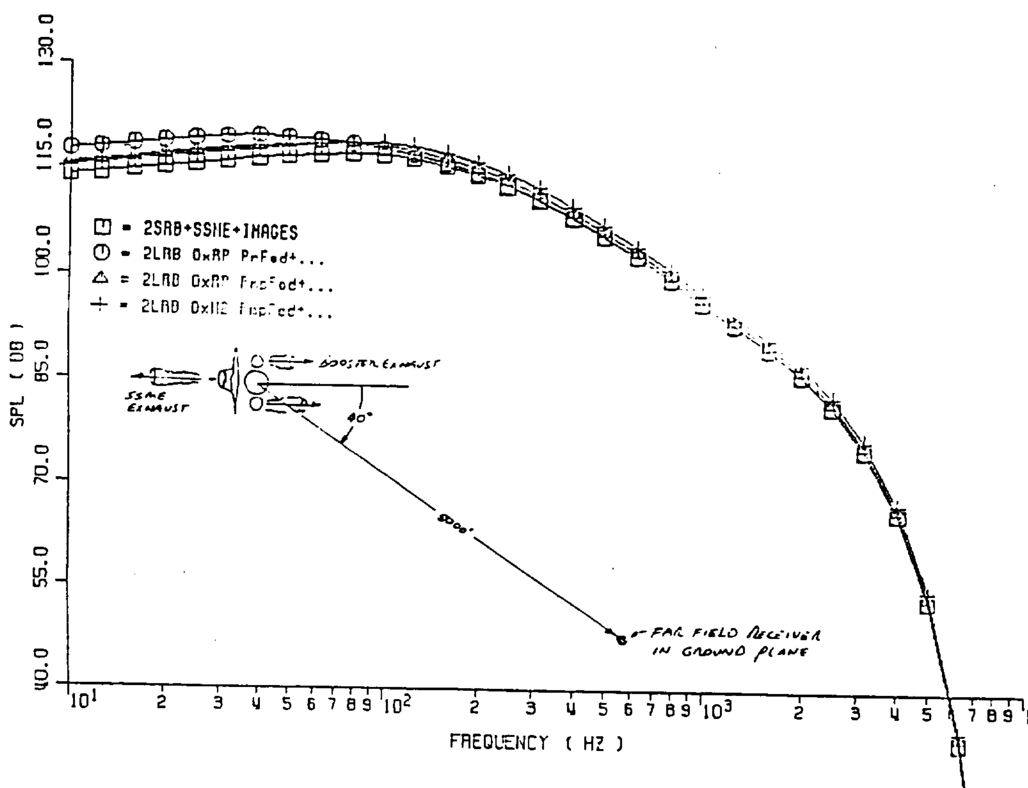

Figure 4-11. Sound Power Levels vs. Frequency Comparison for a Distance of 500 ft. at an Angle of 40°

4.2 POGO SUPPRESSION

The analysis methods shown in the previous report, (the LRB Phase II Study Report) were utilized to arrive at an estimate of the size and weight of the required POGO suppression hardware for the pressure fed and pump fed LRB engine options. However, only a part of the required dynamic characterizations of the vehicle as shown in outline form in Table 4-1, was available.

Table 4-1. Requirements for POGO Evaluation Analysis

*Free-Free Vehicle Mode Information
*Points at elbows and area changes
*Slip condition in propellant ducts
*Tank pressure coefficient
*Propellant Feed System
*Wave equation segments for ducts
*Losses at ends of segments
*Forces and motion at elbows, etc.
*Engine
*Flow line segments dynamic characteristics
*Manifold compliances
*Injector resistances
*Quasi steady state combustor response
*Thrust = (I_{sp}) (flow) assumed applicable

Preliminary analyses were made using available data. A preliminary analysis of the engine and its oxidizer feed system, assumed to be represented as shown in Figure 4-12, was made resulting in the gain vs. frequency characterization shown in Figure 4-13.

The difference in the height of the peaks (the values at the peaks compared with those in parentheses) for a doubling of the chamber pressure are relatively small. The choice of a nominal chamber pressure for the LRB engine is thus not importantly influenced by this part of the vehicle dynamics, at least for these preliminary predictions.

The values in Figure 4-13 are for a fully loaded LOX tank at lift-off. As the liquid level in the LOX tank is lowered, as the flight progresses, the frequency at which the peaks occur will

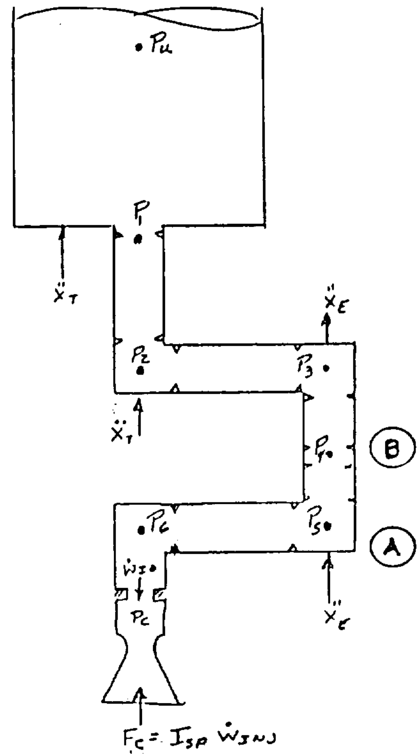

Figure 4-12. Oxidizer System Schematic for POGO Analysis

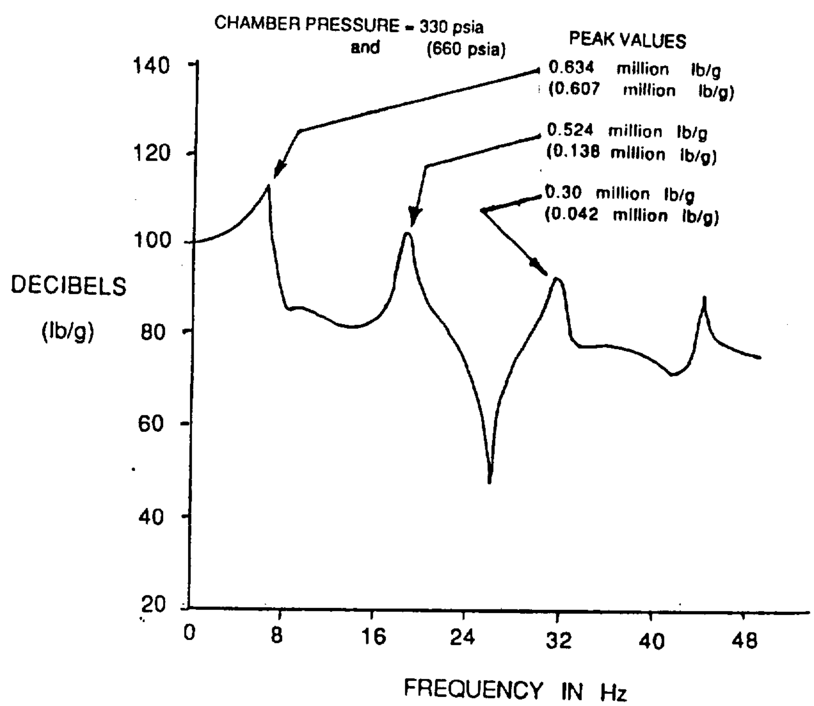

Figure 4-13. LRB Pressure-Fed Engine and LOX Feed System Dynamic Gain vs. Frequency for Two Chamber Pressures.

progressively move to higher frequencies with the possibility that they pass through a resonant frequency of the rest of the vehicle, which will very likely result in a predicted potentially destructive POGO condition. A complete analysis was not possible since the dynamic characteristics (frequency and phase response) of the balance of the vehicle was not available. However, an approximation of the size of a passive POGO suppression device was desired to determine its practicability. Assumptions were made regarding vehicle response based on previous STS POGO studies. This resulted in an estimated POGO suppressor having a volume of about 10 to 15 cubic feet for each engine, located just upstream of the LOX pre-valves, at A in Figure 4-12. To be at all effective, however, a built-in flow resistance in each LOX line located just up-stream of the suppression device (at B in Figure 4-12) is required having about 60 psi pressure drop at the nominal LOX flow rate. This is of course undesirable because it raises the required nominal LOX propellant tank pressure by the same amount. An additional advantage is realized, however, in that it provides an isolating resistance between each of the four engines, decreasing the possibility of pressure variations in one combustion chamber coupling into those of the other engines. The suppression device is more fully described below.

4.2.1 POGO Suppression Devices

The sizes and characteristics of POGO suppressors for the pressure fed and pump fed LRB engines are compared in Figure 4-14, and three slightly different POGO suppressor device concepts are shown in Figure 4-15. In each case a spherical or cylindrical volume surrounds the LOX feed line with connecting ports at the bottom to allow rapid propellant flow in and out of the volume, thus suppressing feed line flow oscillations to the engine. The action is similar to that of a piston accumulator. Concept 1 uses very little helium, since it is filled with helium only once just before liftoff. However, as vehicle acceleration is increased, the gas volume will decrease due to an increase in static head pressure. Counteracting this is a decreased static head due to lowering of propellant level in the tank. Concept 2 maintains the gas volume independent of the static pressure, but requires a small helium bleed flow throughout the boost period. Trade-offs can be made during more detailed design efforts. Meanwhile, concept 2 is considered the suppression sytem of choice at this time. Concept 3 has an active liquid level control to ensure that the static volume remains relatively constant.

- Pressure Fed Engine -- Oxidizer Side Only
  - Approximately 10 to 15 cubic feet of gas (Helium)
  - About 20% liquid for slosh control
  - Size is 3.25 to 3.5 foot diameter sphere concentric with 14 inch diameter line, or a 2.5 foot diameter by 5 foot long annular cylinder
  - Concepts 1,2 and 3 in Figure 4-15 are feasible
  - Upstream flow resistance required
- Pump Fed Engines -- Oxidizer Side Only
  - 1 Cubic foot envelope
  - Concepts 1 and 2 most easily used
  - Difficult to provide low pressure drain for concept 3

Figure 4-14. Outline of Estimated Suppressor Requirements on a Per-Engine Basis Comparing a Pressure Fed Engine with a Pump Fed Engine.

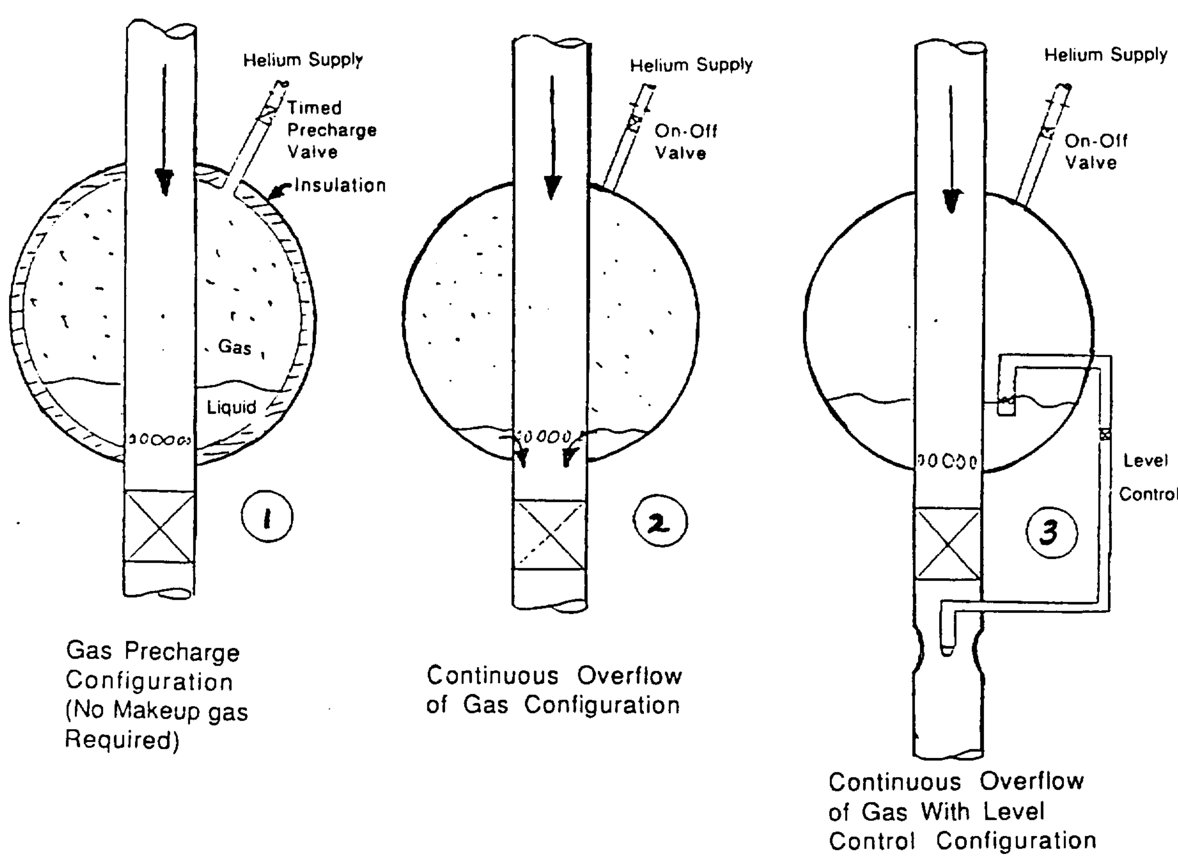

Figure 4-15. Schematics for POGO Suppressor

4.2.2. Influence of Chamber Pressure on POGO Suppression Device Size

To aid in selecting the LRB chamber pressure, the influence of chamber pressure on POGO suppressor size was determined at the lowest frequency peak shown in Figure 4-13. The results are shown in Figure 4-16. For small suppressor sizes below about 3.5 cu. ft., there is an advantage to having a high Pc; however, when the size of the suppressor reaches 3.5 cu. ft. or larger, the required suppressor size for a given thrust per g is smaller for the lower chamber pressures.

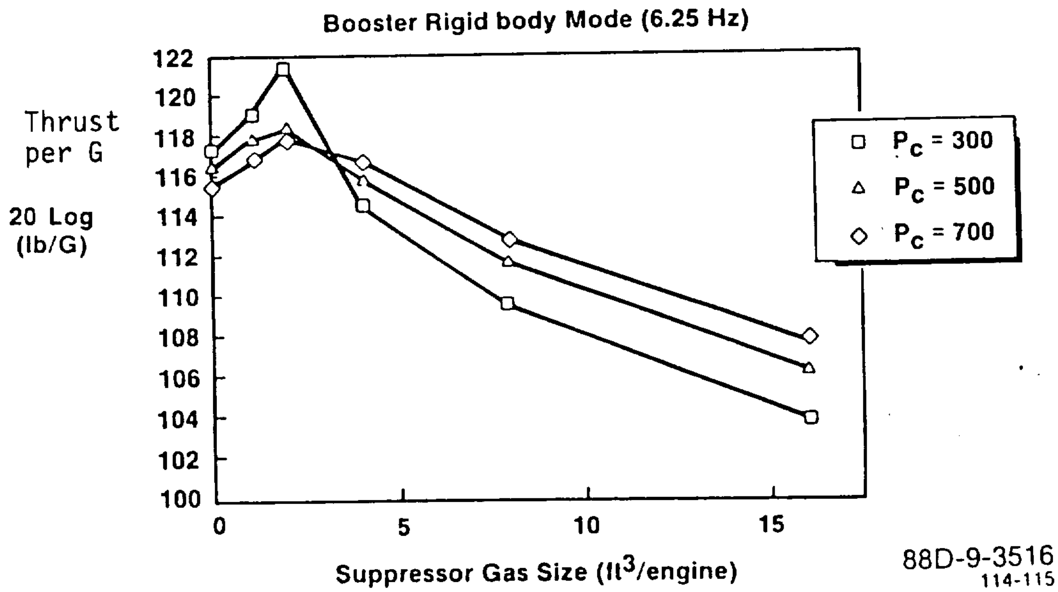

Figure 4-16. POGO Response - Pressure Fed Engine Thrust per G at Lowest Resonance Point

Note that for a pump fed engine the Pc might be 2250 psia, which, if the graph in Figure 4-16 is applied would require a large POGO suppressor. However, a pump fed engine has very low propellant inlet pressures, on the order of 1/10 as great as those for the pressure fed engine. This overrides the influence of chamber pressure on POGO suppressor size.

4.3 CONTROL SYSTEMS

4.3.1. Pressure Fed Engine - Open Loop Control

A simplified block diagram of an open loop control system for an LRB pressure fed engine is shown in Figure 4-17. The main oxidizer and fuel valves are utilized for both shut off and throttling functions and are provided with precise valve positioning controls. The engine is calibrated on a thrust stand so that the map relating valve positions, thrust and mixture ratio as a function of engine propellant inlet pressure and temperature are precisely known. Input to the system is a desired thrust level signal and a mixture ratio signal for implementating the propellant utilization system. Valve positions are automatically calculated based on propellant inlet pressure, temperature and the engine calibration map. Valve position actuators and controls then position the valves to the calculated positions.

An alternate is to use step control by opening and closing by-pass passages in the valve assembly. However, this does not permit independent fine control of the mixture ratio required if a propellant utilization system is utilized.

Another possible system is to utilize step control of the LOX valve to change the thrust level, and utilize a continuously variable fuel valve for implementing the propellant utilization system.

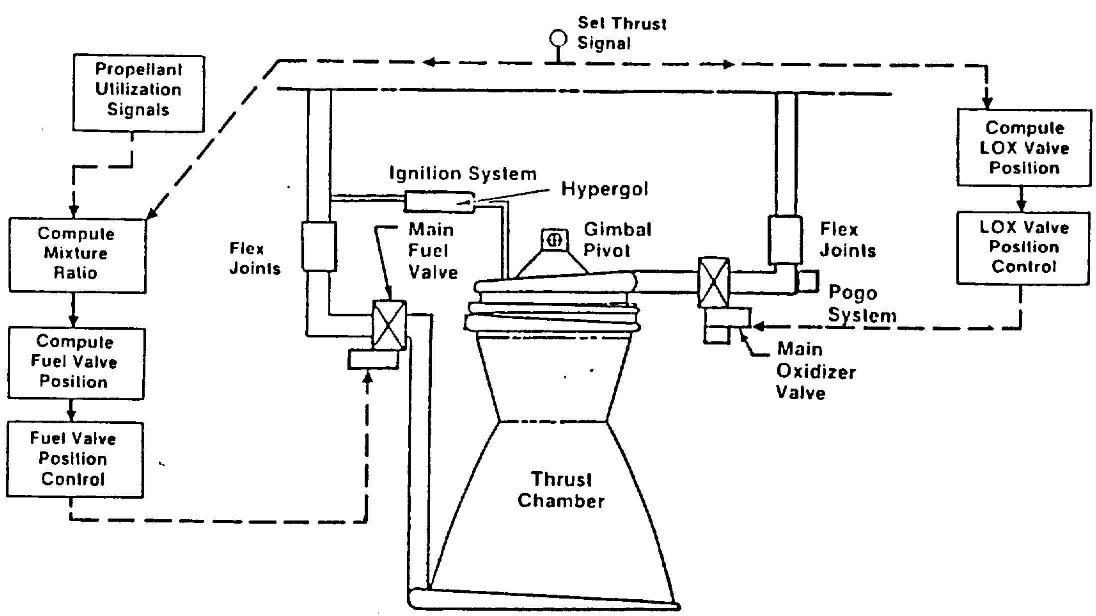

Figure 4-17. Pressured Fed Open Loop Control Block Diagram.

As long as the number of steps in a step control system does not exceed 3 thrust levels, then the step control system is estimated to be less costly than the continuously throttled system.

4.3.2. Pressure Fed Engine - Closed Loop Control

A simplified block diagram of a closed loop version of the system shown in Figure 4-17 is shown in Figure 4-18. A required thrust signal is electronically compared with a calculated thrust based on a chamber pressure measurement; any difference is amplified and utilized to reposition the LOX valve in such a direction as to minimize the difference between the required thrust and the measured thrust. The fuel valve is initially positioned to a calculated position based on the thrust signal. Its final position is however adjusted determined by a required mixture ratio signal based on propellant utilization requirements. This mixture ratio signal is compared with a measured mixture ratio signal based on the measurement of the propellant flow rates. Any difference between the measured and required mixture ratio signals is amplified and causes the fuel valve to be adjusted in such a direction as to reduce the mixture ratio error to near zero.

The main disadvantage of the above system is the complexity and risk associated with the flow and thrust measuring means. The closed loops substantially increase the number of critical failure modes. The chamber pressure measuring sensors must be provided with redundancy and voting circuit capability. The flow sensors must be highly reliable since to make them redundant is probably undesirable from a size, weight and cost standpoint.

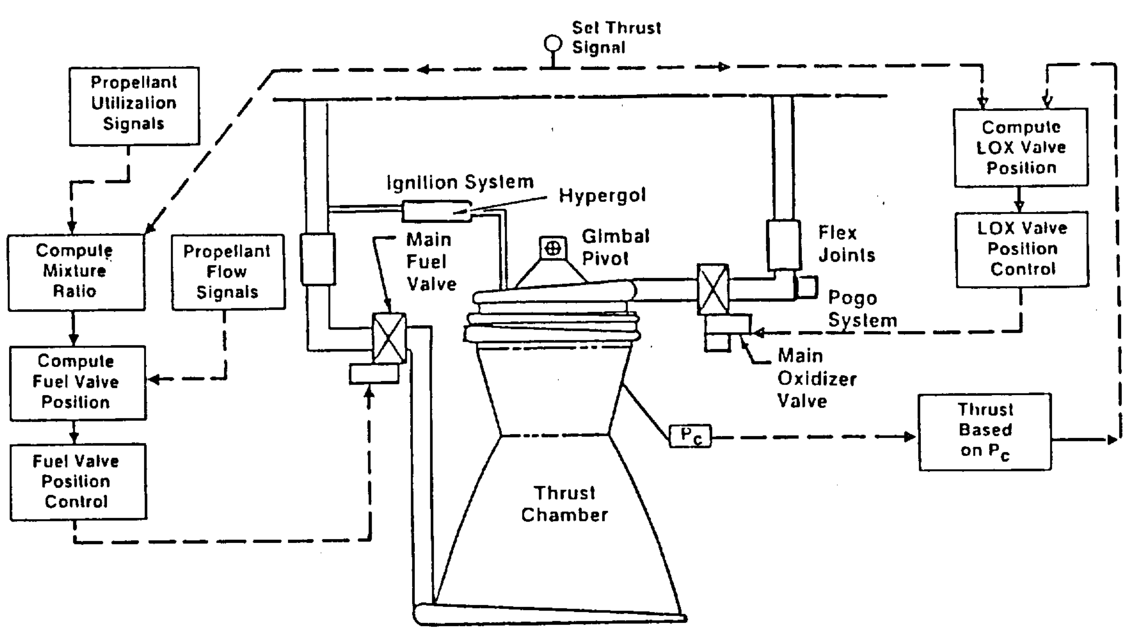

Figure 4-18. Pressure Fed Closed Loop Control Block Diagram

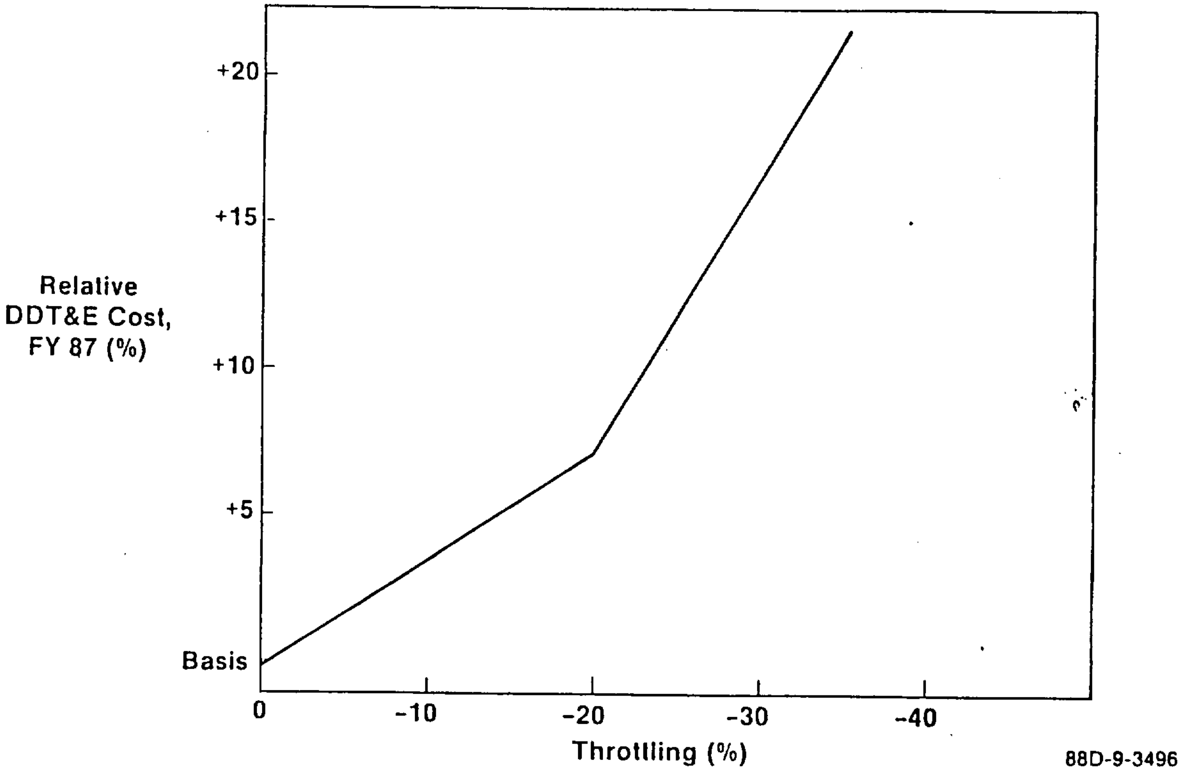

Figure 4-19. Estimated Relative LRB DDT&E Costs of a Pressure Fed LOX/RP-1 Engine vs. Depth of Throttling.

A study was made to determine the relative increased costs associated with engine throttling as related to throttling depth. The results are shown in Figure 4-19. The increased costs for throttling deeper than 20% are due to the relatively little existing test data on pressure fed engines in this range. Extra testing to extend the data base is an anticipated requirement.

4.3.3. Pressure Fed Engine Control

When viewed primarily from an engine standpoint, the system utilizing an open loop control with step throttling of the LOX and fuel valves with provision for continuous adjustment or trimming of the fuel valve within narrow mixture ratio limits around each step throttle point is considered the best. This system provides the advantage of a low number of failure modes and yet allows close adjustment of the mixture ratio.

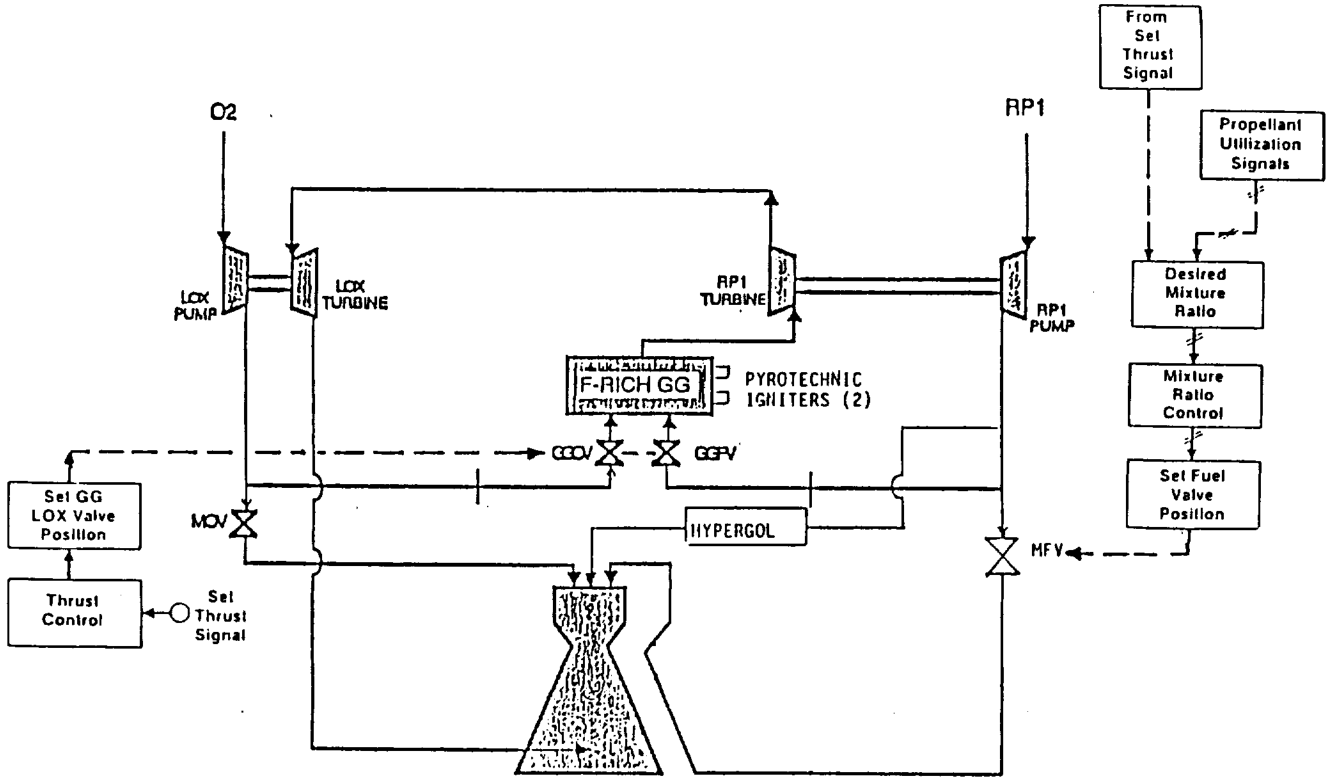

Figure 4-20. LRB LOX/RP-1 Pump Fed Open Loop Control Block Diagram.

4.3.4. Pump Fed LOX/RP-1 Engine - Open Loop Control

A simplified block diagram of an open loop control system for an LRB LOX/RP-1 gas generator cycle utilizing a single shaft turbopump design is shown in Figure 4-20. Throttling is achieved by controlling the flow of LOX to the gas generator (GG). The fuel flow to the GG is not controlled. When a lower thrust is desired, decreasing the LOX flow to the GG results in (1) decreased mass flow rate to the turbine, and (2) decreased temperature of the gas flowing to the turbine due to a decrease in GG mixture ratio. Both simultaneously reduce the power generated by the turbine. Throttling up 10% does not elevate the gas temperature enough to damage the turbine blades. Since the fuel pump and LOX pump are on the same shaft, their respective outputs during throttling tend to track, automatically holding the mixture ratio relatively constant. However, a continuously variable fuel throttle valve in the main fuel line is provided having only limited travel to permit fine adjustments of the mixture ratio based on thrust level and propellant utilization requirements.

Similar to the pressure fed open loop system described previously, the accuracy with which thrust and mixture ratio are controlled depends primarily on the accuracy with which the engine has been calibrated on a test stand. The valve positions are based on the engine performance map, the inlet propellant temperatures and the thrust and mixture ratio demand signals.

4.3.5 Pump Fed LOX/RP-1 Closed Loop Control

Figure 4-21 shows a block diagram for the closed loop control system for the Pump Fed LOX/RP-1 single shaft turbopump engine. Thrust is measured based on a chamber pressure sensor output and compared to the desired thrust level signal. Any difference is amplified and sent to the GG LOX throttle valve to reposition it in such a direction as to decrease the error between the desired thrust and the measured thrust. The main propellant flow rates are measured and the mixture ratio calculated and compared to a desired mixture ratio. Any difference is amplified and utilized to move the fuel throttle valve in such a direction as to minimize any error in mixture ratio. The accuracy and reliability of this system relies heavily on the accuracy and reliability of the chamber pressure measuring means and to the propellant flow rate measuring means. Testing costs of the engine are reduced compared with an open loop system. However, calibration and first cost of flow measuring and chamber pressure measuring systems will off set this.

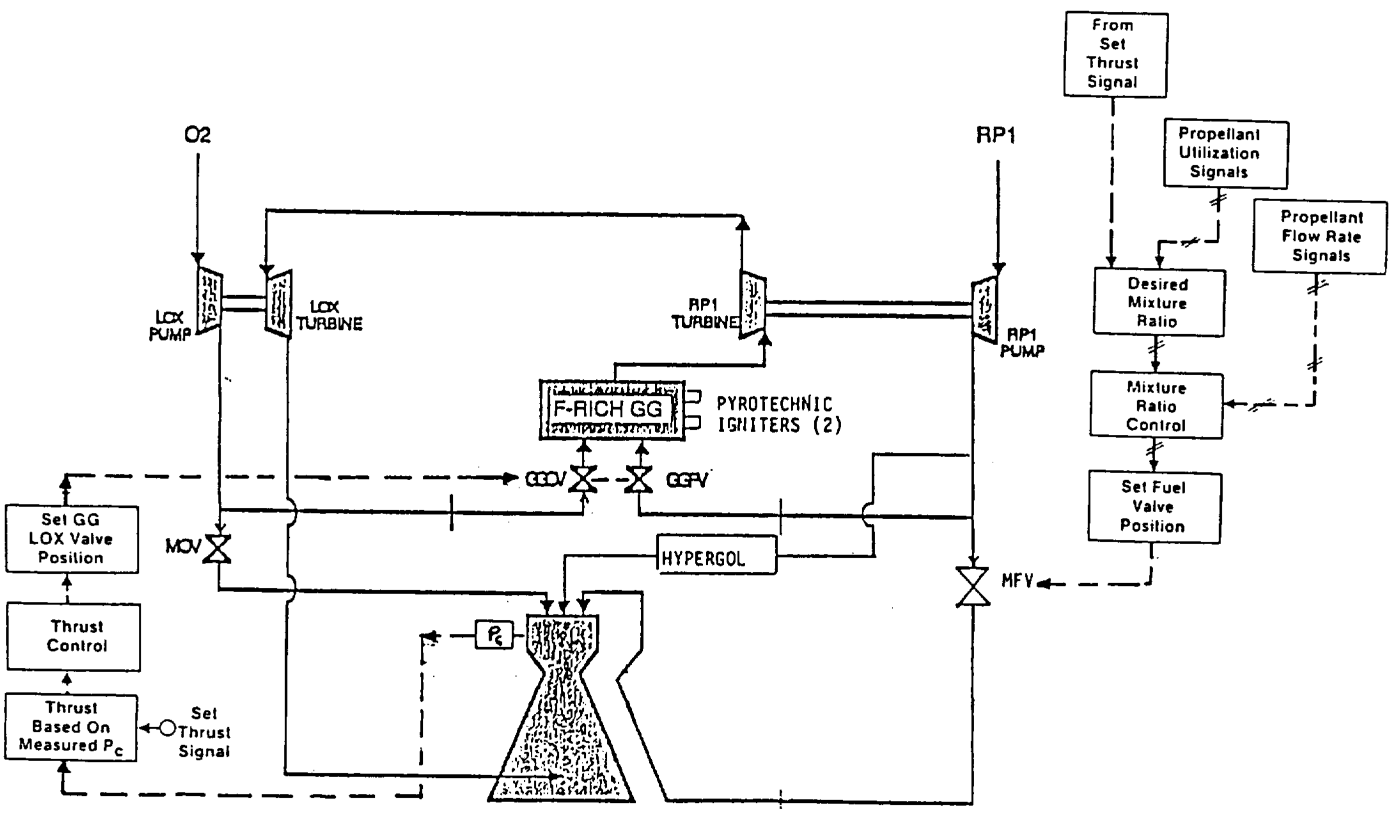

Figure 4-21. LRB LOX/RP-1 Pump Fed Closed Loop Control Block Diagram

4.3.6 Pump Fed LOX/LH_2 Engine - Open and Closed Loop Control

In general, the characteristics and block diagram are essentially the same for this control system as for the pump fed LOX/RP-1 engine described above. The main difference is that there are two separate turbopumps on two different shafts. The control method is essentially the same except that the fuel throttle valve may require a greater range of flow adjustment to compensate for the fact that the two propellant flow rates may not track as accurately under throttled conditions since the two turbopump shafts are physically not linked together.

4.3.7 Control System Conclusions

As for the pressure fed engines, the open loop control system has fewer sensors than the closed loop system since chamber pressure and propellant flow rate measuring devices are eliminated. The open loop control system is recommended with step control on the thrust and vernier adjustment of the main fuel flow rate to permit fine adjustment of the mixture ratio. This will require more extensive test stand calibration than for the closed loop version, but costs are off set by not requiring flyable main propellant flow sensors and a redundant, voting chamber pressure measuring system for thrust determination.

A study was made to determine the relative cost of throttling and how cost is related to depth of throttling. The results are shown in Figure 4-22. The steep upward break in the LOX/RP-1 curve reflects a lack of data regarding stability for throttling deeper than 20%.

The relative cost increase of a closed loop control system as compared to an open loop control system, broken down into several subsystems, is shown in Table 4-2. These, however, are offset by an increased testing and calibration effort required for an open loop system. Closed loop systems have a typical accuracy of ±1%, while open loop systems have only a ±3% accuracy.

4.4 COMBUSTION STABILITY ANALYSIS AND INJECTOR DESIGN

4.4.1 Introduction

The objectives of this analysis were to define the best combustion chamber internal geometry and injector design features for a high performing, stable engine. Through the comparison of the predicted performance and stability characteristics for various design options, the optimum design was selected. Emphasis was placed on the trade off between the design chamber pressure and the expected performance efficiency and stability requirements.

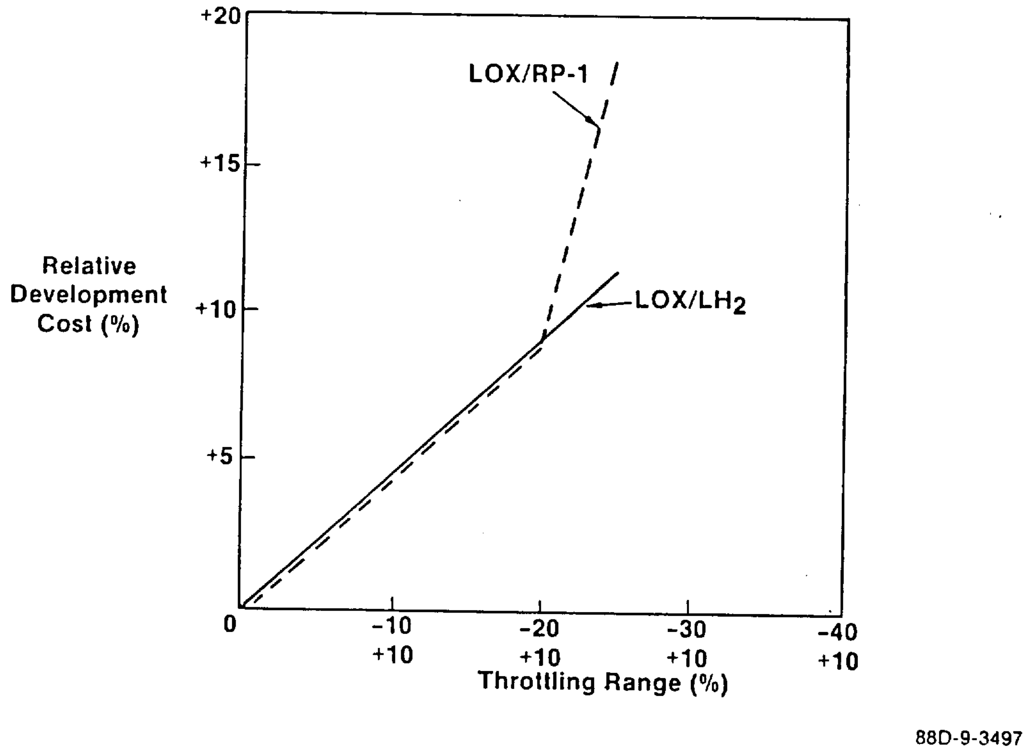

Figure 4-22. LRB DDT&E Cost vs. Throttling Range for a LOX/RP-1 Pump Fed Closed Loop Control Engine

Table 4-2. GG Control System Relative Cost Comparison Summary

	Closed Loop Requirements	Cost Increase Percent for Closed Loop
• Controller -	Increased hardware and software	125
• Sensors -	Propellant flow meters and multiple Pc sensors required -	62
• Propellant Valves -	Continuous valve Positioning Required -	120

The scope of the analysis included chamber sizing, injector element type selection, injector element size vs. pressure drop trade, stability and design choice, and the chamber pressure effect. The design selections for these items are subject to prescribed design goals for performance, stability and cost impact.

The approach to the selection of the injector design and injector operating condition utilizes a non-iterative analysis methodology. The methodology defines an injector design zone, within which the design goals or requirements are satisfied or exceeded. It also allows the display of the effects of all critical design parameters thereby lending itself to the convenience for design optimization. Figure 4-23 is a block diagram illustrating the concept and procedure of the methodology.

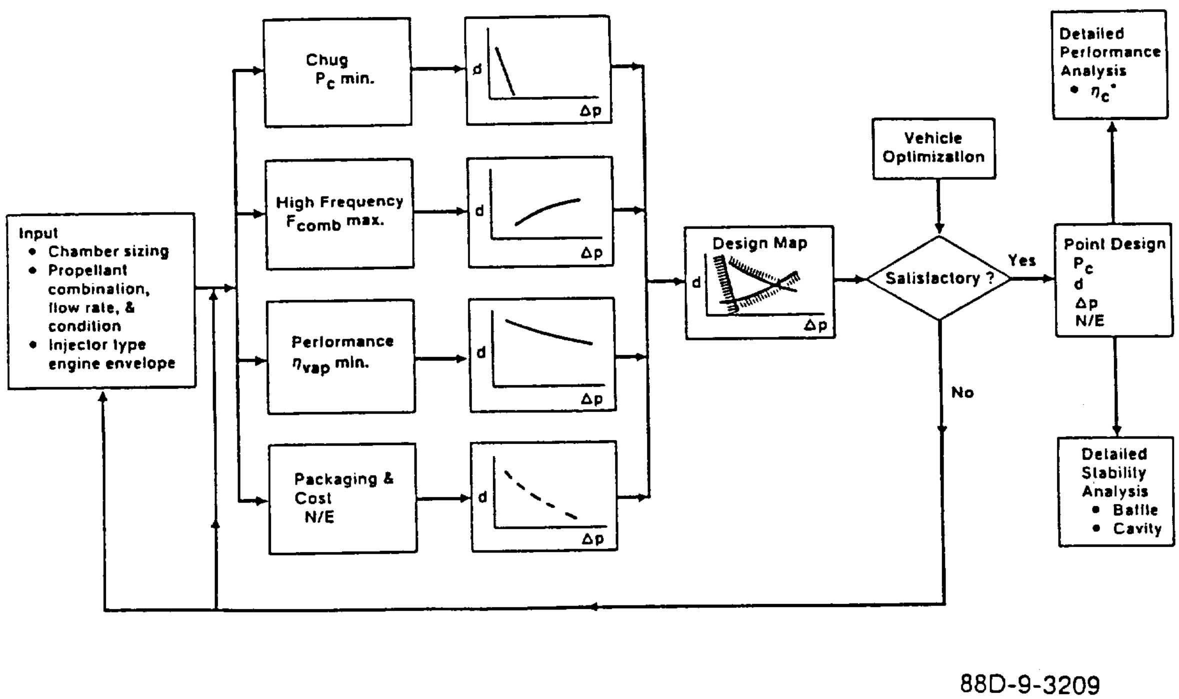

Figure 4-23. Analytical Injector Design Methodology.

The input to injector design analysis consists of chamber internal contour, propellant combination, propellant inlet conditions and flow rates, injector element type under consideration, and the engine envelope (mainly the nozzle exit area and nozzle length limitations). Four design goals are set forth; they are throttling capability limited by the chugging threshold, high frequency acoustic stability, level of combustion performance, and number of injector elements from the standpoint of element packaging and cost to produce. With use of appropriate analytical models, each of these requirements (or goals) is translated into a unique injector element size vs. injector pressure drop relationship for a selected design chamber pressure. Combination of these relationships forms an injector design zone, with which all goals are satisfied or exceeded. By evaluating the implications of vehicle or system requirements and relative technical risks or uncertainties, a point design can then be selected.

4.4.2 Input to Injector Design Analysis

Figure 4-24 is a list of the major input to the analysis for deriving an optimum pressure fed LOX/RP-1 LRB injector design. What were imposed by the vehicle application are a thrust level of 750K lbf at sea level, a maximum nozzle exit diameter of 9 ft (108 in.), and a tank pressure of no higher than 1000 psia (which confines the throat stagnation chamber pressure to a level of approximately 700 psia as maximum).

- Thrust level - 750K, sl
- Chamber sizing

Throat stagnation, pressure, psia	300	500	700
Throat diameter, in	48.3	36.6	30.3
Chamber diameter, in	63.0	47.7	39.5
Chamber residence time, ms	2.05	2.14	2.18
Chamber length, in	40.0	40.0	40.0

- Propellant flowrate

Total flowrate, lbm/sec	3157	3022	2899
Mixture ratio, 2.5			

- Injector types

  OFO triplet

  Like doublet

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Figure 4-24. Input to the Analysis of Chamber Size, Propellant, Flowrate and Injector Type for a Typical LRB Pressure Fed LOX/RP-1 Engine

Aerochemical analysis using the standard TDK computer code was performed to define the other inputs such as mixture ratio, and flowrate vs. size for each of the three reference (throat stagnation) chamber pressures, 300, 500, 700 psia. With a pre-selected contraction ratio of 1.7, the chamber diameters were then determined form the throat sizes.

Two types of injector were considered: impinging type and single element coaxial pintle type. The latter was not selected mainly due to its inherent nature of low performance and its potential for thermal incompatibility at the walls. Figure 4-25 is a summary of the comparison of these two injector types. OFO triplet and like doublet were considered for the impinging type and both were concluded to be suitable for the present application. Only the OFO triplet was carried on for further analysis whose results are reported in the subsequent section.

	Single Element Coaxial Pintle	Multiple Element Impinging

Injector Configuration	• Axial fuel annular
	• Radial outboard oxidizer jets from center body slots
	• Atomization & mixing achieved by coarse oxidizer jets impinging on fuel annular sheet
		• Three common types
		  • Like impinging
		  • Unlike triplet
		  • Unlike Doublet
		• Atomization achieved by fine impinging jets; mixing achieved by unlike spray or unlike impinging jets

Previous Experience		
Data Base	Limited	Extensive
Propellants	N_2O_4/UDMH	Large variety, including LOX/RP-1
Pc	300 psia	<100 to 2500 psia
Thrust	10K to 250K LBF	0.5 to 1.5M LBF
Chamber Length	Up to 36"	up to 40"
C* Efficiency	< 90%	≥90% to ~100%
Technical Evaluation		
Performance	Poor performance due to coarse atomization & non-uniform mixing	Higher performance due to fine atomization and better mixing
Compatibility	Potential problem at wall & downstream of oxidizer holes	Usually not a problem, due to design flexibility
Throttleability	Good throttleability by maintaining high ΔP during throttling	Deep throttle achievable
High Frequency Stability	Susceptible to tangential instability if combustion concentrated near injector periphery	Distributed combustion less susceptible to instability

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Figure 4-25. Impinging Type of Injection Element is Selected

4.4.3 Injector Design Goals and Analytical Methods

The goals to be achieved by the injector for the present study are: a 2-to-1 throttling capability, a safe operation without instability, and a performance level with fuel vaporization efficiency to be no less than 96%. Cost impact is also considered when selecting the number of injection elements. Figure 4-26 summarizes these objectives.

- Chug stability
  2:1 throttling
- High frequency stability
  Potential instability is limited to 3T or below
- Fuel vaporization efficiency
  $\geq 96\%$

88CA0929/2DVb

Figure 4-26. Three Injector Design Goals are Preselected

The main concern for the engine throttling is the chug stability. As the engine is being throttled down, both the chamber pressure and the pressure drops across the injector are reduced, with the latter being reduced at a faster rate. Chug instability can then be induced for an otherwise stable injector, as the ratio of the injector pressure drop to the chamber pressure falls below a certain threshold. This threshold is dependent upon the injector design and chamber geometry. In performing the analysis on this subject and for the high frequency stability, a simplified procedure derived from the modeling approach described in Reference 1 was used. The simplification was made by neglecting the effect of the oxidizer circuit because of its much shorter combustion time lag.

The high frequency acoustic mode stability design was carried out with both an active and passive approach. The passive approach is to utilize the stability aids consisting of acoustic cavities and injector face baffle. The active approach is to limit the potential instability to a frequency of the third tangential (3T) mode or lower, through the injector design. The analysis was performed using the modeling concept described in Reference (1) with the assumption that the fuel is the controlling propellant.

The combustion performance is fuel vaporization controlled since the liquid oxygen vaporizes more rapidly and complete its process before exiting the combustion chamber. The fuel vaporization prediction was made using the generalized length correlation of Reference 2 with modification made to the propellant drop size correlation.

Injector orifice size and number of orifices contribute significantly to the injector cost and fabricability. A large amount of small orifices requires tight tolerance and intense labor. These two design parameters are related to each other through the continuity equation.

4.4.4 Injector Design Results

Analysis to size the injector elements using the methodology shown in Figure 4-23 was made for the OFO triplet for three different (throat stagnation) chamber pressures: 300, 500, and 700 psia. The results are shown graphically in Figure 4-27. At each design chamber pressure, the design zone is identified as a region (in a orifice diameter vs. injector pressure drop diagram), surrounded by the three boundaries based on the throttling (chugging), vaporization and high frequency stability requirements. Any interior point represents a design exceeding those design goals specified in Section 4.4.3 and highlighted in Figure 4-26. That means the throttling ratio is greater than 2 to 1, the fuel vaporization efficiency is greater than 96%, and the acoustic mode instability will have a frequency lower than the 3T frequency. The dashed lines represent the cases of constant number of injection elements. Figure 4-28 illustrates two higher vaporization efficiency curves (98% and 99%) interior to the 96% boundary.

Figure 4-27 indicates the following trends. First, the design zone shifts to the larger orifice size and higher injector pressure drop region as the design chamber pressure is increased. Second, the size of the design zone becomes larger as the chamber pressure becomes higher. Third, the number of elements required reduces as the chamber pressure is increased.

The above observed trends of chamber pressure effects are further elaborated in Figures 4-29 and 4-30. The left side figure of Figure 4-29 shows that the design margin increases with the chamber pressure, hence high chamber pressure improves the level of confidence in meeting the injector design goals. The right side figure shows that, for a specific (96%) fuel vaporization efficiency, fewer and coarser injector elements are permissible as the chamber pressure becomes higher. Figure 4-30 shows the attainability of higher vaporization efficiency with rather few and coarse elements if the chamber pressure is sufficiently high.

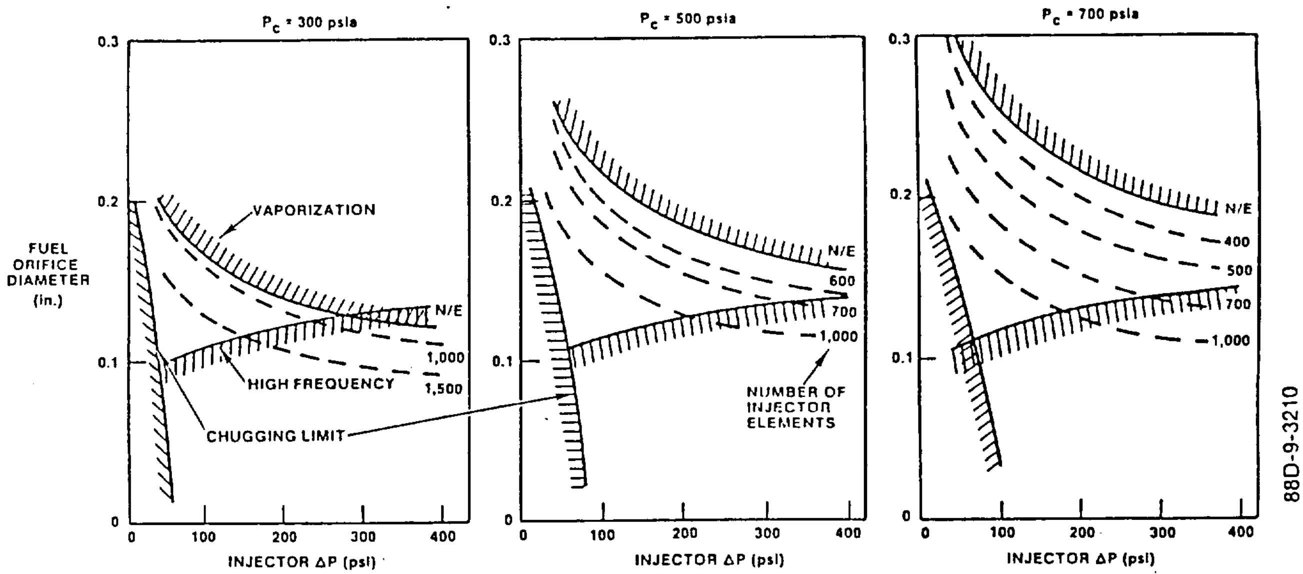

Figure 27. High Performance Injector with Desirable Stability Characteristics is Obtainable over Range of Chamber Pressure, Orifice Size and Injector Pressure Drop.

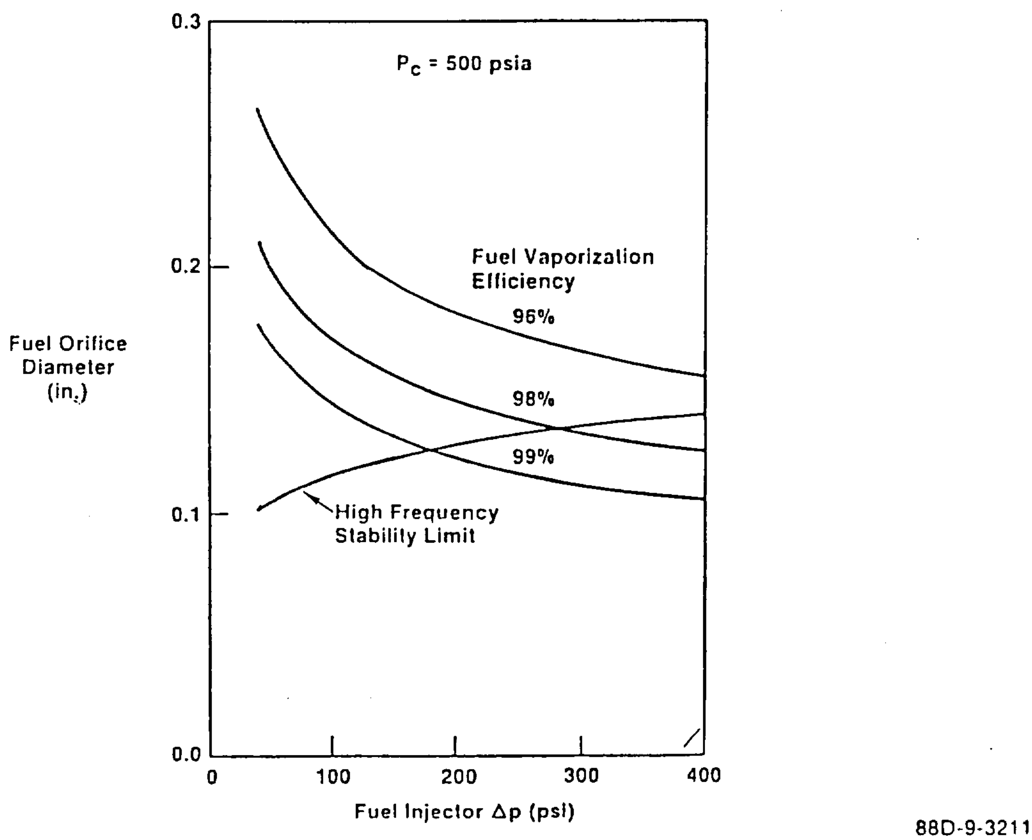

Figure 28. Performance Requirement Effect on Design Zone

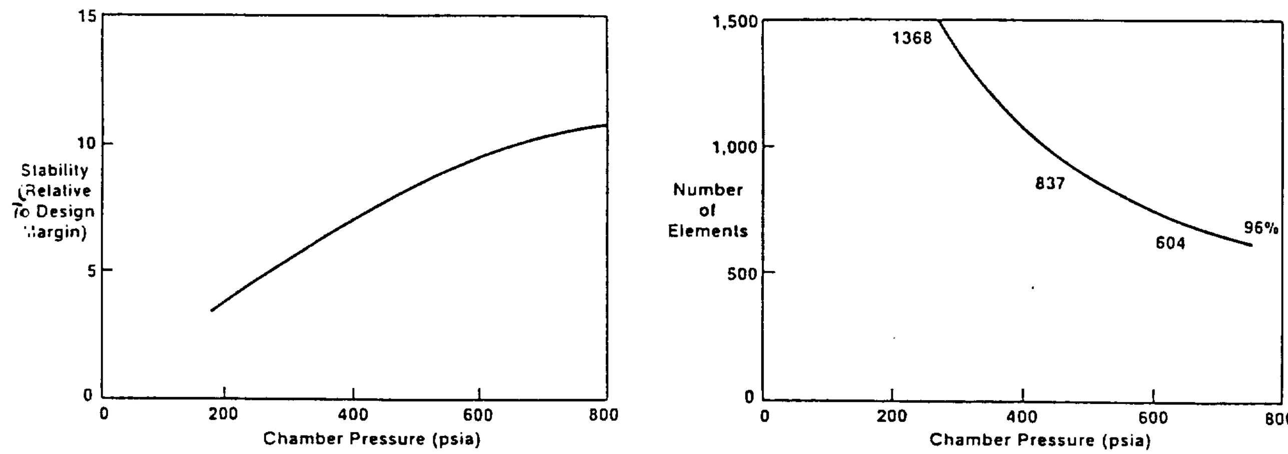

Figure 4-29. Design Margin Increases and Number of Injector Elements Decreases as Chamber Pressure Increase.

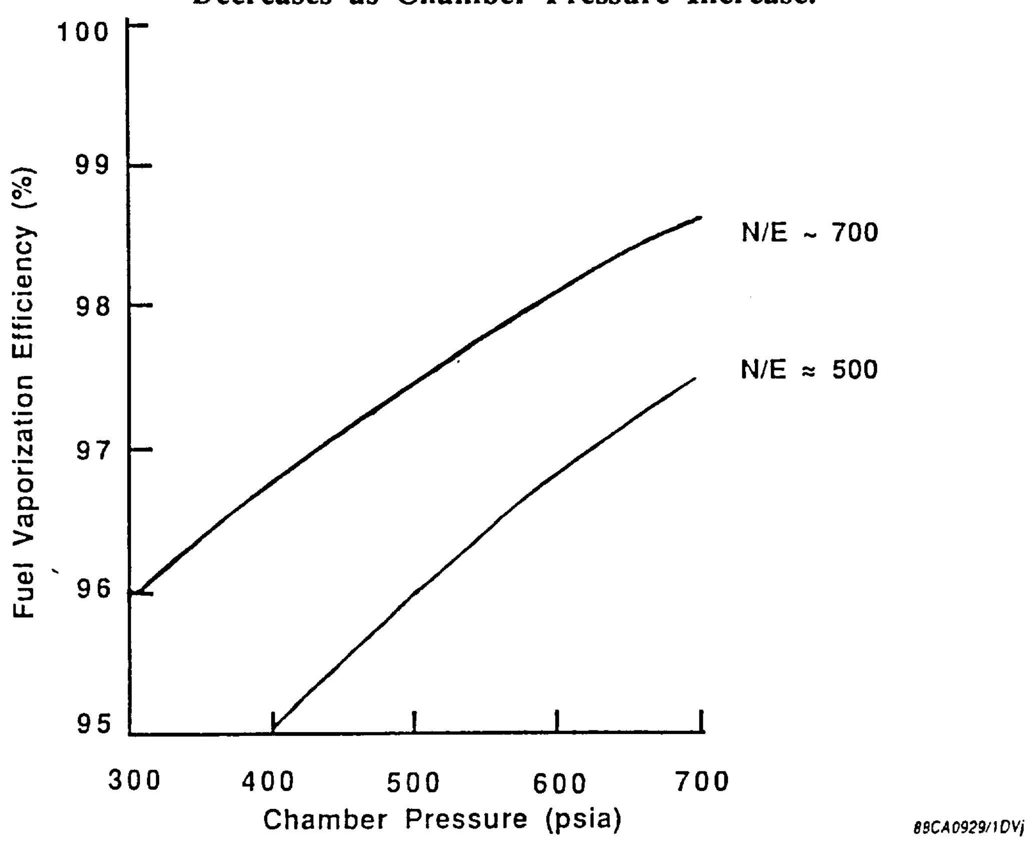

Figure 4-30. Performance Increases with Design Chamber Pressure at the Same Stability (Design) Margin.

4.4.5. Stability Aids Design

The injector design approach used in the present analysis allows the potential instability to occur in the chamber acoustic modes no higher than the 3T mode. Further reduction in the order of the allowable instability requires use of injection orifices too coarse to meet the performance requirements. As a result, stability aids are required to assure the stability over the entire range of operation. The stability aids recommended include both acoustic cavities and injector face baffle. Figure 4-31 summarizes the design specifications. The baffle is the primary stability device using 9 radial blades for the stability of the 1T, 2T, 3T modes with frequency margin. The hub is for structural reason as well as providing partial damping on the 1R mode. The cavities, located on the injector-chamber corner, are tuned to the 1R mode and provide additional stability margin through their tuning capability.

- Baffle for all tangential modes and partially for 1R mode
  - 9 radial blades for 1T, 2T, 3T and additional margin
  - Hub at 0.3 radial location for 1R, compartments
- Cavity for 1R and additional stability margin

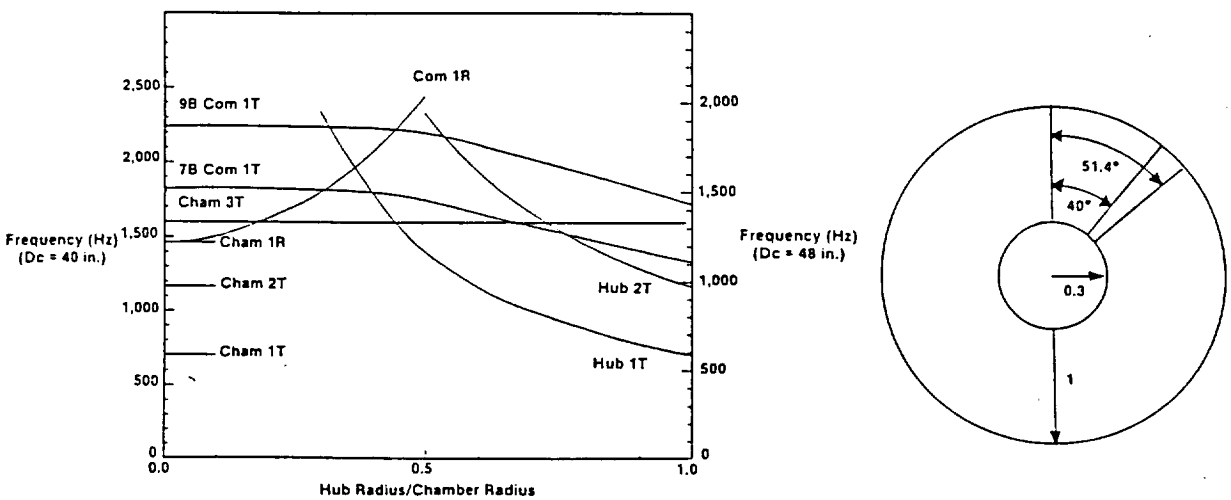

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Figure 4-31. High Frequency Stability can be Optimally Obtained ith Baffle and Cavities

4.4.6 Typical Point Design

Figure 4-27 showed that, using the OFO triplet injector, all the performance and stability goals can be met in the entire range of chamber pressure studied, namely, 300 to 700 psia. The ultimate chamber pressure selection involves considerations from the vehicle stand point of view. However, a typical point design can be selected to demonstrate the representative design features and injector operating condition. Figure 4-32 lists the key parameters for a 500 psia throat stagnation chamber pressure (539 psia injector end chamber pressure) design. An equivalent like doublet injector design is also included.

	OFO Triplet	Like Doublet
Thrust	750K	750K
Contraction Ratio	1.7	1.7
Chamber Diameter	47.7"	47.7"
Chamber Pressure	500 psia (539)	500 psia (539)
Mixture Ratio	2.5	2.5
Injector ΔP	135 psi	135 psi
Injector $\Delta P/Pc$	25%	25%
Fuel Orifice Diameter	0.170"	0.080"
Number of Elements	700	1500
Chamber Length	40"	40"
Baffle	9 blades + 1 hub	9 blades + 1 hub
Acoustic Cavity	1R and Higher Modes except 3T	1R and higher modes except 3T

89DV0109/1005

Figure 4-32. Parameters of Typical Injector Design

5.0 REFERENCES

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2. Priem, R. J. and M. F. Heidmann, "Propellant Vaporization as a Design Criterion for Rocket-Engine Combustion Chambers," NASA TR-R-67, 1960.

3. Gutierrez, B., R. Szabo and D. Roy, "Near-Field and Far-Field Acoustic Levels for Solid and Liquid Rocket Boosters - Revision A", Rocketdyne Division of Rockwell International Internal Report No. 9128-0004 (Dept. 545-128), December 1988.

APPENDIX 7

LRB FINAL REPORT FROM PRATT & WHITNEY

FR-20406
6 June 1988

Split Expander Engine Study for the General Dynamics Liquid Rocket Booster

Final Report

Prepared for
General Dynamics - Space Systems Division
P.O. Box 85990
San Diego, California 92138
In Support of NASA
Liquid Rocket Booster Study Contract
(NAS8-37137)

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UNITED TECHNOLOGIES
PRATT&WHITNEY

Table of Contents

INTRODUCTION

General Dynamics Space Systems Division is under contract to NASA/MSFC to study the feasibility of replacing the Space Transportation System solid rocket boosters with liquid rocket boosters (Contract NAS8-37137). The objective of the study is to define optimum pressure-fed and pump-fed vehicle concepts. The Pratt & Whitney Split Expander Cycle engine is a viable pump-fed liquid rocket engine for the study and General Dynamics has provided a subcontract to Pratt & Whitney to support the study of the Split Expander engine for the liquid rocket booster propulsion system.

SUMMARY

This report documents the results of a study conducted by Pratt & Whitney to supply preliminary design information on various aspects of the Split Expander engine as requested by and in support of the General Dynamics LRB study. Authorization to proceed on the subcontract was received from GDSSD on 5 April 1988.

The initial engine study considered a LO2/LH2 and a LO2/CH4 Split Expander engine and an early downselect to a LO2/CH4 engine was made on 19 April 1988. Engine parametric performance and size information along with vehicle/engine interface information, fuel comparisons, systems safety, environmental factors, reliability, STS compatibility, performance, cost and growth potential were considered when making the downselect. The LO2/CH4 engine was selected over the LO2/LH2 engine based on minimal impact to the Space Transportation System vehicle and launch facilities and the ability to provide improved safety and reliability. Preliminary design efforts for the remaining portion of the Pratt & Whitney contract was totally directed to the LO2/CH4 Split Expander engine following the downselect.

A revised LO_2/CH_4 engine size was supplied to Pratt & Whitney on 11 May 1988 following a parametric study of the engine/vehicle system conducted by General Dynamics. The engine selected on which additional subsystem studies were based is a 756.3K vacuum thrust LO_2/CH_4 Split Expander engine (624.3K sea level thrust) with a mixture ratio of 3.5, a throttling range of 65% to 100% of maximum thrust requirements and a maximum nozzle exit diameter of 106.9 inches. Maximum engine diameter is limited by stage size restrictions.

The body of this report contains a brief description of the parametric analysis conducted to select the engine size followed by detailed discussion sections on engine performance, preliminary design, vehicle to engine interface, engine operation, reliability, cost and engine development definition.

A. Parametric Data

Initial engine sizing information was provided to General Dynamics - Space Systems Division (GDSSD) at a meeting held at GDSSD on 19 April 1988. The initial sizing request was completed by GDSSD using parametric equations previously provided in support of the Advanced Launch System. The initial engines selected for study had the following characteristics using the thrust requirements supplied by GDSSD.

LO2/CH4 Split Expander Engine

Parameter	Baseline Engine	Maximum Pc Engine
Design Thrust (SL/Vac) - K lbs	557/655	592/655
Design Impulse (SL/Vac) - K lbs	281.8/331.0	300.0/332.2
Chamber Pressure (Pc) - psia	700	1075
Mixture Ratio	3.5	3.5
Area Ratio	12.7	12.7
Exit Diameter - in	92	74
Overall Length - in	127	109

The baseline engine utilized a thrust chamber/nozzle assembly produced with 347 stainless steel tubes similar to the RL10. The maximum Pc engine utilized Haynes 230 tubes and shrouded impellers in the fuel pump. The Haynes 230 tubes are nickel based, and they provide greater strength at temperature and they also have a higher conductivity. The combination of improved properties enables Haynes 230 tubes to be designed for lower pressure drop and higher temperature thereby increasing turbine power.

The initial engine sizing was done by using the same area ratio for the baseline and maximum Pc engine to establish relative trends. Increased performance would have resulted by allowing a higher area ratio with increased Pc.

LO_2/LH_2 Split Expander Engine

Parameter	Baseline Engine	Maximum Pc Engine
Design Thrust (SL/Vac) - K lbs	486/580	518/580
Design Impulse (SL/Vac) - K lbs	340/405	362/406
Chamber Pressure (Pc) - psia	534	800
Mixture Ratio	6.0	6.0
Area Ratio	10.2	10.2
Exit Diameter - in	90	74
Overall Length - in	147	131

The meeting at GDSSD resulted in elimination of the LO_2/LH_2 engine as a suitable candidate based on the overall program downselect criteria established by GDSSD relative to safety, reliability, STS compatibility, engine size restrictions,overall vehicle performance and cost.

The parametric equations shown in Table A-1 were provided for the LO_2/CH_4 engine for additional sizing studies to be done by GDSSD following the 19 April meeting at GDSSD. The chamber pressure range in the equations extend to 1200 psia by the use of advanced materials and heat transfer concepts which would require technology demonstration. The current state-of-the-art technology for LO_2/CH_4 was revised to a Pc of 800 psia from 1075 psia with shrouded pump impellers and Haynes 230 material used in the main chamber.

Vacuum specific impulse, engine weight, nozzle exit diameter and engine length can be determined by use of the equations. Area ratio, oxidizer to fuel ratio and chamber pressure effects on engine performance, weight and size can be determined by use of the equations.

In addition, a plot of mixture ratio versus vacuum impulse was provided to GDSSD as shown in Figure A-1. Although the optimum mixture ratio appears to be approximately 3.25 for the area ratio range of interest for this engine, Pratt & Whitney experience indicates that generally a mixture ratio closer to 3.5 is desirable for overall vehicle optimization.

The parametric equations were used by GDSSD to study and finally select a 756.3K lb vacuum thrust engine with a sea level thrust of 624.3K lbs with a chamber pressure of 800 psi. This information was provided to Pratt & Whitney for final detailed definition of the engine cycle. The results of the cycle balance and final performance and size characteristics of the selected engine are discussed in Section D of this report.

TABLE A-1

SPLIT EXPANDER ENGINE PARAMETRIC

EQUATIONS

Propellants LOX/CH4

Vacuum Specific Impulse (Sec)

$$= 938.055 - 640.121\,(1/OF) - 225.796 * \sqrt{OF} + 3.124 * \sqrt{AR} - 534.831\,(OF/PC) - 173.802\,(1/AR) + 0.08315\,(OF * AR)$$

$$\text{Weight (lb)} = 4925 * \left(\frac{FVAC}{600{,}000}\right)^{.95} - \frac{26.83 * FVAC * (25 - AR)}{600{,}000}$$

$$\text{Diameter (in)} = 121 * \left(\frac{FVAC}{600{,}000}\right)^{.5} * \left(\frac{AR}{25}\right)^{.5} * \left(\frac{700}{PC}\right)^{.5}$$

$$\text{Length (in)} = 40.2 + 136.8 * \left(\frac{FVAC}{600{,}000}\right)^{.5} * \left(\frac{AR}{25}\right)^{.738} * \left(\frac{700}{Pc}\right)^{.5}$$

Parameter	Range
FVAC = EPL Vacuum Thrust	350 to 800K lbs
AR = Area Ratio	5.0 to 35.0
OF = O/F Mixture Ratio	3.0 to 4.5
PC = Chamber Pressure	600 to 1200 psia

4/28/88

PRATT & WHITNEY
SPLIT EXPANDER - LOX/CH4
CHAMBER PRESSURE = 800 PSI

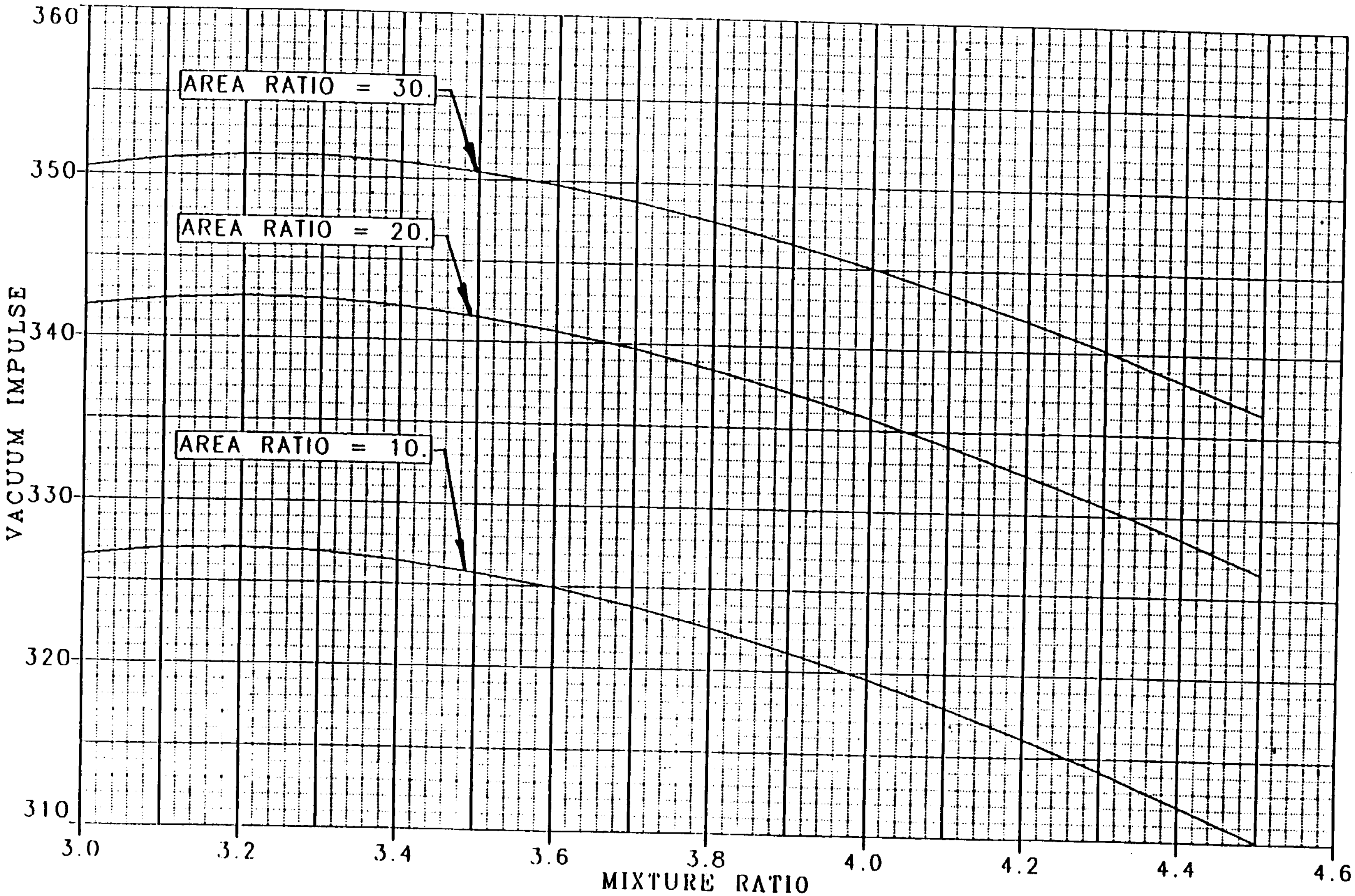

FIGURE A-1

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B. LO2/CH4 and LO2/LH2 Comparison

Early in the split expander study for the LRB a meeting was held at General Dynamics with Pratt & Whitney representatives on 19 April 1988 for the purpose of selecting between LO2/LH2 and LO2/CH4 as the propulsion system propellants. The selection process was done by comparing qualitative factors of each propellant combination on the engine factors, as shown in Table B-1, followed by an overall qualitative assessment of several factors relative to the entire vehicle system. The final selection criteria factors are shown in Table B-2. The criteria are listed in order of importance as determined by General Dynamics.

Evaluation and discussion of each criteria resulted in a mutual selection of LO2/CH4 for the General Dynamics LRB.

TABLE B-1

HYDROGEN VS. METHANE FUEL COMPARISION

Influence on Engine Characteristics

Engine Factor	Methane	Hydrogen
o Ignition	Hard to Lite	Easy to Lite
o Combustion Efficiency	Good	Very Good
o Combustion Stability	Stability Limits Lower-Needs Definition	Stable
o Cooling Capability	Good Coolant	Best Coolant
o Corrosion	None with Planned Mtls.	None
o Hydrogen Embrittlement	None	Engine Requires Protection
o Bulk Density	Higher Bulk Density	Lower Bulk Density
o Soot Formation	Possible	None
o Coking	Possible	None
o Handling/Storage	Cryogenic (-259°F)	Cryogenic (-423°F)

C. Subsystem Selection Information

The discussion of subsystem selection aspects of the engine has been incorporated into Section E, Preliminary Design Analysis, of this report in order to avoid redundant report sections.

D. Selected Engine Configuration Characteristics

The final engine selected by General Dynamics using Pratt & Whitney supplied parametric information has the following characteristics:

Main Propulsion Engine for General Dynamics LRB

$LO2/CH4$ Split Expander Engine

Rated Operating Thrust - k lbs	:	756.3 vacuum
	:	624.3 sea level
Delivered Specific Impulse - sec	:	337.5 vacuum
	:	277.9 sea level
Mixture Ratio (O/F)	:	3.5
Area Ratio	:	16.46
Total Fuel Flow Rate (lbm/sec)	:	499.6
Total Oxidizer Flow Rate (lbm/sec)	:	1754
Total Propellant Flow Rate (lbm/sec) (224.09 lbm/sec burned, 12.7 lbm/sec for tank pressurization)	:	2253.6
Chamber Pressure - psia	:	758.2
Nozzle Exit Diameter - in	:	106.9
Engine Length	:	165.4
Engine Weight	:	5640
Throttle Range	:	65% - 100%

The engine cycle balance sheets are shown in Table D-1 for the 756.3K lb thrust design engine. Table D-2 shows the cycle balance sheet for a 65% throttle setting at sea level. These tables contain pressure temperature flow and density values for the engine propellants at various locations in the engine.

Referring to Table D-1, the LO2 pump consists of a single stage centrifugal pump driven by a single stage turbine. The pump operates at a speed of 7386 rpm with a total flow rate of 1754 lbm/sec and a pump discharge pressure of 1140 psi.

The fuel pump consists of a three stage centrifugal pump driven by a single turbine stage. The pump operates at a speed of 11071 rpm with a total flow rate of 499.6 lbm/sec. Approximately 56% of the discharge flow from the first pump stage is routed to the mixer valve thus bypassing the second and third stage pump. The balance of the methane flow (218.9 lbm/sec) is routed through the second and third stage pumps, discharged at 4480 psia at 218.9 lbm/sec, passed through the chamber and nozzle, through the turbines, mixer and into the injector. Additional details of the pumps such as tip speeds, pressure ratio, efficiency, etc. can be found in Table D-1. Figure D-1 shows the basic cycle characteristics on a flow schematic.

The turbopumps are mounted back to back with counterrotating shafts and they are mounted in a common housing. The pumps use relatively low cost materials and low cost manufacturing techniques to provide a low cost, reliable engine. The low temperature of the turbines allows the use of forged aluminum disk and blades. An integral forged bladed disk turbine wheel, known as a blisk, is under study. Fuel pump impellers are machined from aluminum although studies are planned to produce cast aluminum impellers to further reduce cost. Pump housings are made from cast aluminum. The oxidizer pump impeller is made of forged 347 stainless steel and the integral turbine is made of forged aluminum. The lox pump turbine housing is made of cast aluminum and the main pump housing is made from cast 347 stainless steel.

The injector consists of multiple tangential entry oxidizer elements with a concentric annulus of CH_4. The injector faceplate is a porous material that allows transpiration cooling of the face. This design provides a hollow cone spray of liquid oxygen and is then exposed to high velocity fuel for better atomization. The ignition system consists of an augmented spark igniter (torch) since it can be easily maintained and can be checked out prior to flight.

The thrust chamber and nozzle assembly are fabricated from Haynes 230 tubes brazed together with silver. An INCO 718 structural jacket is then brazed on over the tubes. The engines utilize a dual circuit cooling scheme. Both cooling circuits are single pass with the thrust chamber employing counterflow and the nozzle employing parallel flow. The third stage CH_4 pump discharge flow enters the thrust chamber at its base which is downstream of the throat. After cooling the chamber, the exiting coolant is routed to the top skirt manifold and passes to the end of the nozzle, collected in a manifold and directed to the pump turbine inlet.

A study was performed to find the optimum main propellant valve types for the Split Expander Cycle engine. The study included phases of defining the valve requirements and evaluation criteria, identifying historical rocket data, vendor valve data, comparing valve data to the requirements and criteria and, finally, determining the optimum valves for each location. Results of this study represent the current baseline valve types and will be used in current and future rocket engine requirements.

The results of this study were applied to the LRB main propellant valves to find the optimum valve types for the turbine bypass valve (TBV), jacket bypass valve (JBV), main oxidizer valve (MOV), fuel shutoff valve (FSOV), fuel cooldown valve (FCDV), and the oxidizer cooldown valve (OCDV). Table D-3 lists valve requirements and selected valve types.

Figure D-2 shows a sketch of the mixer concept which will be used for mixing hot and cold CH4 in the split expander engine. This mixer concept provides efficient turbulent mixing between the hot and cold fuel flows with a simple compact configuration and an acceptable pressure drop. The concept was previously used by P&W on the XLR-129 test stand to mix hot and cold hydrogen.

TABLE D-1

PRATT & WHITNEY
GENERAL DYNAMICS LIQUID ROCKET BOOSTER
LOX/CH4 SPLIT EXPANDER ENGINE

ENGINE PERFORMANCE PARAMETERS
****

CHAMBER PRESSURE	758.2
VAC ENGINE THRUST	756300.
S.L. ENGINE THRUST	624300.
TOTAL ENGINE FLOW RATE	2240.9
DEL. VAC. ISP	337.5
THROAT AREA	545.6
NOZZLE AREA RATIO	16.46
NOZZLE EXIT DIAMETER	106.9
ENGINE MIXTURE RATIO	3.50
CHAMBER COOLANT DP	890.
CHAMBER COOLANT DT	644.
CHAMBER Q	114612.

ENGINE STATION CONDITIONS
****

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	40.0	201.0	499.6	123.0	26.39
1ST STAGE EXIT	1134.3	206.2	499.6	132.2	26.56
JBV INLET	1085.7	206.5	280.7	132.2	26.53
JBV EXIT	1002.5	206.9	280.7	132.2	26.48
2ND STAGE EXIT	2806.6	216.8	218.9	148.2	26.69
PUMP EXIT	4479.7	227.1	218.9	164.0	26.82
COOLANT INLET	4307.0	228.1	218.9	164.0	26.72
COOLANT EXIT	3417.2	870.8	218.9	687.5	5.66
TBV INLET	3285.5	869.8	11.1	687.5	5.47
TBV EXIT	1002.5	845.7	11.1	687.5	1.78
CH4 TRB INLET	3285.5	869.8	207.8	687.5	5.47
CH4 TRB EXIT	1535.8	759.8	207.8	632.0	3.01
LOX TRB INLET	1535.8	759.8	207.8	632.0	3.01
LOX TRB EXIT	1012.6	705.2	207.8	604.5	2.15
CH4 TANK PRESS	40.0	685.1	1.7	596.9	0.09
GOX HEAT EXCH	1002.5	705.2	206.2	604.5	2.15
MIXER	1002.5	376.4	498.0	337.0	8.86
FCV INLET	954.5	372.8	498.0	337.0	8.53
FCV EXIT	867.6	366.0	498.0	337.0	7.94
CHAMBER INJ	834.1	363.0	498.0	337.0	7.71
CHAMBER	758.2				

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	60.0	163.0	1754.0	61.7	71.15
PUMP EXIT	1140.0	166.8	1754.0	64.9	71.52
O2 TANK PRESS	60.0	400.0	11.1	204.6	0.45
OCV INLET	1081.4	167.0	1742.9	64.9	71.43
OCV EXIT	877.7	167.9	1742.9	64.9	71.11
CHAMBER INJ	836.4	168.0	1742.9	64.9	71.04
CHAMBER	758.2				

TABLE D-1 (Cont.)

PRATT & WHITNEY
LOX/CH4 SPLIT EXPANDER ENGINE

TURBOMACHINERY PERFORMANCE DATA

CH4 TURBINE

EFFICIENCY	0.743
HORSEPOWER	16322.
SPEED (RPM)	11071.
MEAN DIAMETER	12.20
EFF AREA	4.80
VEL RATIO	0.30
MAX TIP SPEED	590.
STAGES	1.
PRESSURE RATIO	2.14

CH4 PUMP

	STAGE ONE	STAGE TWO	STAGE THREE
EFFICIENCY	0.836	0.722	0.722
HORSEPOWER	6445.	4963.	4914.
SPEED (RPM)	11071.	11071.	11071.
S SPEED	1505.	727.	731.
HEAD	5932	8995	8908.
DIAMETER	13.79	15.62	15.59
TIP SPEED	667.	755.	754.
VOL. FLOW	8443.	3683.	3664.
HEAD COEF	0.430	0.510	0.510
FLOW COEF	0.100	0.100	0.100

O2 TURBINE

EFFICIENCY	0.739
HORSEPOWER	8096.
SPEED (RPM)	7386.
MEAN DIAMETER	12.20
EFF AREA	9.93
VELOCITY RATIO	0.29
MAX TIP SPEED	394.
STAGES	1.
PRESSURE RATIO	1.52

O2 PUMP

EFFICIENCY	0.856
HORSEPOWER	8096.
SPEED (RPM)	7386.
S SPEED	2431.
HEAD	2178.
DIAMETER	13.48
TIP SPEED	435.
VOL. FLOW	11007.
HEAD COEF	0.371
FLOW COEF	0.162

VALVE DATA

	FCV	OCV	JBV	TBV
DELP	86.9	203.7	83.2	2283.1
AREA	41.44	21.62	8.94	0.26
FLOW	497.97	1742.88	280.68	11.13
BYPASS %			56.18	5.08

INJECTOR DATA

	CH4	O2
DELP	75.9	78.2
AREA	46.69	34.98
FLOW	497.97	1742.88

TABLE D-2

PRATT & WHITNEY
GENERAL DYNAMICS LIQUID ROCKET BOOSTER
LOX/CH4 SPLIT EXPANDER ENGINE
35% Down Thrust at S.L.

ENGINE PERFORMANCE PARAMETERS
****

CHAMBER PRESSURE	538.5
VAC ENGINE THRUST	537770.
S.L. ENGINE THRUST	405789.
TOTAL ENGINE FLOW RATE	1597.8
DEL. VAC. ISP	336.6
THROAT AREA	545.6
NOZZLE AREA RATIO	16.46
NOZZLE EXIT DIAMETER	106.9
ENGINE MIXTURE RATIO	3.50
CHAMBER COOLANT DP	484.
CHAMBER COOLANT DT	524.
CHAMBER Q	87243.

ENGINE STATION CONDITIONS
****

* FUEL SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	40.0	201.0	356.5	123.0	26.39
1ST STAGE EXIT	761.9	204.6	356.5	129.2	26.50
JBV INLET	745.2	204.7	164.3	129.2	26.49
JBV EXIT	716.6	204.9	164.3	129.2	26.47
2ND STAGE EXIT	1745.7	211.0	192.1	138.8	26.56
PUMP EXIT	2725.3	217.3	192.1	148.3	26.63
COOLANT INLET	2591.4	218.1	192.1	148.3	26.55
COOLANT EXIT	2107.4	741.5	192.1	602.3	4.35
TBV INLET	1975.5	739.3	54.6	602.3	4.10
TBV EXIT	716.6	712.8	54.6	602.3	1.53
CH4 TRB INLET	1975.5	739.3	137.6	602.3	4.10
CH4 TRB EXIT	987.2	644.0	137.6	559.7	2.36
LOX TRB INLET	987.2	644.0	137.6	559.7	2.36
LOX TRB EXIT	723.9	604.9	137.6	542.3	1.84
CH4 TANK PRESS	40.0	574.7	1.4	534.1	0.10
GOX HEAT EXCH	716.6	604.9	136.2	542.3	1.84
MIXER	716.6	359.9	355.1	357.2	5.39
FCV INLET	676.6	355.3	355.1	357.2	5.12
FCV EXIT	618.3	348.0	355.1	357.2	4.70
CHAMBER INJ	589.5	343.8	355.1	357.2	4.48
CHAMBER	538.5				

* OXYGEN SYSTEM CONDITIONS *

STATION	PRESS	TEMP	FLOW	ENTHALPY	DENSITY
PUMP INLET	60.0	163.0	1250.7	61.7	71.15
PUMP EXIT	687.2	165.3	1250.7	63.6	71.36
O2 TANK PRESS	60.0	400.0	7.9	204.6	0.45
OCV INLET	657.3	165.4	1242.8	63.6	71.31
OCV EXIT	599.2	165.6	1242.8	63.6	71.22
CHAMBER INJ	578.2	165.7	1242.8	63.6	71.18
CHAMBER	538.5				

TABLE D-2 (Cont.)

PRATT & WHITNEY
LOX/CH4 SPLIT EXPANDER ENGINE

TURBOMACHINERY PERFORMANCE DATA

CH4 TURBINE

EFFICIENCY	0.737
HORSEPOWER	8290.
SPEED (RPM)	8903.
MEAN DIAMETER	12.20
EFF AREA	4.80
VEL RATIO	0.28
MAX TIP SPEED	474.
STAGES	1.
PRESSURE RATIO	2.00

CH4 PUMP

	STAGE ONE	STAGE TWO	STAGE THREE
EFFICIENCY	0.818	0.715	0.714
HORSEPOWER	3110.	2602.	2578.
SPEED (RPM)	8903.	8903.	8903.
S SPEED	1396.	814.	819.
HEAD	3923	5323	5272.
DIAMETER	13.79	15.62	15.59
TIP SPEED	536.	607.	606.
VOL. FLOW	6038.	3247.	3238.
HEAD COEF	0.440	0.466	0.464
FLOW COEF	0.089	0.110	0.110

O2 TURBINE

EFFICIENCY	0.736
HORSEPOWER	3394.
SPEED (RPM)	5580.
MEAN DIAMETER	12.20
EFF AREA	9.93
VELOCITY RATIO	0.27
MAX TIP SPEED	297.
STAGES	1.
PRESSURE RATIO	1.36

O2 PUMP

EFFICIENCY	0.848
HORSEPOWER	3394.
SPEED (RPM)	5580.
S SPEED	2331.
HEAD	1267.
DIAMETER	13.48
TIP SPEED	328.
VOL. FLOW	7867.
HEAD COEF	0.378
FLOW COEF	0.154

VALVE DATA

	FCV	OCV	JBV	TBV
DELP	58.3	58.1	28.6	1258.8
AREA	41.44	28.89	8.94	1.91
FLOW	355.07	1242.75	164.35	54.57
BYPASS %			46.10	28.40

INJECTOR DATA

	CH4	O2
DELP	51.0	39.7
AREA	46.69	34.98
FLOW	355.07	1242.75

SPLIT EXPANDER CYCLE ENGINE

Conditions at 756.3K Vacuum Thrust (624.3K S.L.)

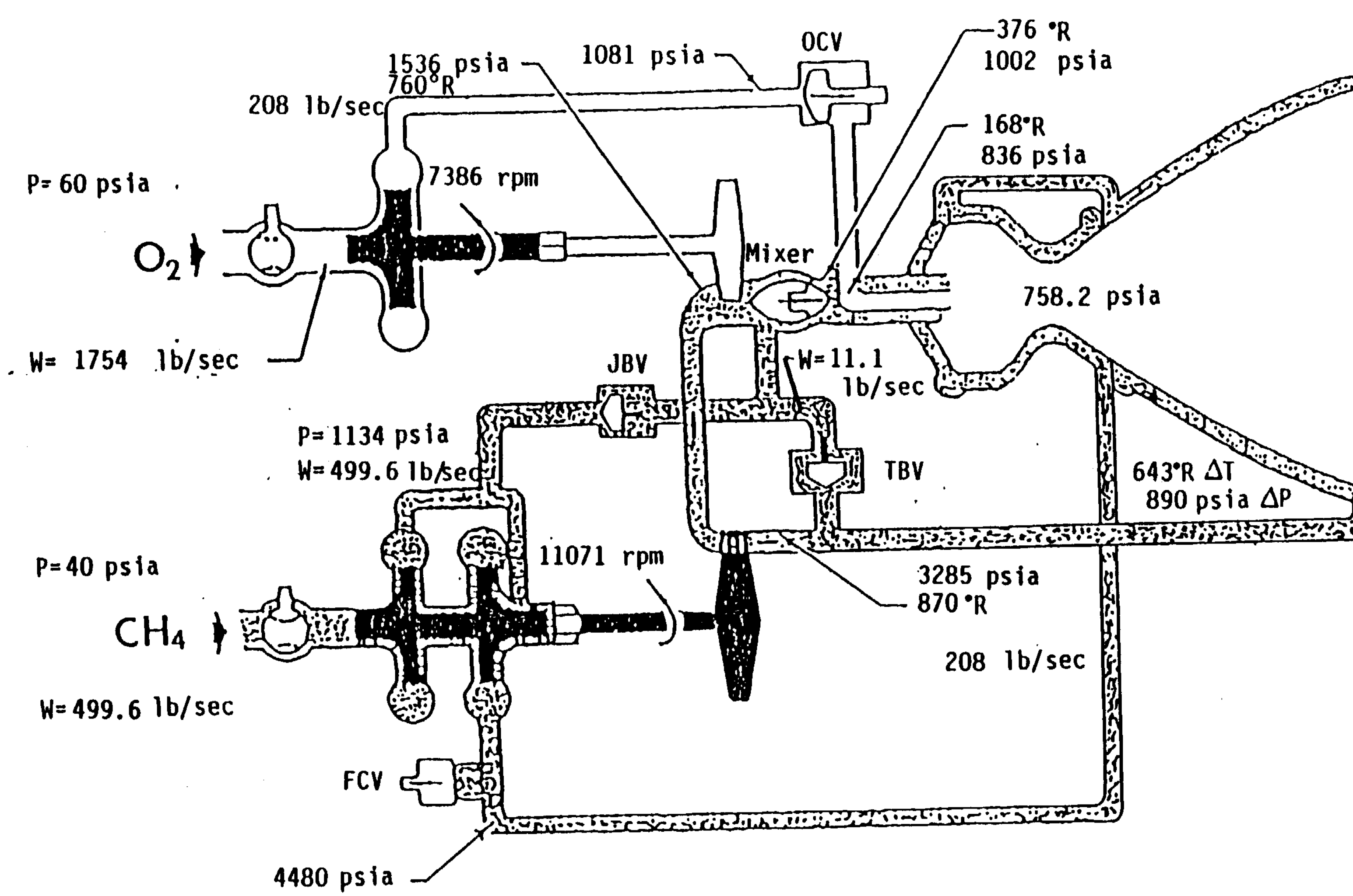


FIGURE D-1

LRB PROPULSION SYSTEM VALVES

Split Flow Expander Cycle Requirements

Valve Name	Sealing Capability	Safe Position	Turndown Ratio	Valve Types	Actuation Method
TBV	No	Open	10-1	Sleeve	EM Actuated
JBV	Yes	Closed	5-1	Butterfly	EM Actuated
MOV	No	Closed	5-1	Butterfly	EM Actuated
FSOV	Yes	Closed	Open/Closed	Gate	Start Sol.
FCDV	Yes	Open	Open/Closed	Poppett	Start Sol.
OCDV	Yes	Open	Open/Closed	Poppett	Start Sol.

TABLE D-3

CH_4 MIXER

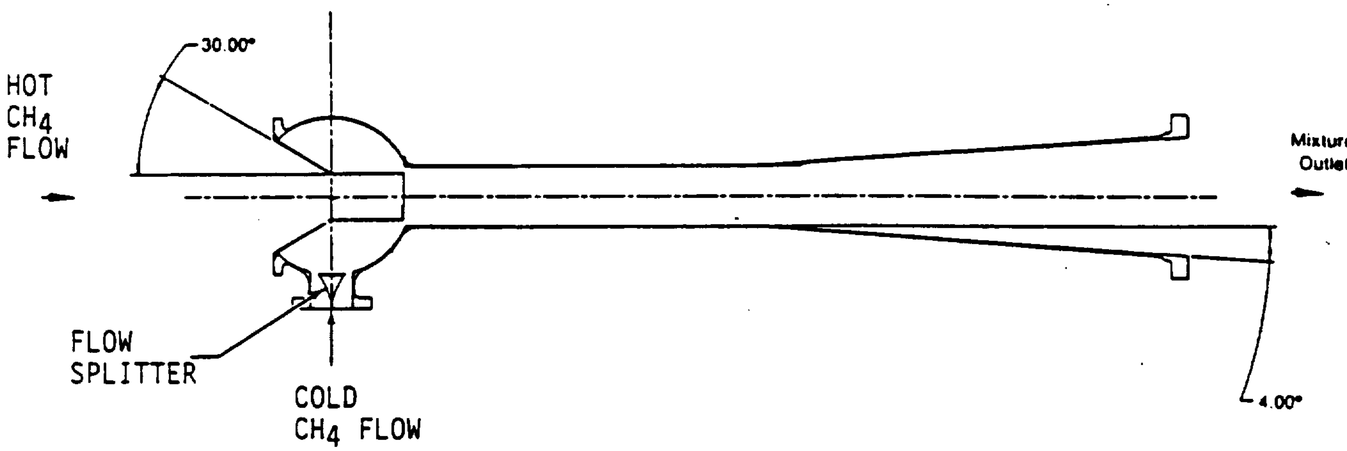


FIGURE D-2

E. Preliminary Design Analysis

The baseline fixed thrust engine flow schematic is shown in Figure E-1. This engine configuration employs simple open loop pneumatically operated engine valves. The main oxidizer valve (MOV), jacket bypass valve (JBV), and turbine bypass valve (TBV) are spring loaded, adjustable stop, poppet-type valves actuated by internal engine pressures. The fuel cooldown valve (FCV), oxidizer cooldown valve (OCV), and fuel shutoff valve (FSOV) are helium actuated, on-off valves. The adjustable stops of the control valves will be trimmed and locked during acceptance testing with the JBV set to provide the proper fuel jacket bypass flow split, the MOV set to provide the proper mixture ratio, and the TBV setting thrust. The OCV provides a path for recirculation of oxidizer flow when in the open position and when closed provides a path for the starting oxidizer flow to the igniter and chamber.

The POGO system is located just upstream of the oxidizer turbopump inlet. This system will be similar to the one employed on the SSME using a gas filled plenum to decouple the engine and feedline.

The horizontally oriented turbopumps provide a significant length of inlet duct on the engine side of the vehicle/engine interface. This provides a great deal of flexibility in the vehicle propellant duct, straight duct requirements at inlet, etc. without impact on the engine.

LO2/CH4 Split Expander Cycle

Propellant Flow Schematic
(Fixed Thrust Configuration)

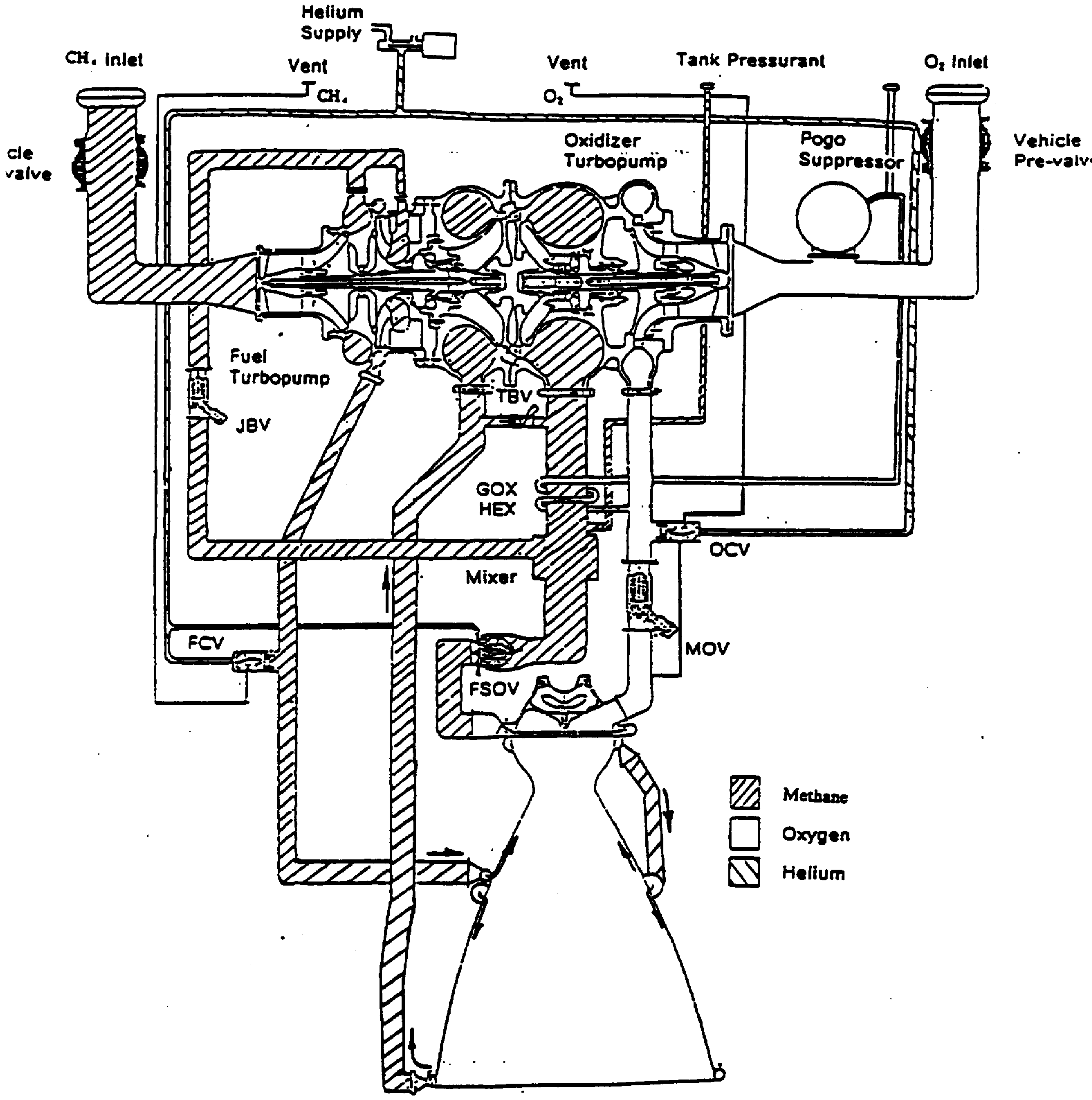


FIGURE E-1

The engine configuration shown depends on a forced convective cooldown of limited duration (less than one hour) and the following operation description is addressed to this configuration.

COOLDOWN

The engine valve requirements are somewhat dependent on the mode of cooldown selected for the engine. The current configuration assumes a limited cooldown duration (1 hour or less) with forced flowrate. Cooldown propellants can be recirculated through the fuel cooldown valve (FCV) and the oxidizer cooldown valve (OCV) to the vehicle tanks, recirculated to external tanks, or dumped, whichever is optimum for the vehicle system. If a cold soak type, passive cooldown procedure is desirable, an additional valve would be required downstream of the fuel pump to prevent cooldown of the thrust chamber.

The fuel shutoff valve (FSOV), main oxidizer valve (MOV), jacket bypass valve (JBV), and turbine bypass valve (TBV) are normally closed and the fuel cooldown valve (FCV) and oxidizer cooldown valve (OCV) are normally open. Cooldown is accomplished by opening the prevalves at the engine inlet and circulating propellants through the turbopumps and FCV and OCV until the pumps are properly conditioned which will probably be determined by cooldown time and checked by housing temperature measurements.

Figures E-2 and E-3 present estimated fuel and oxidizer cooldown time requirements and consumption as a function of flowrates for the baseline cooldown procedure.

START

Engine start is accomplished by opening the start solenoid valve which ports helium to the FCV, OCV and FSOV. This closes the FCV and the OCV and opens the FSOV. The cooldown flow is shutoff, starting oxidizer flow is directed through the OCV, around the MOV to the combustion chamber and igniter. Opening the FSOV supplies fuel flow to the combustion chamber and igniter and allows the gaseous fuel in the chamber coolant passages to flow through the turbines, initiating turbopump rotation. OCV actuation is timed to occur faster than FSOV actuation, setting up an initial.oxidizer rich atmosphere in the igniter and thrust chamber. When the fuel from the FSOV actuation reaches the igniter and combustion chamber, mixture ratio drops rapidly with ignition occurring first in the igniter and then in the main combustion chamber as ignitable conditions, which are a function of O/F ratio and pressure, are achieved.

The fire in the combustion chamber continues to vaporize the fuel as turbopump speed, fuel flowrate and system pressures increase. When turbopump speeds get to approximately 50% of steady state operation levels, the JBV is opened allowing bypass around the jacket and

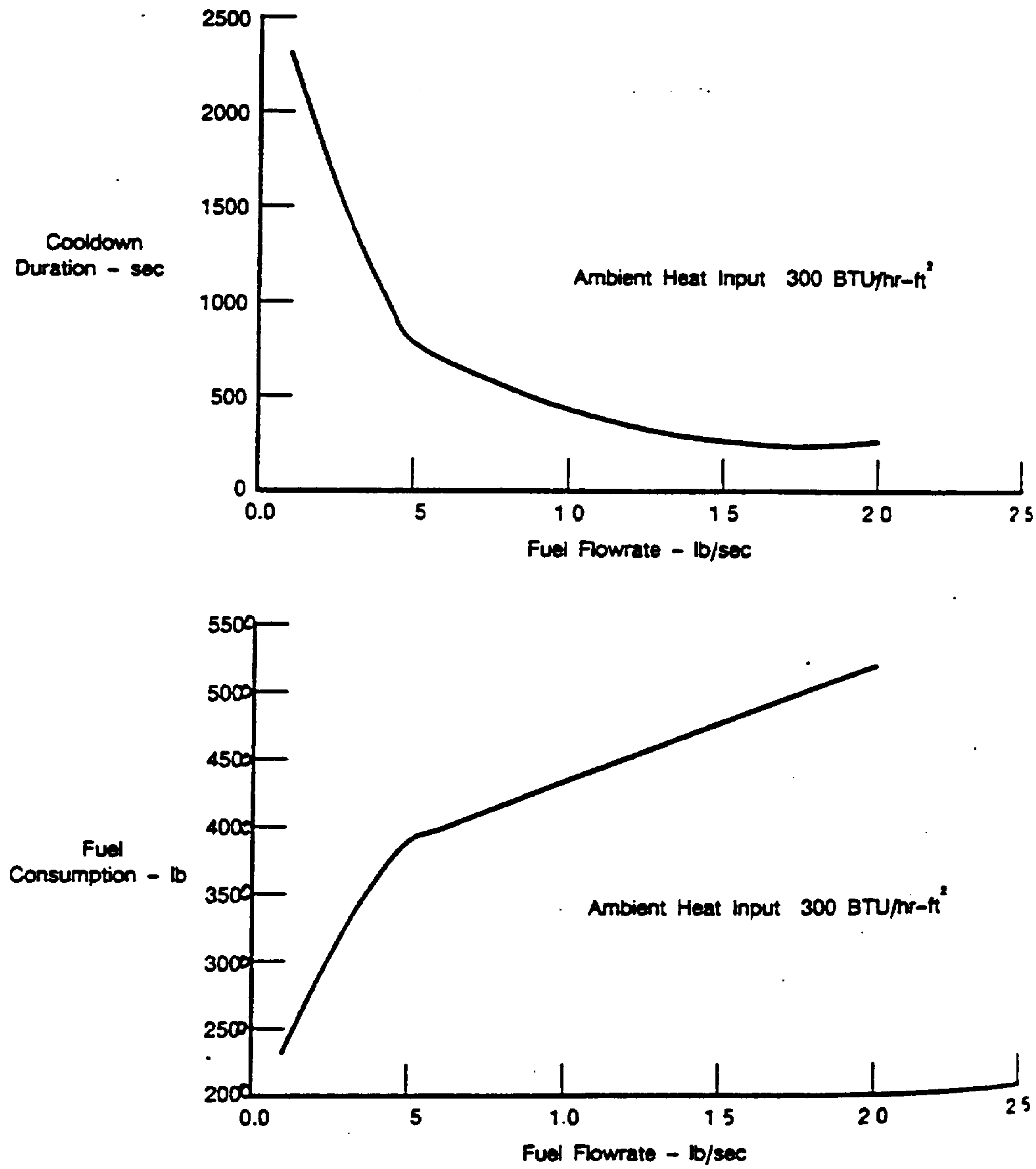


Figure E-2 Fuel System Cooldown Characteristics

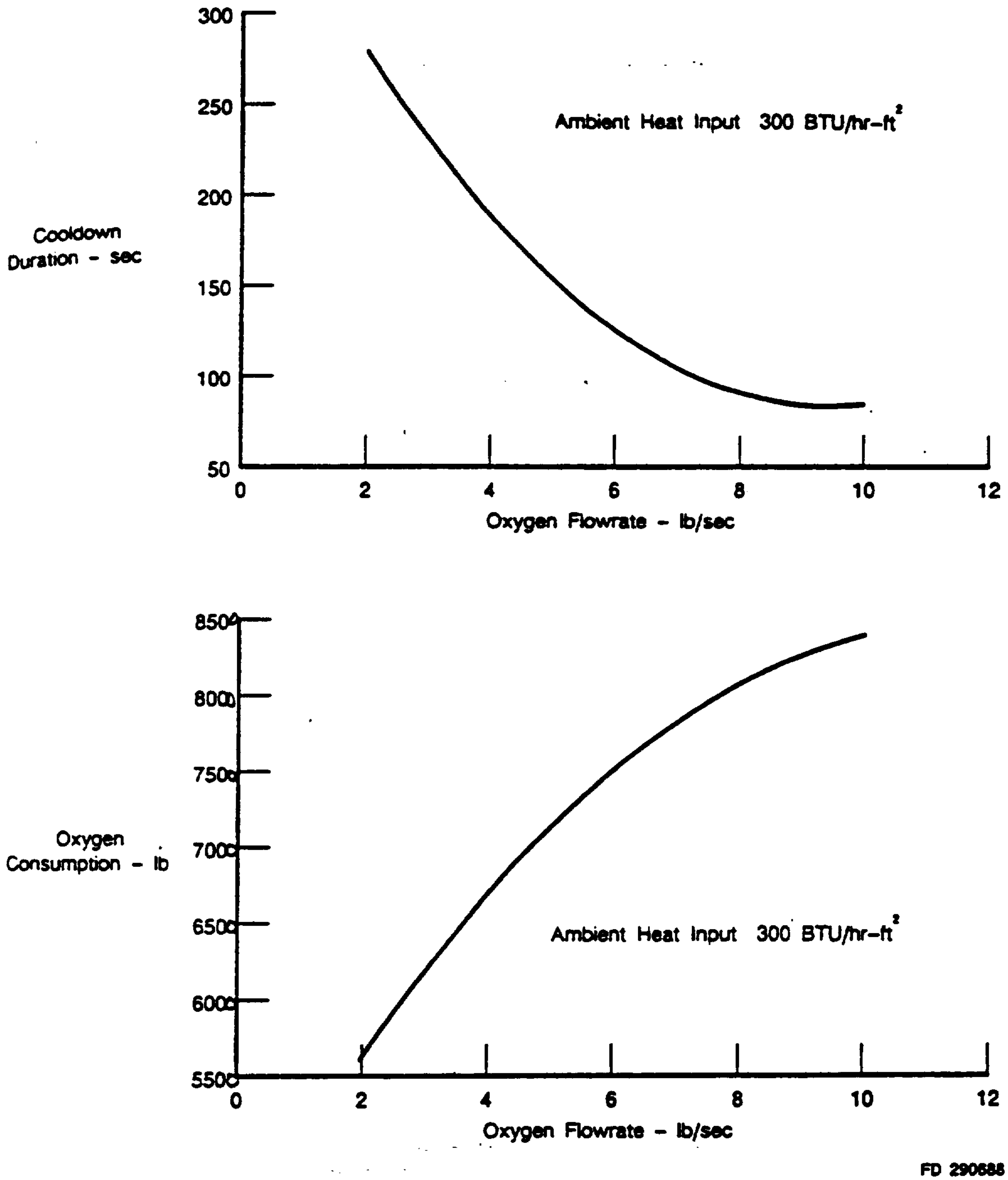


Figure E-3 Oxidizer System Cooldown Characteristics turbines and providing some turbine back pressure to slow the acceleration. At 60 to 80% of steady state operational turbopump speeds the MOV is opened, providing high oxidizer flowrates to the combustion chamber and resulting in a rapid increase in chamber pressure (thrust) and mixture ratio. When the engine reaches approximately 85% of normal operating level, the TBV opens bypassing fuel flow around the turbines and preventing a thrust overshoot.

Initial ignition of the main combustion chamber is to a chamber pressure of approximately 25 psia which, because of choking and recovery characteristics of converging-diverging nozzles, makes the engine acceleration independent of ambient pressures less than 16 psia. A two to three second acceleration from start signal to 95% thrust is expected with a ± 0.3 second variation. Propellant consumption during the start transient (start signal to 100% thrust) is estimated to be 400 lbs of CH_4 and 2200 lbs of LO_2.

STEADY STATE

Steady state jacket bypass flow, thrust level, and mixture ratio are set by adjustment of the JBV, TBV, and MOV valves during engine acceptance testing. The engine as configured has a fixed thrust and mixture ratio capability.

SHUTDOWN

Shutdown is accomplished by closing the start solenoid valve and venting the helium. This closes the FSOV resulting in a rapid thrust decay (less than 0.15 sec to 1% thrust) and opens the FCV and OCV to vent the high pressure propellants. Engine shutdown time can be lengthened if required. The removal of turbine power by closing the FSOV results in a rapid deceleration of the engine. The FCV will vent the high pressure fuel, preventing any system overpressure from the sudden flow stoppage. The engine is ready for another start if required as soon as the pumps windmill down. Propellant consumption during the shutdown transient is estimated to be 36 lbs of CH_4 and 126 lbs of LO_2.

ABORT

If the safety monitoring system indicates a problem, the engine can be shutdown in less than 0.15 seconds. A longer abort shutdown can be provided if required. If abnormal operation is experienced during the start sequence (slow speed buildup, etc.), ground monitoring should identify the problem and shutdown all engines.

THROTTLING ENGINE CONFIGURATION

The baseline engine as described above is a fixed thrust simple control system configuration. An optional engine configuration employing a complex closed loop control system to allow continuous throttling capability over its thrust range is also available at increased cost. For this configuration, valves are actuated using a sophisticated electromechanical system and are able to be set at any area as directed by the command signal. The valve command signals are generated by an on-engine computer system that uses measured engine parameters such as chamber pressure, turbine temperatures and flowrates to calculate thrust and mixture ratio and control to the desired vehicle levels. Redundant instrumentation is provided for all control parameters to assure the correct measurements are used to position the valves.

The engine operation, as described for the fixed thrust engine, is essentially unchanged except the valves are actuated by electromechanical actuators instead of pneumatic, and variable thrust and mixture ratio capability is available during steady state operation.

Engine thrust level is controlled by utilizing the TBV to maintain chamber pressure and therefore thrust. To throttle the engine down to a lower thrust level, the TBV will open up and reduce the amount of turbine flow and available horsepower. The pumps spin down and pressures decrease throughout the engine system until the desired chamber pressure (thrust level) is attained. The JBV remains at its 100% rated power level value throughout the throttling range and mixture ratio is maintained using the propellant utilization portion of the oxidizer control valve.

PRELIMINARY CONTROL COMPARISON

For the General Dynamics' Liquid Rocket Booster (LRB) program, three control system concepts were studied to compare production cost and mission reliability differences relative to system capability. The lowest cost and highest mission reliability system was a pneumatically actuated, single thrust set point system. Two variable thrust, closed loop thrust and mixture ratio systems were considered for throttling capability from -35% to +10% of the engine design point. Production cost and mission reliability were then estimated to determine the cost versus benefit of each system.

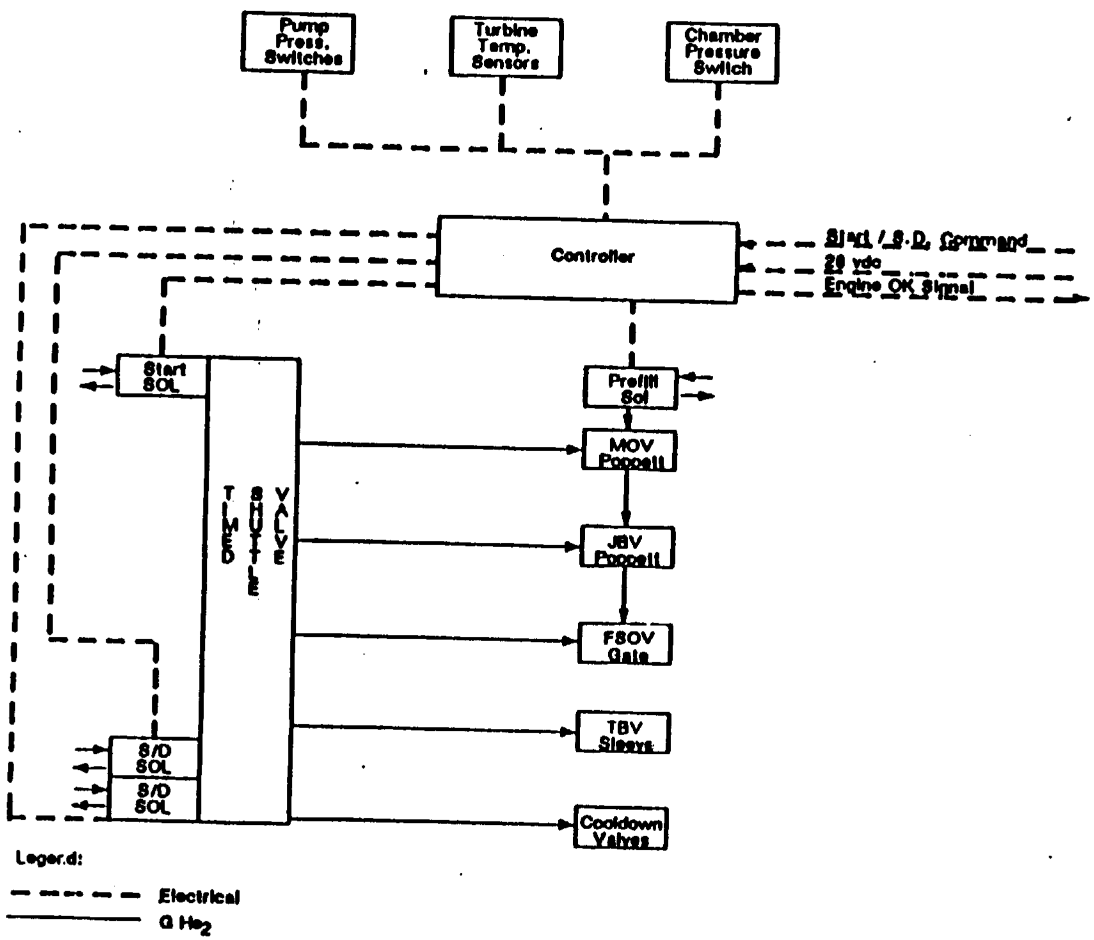
LRB Control System Study Overview

- Three Control System Configurations Reviewed
  - Single Point Thrust/Mixture Ratio
  - Variable FN, Closed Loop FN/MR, Single String Controller
  - Variable FN, Closed Loop FN/MR, Dual Channel Controller
- Mission Reliability Comparison Performed
- Production Cost Comparison Performed
- Expendable LO2/CH4 Split Flow Expander Cycle Engine Application

The single point system shown in Figure E-4 represents an RL10-derived concept in which all controlled parameters are "ON" or "OFF" type elements. An electronic controller provides engine/vehicle interface, engine safety monitoring, prestart "ON" or "OFF" control, start "ON"

LRB CONTROL SYSTEM STUDY

SINGLE POINT FN/MR CONCEPT



Provides:

- Limited Preflight Checkout (Controller & Solenoid Only)
- Start Sequencing
- Shutdown
- One Power Level
- On/Off Interfaces With Vehicle
- Switched Parameter Safety Monitors
- Fail Safe – Shutdown

Requires:

- High Pressure Helium Supply
- 28 VDC Power Supply

TOTAL SYSTEM COST

BASELINE

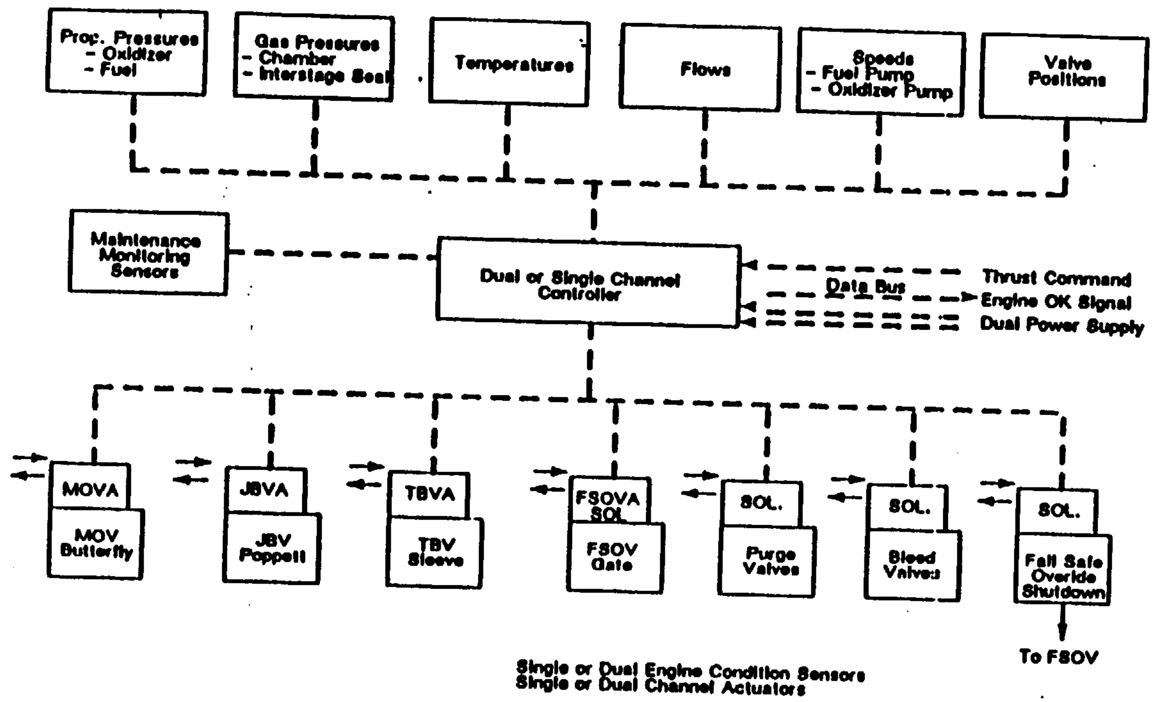
Figure E-4

or "OFF" control and emergency shutdown capability. This system has shutdown failsafe capability in the event that the control system is inoperable. Control valves are pneumatically actuated, solenoid actuated, or propellant pressure actuated and thus require no independent actuation system. The result is a simple, low cost, highly reliable system.

The variable thrust control system shown in Figure E-5 includes a controller with closed loop thrust and mixture ratio control and the necessary sensors and actuators to effect closed loop control. For both the dual channel (DC) and the single channel (SC) controller system concepts, dual sensors, dual actuator interface coils (electric motor stator coils for electromechanical actuation or electrohydraulic servovalve torque motor coils for hydraulic actuation), and dual power supplies are used. Also, in both cases, all prestart activities and engine "ON" activities are scheduled by the controller. The dual channel controller concept provides two active control channels which drive each actuator coil simultaneously. In the event that one channel becomes inoperative, the operational channel provides complete control capability. The dual channel system is considered fail operational/fail safe. The single channel controller system concept provides operational capability identical to that of the dual channel controller; however, only one controller channel drives both actuator coils. In the event that the single controller channel becomes inoperative, failsafe shutdown is effected.

LRB CONTROL SYSTEM STUDY

VARIABLE, CLOSED LOOP FN/MR CONCEPT



Legend:

- – – – – Electrical
- ——— Hydraulic Supply (Optional High Power Electrical Supply, If EMA 's Used)

NOTE: System Cost Differences Are Based On Lot Sizes of 100.

Provides:

- Preflight Checkout
- Start
- Shutdown
- Variable Thrust Levels
- Closed Loop Thrust Control
- Closed Loop MR Control
- Interface With Vehicle
- Safety Monitoring
- Fail Operational/Fail Safe With Dual Channel Controller
- Fail Safe With Single Channel Controller

Requires:

- Hydraulic Supply (Dual)
- Optional High Power Electrical Supply
- 28 VDC Power Supply
- 400 Hz. AC Power Supply

Dual Channel Controller System Cost

- Baseline + $.67M '87$

Single Channel Controller System Cost

- Baseline + .34M 87$

Figure E-5

Mission reliability for each of the systems is shown in Figure E-6. The single point system provides an estimated failure rate of .023 failures per 1000 engine starts or a control system mission reliability of .99999 with the system failures resulting in safe engine shutdown. All engine components other than the control system combine for a predicted failure rate of .164/1000 starts, or .99984 mission reliability, for comparison purposes. The DC variable system has a predicted failure rate of .030/1000 starts, while the SC variable system has a predicted rate of .131/1000 starts. Again, all failures result in safe engine shutdown.

Production costs were estimated for each of the three systems assuming a lot size of 100 with a total buy of 1000 engines. As shown in Table E-1, the DC variable system shows a cost of $670K over the baseline single point control system and the SC channel variable throttle system shows a cost of $345K above the baseline. In each case, the costs were compiled independent of any maintenance monitoring system which may be assumed to add $250K per system, if used. An SC channel, variable throttle system may be obtained at reduced cost, compared to the dual channel system; however, mission reliability is impacted to an extent, while the single point system provides low cost and high reliability with limited capability. The most desirable system may only be determined by weighing the impact on vehicle design, yielded by the variable system versus single point control, against the cost and reliability differences associated with each system. Table E-2 shows a comparison of the various types of control systems for a Split Expander engine along with the estimated accuracy.

LRB THROTTLING STUDY

System Mission Reliability Comparison

Single Point System

Emergency Shut Downs Per 1000 Starts										
0.023	=	Controller 0.010	+	Shuttle Valve 0.001	+	4 Main Valves 0.008	+	Sensors Negligible (Dual Sensors Used)	+	Auxiliary Valves 0.004

Variable FN, Single Channel Controller

0.131	=	Controller 0.104	+	Actuators 0.015	+	4 Main Valves 0.008	+	Sensors Negligible	+	Auxiliary Valves 0.004

Variable FN, Dual Channel Controller

0.030	=	Controller 0.003	+	Actuators 0.015	+	4 Main Valves 0.008	+	Sensors Negligible	+	Auxiliary Valves 0.004

Figure E-6

C-4

LRB CONTROL SYSTEM STUDY

Capability vs. Mission Reliability & Cost

System Concept	Capability	Mission Reliability Shutdowns Per 1000 Starts	Production Cost (1987 $)
Single Point	1 Set Point	0.023	Baseline
Variable Thrust			
Single Channel Controller	Variable	0.131	Baseline + $345K
Dual Channel Controller	Variable	0.030	Baseline + $670K

- o Reliability Prediction Indicates All Non-Control Engine Elements Contribute A Total Of 0.168 Shutdowns Per 1000 Starts Or A Mission Reliability Of 0.99983
- o Maintenance Monitoring May Be Added For $250K, Regardless Of System Concept
- o Production Costs Are Based On Lot Size Of 100 For Total Buy Of 1000 Engines

TABLE F-1

CONTROL SYSTEM STUDY

Control Type Comparison

Control Type	Vehicle/Engine Interface	Method	Est. Accuracy	Measurements
Single Set Point FN & MR	Relays,Switches, Data Bus,Power	Two Position Valves Solenoid Actuated	+/- 2.5%	None
Variable FN, Fixed MR	Relays,Switches, Data Bus,Power	Closed Loop FN/ Open Loop MR	+/- 1.0% FN +/- 2.5% MR	Two Of Any Three Listed 1.Fuel And Oxidizer Flo Rates Measured At The Engine/Vehicle Interface 2.Fuel Flow And Main Pc 3.Oxidizer Flow And Main Pc
		Closed Loop MR/FN	+/- 1.0%	
		Open Loop FN/ Closed Loop MR	+/- 2.5% FN +/- 1.0% MR	
Fixed FN, Variable MR	Relays,Switches, Data Bus,Power	Closed Loop MR/ Open Loop FN	+/- 1.0% MR +/- 2.5% FN	Same As Above
		Closed Loop FN/MR	+/- 1.0%	
		Open Loop MR/ Closed Loop FN	+/- 2.5% MR +/- 1.0% FN	
Variable FN, Variable MR	Relays,Switches Data Bus,Power	Closed Loop FN/MR	+/- 1.0%	Same As Above
		Closed Loop FN/ Open Loop MR	+/- 1.0% FN +/- 2.5% MR	
		Closed Loop MR/ Open Loop FN	+/- 1.0% MR +/- 2.5% FN	

TABLE E-2

EXPANDER CYCLE POWER MARGIN

This discussion uses the H2/O2 expander cycle as an example but applies equally well to hydrocarbon engine configurations. The power margin discussion which follows is included per the request of GDSSD and it is similar to the discussion previously provided to GDSSD. All engines independent of cycle face the challenge of reaching rated thrust during their development program. In the development phase, the components rarely meet all of their performance goals in the first engine build. Some modifications and/or minor redesigns are normally needed to achieve rated engine operational capability by the end of the development program. While gas generator and staged combustion cycle engines are initially plagued with having to run too hot a turbine temperature to meet rated thrust, the expander cycle engine could possibly have too low a turbine temperature.

The attainment of rated thrust in expander cycle engines which depend upon the regenerative heat in the nozzle for turbine power, is impacted by both the heat picked up (ΔT) as well as the pressure loss (ΔP) in the nozzle tubes and manifolds. The expected impact of these heat exchanger characteristics on the engine during development and production phases is described in the following sections.

1. Development Program Uncertainties

Figure E-7 shows the predicted nozzle ΔT and ΔP design uncertainties for the initial set of H2/O2 engine development hardware. The pressure drop is a function of density and; therefore, these parameters are interdependent, and a concurrent worst case for both parameters is not practical to consider. A reasonable level of uncertainty is estimated to be $\pm 10\%$ of both parameters.

The design point for the expander cycle is currently set with a turbine bypass margin of 10.6% (excess available horsepower). This excess power capability can also be expressed in terms of excess chamber pressure margin. Figure E-8 shows the effect of ΔP and ΔT deviation on the level of engine chamber pressure margin. Chamber pressure margin is the chamber pressure level available above the normal operating point for a fixed set of hardware. It is a capability designed into the hardware and is readily available for engine thrust growth or compensation for component variations. As shown, the current engine design margin is 110 psi. The lowest expected chamber pressure margin resulting from design uncertainties (most likely extreme) is 40 psi; however, the margin is just as likely to be 170 psi.

The O2/H2 expander engine is being designed with a 110 psi chamber pressure margin to cover both a 10% thrust growth and the predicted engine to engine variations due to manufacturing tolerances. In addition to the design margin just described, there are a number of paths available to recover chamber pressure during the development phase. They include redesign of chamber/nozzle to obtain original design goals, redesign of the chamber to increase length for greater ΔT, redesign of manifolds to split the chamber and nozzle coolant paths to reduce pressure loss, redesign of the chamber to add trips or fins to enhance hot side heat transfer, redesign of chamber to substitute higher strength, higher conductivity tube material to decrease pressure loss and redesign of the chamber to increase throat area which would allow an increase in thrust without increasing chamber pressure.

The predicted effect on chamber pressure margin of these various redesigns is shown in Table E-3. A review of these data shows that the various development methods available to compensate for the projected level of heat exchanger performance deviations is adequate to cover even worst case scenarios.

2. Production Program Uncertainties

In the production phase all of the delivered engines must demonstrate rated thrust during the acceptable test. This is accomplished by providing sufficient margin in the engine design to cover engine to engine variations due to manufacturing tolerances.

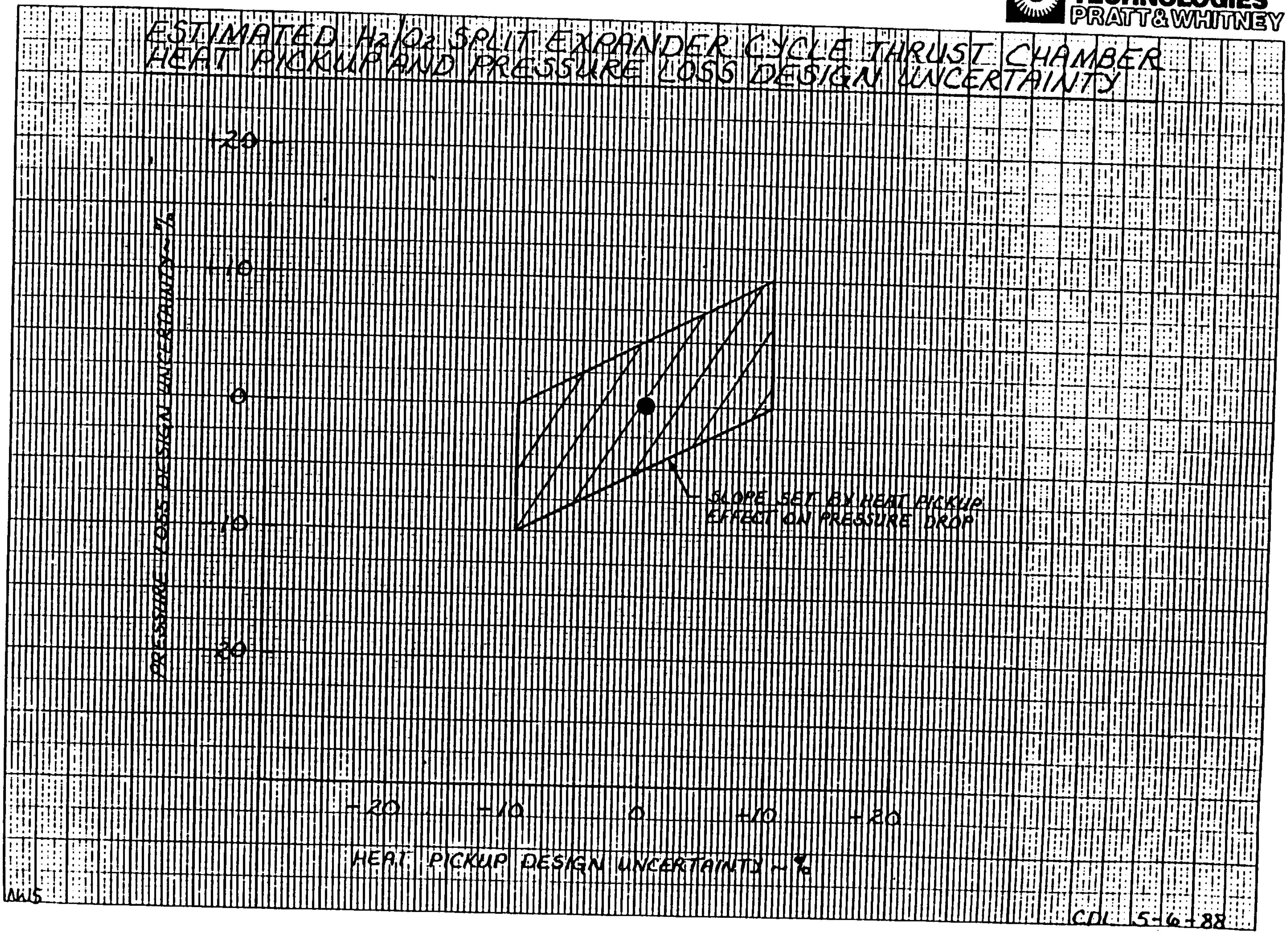
RL10A-3-3 production engine data was surveyed to determine the historical expander cycle power variation. Eighteen RL10 production engines had been run during acceptance testing with the necessary measurements to determine turbine excess power. The variation in turbine power was determined to be only 2.93% (2 sigma variation 95% confidence). This represents the total turbine power variation experienced due to variations in all component performance (chamber, nozzle, pumps, turbine, etc.) between production engines. Accordingly, the H2/O2 expander engines have been designed with 3% of the turbine power margin consigned for expected deviations from nominal component operating characteristics.

TABLE E-3

Development Parameter	Addition To Pc Margin At Operating Point
o Use design margin	110 psi
o Correct original design	Up to total error
o Add 5 inch chamber length	42 psi
o Split flow manifold	48 psi
o Enhancement features	Effects not quantified
o Substitute tube material	57 psi
o Increase throat area	30 psi

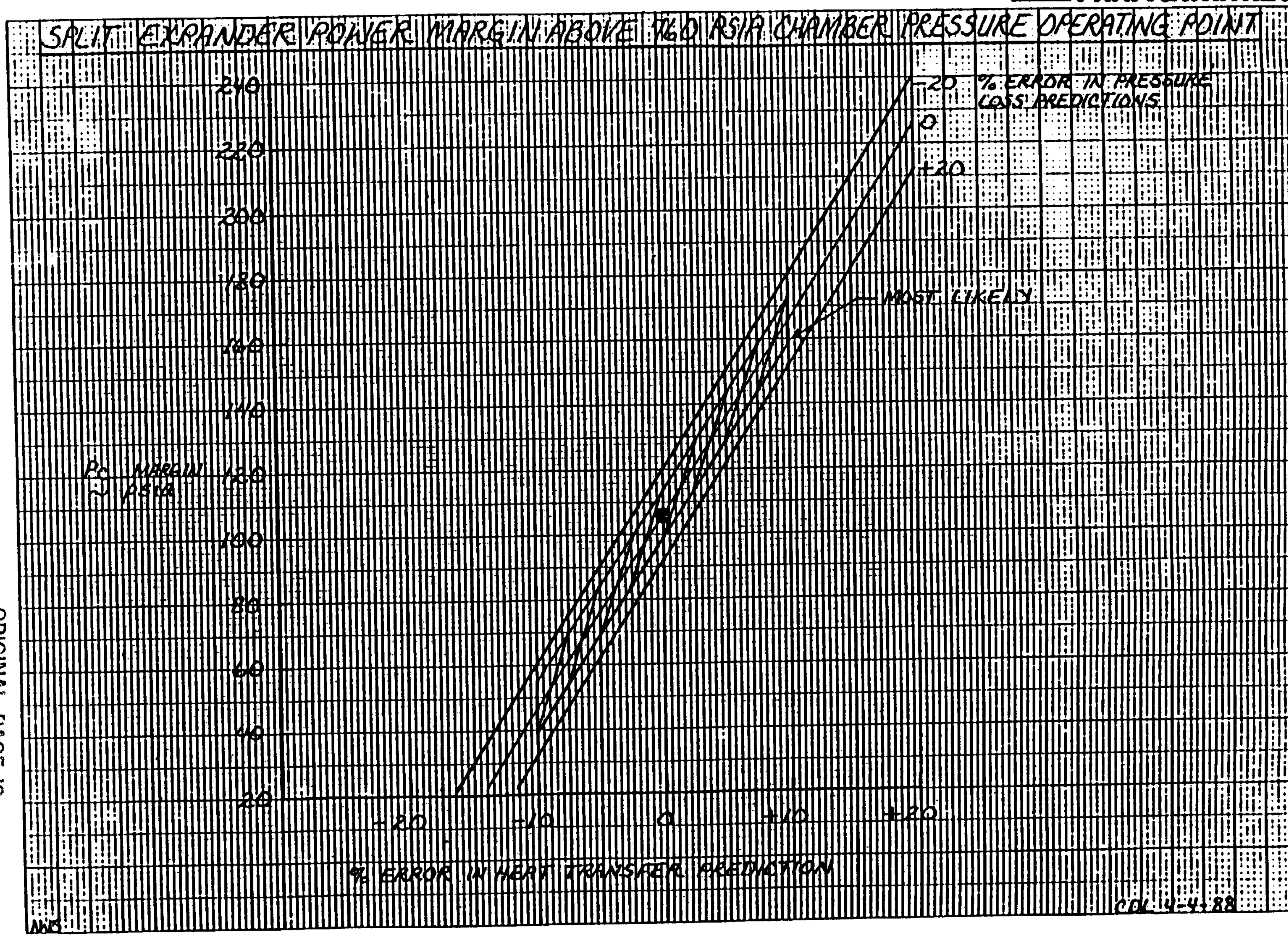
46 1320

FIGURE E-7



UNITED TECHNOLOGIES
PRATT & WHITNEY

FIGURE E-8



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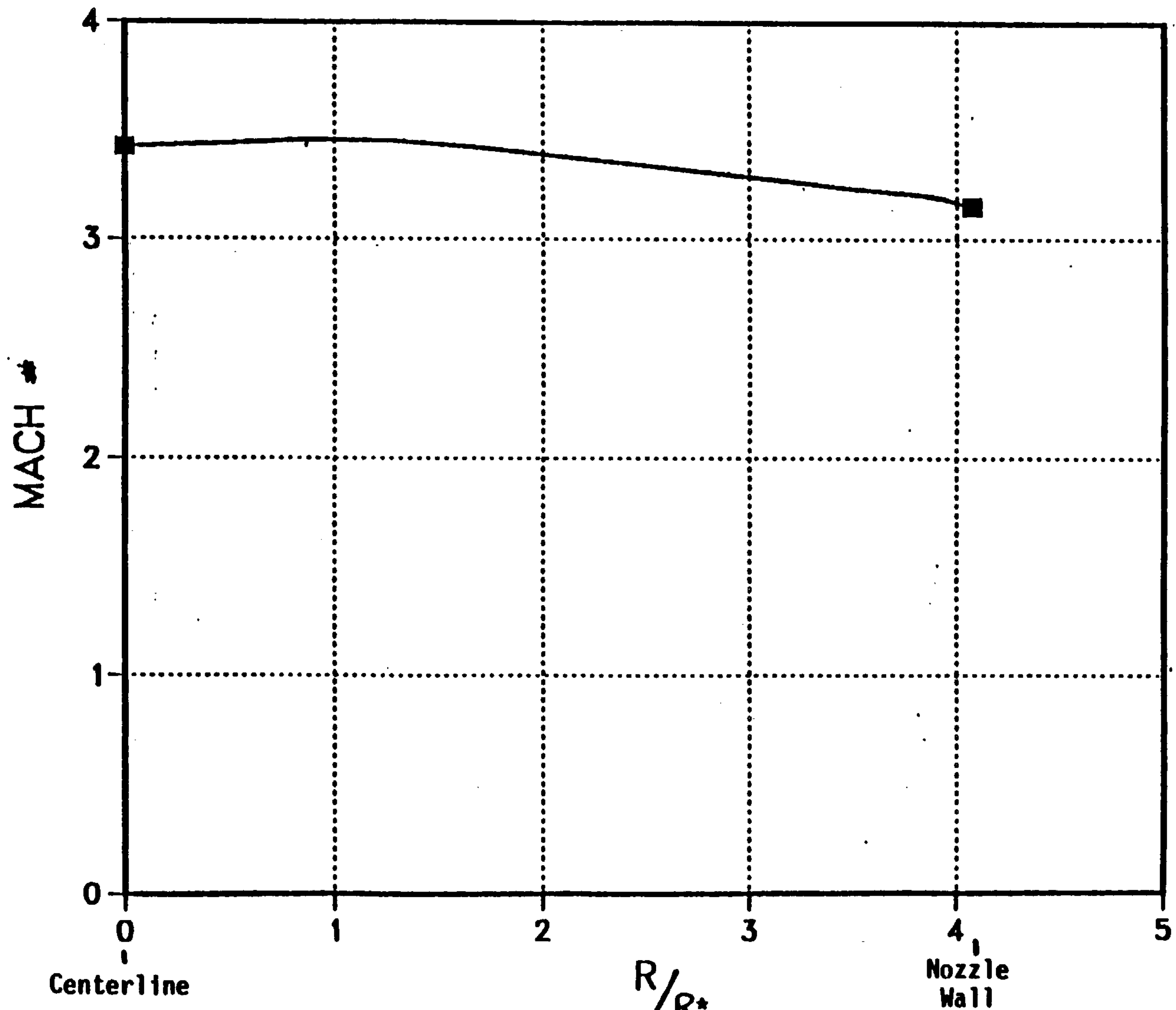
NOZZLE EXIT PLANE FLOW CONDITIONS

Flow characteristics at the nozzle exit plane are presented in Figures E-9 through E-13 for the final LO2/CH4 756.3K vacuum thrust engine. These profiles were generated using the JANNAF TDK and TBL computer models. Combustion efficiency of 99% and nozzle efficiency of 97.4% were calculated for the 758 psia chamber pressure mixture ratio of 3.5 base case. Boundary layer thickness at the nozzle exit was calculated to be 0.16 inch and the average specific heat ratio at the nozzle exit 1.2.

NOZZLE EXIT PLANE SUMMARY



GD LRB engine
LOX/CH4
Pc 758 psia

FIGURE E-9

R* = 13.18 in

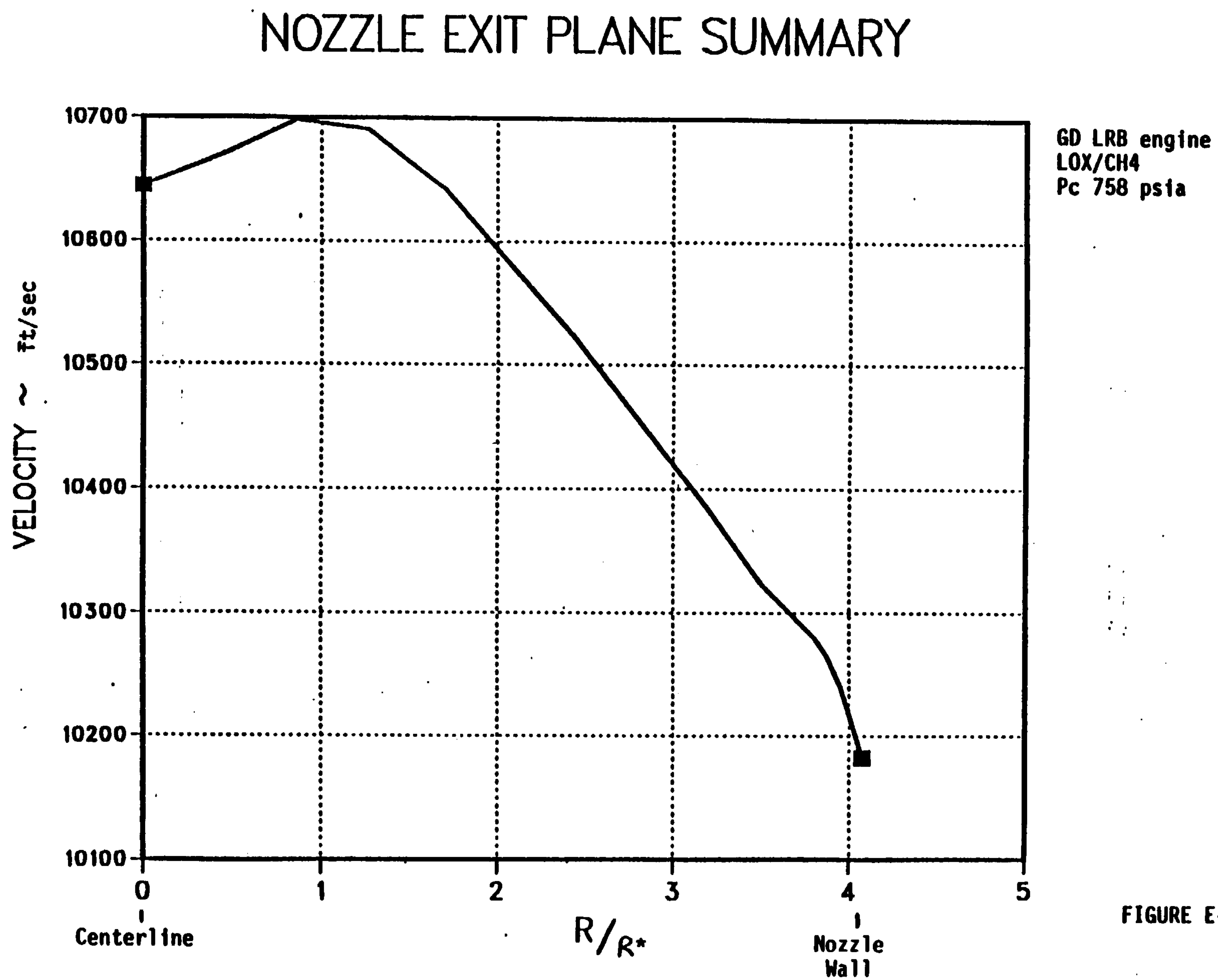


FIGURE E-10

R* = 13.18 in

DEG 5/20/88

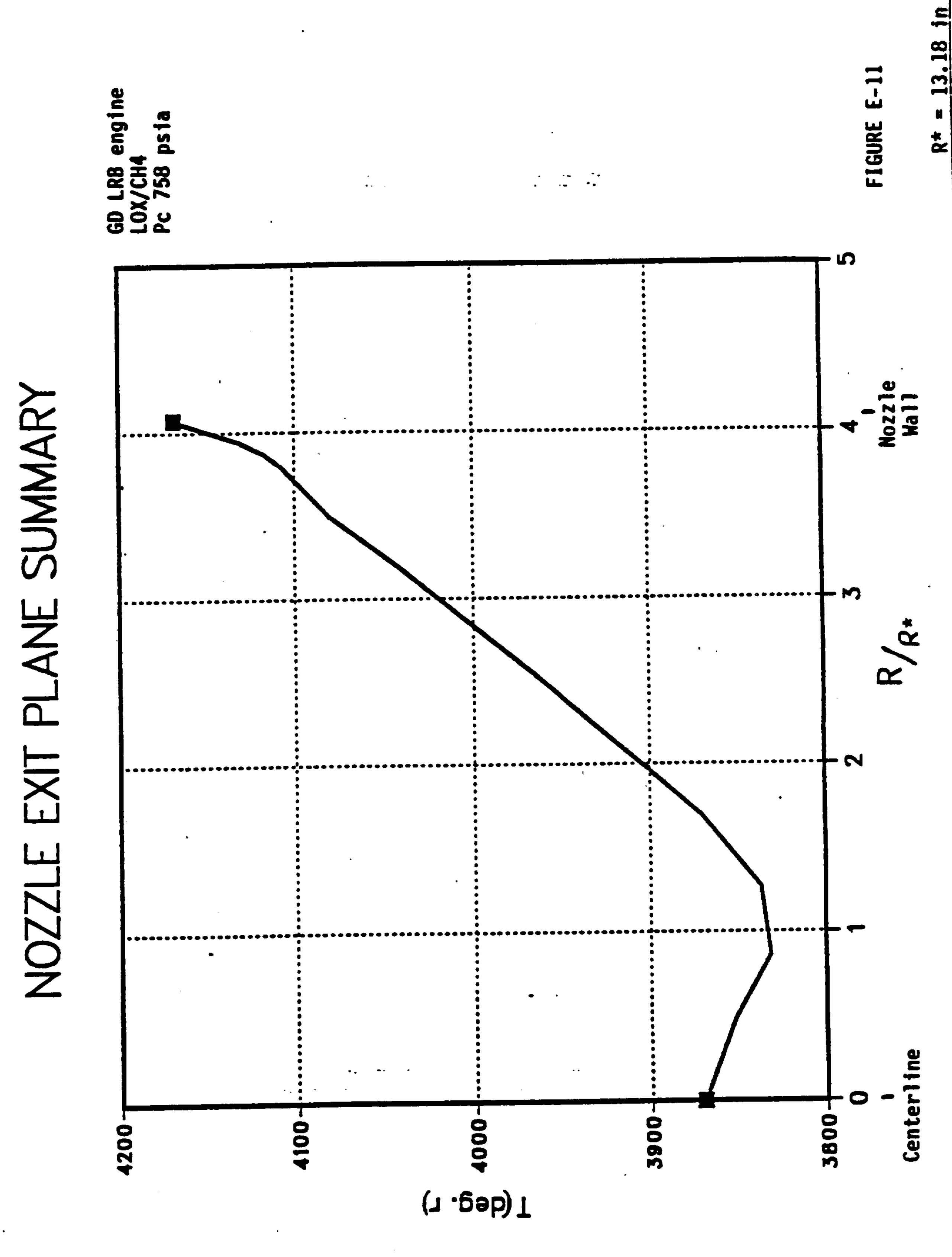

NOZZLE EXIT PLANE SUMMARY
GD LRB engine
LOX/CH4
Pc 758 psia
T(deg. r)
4200
4100
4000
3900
3800
0
1
2
3
4
5
Centerline
Nozzle Wall
R/R*
R* = 13.18 in

FIGURE E-11

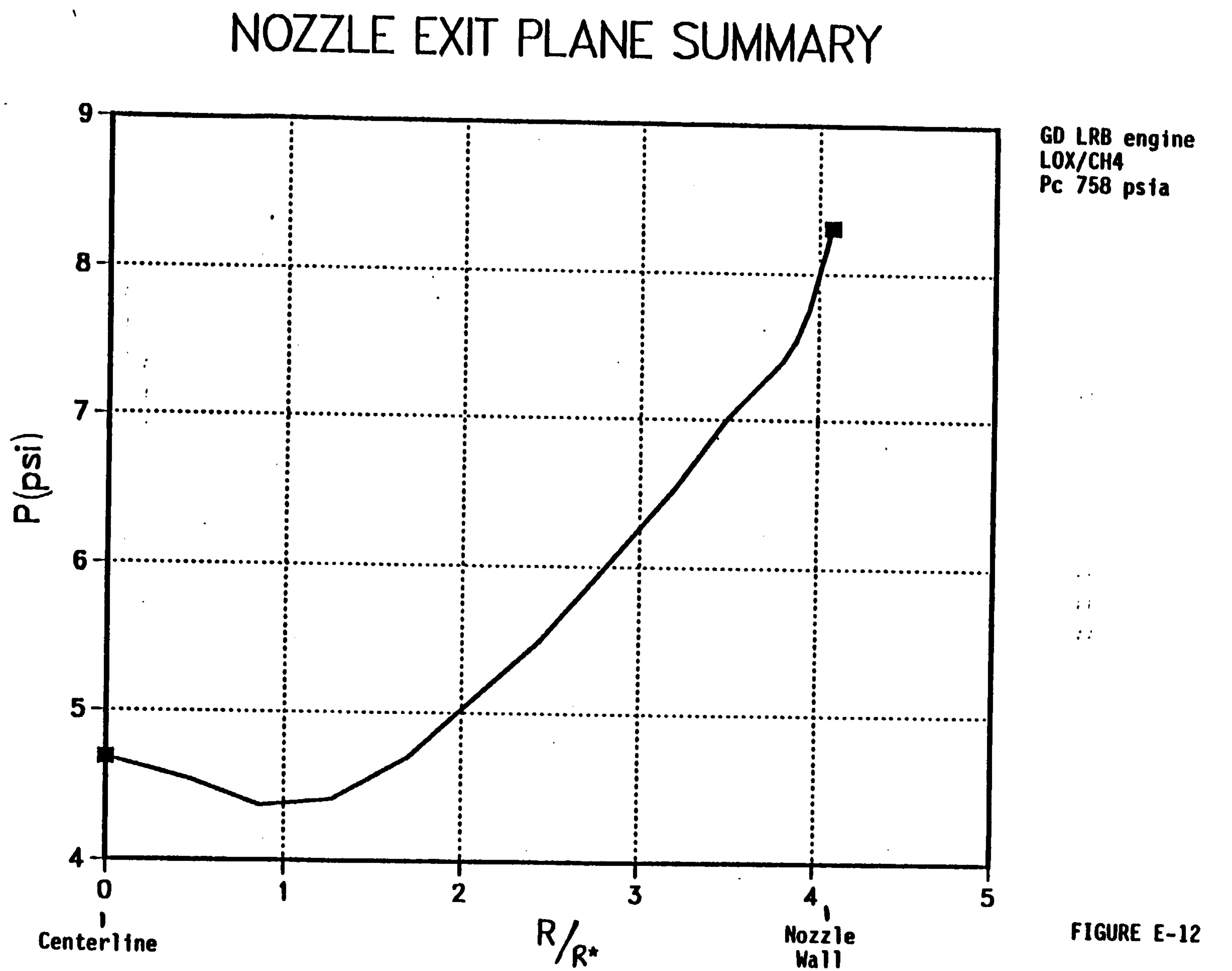


FIGURE E-12

R* = 13.18 in

DEG 5/20/88

NOZZLE EXIT PLANE SUMMARY

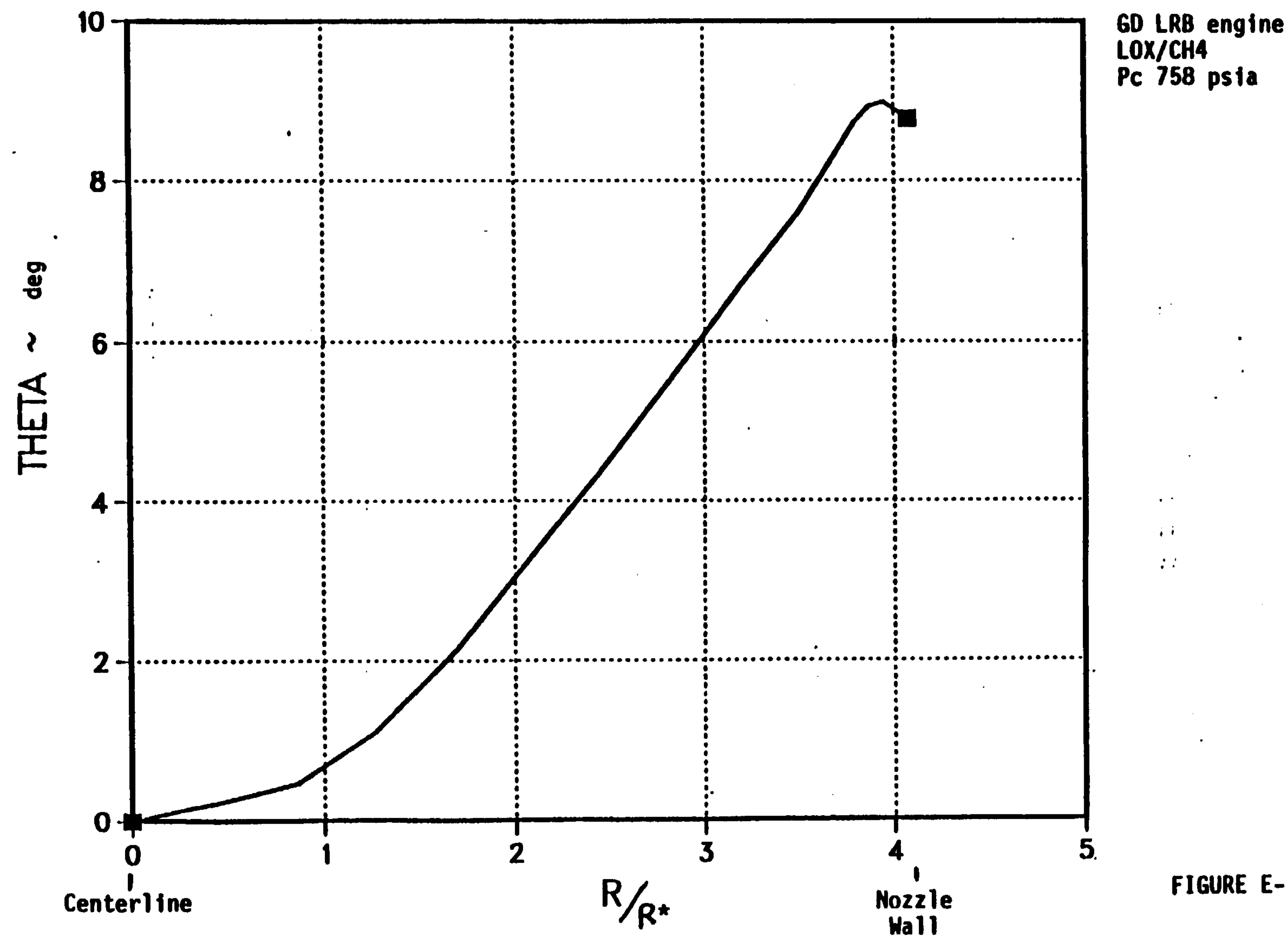


FIGURE E-13

R* = 13.18 in

FLIGHT INSTRUMENTATION

Flight instrumentation requirements will not be fully defined until the development program is underway and safety monitoring requirements have been fully defined. The following list is the best estimate at this time.

Preliminary GDSSD LRB Flight Instrumentation

Chamber Pressure

Fuel Pump Inlet Pressure

Fuel Pump Inlet Temperature

Fuel Pump Housing Temperature

Fuel Pump Vibration

Fuel Pump Speed

Oxidizer Pump Inlet Pressure

Oxidizer Pump Inlet Temperature

Oxidizer Pump Housing Temperature

Oxidizer Pump Vibration

Oxidizer Pump Speed

Fuel Turbine Inlet Pressure

Fuel Turbine Inlet Temperature

INTERFACE REQUIREMENTS

The engine was designed with propellant conditions assumed to be at 40 psia on the CH4 line interface to the engine and 60 psia at the LO2 interface. The engine propellant line diameters are approximately 10 inches for the LO2 line and 9.3 inches for the CH4 line. The horizontal mounting of the turbopump assembly results in sufficient line length on the engine flow ducts. No additional straight line length is required on the vehicle side of the interface.

The preferred electrical interface with the vehicle requires dual power supplies of 135 or 270 VDC with a maximum combined power of 5 kw for a fully variable thrust control. For a single point thrust control system a 28 VDC dual power supply with a maximum combined power of 0.5 kw would be preferable.

Six line interfaces are required for tank pressurization, engine purge and valve actuation. Two pneumatic lines are required for LO2 and CH4 tank pressurization, two propellant recirculation lines and an N2 supply and He supply line to the engine are also required. The nitrogen source line can be routed to the launch facility for a supply source. Two actuator attachments, spaced 90° apart, are provided on the engine for the vehicle supplied actuators. The maximum actuator load is estimated to be 2700 lbs for the gimbal conditions of ±6° gimbal angle, 10° per second velocity and 40° per second squared acceleration. A sketch of the assumed actuator attach point in the skirt is shown in Figure E-14.

ACTUATOR LOCATION

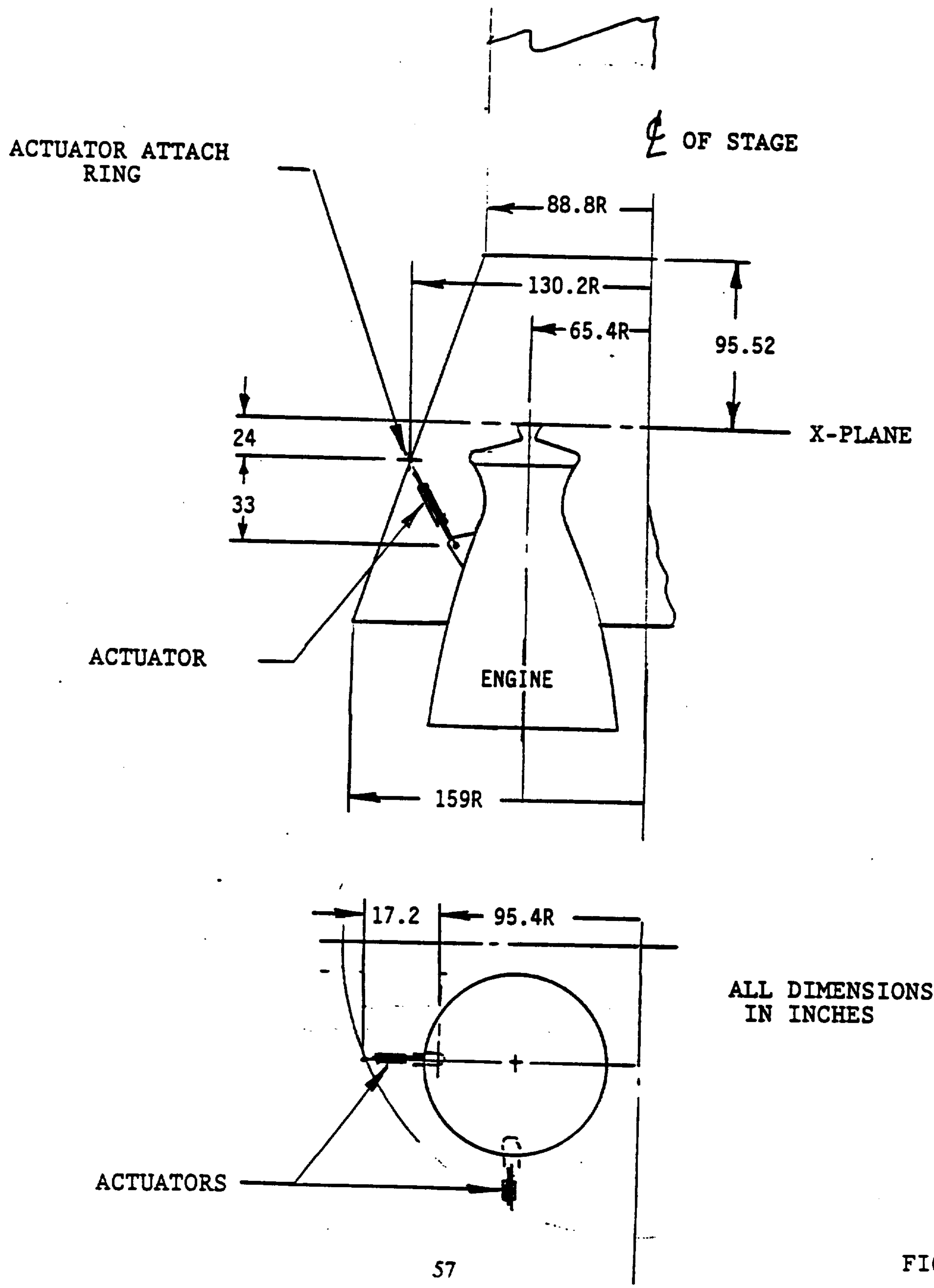


FIGURE E-14

The main interfaces are shown on the following single engine drawing (Figure E-16).

SINGLE ENGINE DRAWING

Figures E-15 and E-16 show two views of the selected 756K vacuum thrust engine design. The turbopumps are mounted back to back (turbines facing each other) with the rotors counterrotating. This configuration reduces the number of parts in the turbine areas by providing smaller diameter turbines with less blades and the future potential of eliminating the vanes between the turbine rotors. Turbine flow ducting is also reduced to a minimum. The engine area ratio was selected at 16.46 to maintain the nozzle exit diameter at a size compatible with the vehicle diameter and gimballing clearance requirements. The inlet flow ducts utilize scissor-type bellows to maintain simplicity and low weight on the engine. The scissor joints can accommodate up to approximately ±6° of gimbal angle. Figure E-16 shows the plan view of the engine with all interface points.

MULTIPLE ENGINE DRAWING

Figure E-17 shows the mounting arrangement for the four engines in the LRB stage. The centerline spacing of 130.8 inches was used as suggested by General Dynamics to allow sufficient clearance at the engine nozzle exit plane when gimballing the engines. The turbopump assemblies face the inside of the stage and the actuator attach points are arranged on the outside of the engine cluster to accommodate engine actuator mounting to the stiffener ring in the vehicle skirt.

GENERAL DYNAMICS LRB CH4/LOX SPLIT EXPANDER

756K VACUUM THRUST, AREA RATIO = 16.46

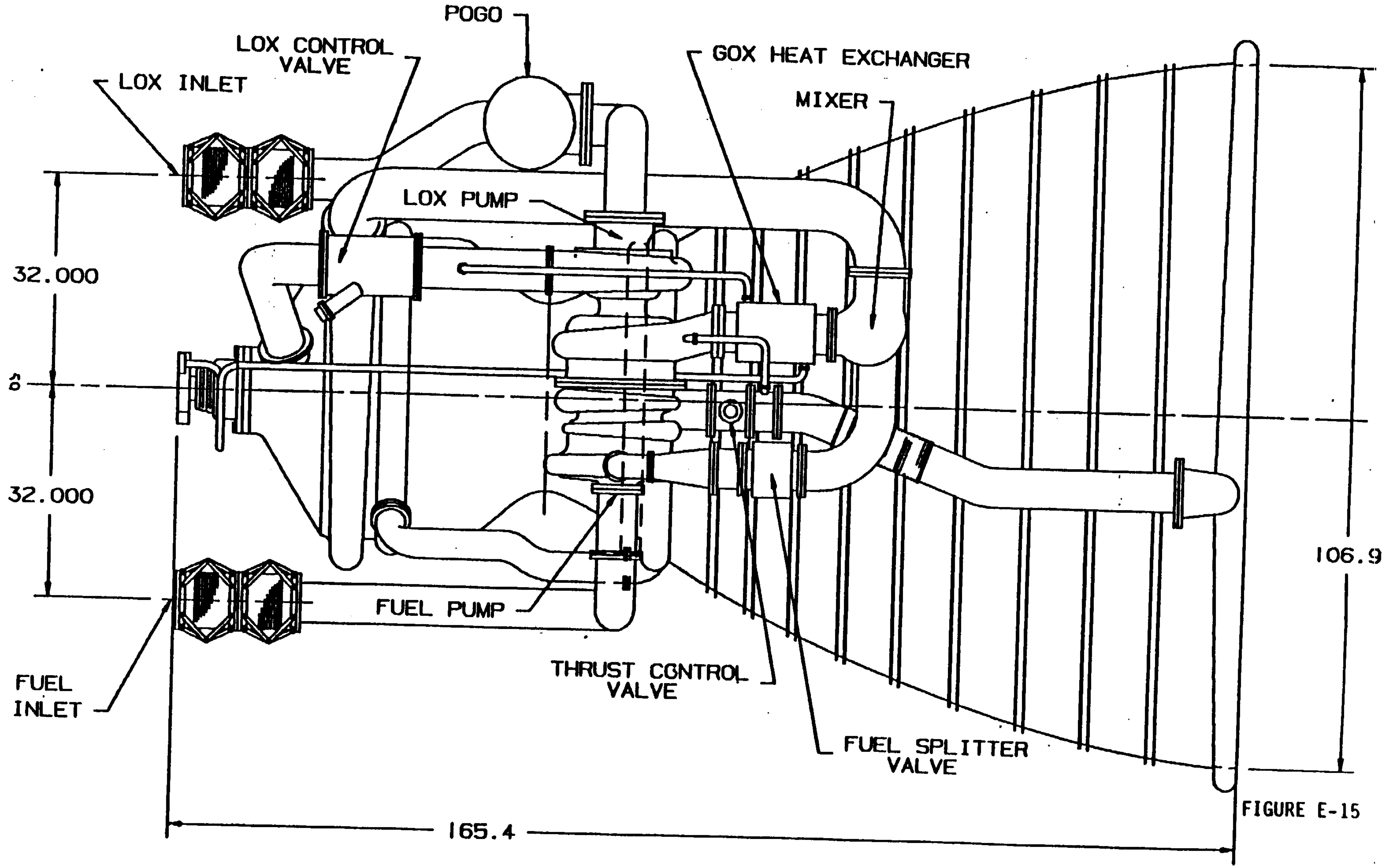


FIGURE E-15

GENERAL DYNAMIC

756K VACUUM THRUST

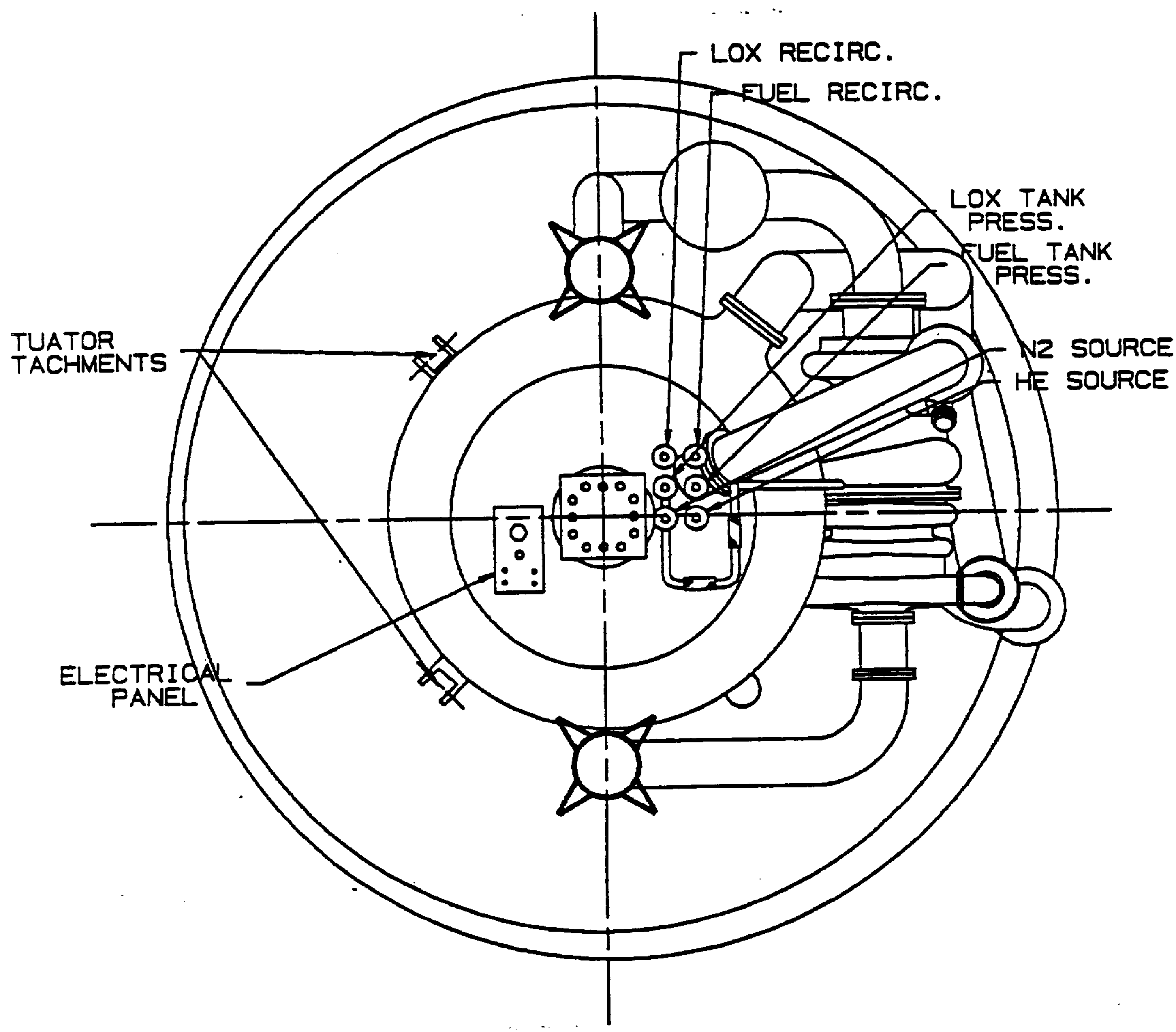


FIGURE E-16

GENERAL DYNAMICS LRB CH4/LOX SPLIT EXPANDER
CLUSTER ASSEMBLY

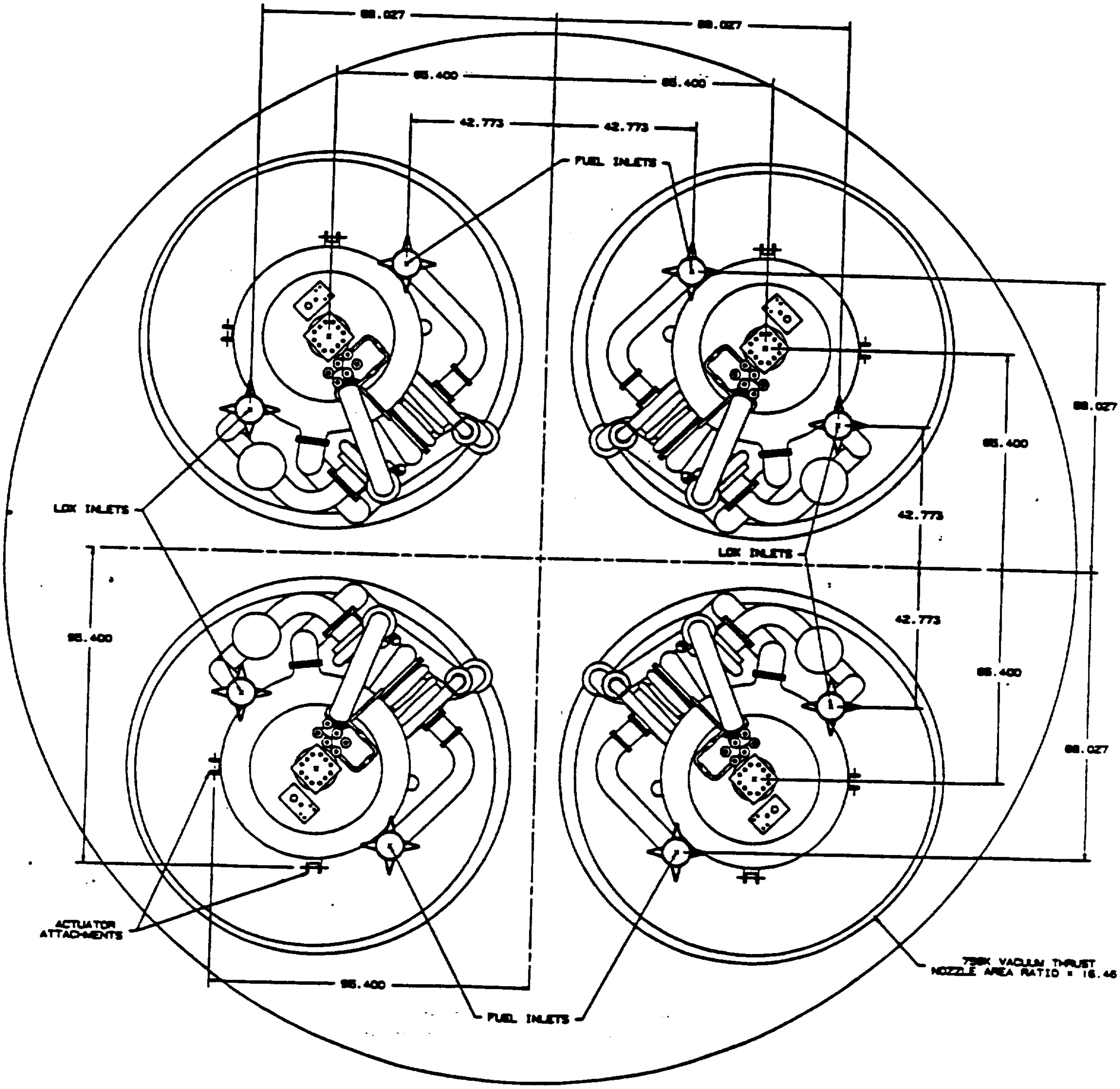
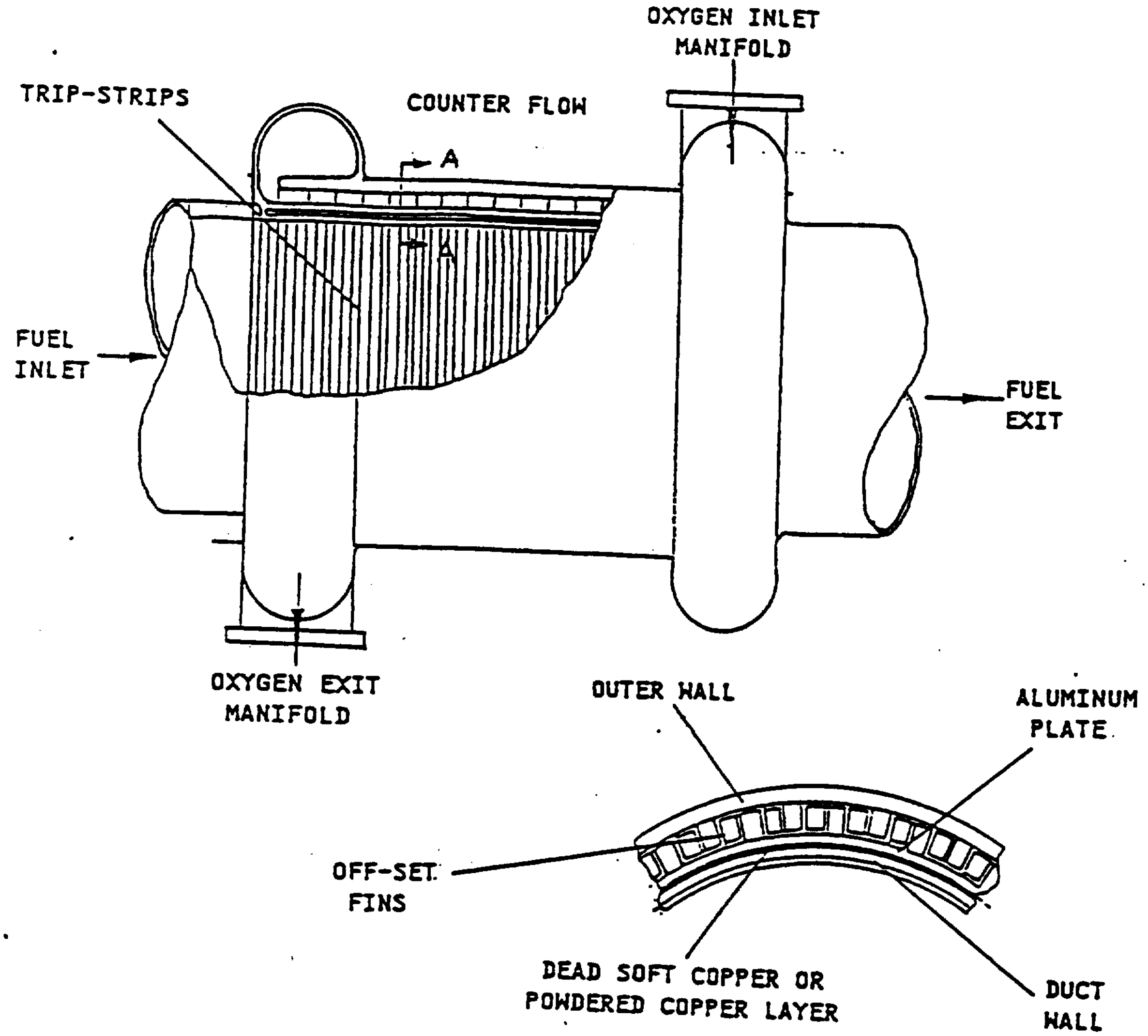


FIGURE E-17

F. Pressurization

The engine is configured to provide gaseous methane at (1000 psia and 705°R) a flowrate of 1.7 lbs/sec and gaseous oxygen at (1100 psia and 400°R) a flowrate of 11.1 lbs/sec. The gaseous methane is bled off the engine between the turbine discharge and the mixer. The gaseous oxygen is produced in a gox heat exchanger which utilizes the hot gaseous methane to vaporize the oxidizer.

The gox heat exchanger, which gasifies liquid oxygen for tank pressurization, consists of a counter flow offset finned heat exchanger that is wrapped around the turbine exhaust duct. Figure F-1 presents the gox heat exchanger configuration. The gox heat exchanger consists of an aluminum duct wall that has trip-strips on the turbine exhaust side wall for improved convective heat transfer film coefficients. The oxygen passages are constructed of offset fins that are bonded to the high strength outer wall and the inner aluminum plate. The offset fins enhance the oxygen side convection heat transfer film coefficients which will reduce the size of the heat exchanger. The aluminum plate is separated from the duct wall by a highly conductive layer of either dead soft copper or powder copper in colloidal suspension. The copper layer has been incorporated into the design to stop crack propagation from the inner plate to the duct wall.



VIEW A-A OXYGEN PASSAGE GEOMETRY

FIGURE F-1

G. POGO And Stability Analysis

The LRB POGO system will be similar to the SSME POGO system which uses a gas filled plenum to isolate engine feedline oscillations from the engine. Pressurized gox is supplied by the gox heat exchanger and the gas is used to energize the POGO suppressor and the LO2 propellant tank.

H. Reliability and Safety

The initial fault tree for the split expander engine is shown in Figure H-1. The fault tree lists all significant abnormal events which could happen to the primary components on the engine and the resulting effect on the engine system. Figure H-2 is a legend which is to be used to properly interpret the fault tree notations.

Figure H-3 shows the reliability of the RL10 engine down to the component level. The reliability values shown are based on a total of 1470 accountable engine firings, both during ground test and operational flights.

Figure H-4 is an estimated component reliability assessment of the Split Expander engine using the RL10 demonstrated reliability as a basis but making adjustments for higher pressures and temperatures.

System reliability relation of the liquid rocket booster stage propulsion system, based on eight total engines with one engine out capability, is shown in Figure H-5. This figure can be used to determine the necessary single engine reliability based on the overall booster propulsion system reliability.

SPLIT EXPANDER ENGINE

FAULT TREE ANALYSIS

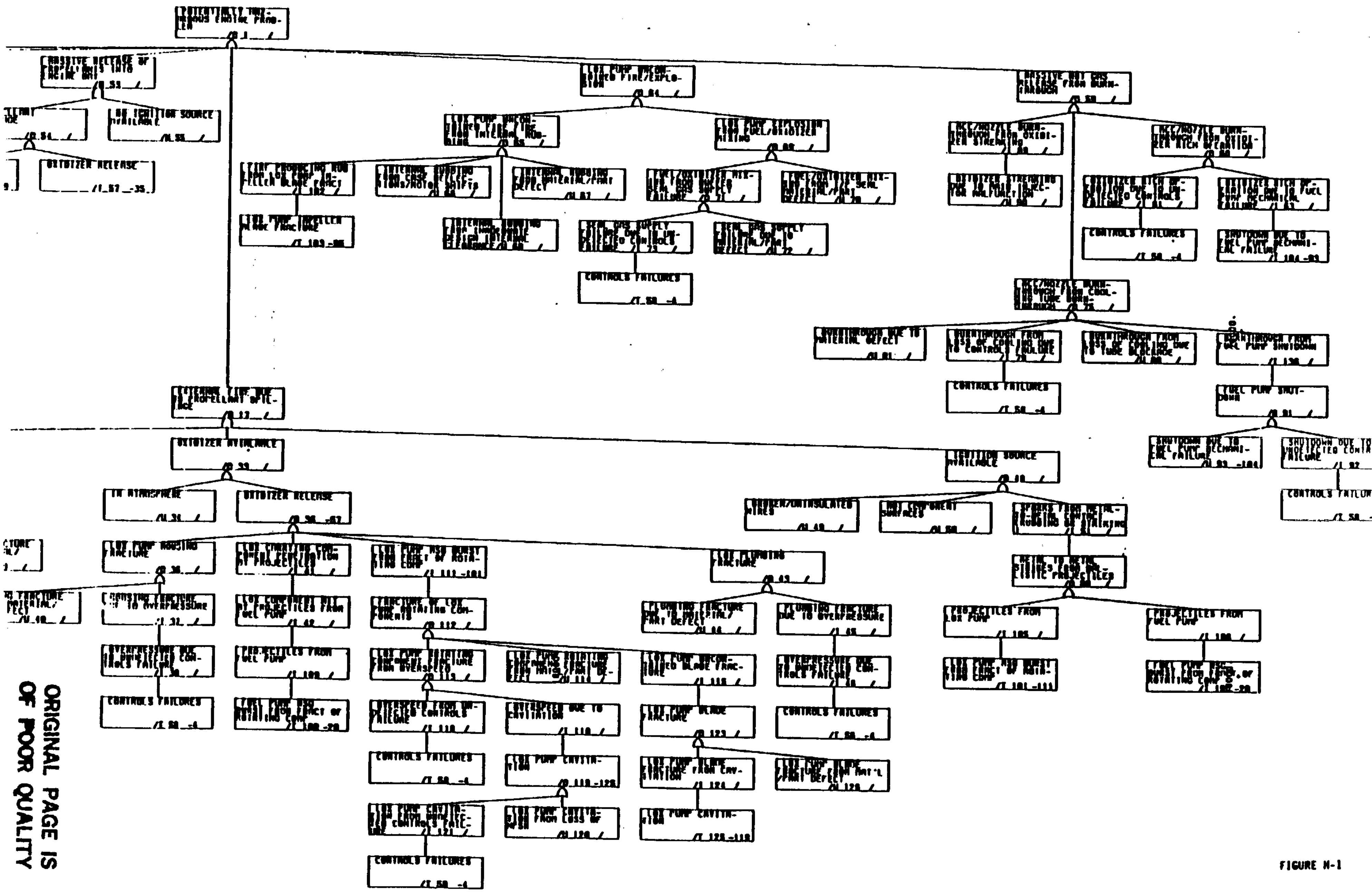
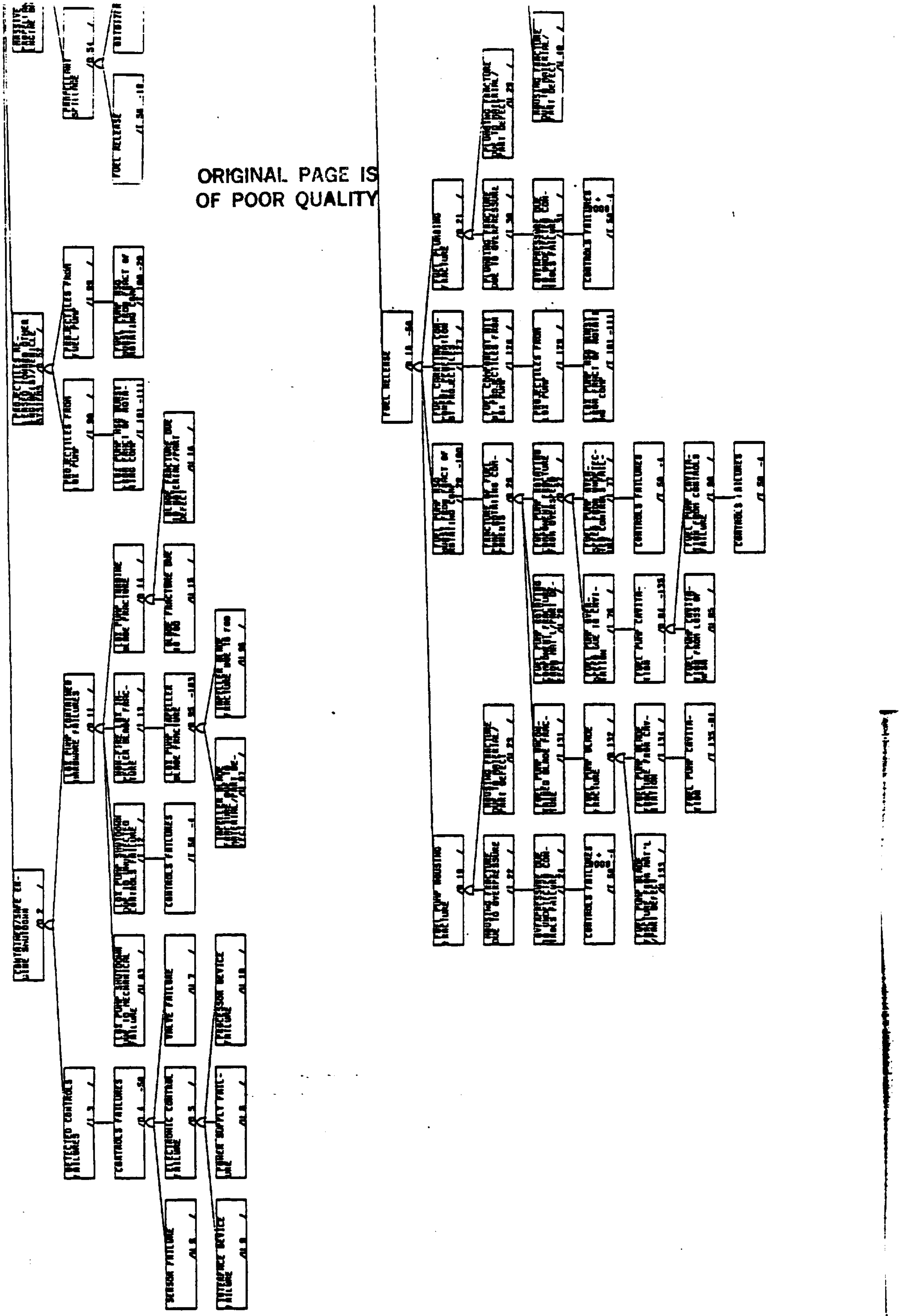


FIGURE H-1



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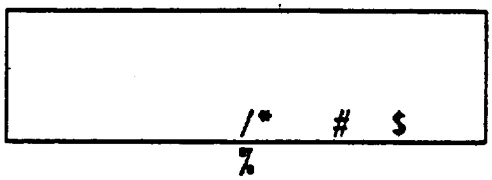
FAULT TREE NOMENCLATURE



*	TYPE GATE	A - "And" Gate O - "Or" Gate U - "Undeveloped" Gate T - "Transfer" Gate H - "House" Gate I - "Inhibit" Gate
#	Node Number	
$	Node Modifer	/ - No Additional Info - XX - Transfer from/to Gate
%	Node Identifier	- "And" Gate - "Or" Gate

FIGURE H-2

[redacted]

RL10 RELIABILITY

COMPONENT	LOSS RATE (/1000 firings)	SOURCE
-Oxidizer Pump	0.011	RL10, w/SSME split
-Hydrogen Turbopump	0.021	RL10, w/SSME split
-Gearbox	0.000	RL10, w/PW5000 prediction
-Thrust Chamber and Injector	0.021	RL10 history
-Oxidizer inlet valve	0.004	RL10 history
-Fuel inlet valve	0.022	RL10 history
-Main fuel shutoff valve	0.075	RL10 history
-Oxidizer flow control valve	0.000	RL10 history
-Fuel pump interstage valve	0.069	RL10 history
-Fuel pump discharge valve	0.038	RL10 history
-Start solenoid valve	0.000	RL10 history
-Prestart solenoid valves(2)	0.000	RL10 history
-Ignitor Oxidizer supply valve	0.002	RL10 history
-Prelaunch cooldown check valve	0.000	RL10 history
-Thrust control	0.025	RL10 history
-Ignition system	0.133	RL10 history
-Gimbal	0.000	RL10 history
-Engine plumbing	0.005	RL10 history
-Propellant plumbing	0.047	RL10 history
TOTAL	0.471	

$\bar{R} = 0.999529$

NOTES:

-Reliability based upon 1470 accountable engine firings without failure and calculated at the 50% confidence level for the zero failure case.

-Component breakdown based upon prior study that evaluated 4000 development test firings containing 594,000 seconds of run time (Ref: PWA FR-4657 dated 10/26/71)

-Turbopump assembly failure rate of 0.032 broken down to pump and gearbox level based upon PWA SSME-ATD prediction and using PW5000 gearbox prediction.

FIGURE H-3

SPLIT EXPANDER CYCLE RELIABILITY PREDICTION

COMPONENT	LOSS RATE (/1000 firings)	SOURCE
-Oxidizer Turbopump	0.033	RL10 Turbopump w/temp, spd,press bias
-Hydrogen Turbopump	0.039	RL10 Turbopump and temp, spd,press bias
-Thrust Chamber and Injector	0.023	RL10 history w/temp,press bias
-Fuel split valve	0.060	RL10 MFSOV w/temp,press bias
-Oxidizer inlet valve	0.004	RL10 w/temp,press bias
-Fuel inlet valve	0.022	RL10 w/temp,press bias
-Main fuel shutoff valve	0.070	RL10 w/temp,press bias
-Oxidizer flow control valve	0.004	RL10 w/temp,press bias
-Fuel pump interstage valve	0.090	RL10 w/temp,press bias
-Fuel pump discharge valve	0.059	RL10 w/temp,press bias
-Start solenoid valve	0.000	RL10
-Prestart solenoid valves(2)	0.000	RL10
-Ignitor Oxidizer supply valve	0.004	RL10 w/temp,press bias
-Prelaunch cooldown check valve	0.000	RL10
-Thrust control	0.049	RL10 w/temp,press bias
-Ignition system	0.133	RL10 history
-Gimbal	0.000	RL10 history
-Engine plumbing	0.003	RL10 w/temp,press bias
-Propellant plumbing	0.045	RL10 w/temp,press bias
	TOTAL = 0.638	

$\bar{R}$ = 0.999362

FIGURE H-4

EIGHT ENGINE LIQUID BOOSTER SYSTEM
ONE ENGINE OUT CAPABILITY, CORRELATION = 0.01

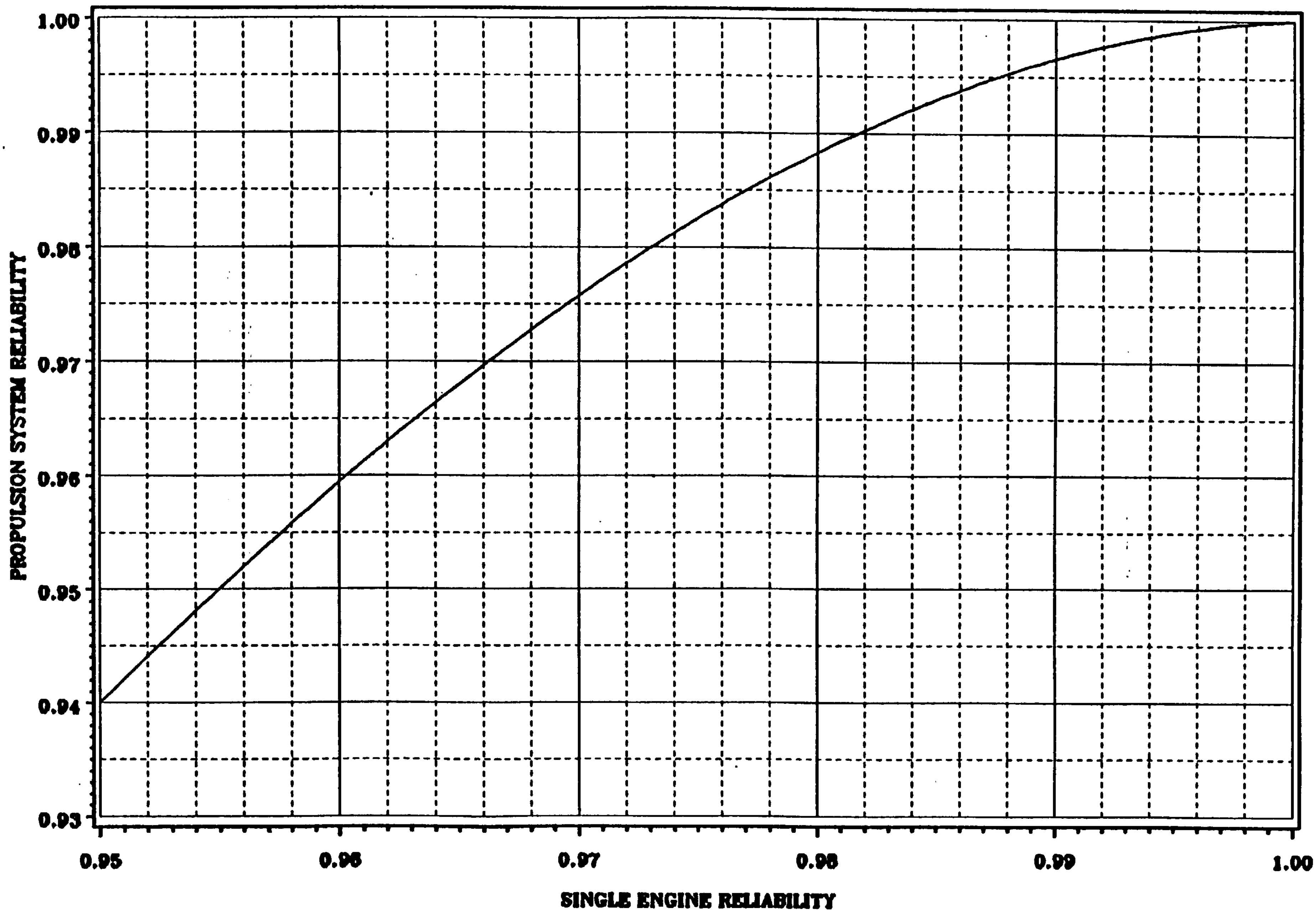


FIGURE H-5

PAGES 71 THRU 74 CONTAINING COST DATA HAVE NOT BEEN INCLUDED IN THIS APPENDIX DUE TO THEIR PROPRIETARY NATURE. THESE PAGES HAVE BEEN SUBMITTED SEPARATELY TO GENERAL DYNAMICS.

J. Programmatics

DEVELOPMENT SCHEDULE

The engine development schedule for the LO2/CH4 Split Expander engine up to the Preliminary Rating Test (PRT) which qualifies the engine for test flight is 4 years and 3 months as shown in Figure J-1. Additional engine testing required beyond PRT is additional qualification testing to demonstrate reliability beyond the .99 (90% confidence) level at PRT and an engine cluster test. The first flight of the engine could occur as early as 4 years and 9 months from full scale development start. The completion of the flight rating test is estimated at 5 years, 2 months following initiation of the full scale development program.

DEVELOPMENT DESCRIPTION

Development of the engine would begin with a parallel development effort for the main chamber, LO2 pump and CH4 pump. Test stand firings of the main combustion chamber would occur at 19 months into the program followed by test runs of each pump individually. The three major hardware development components would be integrated with the other engine hardware and control system and the first full engine test firing would take place approximately 25 months following FSD.

A total of ten engines are required through PRT--five for development and five for flight configuration testing. Four engines in each category are considered active with one engine being used as a spare. In addition, 3 complete sets of major components are to be used during the program as well as spare parts required for test engine overhaul. Three major failures are assumed during the development schedule which will consume the spare sets of hardware. Each test engine will be overhauled for bearings and seals after 35 firings and a chamber and pump overhaul will occur after 70 firings on each engine.

A total of two development test stands were assumed to be supplied by the Government with testing occurring at scheduled rate of five firings/week/stand. Test stands are assumed to be utilized 3 shifts/day for 6 days/week. In addition, it is assumed that the Government will supply the propellants for the test firings. Turbopump test stands at P&W facilities in Florida would be utilized for the program. However, a Government supplied pump stand would also be used. Government test facilities are expected to be utilized at NASA's NSTL located in Mississippi.

DEVELOPMENT SCHEDULE

Same Schedule for LOX/H_2 and LOX/CH_4

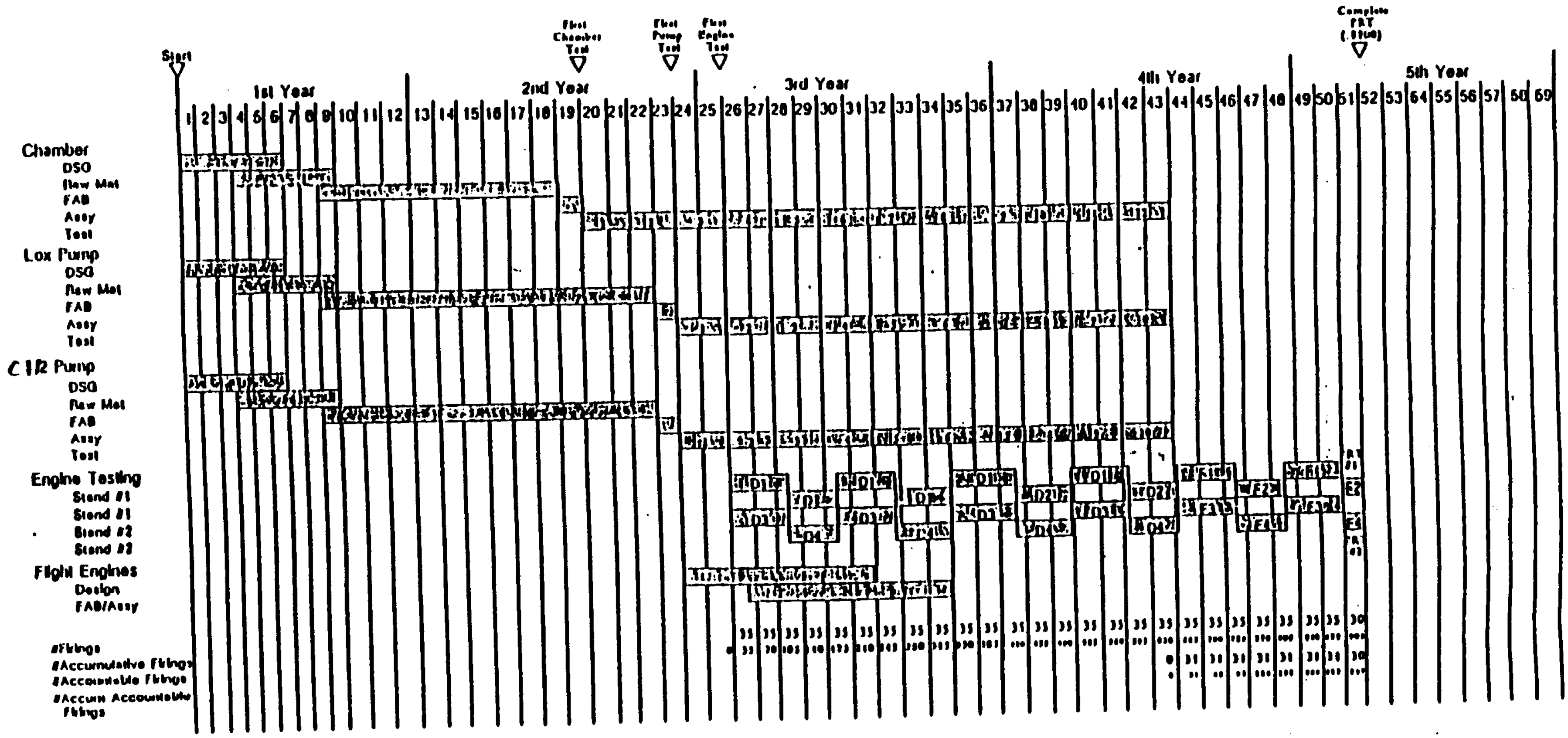


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FIGURE J-1

ENGINE PRODUCTION AND INTEGRATION WITH VEHICLE

Pratt & Whitney plans to produce the split expander cycle rocket engine at its existing manufacturing facilities. Primary production of components and subassemblies would be done at the Connecticut facilities of P&W with support from P&W plants located in Georgia and Maine as well as from the P&W vendor base. The P&W production facilities have over ten million square feet devoted to fabrication and assembly of aerospace engines. These facilities currently produce more than ten times the quantity of engines needed for the LRB program per year. The majority of these engines are intricate, high technology gas turbine engines which require well controlled, cost effective production methods. Parts which lend themselves to automation techniques would be produced using automation. In all cases, the most cost effective methods for producing engine parts would be utilized. The current production capability at P&W facilities can absorb the production requirements for the split expander engine.

Production engine testing would be conducted at NASA/NSTL facilities located in Mississippi. These facilities would be used for engine development testing and tests required before production engines are shipped to General Dynamics. P&W currently envisions acceptance testing each production engine prior to shipment.

An assembly facility would be built and located at the engine test facility for the LRB engines. This assembly facility would be used for the final assembly of both development and production engines thus reducing costs by reducing logistic costs associated with shipping complete engines to and from the Connecticut manufacturing facility to the Mississippi test facility. Engine teardown and repair would be conducted at the assembly facility.

The integration of the engines with the booster stage would take place at the launch facility at the Kennedy Space Center. Vehicle integration at General Dynamics facilities in California is not desirable due to the relatively large size of the engines and booster stage.

No additional technology is required prior to the start of the engine development program. However, specific technology programs to reduce manufacturing costs could be conducted in parallel with the development program. The technologies are similar to those suggested by P&W for the Advanced Launch System program with some modifications necessary to meet the target launch date and schedule for the LRB. The preliminary list of technology programs to reduce the cost of the split expander engine components are identified as follows: cast injector element technology, chamber/nozzle materials technology, tubular nozzle fabrication technology, cast fuel and lox pump impeller fabrication, integrally bladed turbine disk/blade technology, low cost poppet-type valves for metering and shutoff of engine propellants and, finally, low cost diagnostic system technology for expendable engines.

APPENDIX 8

LRB FINAL REPORT
FROM TRW

TRW

Liquid Rocket Booster For STS Feasibility Study

Phase 1 Final Report

Contract No. L/C 08-01289

15 February 1988

Prepared for:
General Dynamics Space Systems Division
San Diego, California

In support of:
NASA/MSFC Contract NAS8-37137

01-004-3682/CV (02-16-88)

CONTENTS

CONTENTS (Continued)

FOREWORD

This document summarizes the work performed by TRW between 12 November 1987 and 15 February 1988 under subcontract to General Dynamics Space Systems (GDSS) Division on a Phase I study of the feasibility of a liquid propellant rocket booster for STS. The prime contract was issued by NASA/MSFC under Contract Number NAS8-37137 and the TRW work was performed under GDSS Contract Number L/C 08-01289. The technical monitor for GDSS was Gopal Mehta and the TRW program manager was Frank J. Stoddard of the Engineering Operations of TRW's Applied Technology Division.

Under the terms of the subcontract, TRW was to supply GDSS with information on pressure-fed liquid propellant booster engine and pressurization system technology. The resultant information and studies, summarized herein, represent the technical contributions of a number of TRW engineers. The primary contributors were:

Charles Greenwald, Project Engineer
James Reeve, Thermal/Fluids Analyst
Dr. Lee Dailey, Propulsion System Analyst
Kenneth Trowbridge, Design
Herman Kreig, Propulsion Engineer
Dr. Jack Blumenthal, Chemist
James Wells, Propulsion Engineer
Gordon Dressler, Propulsion Engineer
Dean Cheng, Mass Properties
John Hardgrove, Propulsion Manager
Joseph Lewis, Tank Specialist
Dick Weiner, Components Engineer
Dr. Thomas Fitzgerald, Physical Chemist
Dr. Erwin Fishman, Chemist
Dr. Edward Barrish, Chemical Engineer
Dr. David Lewis, Gasdynamicist
Benjamin Cruz, Test Engineer
Maurice O'Brien, Components
Edward Brooks, Chemical Engineer

1. INTRODUCTION AND SUMMARY

With the recent failures of U.S. launch vehicles it was made apparent that launch reliability is the number one criteria for launch vehicle design. It is therefore important that any new space transportation system consider reliability/safety first, followed by low cost. It is expected that the high reliability launch vehicles of the future will be of simple design and will use both old and new technology as necessary. Launch vehicle performance (i.e., I_{sp}, mass fraction, etc.), while third in order of importance, is still important to many applications such as Space Shuttle, where a PFLRB is being considered as a replacement for an existing SRB.

It was decided in the Phase I studies that the propellant for the liquid propellant booster would be LOX/Hc. Both pump-fed and pressure-fed liquid propellant engines have been operated on this propellant with high performance. The major reason for considering a pump-fed engine system is to reduce propellant tank weight. For a system such as the Space Shuttle booster, where the propellant tanks are designed to support the entire launch stack, including the Shuttle itself, the propellant tanks are thick-walled for both pump-fed and pressure-fed engines. With similar weight tanks, the pump-fed system has only two advantages over a pressure-fed system: its engine is smaller and it is able to employ a more efficient, higher area ratio nozzle. In all other important respects, the pressure-fed system is superior.

For example:

The historical reliability of pressure-fed systems is much higher than that for pump-fed systems.

The pressure-fed systems are much less complex with far fewer failure modes.

The cost of pressure-fed engines is much lower than that of pump-fed engines.

- The cost/pound of payload is much lower for a pressure-fed booster system than for a corresponding pump-fed booster.

- A pressure-fed engine is more compatible with recovery and reuse.

The reliability issues that take precedence in the design of future launch vehicles all favor pressure-fed systems:

- Simplicity

- Verification of design margins

- Preflight system verification

- Redundancy in critical components

The unique problems of current pump-fed engines which drive their costs up and their reliability down are:

- High speed propellant pumps with dynamic seals and multiple valving

- High speed hot gas turbines with dynamic seals and multiple valving

- Hot gas generators (an extra rocket engine)

- Regeneratively cooled rocket engine chamber with very complex structural/cooling tubes system

- High pressure combustion chamber shell and nozzle

- Complex injector system with thousands of small orifices

- Combustion stability problems that require unique baffling systems to suppress instabilities

Some claim that the pressure-fed launch vehicle would not be more reliable than a pump-fed launch vehicle. This is simply not true based on any acceptable method of calculating reliability. There are two orders of magnitude more parts in a pump-fed engine, the pump-fed propellant tanks are more fragile, and both have pressurization systems and TVC systems. Starting with the failure mode and effect analysis through the actual reliability calculations, a simple pressure-fed LCLV is more reliable. The prelaunch verification testing is much more complex with pump-fed engines than with pressure-fed engines where almost all dynamic elements can easily be tested prior to flight. According to all Shuttle experts, the most critical subsystem in the launch of the Shuttle is the pump-fed SSME. It was always believed that these engines would be more apt to fail than the SRBs.

Once one accepts the rationale for pressure-fed launch vehicles being more reliable and less costly than pump-fed launch vehicles, it follows that TRW's simple coaxial injector engine is more reliable and less costly than its competition for many of the same arguments:

- Much simpler injector concept than shower head, doublets, triplets and radial injection and other injectors with thousands of small orifices
- Demonstrated inherent combustion stability
- Simple ablative-cooling system that avoids the burn-through risks and added pressure drop of the regeneratively cooled chambers
- Adaptable to any of the three candidate TVC systems (differential throttling, LITVC or gimballing)
- Face shutoff capability to protect the injector during water landings
- Very low cost manufacture, integration and test
- Low development cost and low technical and schedule risks
- Only injector concept with deep throttling capability

- Most reliable of any engine concept due to its simple injector and ablative-lined combustor chamber
- Readily scaled to larger sizes.

Past experience in testing this engine has demonstrated high performance in an engine designed for earth storable propellants that was tested with LOX/RP-1. There is no reason that TRW's coaxial engine should have lower performance than another type of pressure-fed engine with the same area ratio nozzle. Also, the propellant tank weights for the ablative-lined engine should be lower since there is no added cooling system pressure drop.

The TRW engine is easily adapted to recovery and refurbishment. The injector is shut off at its face before descent, thus trapping residual propellant and pressurized gas in the tanks. After water recovery, the ablative liner is replaced, the injector refurbished (if necessary) and the engine is acceptance tested.

During the LRB Phase I contract, TRW provided GDSS data to support booster systems studies. This included propellant data, both theoretical and experimental on both neat and gelled propellants. GDSS has since selected LOX/RP-1 for the LRB. TRW engine experience indicates that our coaxial injector is compatible with LOX/RP-1 and that stable combustion can be sustained with acceptable performance. The coaxial injector eliminates the combustion instability problem that has plagued other types of injectors and contributed to long and expensive engine development programs. Demonstrated C* efficiencies in TRW engine configurations not optimized for LOX/RP-1 have been as high as 95.6 percent. An optimized engine should be capable of 97 percent based on our experience with optimized engines using storable propellants. In most cases, the combustion efficiency of the coaxial injector has been found to rival that of doublet or triplet injectors. Consequently, we expect to be able to achieve combustion performance comparable to the doublet or triplet injectors using LOX/RP-1 without having the instability problems of those injectors.

In addition to its inherent stability, the coaxial injector provides easily mechanized, excellent throttling performance over a 100 percent to 70 percent thrust range (<0.5 percent rolloff). Deep throttling (10:1) performance of greater than 92 percent C* is easily achieved with this injector. It is impossible to produce this throttling performance with other types of injectors.

This coaxial injector capability can be utilized in an LRB application to eliminate the need for a TVC system. In that case, effective thrust vectoring is produced by differential throttling. The required throttling rates for this purpose are readily achieved with a throttling coaxial injector. Elimination of a separate TVC by using differential throttling provides a significant weight reduction and increased reliability. Data on LITVC and gimballing were provided to GDSS for comparison purposes. These techniques for thrust vector control are readily incorporated on the TRW coaxial engine. Most of our previous booster engine studies showed LITVC as the control technique. With redundant valve injectors, LITVC is a relatively reliable control technique. Gimballed TRW engines were flown on the Apollo LMDE and the MDAC Delta. This is also an acceptable TVC system for TRW's engine and must be traded off based on both engine and overall system considerations. All three TVC techniques are acceptable to TRW and were presented to GDSS for final selection.

In addition to assisting GDSS in making the propellant selection, we provided weight and cost scaling data to support booster trade studies. This illustrated the major advantages of the coaxial injector: it is lightweight and low cost. Its demonstrated scalability and stability, coupled with low cost ablative material technology and engine design, result in low development costs and much lower recurring costs than those for other engine/injector types. Flat face, radial and coaxial injectors were ranked with respect to stability, throttling capability, face shutoff capability, manufacturing and development costs, combustion performance, chamber size, engine weight, reliability, and compatibility with water recovery. Overall ranking winner was the coaxial injector.

A weight scaling relationship was provided to GDSS based on existing 50K ablative engine weights as a baseline and analytical scaling relationships for each engine component. Once GDSS established 619K pounds thrust at a chamber pressure of 400 psia as the baseline engine and fixed the nozzle exit diameter at 90 inches, we performed a preliminary conceptual design of an ablative-cooled engine. Weight estimates for this design were obtained and found to be somewhat lower than predicted by the weight scaling relationship. Layout drawings of the conceptual engine design were provided to GDSS.

We also provided GDSS with weight and cost estimates for a regeneratively cooled engine, for both regeneratively and ablatively cooled engines with LITVC, and for recoverable and expendable versions of these engine/TVC combinations. The weight impact of gimballing the engine was also provided. Our recommendation on the basis of maximum reliability and minimum cost, weight and risk is to baseline the coaxial injector, ablatively cooled engine.

During Phase I, we supported GDSS in studying the pressurization system and provided data on lightweight propellant and pressurant tank manufacturing technology. Information on the relative benefits of KEVLAR and graphite epoxy overwraps was provided. Our position is that graphite epoxy overwrap is the clearly superior material. It provides lighter weight tanks while avoiding the water absorption and creep problems of KEVLAR.

We analyzed a number of candidate pressurization systems at the request of GDSS and added a catalyzed $He/H_2/O_2$ cascaded tank system that appeared promising. The latter system, consisting of two pressurant tanks, proved to be significantly lighter than the other candidates and has the potential for downsizing the booster. A number of issues with respect to the catalyst bed were examined. It is known that commercial catalysts can be used but their cost makes it desirable to find a lower cost alternative. TRW has done exploratory work in that area. We have determined that a packed bed catalytic reactor cannot be used because of the large pressure drop that would exist at flow conditions appropriate to the LRB. Other catalyst bed configurations, however, can be used to eliminate this problem while maintaining high catalyst efficiency.

Reliability data from flight experiences with pump-fed and pressure-fed engines were provided to GDSS. The most significant advantage of TRW's pressure-fed coaxial engine is its high reliability and lack of failure modes.

TRW appreciates having the opportunity of supporting GDSS in the Phase I LRB Study. We have long been advocates of low cost, high reliability liquid rocket boosters and over the years have developed a unique engine technology that we believe is ideal for this application.

The following text is divided into seven major sections. The first gives a historical review of TRW efforts in low cost, high reliability, pressure-fed, liquid propellant rocket booster engine developments. The next two sections provide propellant selection considerations and a review of TRW combustion stability experience with coaxial injectors. Then, a high reliability, low cost engine design concept for the STS application is discussed. In that section, options for engine cooling, throttling, injectors, TVC, etc., are ranked, engine and injector concept drawings and schematics are presented, weights are given, reliability is discussed, the influence of water recovery on engine design is considered and cost projections are presented. The section following that deals with a study of candidate pressurization systems. The sixth section following describes the status of composite tank technology and the last major section presents cost data. References are listed in the eighth section.

In addition to the main body of the text, two appendices of supporting data are provided. They cover weight and cost modeling.

2. HISTORICAL REVIEW OF TRW LARGE PRESSURE-FED LIQUID PROPELLANT ROCKET BOOSTER AND ENGINE TECHNOLOGY

TRW experience with large pressure-fed liquid propellant rocket engines and launch vehicles dates back to the early 1960's. Definition studies were performed under contract for NASA in 1962 on a heavy-lift sea launched two-stage-to-orbit launch vehicle. This concept, known as Sea Dragon, employed pressure-fed LH_2/LO_2 first and second stage engines. Tankage, thrust vector control, pressurization systems and engines were studied as well as complete fabrication-to-launch systems engineering analysis.

In the early 1960's TRW started engine design work leading to contract award in 1965 for the Lunar Module Descent Engine (LMDE). This highly successful engine landed the Apollo astronauts on the moon. The LMDE is a 10,500 lb_f thrust design using pressure fed storable bipropellants and is throttleable over a 10-to-1 thrust range. Aerospace/SAMSO studies during this period identified this engine design, employing a unique centrally located coaxial injector, as having demonstrated an unprecedented record of complete dynamic stability and scalability throughout its entire development history. Further consideration of the LMDE indicated that the design should be amenable to low cost fabrication for large pressure-fed fixed-thrust engines for booster systems.

In 1965 TRW was asked to determine the feasibility of scaling the LMDE to the multimillion lb_f thrust level, while maintaining previously demonstrated performance and combustion stability characteristics. Previous TRW experience had included the scaling of the basic engine concept from 25-500 lb_f to 500-5000 lb_f (MIRA-5K) and the LMDE was scaled from the MIRA-5K to the 1000-10,000 thrust level. Figure 2-1 shows engine C* efficiency test data as a function of thrust through our 250,000 lb_f engine.

Coaxial Injector Scaling

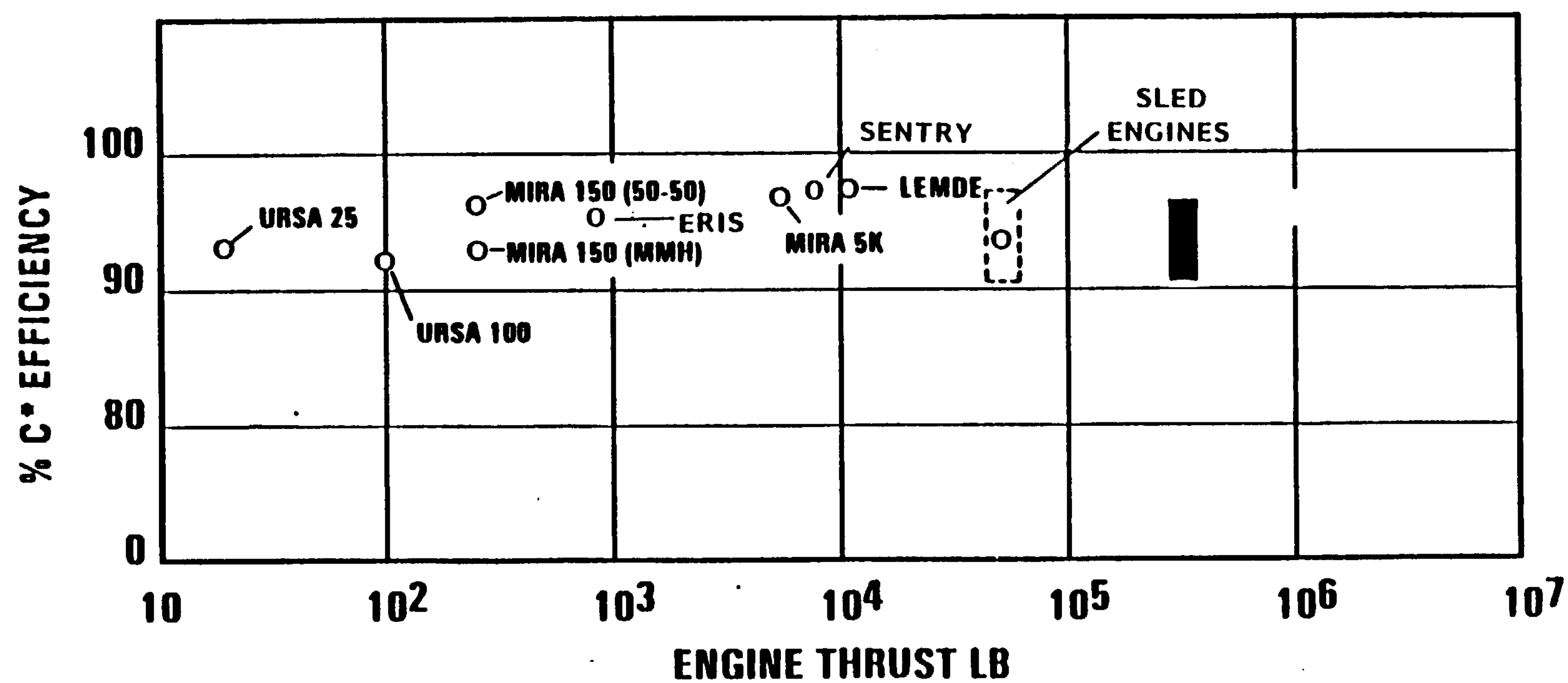

Figure 2-1. Coaxial Injector Combustion Efficiency

The engines shown in Figure 2-2 have all evolved from the 10,500 lb_f LMDE, shown in the upper right corner. The engines all use storable and cryogenic propellants and all use the same coaxial injector system. The engine in the lower left corner has been fired at thrust levels of 50,000 to 250,000 lb_f. The engine in the lower middle has been fired to thrust levels greater than 50,000 lb_f (sea level), while the engine in the lower left has been fired at thrust levels of 22,000 to 35,000 lb_f.

Following completion of successful hydraulic tests of an element of a 1.5 x 10^6 lb_f pintle injector, TRW independently undertook the fabrication and test of 250,000 lb_f thrust low cost ablative lined engines. The test hardware was built by commercial piping fabrication to demonstrate manufacturing techniques and design philosophy.

Figure 2-3 shows a firing of TRW's 250,000 lb_f pressure fed engine. This engine completed a successful development test program at the Air Force Rocket Propulsion Laboratory (AFRPL) during 1968-69. This project, a task under the overall Minimum Cost Design Space Launch Vehicle (MCD/SLV) Program, had as a goal the development of low-cost injectors capable of performing at 90 percent theoretical I_{sp} (shifting), 250,000-pound thrust using N_2O_4/UDMH propellants, and to evaluate their scalability up to the multimillion-pound-thrust class.

During these tests, several design configurations were evaluated which provided design data for demonstration injector tests later in the project. Several injector and chamber configurations were tested. Design condition performance obtained was approximately 94 percent of theoretical velocity (C*) and 90 percent vacuum I_{sp} (low C_F was acceptable since a cutoff cone was used for the nozzle). Dynamic combustion characteristics of this concept were evaluated by artificially inducing chamber pressure overpressures of 100 percent or greater. In all tests, the chamber pressure recovered to within 10 percent of the original value within 30 to 40 milliseconds, and the engine was considered dynamically stable.

TRW

TRW Low-Cost Design Engines

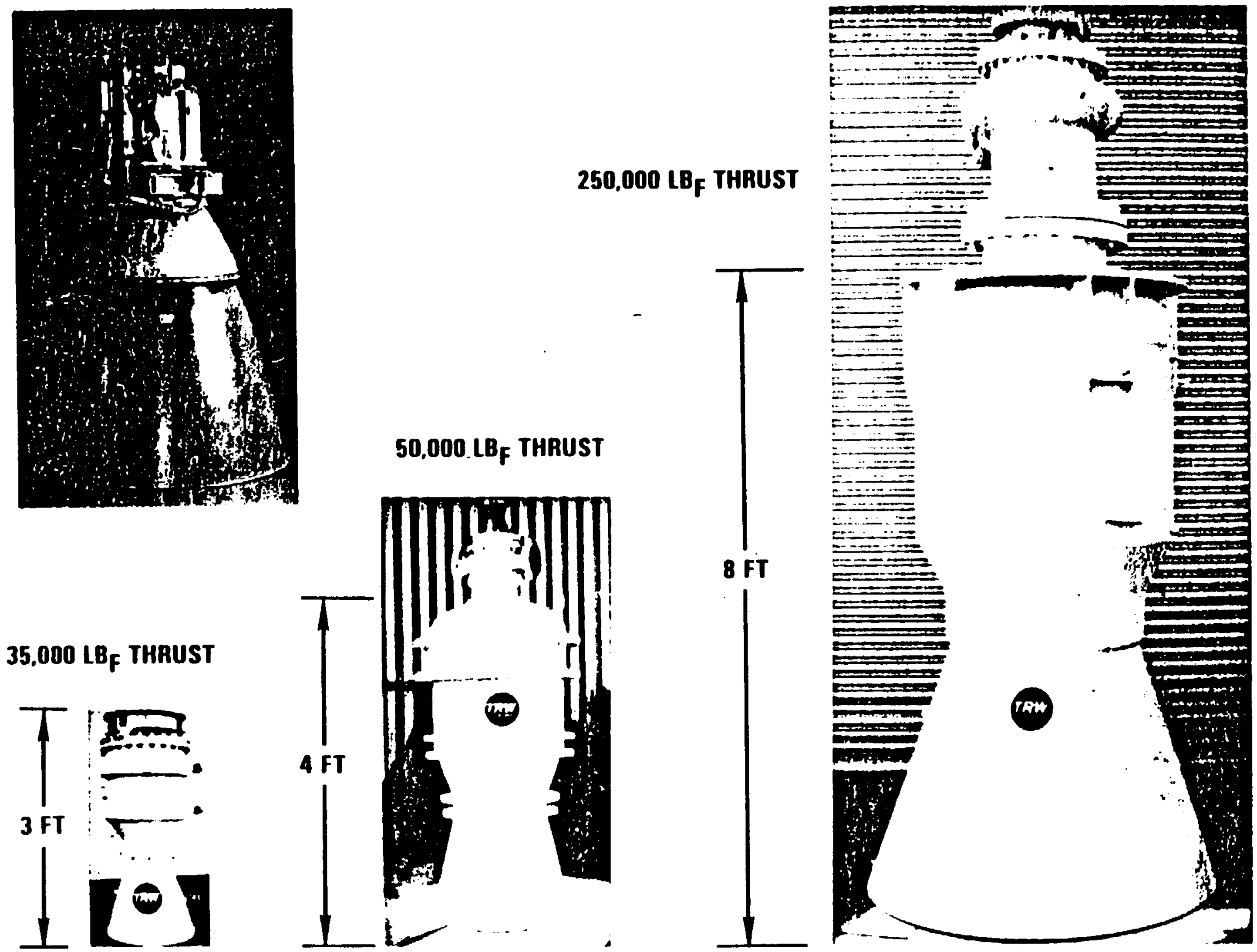

Figure 2-2. Evolution of TRW Low Cost Engines

250,000 Pound Thrust Engine Firing at Full Thrust at AFRPL

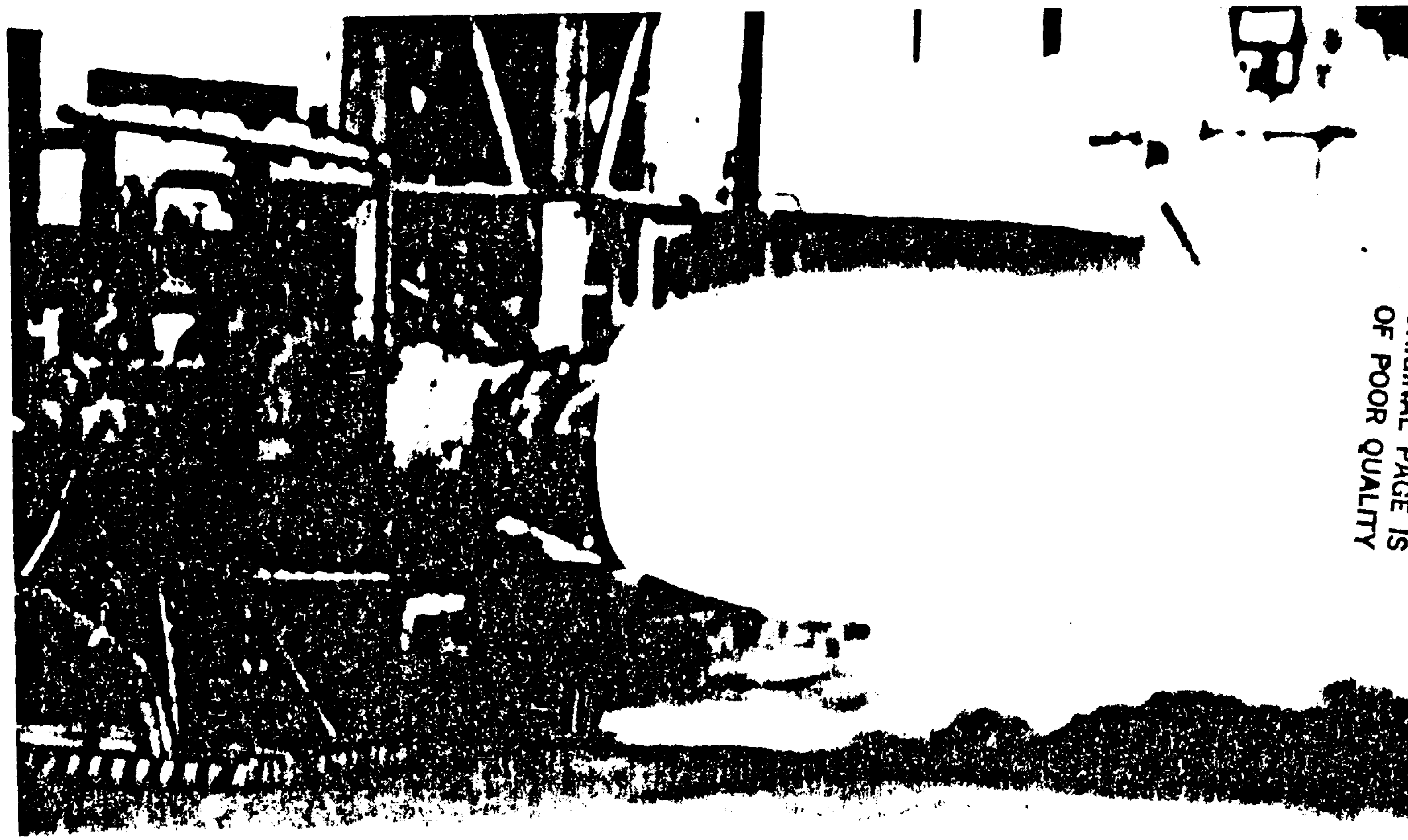

Figure 2-3. 250,000-Pound Thrust Engine Firing at AFRPL

These tests conclusively demonstrated that scalable, dynamically stable engines could be manufactured using commercial fabricating techniques and procedures at costs heretofore unobtainable for high thrust rocket engines. In addition, the experimental test program proved the feasibility of good engine performance and extended service life ablative liners with minimal development time and resources required.

In addition to testing with storable propellants, this family of engines was also tested with LOX/RP-1 and LOX/propane. In 1971-72 tests were conducted at thrust levels of 2,000 and 50,000 lb_f using these cryogenic propellants. Performance results very similar to those for storable propellants were observed and the feasibility of using this engine design with storable or cryogenic propellants was established.

TRW also participated in several funded pressure-fed launch vehicle and strap-on booster studies for NASA and DOD in the late 1960's to early 1970's. A major study was conducted for NASA in 1968-69 as part of the National Space Booster Study. In this study TRW generated a great amount of parametric data on pressure fed engines, pressurization systems, thrust vector control systems, tankage materials, etc. Complete three-stage launch vehicles were designed and evaluated. Much of this data is still useful for current studies.

In this time period TRW developed and built a bipropellant gas generator heated steam (GGHS) launch vehicle tank pressurization system for full scale testing. Figure 2-4 shows a TRW 250,000 lb_f thrust mock-up engine mounted on the Boeing-X stage including mounted GGHS system hardware. Tank expulsion tests were successfully completed using referee fluids with pressurization from a TRW-supplied bipropellant/water gas generator. Project funding was terminated prior to full-up hot-fire expulsion testing.

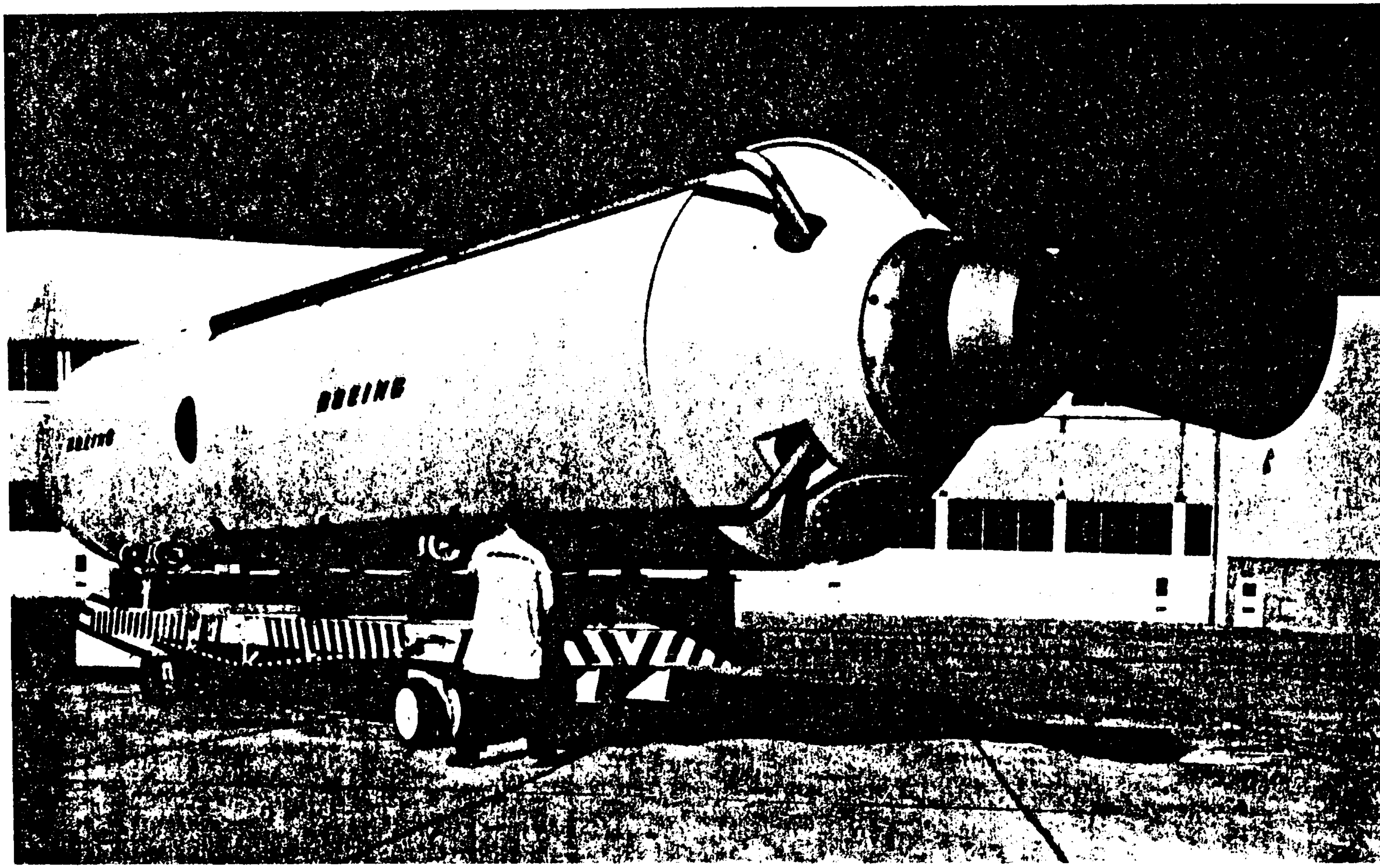

Figure 2-4. TRW 250,000 lb_f Thrust Engine Mock-Up Mounted on Boeing-X Stage with Mounted GGHS Hardware

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In 1971-72 TRW conducted a study for NASA/MSFC to design a pressure-fed engine using LOX/RP-1 and LOX/propane for a water recoverable Space Shuttle booster. This engine, shown in Figure 2-5, has a sea level thrust of 1.2×10^6 lb_f. Also in 1971, TRW completed a series of sled engine tests at Holloman Air Force Base with our storable engines in the thrust range of 35,000 to 50,000 lb_f.

In support of engine and launch vehicle studies in this time period, TRW conducted thrust vector control (TVC) studies. For most vehicles liquid injection (LITVC) and thrust modulation (for clustered engine boosters) were selected in trades over gimballing the engines for thrust vector control for minimum cost and weight considerations.

The early 1970's studies culminated in a major NASA review at MSFC in 1972 to evaluate this simple engine design with its inherent high reliability. Pressure-fed liquid rocket boosters (PFLRB) were considered serious contenders for Shuttle (STS) strap-ons. NASA's decision to baseline solid rocket boosters for STS in 1972 ended all TRW PFLRB work for that period.

In 1981 TRW performed a funded study to update the late 1960's pressure-fed launch vehicle designs for use as a backup for the Shuttle. This was to be a three-stage expendable unmanned launch vehicle with the same payload performance. The study concluded that the basic design concept is still valid.

Recent TRW launch vehicle studies include the following:

- 1985-86, Space Transportation Architecture Study - an Air Force study contract to develop architectural level recommendations for methods and investments to reduce space transportation costs. Pressure-fed designs were compared with other launch vehicle concepts.
- September 1986, Space Based Laser Launch Vehicle - preliminary design of a heavy lift launch vehicle with PFLRB strap-ons for large laser payloads into highly inclined orbits.

TRW

Low Cost Pressure-Fed Thruster

Sizes: 10K to 2M Pound Thrust

Simple valves, explosively actuated burst diaphragms to open, propellant plug valves to close

Single element coaxial injector

- **Scalable propellant mixing**
- **Dynamically stable combustion through central injection location**

Rolled and welded steel chamber and nozzle

- **Rubber insulated chamber and nozzle**
- **Ablative throat**

Figure 2-5. TRW Low Cost Engine Concept

3. PROPELLANT SELECTION CONSIDERATIONS

Candidate propellants for the STS LRB can be loosely grouped into cryogenic (including cryogenic/hydrocarbons), earth storables and gels. The pros and cons of each category are listed in the facing charts. Gels have longer term promise of greatly reducing the required booster size because of the very high density specific impulses that they provide. They also promise to greatly increase safety. In the near term, however, there are a number of issues that need to be resolved before they should be selected for the STS LRB. These issues are primarily tankage and transfer issues.

Propellant Characteristics – Pros and Cons

TRW

ITEM	CRYOGENIC	EARTH STORABLE	GELS
TANKS	• MORE COMPLEX • FRACTURE MECHANICS DRIVEN DESIGN • NARROW MATERIAL SELECTION • LARGER VOLUME	• SIMPLE TANK • WIDE MATERIAL SELECTION • SMALLER VOLUME	• UNRESOLVED ISSUES
ENVIRONMENTAL	• LOWER IMPACT	• HIGHER IMPACT	• BETTER THAN EARTH STORABLES
COST	• LESS	• GREATER	• MORE EXPENSIVE THAN STORABLES
AVAILABILITY	• ABUNDANT	• NEED NEW SUPPLY PLANTS	• NEED NEW SUPPLY PLANTS
PERFORMANCE	• HIGHER	• LOWER	• LOWER THAN STORABLES
STS FACILITY COMPATIBILITY	• COMPATIBLE	• NEED NEW PROPEL-LANT HANDLING EQUIPMENT	• NEED NEW PROPEL-LANT HANDLING EQUIPMENT
PRESSURIZATION SYSTEM	• MORE COMPLEX	• LESS COMPLEX	• LESS COMPLEX

Propellant Characteristics – Pros and Cons (Continued)

TRW

ITEM	CRYOGENIC	EARTH STORABLE	GELS
RELIABILITY	• LOWER THAN STORABLES	• HIGHEST	• MAY BE LOWER THAN STORABLES - REQUIRES MORE STUDY
ENGINE	• INJECTOR DESIGN IMPACT	• SIMPLER INJECTOR	• SAME AS STORABLES BUT REQUIRES FACE SHUT-OFF INJECTOR
STARTING	• REQUIRES STARTING SYSTEM	• NO STARTING SYSTEM REQUIRED (HYPERGOLIC)	• NO STARTING SYSTEM REQUIRED (HYPERGOLIC)
PROPELLANT DELIVERY SYSTEM	• MORE COMPLEX VALVES	• LESS COMPLEX	• SAME AS STORABLES
SAFETY	• WELL ESTABLISHED PROCEDURES • EXTENSIVE EXPERIENCE BASE	• MORE TOXIC • SPILLS MORE HAZARDOUS • NEW STS PROCEDURES REQUIRED	• BETTER THAN EITHER EITHER STORABLES OR CRYOGENIC PROPELLANTS
HARDWARE DEVELOPMENT RISK	• MODERATE	• LOW	• HIGH

The remainder of this section summarizes the physical properties of several liquid propellants and compares the performance of 12 bipropellant combinations at a nozzle area ratio $\epsilon = 6$ which is typical of a booster. Performance parameters for payload effectiveness and propellant volume to payload ratio are developed and used to compare propellants.

Approach

The relative effectiveness of various candidate propellants can be judged by developing a first-order figure of merit that incorporates the adverse effects of engine, propellant tank, pressurant and pressurant tank masses, expressed in a way that shows the influence of pressure, thrust level and propellant mass. A payload fraction parameter is defined as the ratio of payload plus all stage inert mass other than engine, tank, pressurant and pressurant tank masses, to the total initial vehicle mass:

$$PLF = \frac{M_{PL} + M_I}{M_o} \tag{1}$$

Since

$$M_o = M_{PL} + M_I + (M_{pp} + M_T) + (M_{PR} + M_{PT}) + (M_E + M_{SK})$$

the payload fraction can be expressed as

$$PLF = 1 - \frac{M_{PP}}{M_o} - \left(\frac{M_T + M_{PR} + M_{PT} + M_E + M_{SK}}{M_o}\right) \quad (2)$$

It is convenient to modify this, using the rocket equation

$$\frac{M_{PP}}{M_o} = 1 - e^{-\frac{\Delta V}{gI_{sp}}} \quad (3)$$

to give

$$PLF = e^{-\frac{\Delta V}{gI_{sp}}} - \left(1 - e^{-\frac{\Delta V}{gI_{sp}}}\right)\left(\frac{M_T}{M_{PP}} + \frac{M_{PR}}{M_{PP}} + \frac{M_{PT}}{M_{PP}} + \frac{M_E}{M_{PP}} + \frac{M_{SK}}{M_{PP}}\right) \quad (4)$$

where subscript T is propellant tank, PP is propellant, PR is pressurant, PT is pressurant tank, E is engine and SK is nozzle extension (beyond $\epsilon = 6$).

Propellant Tank

The tank mass ratio for a given geometry, depends only on the ratio of tank to propellant density multiplied by the ratio of tank pressure to wall stress. This is illustrated by the spherical tank for which

$$\frac{M_T}{M_o} = \frac{3}{2} \left(\frac{vol}{M_o}\right) \left(\frac{P\rho}{\sigma}\right)_T = \frac{3}{2} \frac{M_{PP}}{M} \left(\frac{\rho_T}{\rho}\right) \left(\frac{P}{\sigma}\right)_T$$

For a general geometry, the coefficient 3/2 is replaced by an empirical constant K_T. From recent LRB design studies, this constant is 15 and

$$\frac{M_T}{M_{PP}} = 15 \left(\frac{\rho_T}{\rho}\right) \left(\frac{P}{\sigma}\right)_T \tag{5}$$

<u>Pressurization System</u>

Pressurant mass for unheated helium is

$$\frac{M_{PR}}{M_{PP}} = \frac{P_T/RT_{PR}}{\rho} \left(\frac{1.66}{1 - \frac{1}{\lambda} \frac{Z_i}{Z_f}}\right) \tag{6}$$

where λ is blowdown pressure ratio, T_{PR} is initial helium temperature and Z_i and Z_f are initial and final helium compressibility factors. A spherical pressurant tank is assumed, so that

$$M_{PT} = \frac{3}{2} \left(\frac{M_{PR}}{\rho_{PR}}\right) (P_{PR}) \left(\frac{\rho}{\sigma}\right)_{PT} = \frac{3}{2} M_{PR} Z_i RT_{PR} \left(\frac{\rho}{\sigma}\right)_{PT}$$

By increasing the coefficient to 4, this relation is brought into agreement with recent LRB helium tank designs. Then

$$\frac{M_{PT}}{M_{PP}} = 4\ (RT)_{PR} \left(\frac{\rho}{\sigma}\right)_{PT} \frac{M_{PR}}{M_{PP}} \tag{7}$$

Engine

A similar approach is used for engine mass. It is represented as the mass of a cylinder closed at both ends, with an empirical coefficient based on a particular design.

The thickness is assumed uniform and given by

$$\tau = \left(\frac{PD}{2\sigma}\right)_c$$

giving a wall mass

$$M_e = k\pi \left(\frac{L}{D} + \frac{1}{2}\right) \left(\frac{\rho D}{\sigma}\right)_c \left(PD^2\right)$$

where L and D are combustor length and diameter, and ρ and σ are wall density and design stress.

The chamber diameter is eliminated by using the thrust relation,

$$F = \dot{m} g I_{SP} = \rho\ AV\ (gI_{sp}) = \frac{\pi}{4} \left(\frac{PD^2}{RT}\right) \left(M \sqrt{\gamma RT}\ \right)_c gI_{sp}$$

from which

$$\left(PD^2\right)_c = \frac{4}{\pi} \frac{F/I_{sp}}{gM_c} \left(\sqrt{\frac{RT}{\gamma}}\right)_c \tag{8}$$

The chamber mass is then

$$M_E = \frac{k8/\sqrt{\pi}\left(\frac{L}{D} + \frac{1}{2}\right)}{\left(M_c g\right)^{3/2}} \left(\frac{\rho}{\sigma}\right)_c \left(\frac{RT}{\gamma}\right)_c^{3/4} \frac{\left(F/I_{sp}\right)^{3/2}}{\sqrt{P_c}} \tag{9}$$

Differences in chamber L/D and Mach number for different engines are assumed to be absorbed in the coefficient in Equation (9) calculated for the following engine design:

$F = 2.49 \times 10^6$ N (M_{pp} = 323,000 kg)
I_{sp} = 310 (ideal equilibrium, $\epsilon = 6$)
P_c = 20.4 atm
ρ_c = 8,000 kg/m^3
σ_c = 3,000 atm
$(RT/\gamma)^{3/4}$ = 33,100
M = 1,680 kg

Using these values, the coefficient in Equation (9) becomes 3.8, and the engine mass ratio is

$$\frac{M_E}{M_{PP}} = 3.8 \left(\frac{\rho}{\sigma}\right)_c \left(\frac{RT}{\gamma}\right)_t^{3/4} \frac{\left(F/I_{sp}\right)^{3/2}}{M_{PP}\sqrt{P_c}} \tag{10}$$

where the coefficient 3.8 is in SI units.

Nozzle Skirt

The skirt mass, for area ratios larger than 6, is estimated by approximating the skirt with a 20^0 half-angle cone. The nozzle exit diameter at $\epsilon = 6$ is assumed to be 1.25 times the chamber diameter. Then for any larger diameter.

$$\frac{D}{D_c} = 1.25 + 2\,\frac{L}{D_c}\,\mathrm{Tan}\,20^0 = \frac{D}{D_*}\,\frac{D_*}{D_o}\,\frac{D_o}{D_c} = 1.25\,\sqrt{\frac{\epsilon}{\epsilon_o}}$$

so that

$$\frac{\epsilon}{\epsilon_o} = \left(1 + 0.58\,\frac{L}{D_c}\right)^2 = 0.64\,\left(\frac{D}{D_c}\right)^2 \qquad (11)$$

With ϵ determined for any skirt length L, the Mach number and pressure ratio P/P_c are known. Then the local skirt thickness τ_{sk} can be calculated from the stress relation.

$$\tau_{sk} = \left(\frac{PD}{2\sigma}\right)_{sk}$$

so that the local differential mass is

$$dM_{sk} = \frac{\pi}{2}\,P_c D_c^3\,\left(\frac{\rho}{\sigma}\right)_{sk}\,\frac{P}{P_c}\,\left(\frac{D}{D_c}\right)^2 d\,\left(\frac{L}{D_c}\right)$$

The dimensionless skirt mass for any length L/D_c is then

$$\frac{M_{sk}}{\frac{\pi}{2}\,D_c^3\,\rho_{sk}\,(\rho_c/\sigma_{sk})} = \int_0^{\frac{L}{D_c}} \frac{P}{P_c}\,\left(\frac{D}{D_c}\right)^2 d\,\left(\frac{L}{D_c}\right) = f\,\left(\frac{\epsilon}{\epsilon_o}\right) \qquad (12)$$

This is integrated numerically, using Equation (11) for ϵ, which determines Mach number, and consequently P/P_C. The integrand is plotted in Figures 3-1 vs. L/D_C and the integral is shown in Figure 3-2 with ϵ/ϵ_o as independent variable

The skirt mass can then be expressed as

$$M_{sk} = \frac{\pi}{2} \left(\frac{\rho}{\sigma}\right)_{sk} P_c D_c^3 \, f \left(\frac{\epsilon}{\epsilon_o}\right)$$

or, with Equation (8), and the factor k in Equation (9)

$$M_{sk} = \frac{4k}{\sqrt{\pi}} \left(\frac{\rho}{\sigma}\right)_{sk} \frac{(F/I_{sp})^{3/2}}{\sqrt{P_c}} \left(\frac{RT}{\gamma}\right)_c^{3/4} \frac{f(\epsilon/\epsilon_o)}{(gM_c)^{3/2}}$$

and with Equation (11), and assuming $gM_C = 6$,

$$\frac{M_{sk}}{M_E} = 0.15k \frac{(\rho/\sigma)_{sk}}{(\rho/\sigma)_c} f(\epsilon/\epsilon_o)$$

A comparison of Equations (9) and (10) shows the constant k to be 6.2. This experience factor is applied to the skirt mass equation to give

$$\frac{M_{sk}}{M_\epsilon} = 0.93 \frac{(\rho/\sigma)_{sk}}{(\rho/\sigma)_c} f(\epsilon/\epsilon_o) \tag{13}$$

Figure 3-1. Dimensionless Nozzle Skirt Mass Versus Skirt Length Ration

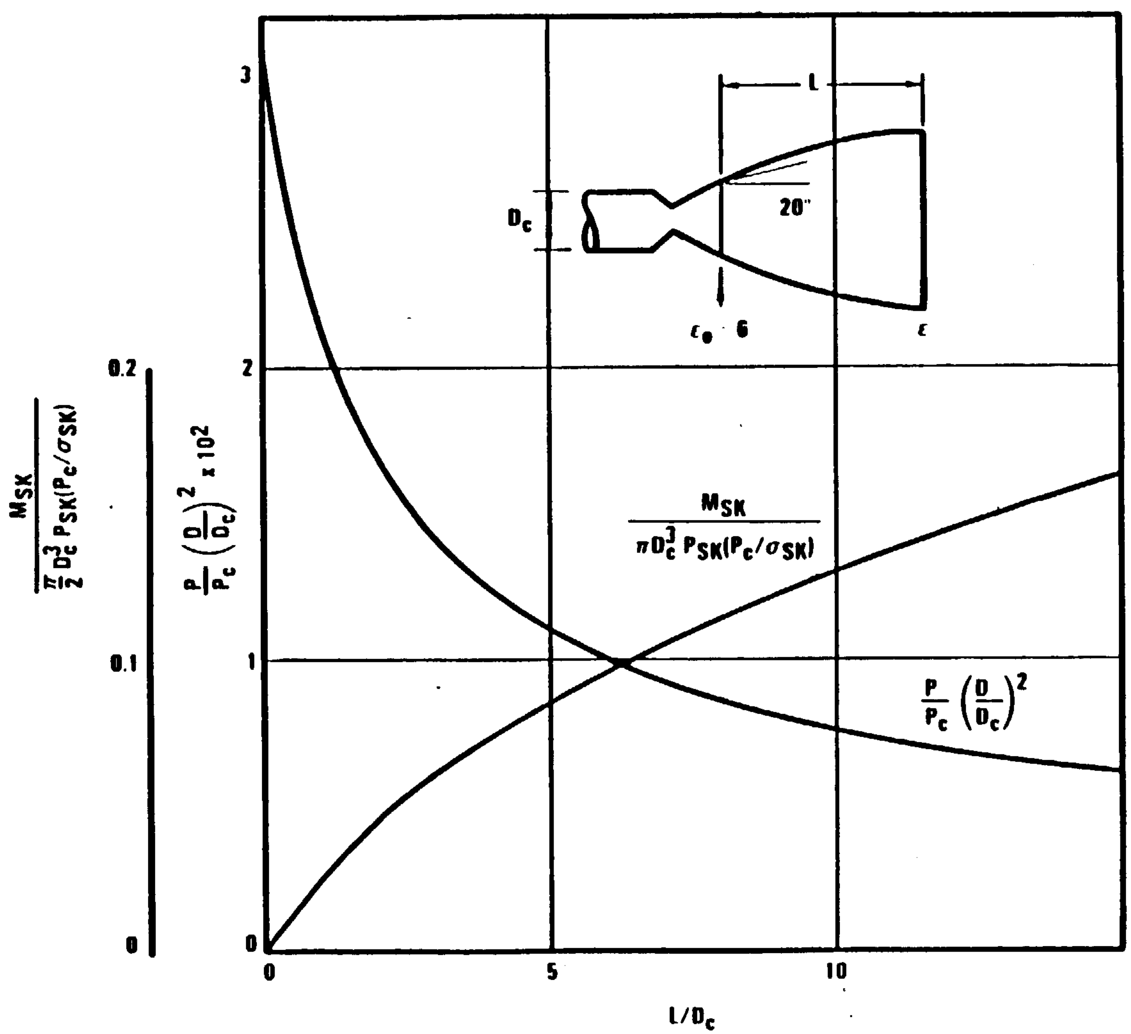
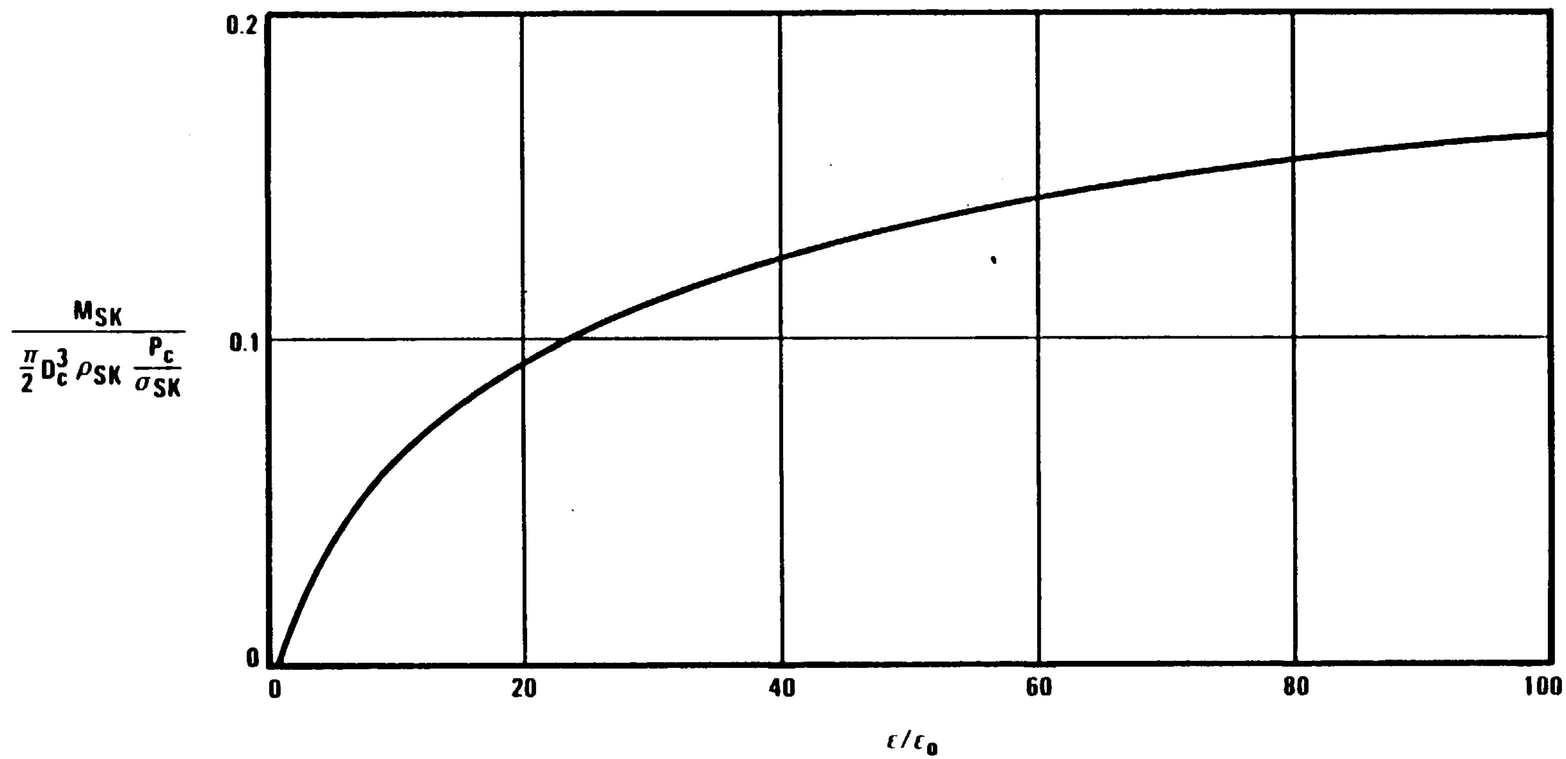


Figure 3-2. Nozzle Skirt Mass Versus $\varepsilon/\varepsilon_0$
($\varepsilon_0 = 6$)



Complete Stage

Equations (5), (6), (7), (10) and (13) can now be used in Equation (4) to give the payload fraction:

$$PLF = e^{-\frac{\Delta V}{gI_{sp}}} - \left[1 - e^{-\frac{\Delta V}{gI_{sp}}}\right] \left[15 \frac{\rho_T}{\rho} \left(\frac{P}{\sigma}\right)_t + \frac{P_T/RT_{PR}}{\rho} \frac{1.66}{\left(1 - \frac{Z_i/Z_f}{\lambda}\right)} \left(1 + 4(RT)_{PR} \left(\frac{\rho}{\sigma}\right)_{PT}\right.\right. \tag{14}$$

$$\left.\left. + 3.8 \left(\frac{\rho}{\sigma}\right)_c \left(\frac{RT}{\gamma}\right)^{3/4} \frac{(F/I_{sp})^{3/2}}{M_{PP}\sqrt{P_c}} \left[1 + 0.93 \frac{(\rho/\sigma)_{sk}}{(\rho/\sigma)_c} f\,(\epsilon/\epsilon_o)\right]\right)\right]$$

I_{sp} is the only propellant parameter in the first term. This indicates the ideal payload that would be delivered at the velocity increment ΔV if payload and propellant were the only masses. The propellant density ρ (in addition to I_{sp}) appears in the propellant and pressurization terms but not explicitly in the engine and skirt terms, although the propellant is involved there also through the combustion product density.

Propellant tank and pressurization masses scale directly with combustor pressure while those due to the engine and nozzle vary inversely with the square root of pressure. The engine mass increases with the 3/2 power of thrust and varies inversely with propellant total mass.

<u>Propellant Performance Comparison</u>

Eleven liquid bipropellants have been examined using the evaluation approach described above. The physical characteristics of the oxidizers and fuels used in these propellants are listed in Table 3-1. Combustor pressures of 300, 700 and 1,500 psia were used with a propellant tank pressure 50 psi greater than the combustor pressure for a typical booster with ϵ = 6 and ΔV = 10,000 ft/sec.

Table 3-2 lists average propellant density $\bar{\rho}$, specific heat ratio γ, molecular weight w and temperature in the combustor together with the O/F ratio for maximum I_{sp} and the value of that maximum for the ϵ = 6 case. These calculations were made assuming ideal, one-dimensional, equilibrium nozzle flow for 300, 700 and 1,500 psia pressure. Table 3-3 lists the combined mass fraction for the propellant tanks, pressurant and pressurant tank under the heading MF for the tank. Similarly MF for the engine combines tank and nozzle masses. The resulting payload parameter PLF is also shown with another parameter which is the ratio of propellant volume to the sum of payload plus other inert mass. This ratio is

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Table 3-1. Physical Characteristics of Fuels and Oxidizers

	DENSITY AT T&P			ONE ATMOSPHERE		CRITICAL POINT			
PROPELLANT	LB/FT^3	T (F)	P (psia)	BOILING POINT (F)	FREEZING POINT (F)	TEMP (F)	PRESSURE (psia)	CRYOGENIC	TOXICITY (OF LIQUIDS)
ClF_5	111	76	50	7	-153	289	770	NO	VERY HIGH
ClF_3	98	76	25	53	-105	345	838	NO	VERY HIGH
N_2F_4	94	-149	400	-99	-261	97	1132	MILD	HIGH
NF_3	84	-150	115	-200	-340	-39	657	MILD	HIGH
N_2O_4	90	70	15	70	12	316	1470	NO	HIGH
LOX	71	-297	15	-297	-362	-182	730	DEEP	NONE
N_2H_4	62	76	15	236	35	716	2131	NO	HIGH
MMH	55	76	15	189	-63	609	1195	NO	HIGH
RP-1	48	76	15	422	-50	758	315	NO	MILD
ETHANOL	50	76	15	173	-174	469	925	NO	VERY MILD
PROPANE	31	76	136	-44	-306	206	617	NO	NONE
METHANE	26	-259	15	-259	-297	-116	673	DEEP	NONE
NH_3	42	-28	15	-28	-108	270	1640	MILD	MILD
H_2	44	-423	15	-423	-435	-400	188	DEEP	NONE

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Table 3-2. Propellant Performance Characteristics, $\varepsilon = 6$

(Equilibrium one-dimensional flow)

		P_c 300 psia						P_c 700 psia						P_c 1500 psia				
PROPELLANT	O/F	$\bar{\rho}$	Isp	γ	w	T_c (k)	O/F	$\bar{\rho}$	Isp	γ	w	T_c (k)	O/F	$\bar{\rho}$	Isp	γ	w	T_c (k)
LOX/H_2	6.61	23.8	389	1.202	14.1	3404	6.61	23.8	391	1.198	14.3	3515	6.61	23.8	394	1.195	14.4	3613
N_2F_4/N_2H_4	3.10	66.9	348	1.306	20.9	4332	3.18	67.0	350	1.304	21.1	4544	3.25	67.0	352	1.306	21.4	4555
NF_3/N_2H_4	2.95	84.4	338	1.303	21.1	4113	2.95	84.4	340	1.300	21.3	4240	2.95	84.4	341	1.297	21.4	4351
ClF_5/N_2H_4*	2.59	91.5	326	1.311	21.7	3964	2.59	91.5	328	1.307	21.9	4094	2.71	92.1	330	1.304	22.3	4245
LOX/MMH	1.28	62.7	323	1.222	19.9	3369	1.28	62.7	324	1.220	20.1	3469	1.28	62.7	326	1.217	20.2	3555
LOX/METHANE	2.81	49.4	320	1.210	19.6	3317	2.95	49.9	322	1.207	20.2	3432	3.07	50.4	323	1.202	20.8	3544
LOX/PROPANE	2.59	52.4	315	1.217	21.2	3411	2.59	52.4	317	1.214	21.4	3516	2.69	52.8	318	1.209	22.0	3640
LOX/ETHANOL	1.67	61.2	307	1.193	21.2	3179	1.67	61.2	309	1.190	21.4	3267	1.67	61.2	310	1.187	21.6	3344
LOX/RP-1	2.43	62.5	310	1.222	22.5	3480	2.43	62.5	312	1.218	22.7	3595	2.43	62.5	314	1.215	23.0	3697
ClF_3/N_2H_4*	2.62	62.9	307	1.307	22.7	3705	2.62	62.9	308	1.302	22.9	3819	2.75	63.7	309	1.301	23.3	3949
LOX/NH_3	1.41	55.2	305	1.204	19.7	2988	1.41	55.2	306	1.202	19.8	3049	1.41	55.2	307	1.200	19.9	3100
N_2O_4/MMH*	1.92	73.6	300	1.227	21.7	3231	2.08	74.3	301	1.222	22.5	3346	2.08	74.3	302	1.219	22.6	3422

*HYPERGOLIC

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Table 3-3. Propellant Characteristics, $\varepsilon = 6$, $\Delta V = 10{,}000$ ft/s

(Equilibrium one-dimensional flow)

	P_c = 300 psia					P_c = 700 psia					P_c = 1500 psia				
PROPELLANT	MF TANK	MF ENGINE	$e^{-\Delta V / g I_{sp}}$	PLF	PPVOL/PL $\frac{FT^3}{LB}$	MF TANK	MF ENGINE	$e^{-\Delta V / g I_{sp}}$	PLF	PPVOL/PL $\frac{FT^3}{LB}$	MF TANK	MF ENGINE	$e^{-\Delta V / g I_{sp}}$	PLF	PPVOL/PL $\frac{FT^3}{LB}$
LOX/H_2	0.074	0.005	0.450	0.409	0.0565	0.158	0.003	0.452	0.365	0.0631	0.326	0.002	0.455	0.277	0.0827
N_2F_4/N_2H_4	0.026	0.005	0.410	0.395	0.0223	0.056	0.003	0.412	0.379	0.0232	0.116	0.002	0.414	0.346	0.0253
NF_3/N_2H_4	0.021	0.005	0.399	0.386	0.0184	0.044	0.003	0.401	0.375	0.0189	0.092	0.002	0.402	0.347	0.0204
ClF_5/N_2H_4*	0.019	0.005	0.386	0.374	0.0178	0.041	0.003	0.388	0.363	0.0184	0.084	0.002	0.390	0.339	0.0195
LOX/MMH	0.028	0.005	0.382	0.365	0.0270	0.060	0.003	0.383	0.346	0.0284	0.124	0.002	0.386	0.310	0.0316
LOX/METHANE	0.035	0.005	0.379	0.357	0.0352	0.075	0.003	0.381	0.335	0.0370	0.149	0.002	0.382	0.290	0.0423
LOX/PROPANE	0.034	0.005	0.373	0.352	0.0361	0.072	0.003	0.375	0.330	0.0361	0.154	0.002	0.377	0.281	0.0420
LOX/ETHANOL	0.029	0.005	0.364	0.346	0.0300	0.061	0.003	0.368	0.327	0.0317	0.127	0.002	0.367	0.287	0.0360
LOX/RP-1	0.028	0.005	0.367	0.349	0.0290	0.060	0.003	0.370	0.332	0.0304	0.124	0.002	0.372	0.284	0.0354
ClF_3/N_2H_4*	0.028	0.005	0.364	0.346	0.0292	0.060	0.003	0.365	0.327	0.0309	0.122	0.002	0.366	0.289	0.0344
LOX/NH_3	0.032	0.005	0.361	0.341	0.0339	0.068	0.003	0.362	0.319	0.0366	0.141	0.002	0.364	0.274	0.0421
N_2O_4/MMH*	0.023	0.005	0.355	0.340	0.0258	0.050	0.003	0.356	0.324	0.0268	0.104	0.002	0.358	0.291	0.0297

*HYPERGOLIC

$$\frac{PPVol}{PL} = \frac{M_{PP}/M_o}{\rho(M_{PL} + M_I)/M_o} = \frac{1 - e^{-\frac{\Delta V}{gI_{sp}}}}{\rho(PLF)} \quad (15)$$

The PLFs at P_C = 300 psia are plotted against I_{sp} in Figure 3-3. For all propellants except LOX/H_2, PLF correlates very well against I_{sp} for propellant densities ranging roughly from 50 to 90 lb/ft^3. Figure 3-4 shows the same results for P_C = 1,500 psia, although the scatter is greater at the higher pressure.

While the performance payload parameter does correlate against I_{sp}, scatter due to density differences can be seen in Figure 3-3 for P_C = 300 psia and more strongly in Figure 3-4 at P_C = 1,500 psia. Propellant density has a stronger influence on the PP VOL/PL ratio. In cases for which the booster is volume-constrained it is more useful to present the data in terms of payload fraction PLF as a function of the ratio of propellant volume to payload, as in Figures 3-5 and 3-6. This allows the trade-off between booster volume and payload fraction for various propellants to be visualized more easily.

Propellant Selection

Figures 3-5 and 3-6 show propellant performance for a booster having an area ratio ϵ = 6 and P_C = 300 and 700 psia. LOX/H_2 has the highest payload fraction but a very large volume-to-payload ratio which would not be acceptable for a strap-on booster where volume is critical. The three fluorinated oxidizers would not be used for a first stage booster where toxic products

Figure 3-3. Correlation of PLF with Isp
(P_c = 300 psia)

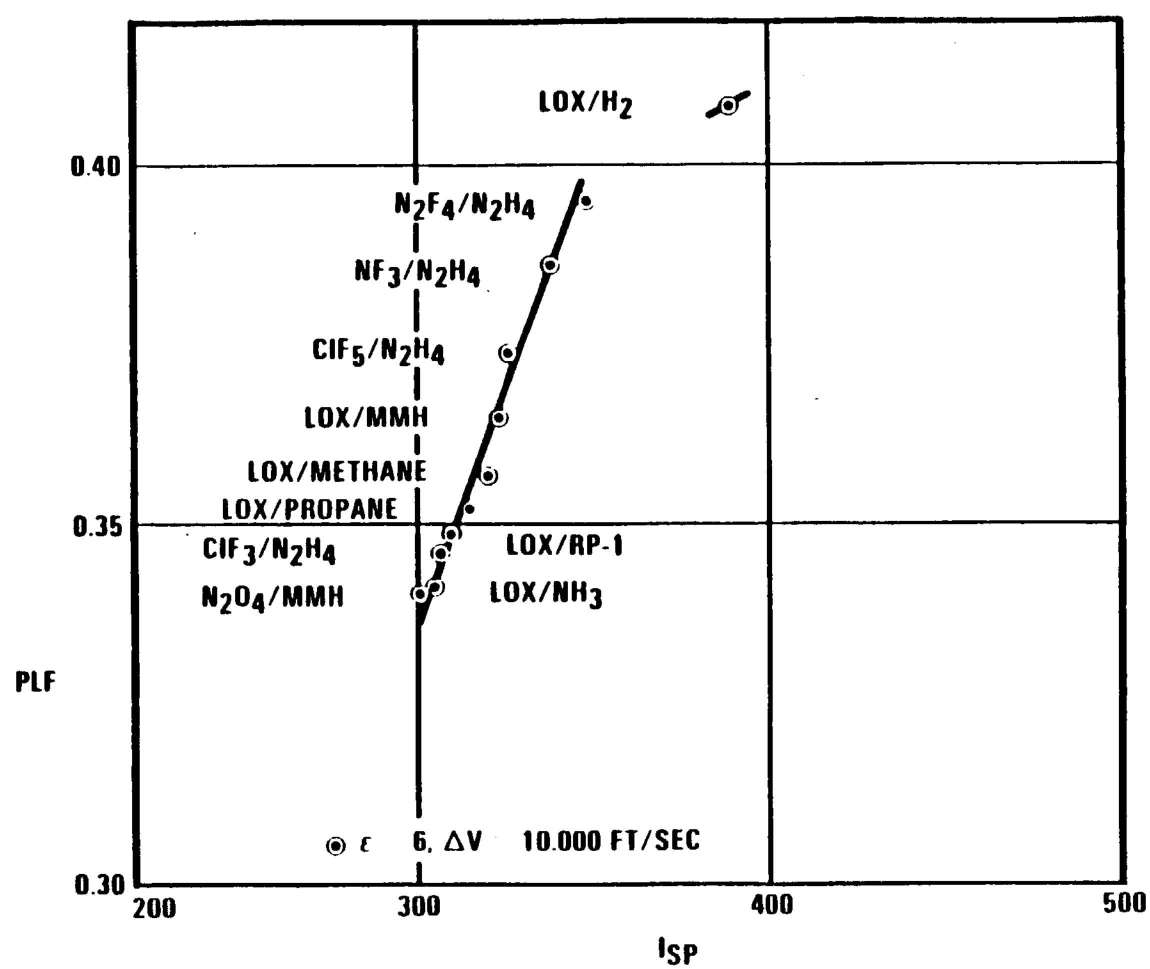

Figure 3-4. Correlation of PLF with Isp
(P_c = 1500 psia)

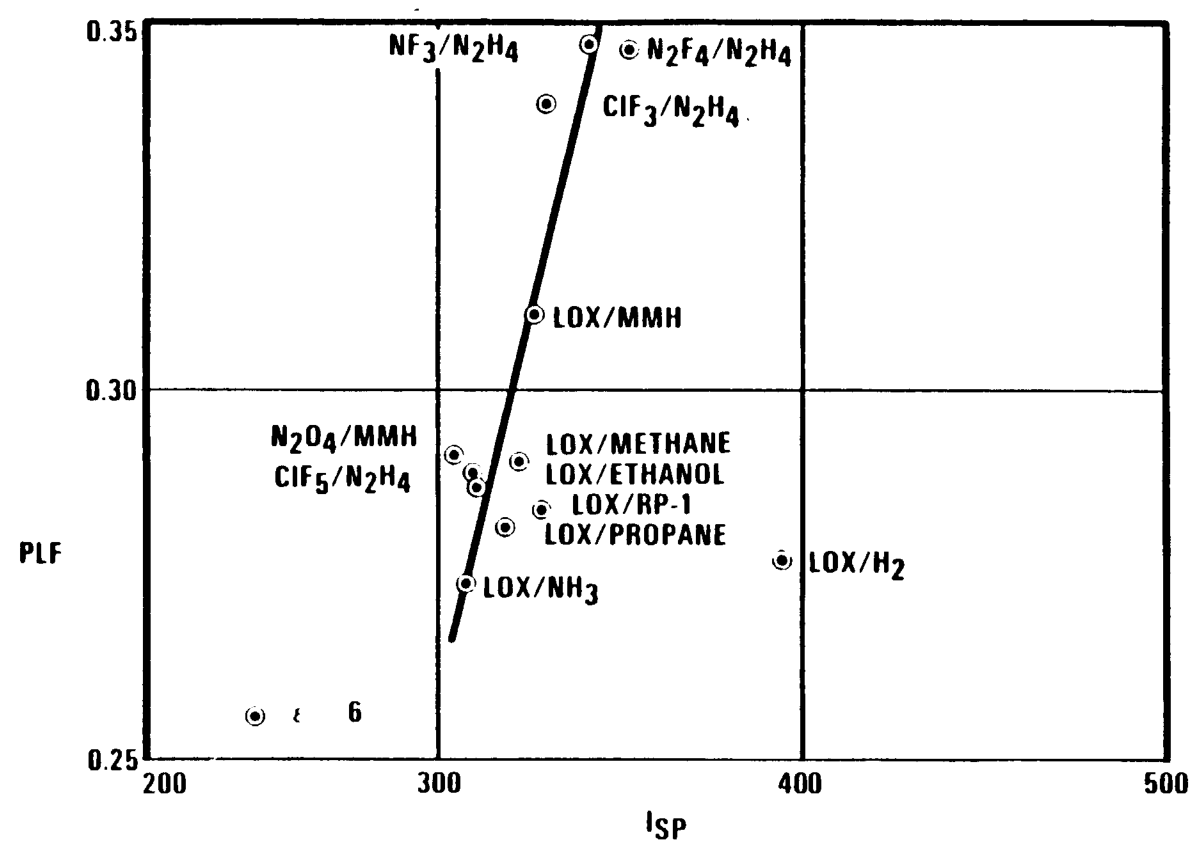

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Figure 3-5. Propellant Volume and Payload Parameters
$P_c = 300$ psia, $\varepsilon = 6$, $\Delta V = 10{,}000$ ft/s)

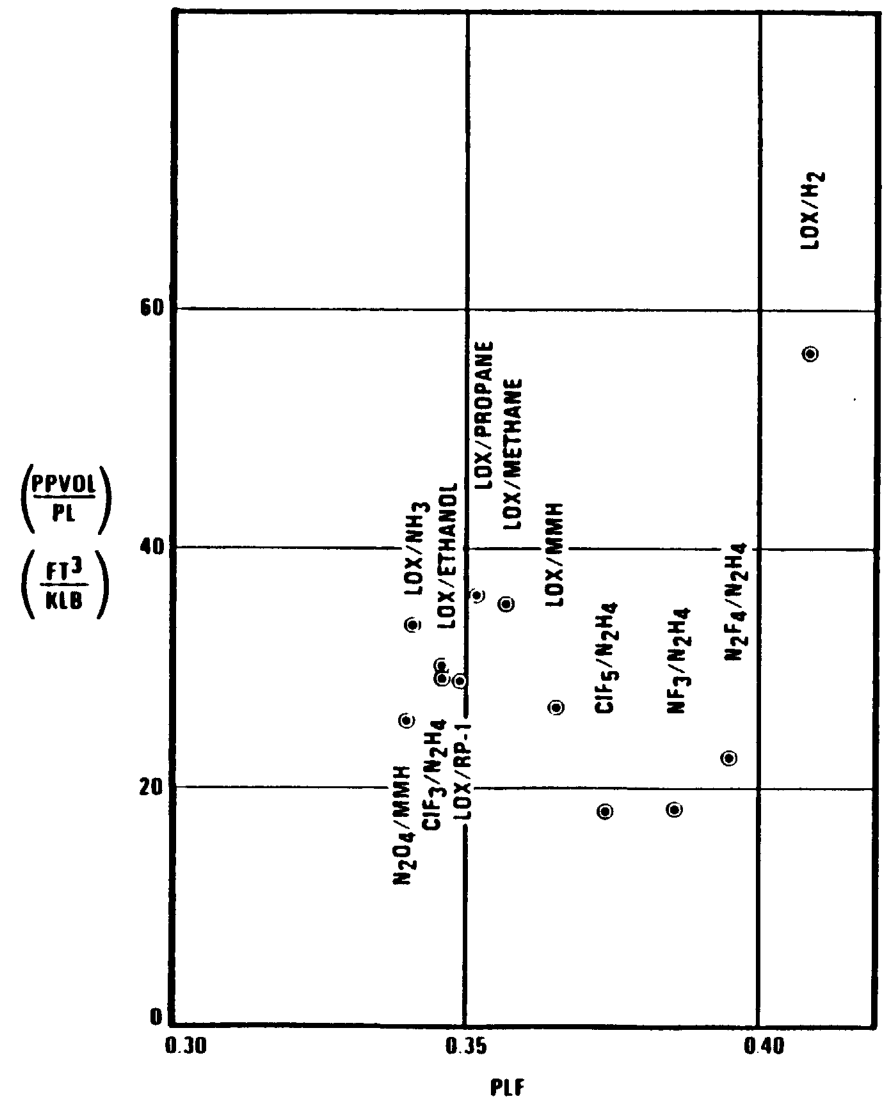
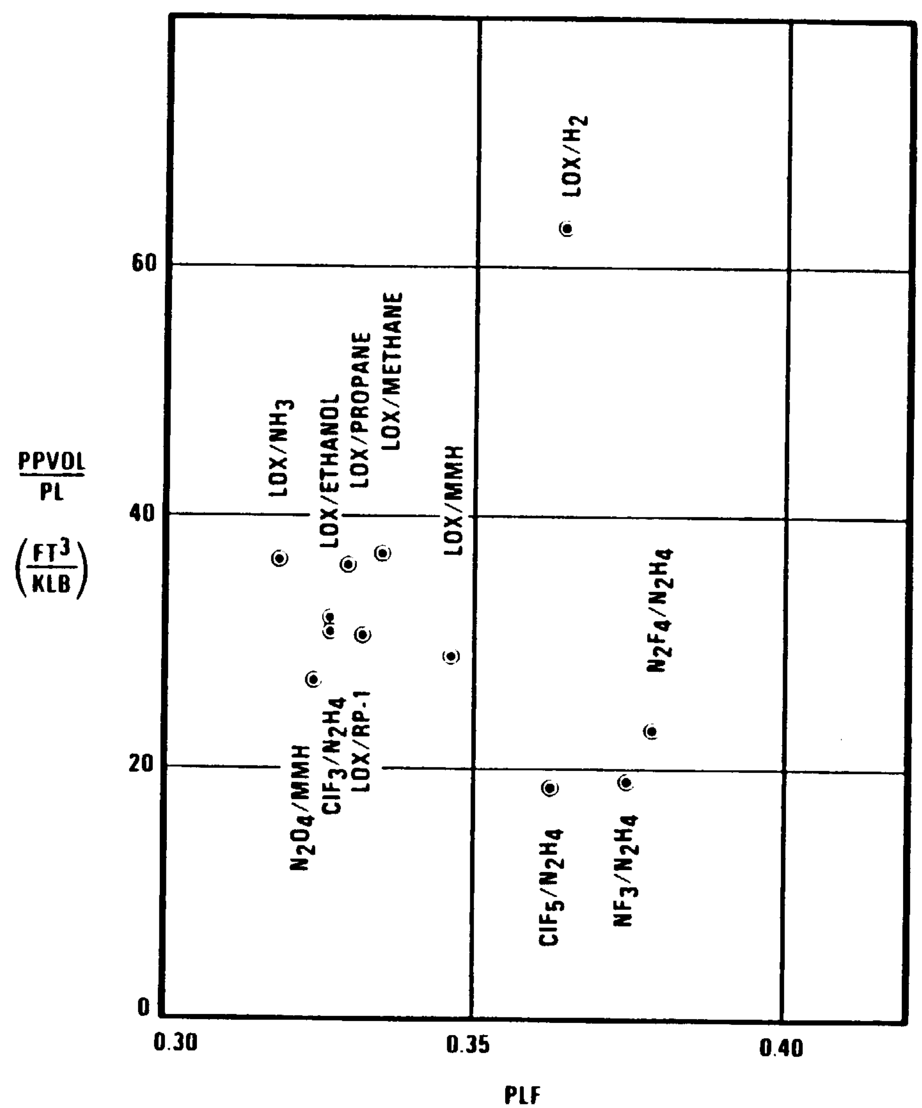


Figure 3-6. Propellant Volume and Payload Parameters
$P_C = 700$ psia, $\varepsilon = 6$, $\Delta V = 10{,}000$ ft/s)



such as HF and ClF would not be allowed. The next best performer is LOX/MMH. This has the drawback of a toxic fuel, which would increase operational hazards. A very large spill, although not hypergolic, would be disastrous if ignited. A mixture of the fuel and oxidizer forms a detonable gel that would decompose to MMH which would have to be cleaned up later.

If this hazard is not acceptable, the next most attractive candidate is LOX/methane, although its volume is considerably greater than that of LOX MMH. If the larger volume were clearly unacceptable, the next selection would be LOX/RP-1 which has a volume parameter only 7% larger than that of LOX/MMH. LOX/ethanol is also an interesting possibility. As seen in Table 2, its combustion temperature, at the O/F ratio for maximum I_{sp}, is 540°F lower than that of LOX/methane. LOX/NH_3 is 340°F cooler than LOX/ethanol, but it has more volume and less payload capability.

From these considerations, the preferred propellant for a volume-constrained strap-on booster taking into account environmental and safety concerns is LOX/RP-1. LOX/ethanol would be the next best choice. When the volume constraint can be relaxed enough, LOX/methane is the best selection.

4. COMBUSTION STABILITY OF TRW COAXIAL INJECTOR LIQUID PROPELLANT ROCKET ENGINES

Examination of the past United States major engine development results show that combustion instability problems have been encountered in over two-thirds of the major engine development. For STS LRB propulsion systems, the choice of an injector concept is the single most important design decision. The injection design directly affects the overall dynamic stability, performance and thrust chamber life. Minimization of engine development costs is dependent on the inherent combustion stability of the injector employed on the LRB engine.

Analytical studies made of the tangential and radial modes of instability (modes of greatest concern) indicate these modes will not be a problem with the coaxial pintle injector. Examination of the antinodes for various acoustic modes show the location of possible abnormally high energy release zones which could sustain combustion instability once initiated. Comparison of the responses between a centrally located injector and a distributed injector to a tangential disturbance shows that the zone of maximum rate of energy release in the distributed injector moves outward toward the chamber periphery, thus sustaining the instability. In the centrally located injector, the maximum energy release zone moves toward the chamber axis, or nodal location, and is therefore ineffective in sustaining the spinning or tangential wave front. Similar arguments can be offered in support of the stability of the centrally located injector to the first radial mode of combustion instability.

Major Engine Development History

Engine	Thrust Level	Instability
F1	1.5×10^6	Yes
H-1	188K	Yes
J-2	205K	Yes
Thor	150K	Yes
Titan II	215K	Yes
Titan II	100K	Yes
LMDE	10.5K	None
LM Ascent (Bell)	3.5K	Yes
Apollo SM	21.9K	Yes
Transtage	8K	Yes

Summary of Coaxial Injector, Pressure-Fed Engine Combustion Stability Experience

ALL EXISTING INFORMATION INDICATES THAT THE TRW COAXIAL INJECTOR IS INHERENTLY STABLE

EXPERIMENTAL DATA

- THRUST RANGE 25 TO 250,000 LB_F
- DYNAMIC TESTS
  - A) NON-DIRECTIONAL BOMBING
  - B) PULSE GUN
  - C) MAX ΔP ACHIEVED >200% P_C
  - D) ΔP >150% P_C IN ALL CASES
  - E) P_C RECOVERY TIMES ~10 TO 15 MS IN ALL CASES
- NOT A SINGLE CASE OF COMBUSTION INSTABILITY IN OVER 20,000 TESTS AND HUNDREDS OF THOUSANDS OF SECONDS OPERATION AT BOTH FULL THRUST AND DEEP THROTTLE (TO ~10%) CONDITIONS
- BOMB TESTS PERFORMED ON LOX/RP-1 (50K, O/F = 2.4, P_C = 250 PSIA) AND STORABLE PROPELLANTS (TO 250K)
- LOX C_3H_8 TESTED AT 2K WITH NO EVIDENCE OF INSTABILITY

THEORETICAL

- 1972 DYNAMIC SCIENCE ANALYSIS OF TRW 1500K, LOX/RP-1 ENGINE DESIGN

In 1969, tests were performed at AFRPL on a TRW-supplied 250K coaxial injector engine using NTO and UDMH propellants. The tests included parametric variations of injector and chamber geometry. As part of the tests, a series of combustion stability evaluations were performed. No instabilities were observed in any of the tests. Documentation of the tests is provided in Ref. 1.

250,000 lb_f Stability Rating Tests at AFRPL (NTO/UDMH)

NUMBER OF PULSE-GUN TESTS (RADIAL AND TANGENTIAL)	13
NUMBER OF NON-DIRECTIONAL BOMB TESTS	8
MAXIMUM PULSES PER TEST	3
MAXIMUM OVERPRESSURE	125%

NO HIGH FREQUENCY INSTABILITIES

TRW test experience with LOX/RP-1 was gained in a 1971 IR&D series of test firings at the 2K and 50K lb thrust levels. Data from the 2K tests are shown in this table. Both engines used fixed-gap TRW coaxial injectors designed for storable propellants and consequently were not optimized for LOX/RP-1. Nevertheless a combustion efficiency of 95.6% was achieved. Stable combustion was achieved in all cases.

Ignition of LOX/RP-1 was successfully accomplished using GF_2 in the 2K tests. A TEA (triethylaluminum) slug start system, in which a TEA cartridge was placed in the RP-1 delivery line between burst discs, was successfully demonstrated in LOX/propane tests in the 2K engine.

2K LOX/RP-1 Test Results

TRW

Test No. HB2	Fuel Gap (in)	Ignition System	Test Duration (sec)	P_{of} (psia)	P_{ft} (psia)	MR (O/F)	W_t (lb/sec)	ΔP_o (psi)	ΔP_f (psi)	T_{ol} (°F)	T_{fl} (°F)	P_{cns} (psia)	C* (fps)	nC* (%)
224	0.016	GF_2	1.0	765	865	2.80	6.844	122	42	-297	60	292	5445	94.3
225	0.016	GF_2	2.1	765	865	2.91	6.840	142	48	-299	60	295	5485	95.6
226	0.016	GF_2	2.2	820	720	3.30	6.894	152	41	-299	60	287	5305	95.0
227	0.016	GF_2	2.1	735	965	2.70	6.880	117	50	-300	60	295	5470	94.4

Legend

GF_2: Gaseous fluorine

P_{of}: Oxidizer tank pressure

P_{ft}: Fuel tank pressure

ΔP_o: Injector pressure drop (ox)

ΔP_f: Injector pressure drop (fuel)

T_{ol}: Oxidizer line temperature

T_{fl}: Fuel line temperature

P_{cns}: Nozzle stagnation chamber pressure

C*: Measure characteristic velocity

nC*: Combustion efficiency, percent of theoretical shifting equilibrium

TRW test experience with LOX/RP-1 at the 50K lb thrust level is shown in this table. The fixed-gap TRW coaxial injector was designed for storable propellants and consequently was not optimized for LOX/RP-1. As a result, significant improvement in performance over that shown is to be expected. Ignition of LOX/RP-1 was successfully accomplished using TEA (triethylaluminum) injection. Five bomb tests were performed in the 50K engine tests that demonstrated the inherent stability of the engine.

50K LOX/RP-1 Test Results

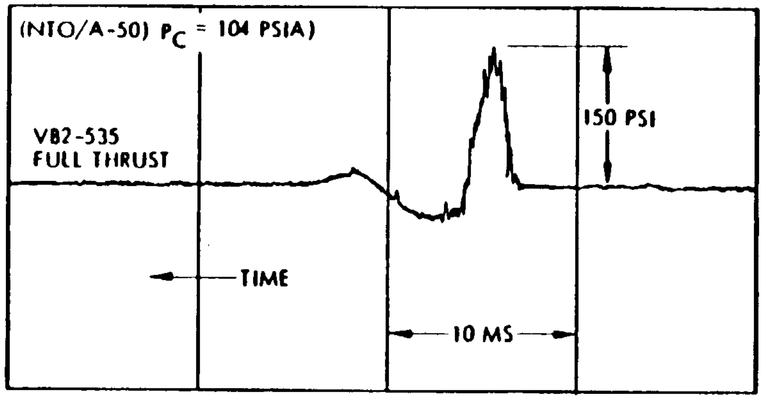
Test No. VB-1	Test Duration (sec)	P_c (psia)	W_f (lb/sec)	MR (O/F)	ΔP_o (psi)	ΔP_f (psi)	nC* (%)	Remarks
647	1.0	-	-	-	-	-	-	LOX/TEA ignition test only
648	1.0	-	-	-	-	-	-	LOX/TEA ignition test only
649	1.2	197	151.8	2.29	80	47	84	Data with RP-1 flowing through TEA systems
650	1.2	235	169.0	2.65	108	52	90.5	RP-1 not flowing through TEA system
651*	1.8	234	170.6	2.65	111	49	90.0	RP-1 not flowing through TEA system
652*	1.8	243	174.1	2.44	112	56	90.6	RP-1 not flowing through TEA system
653*	1.8	248	179.9	2.21	116	81	89.0	RP-1 not flowing through TEA system
654	1.8							RP-1 not flowing through TEA system
655*	1.8	231.5	172.7	2.36	102.8	63.8	87.0	RP-1 not flowing through TEA system

*Bomb tests.

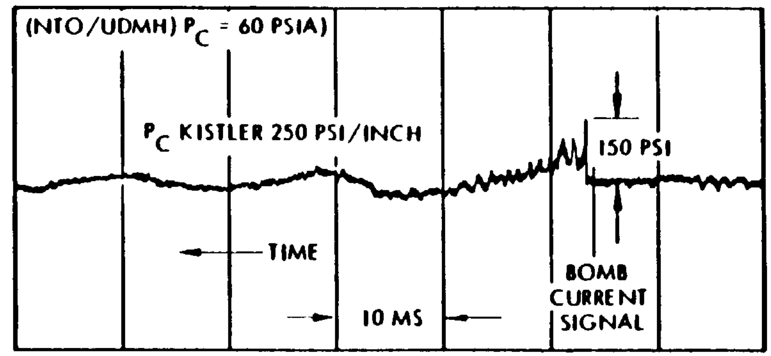
TRW has conducted stability evaluations on a number of engines at different thrust levels with a variety of propellants. Of particular interest is the 50K LOX/RP-1 bomb-test stability evaluations. In both the 50K engine stability tests and in firings of a 2K engine, no evidence of combustion instability was found with LOX/RP-1. Furthermore, TRW has never observed any combustion instabilities with any coaxial injector/propellant combination.

TRW Engine Stability Test Results

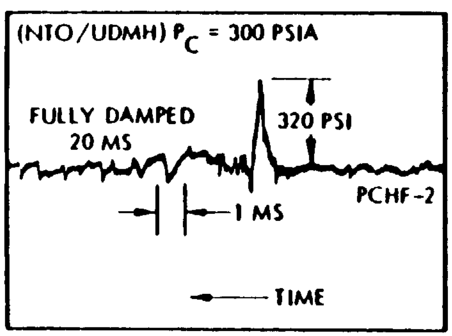
LMDE INJECTOR



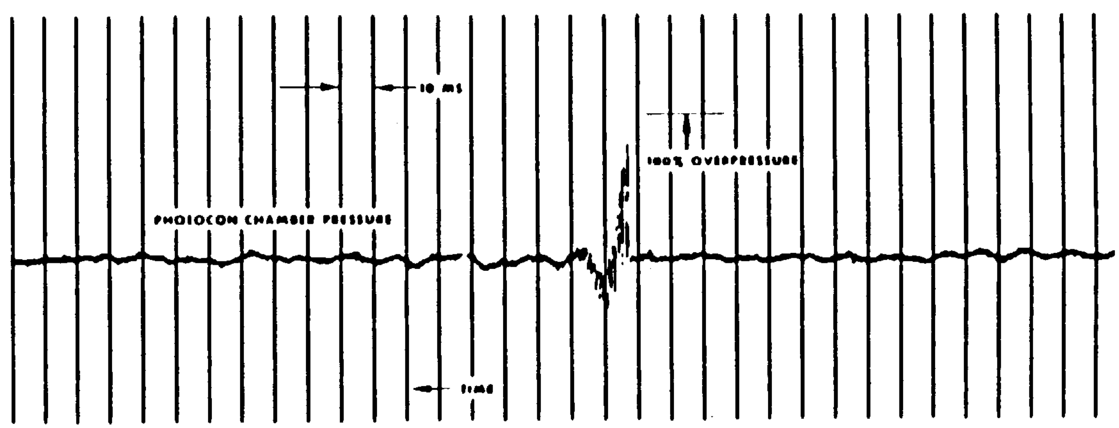
250K INJECTOR THROTTLED TO 50K



250K INJECTOR STABILITY TESTS



50K INJECTOR (LOX/RP-1 P_C, 250 PSIA)



Test data on combustion stability of TRW engines operating with LOX/RP-1 is limited to 50K lb_f thrust level. Analyses however, have been performed for LOX/RP-1 at thrusts up to 2000K lb_f. These analyses, documented in Ref. 2, show that the TRW coaxial injector engine is absolutely stable while conventional injectors are predicted to be unstable.

Low Cost Booster PF Engine Stability Analysis

PERFORMED BY DYNAMIC SCIENCE (1972)

- NONLINEAR INSTABILITY ANALYSIS BASED ON PRIEM-GUENTERT MODEL

ANALYSIS PERFORMED FOR

- 1500K TO 2000K (VAC) THRUST TRW ENGINE DESIGN OPERATING ON LOX/RP-1
  - P_C = 250 AND 300 PSIA
  - TWO PINTLE DIAMETERS
  - REGENERATIVE AND ABLATIVE COOLING
- 1500K LOX/C_3H_8 NOMINAL ENGINE
- COAXIAL AND CONVENTIONAL INJECTORS

RESULTS

- HIGH FREQUENCY DISTURBANCES
  - CONVENTIONAL INJECTOR UNSTABLE
  - COAXIAL INJECTOR ABSOLUTELY STABLE
- LOW FREQUENCY CHUGGING MODE
  - 5-7 MS COMBUSTION DELAY CHARACTERISTIC OF COAXIAL INJECTOR RESULTS IN CRITICALLY DAMPED SYSTEM
  - NO RESONANT CONDITION EXISTS FOR COAXIAL INJECTOR
  - NO INSTABILITY TO AT LEAST 40% THROTTLING

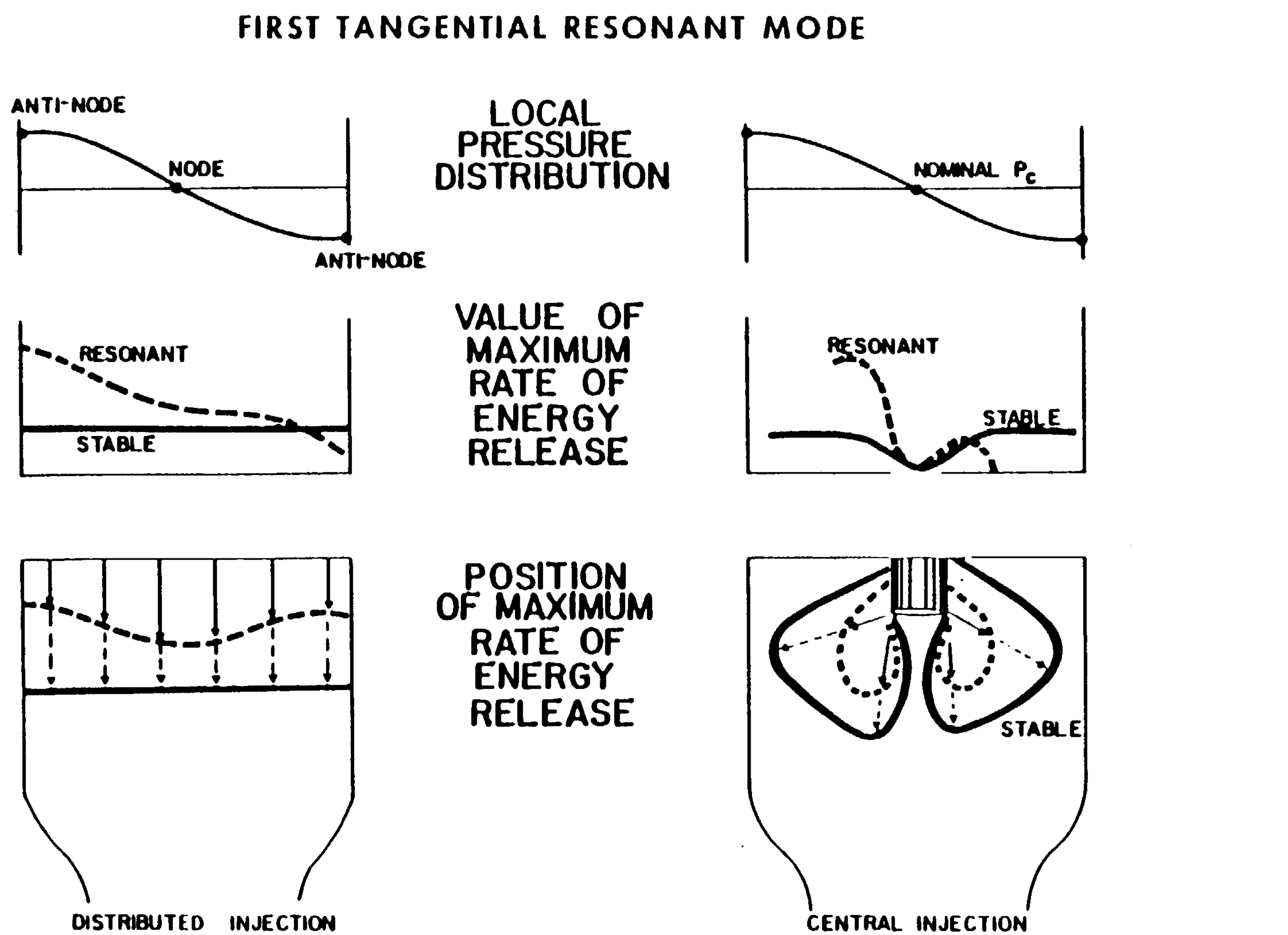
POGO

- LOW PROBABILITY OF OCCURRENCE WITH GOOD DESIGN

To sustain a disturbance in a combustion chamber, it is necessary to supply combustibles in and near the antinode regions. A characteristic disturbance always drives the resulting increased combustion rate flame towards the supply source. In the case of the conventional injector unit, this results in flame movement towards an antinode. In the case of the coaxial unit, this results in flame movement towards a nodal zone with decreased pressure disturbance and, hence, to a combustion zone of stabilizing influence. With respect to radial modes, it is conceivable that radial modes are possible, although to date they have not been observed. If they do occur, no hardware damage is possible with the coaxial injection; the head end zones are cold zones, and no scrubbing of hot gases across any critical zones is possible.

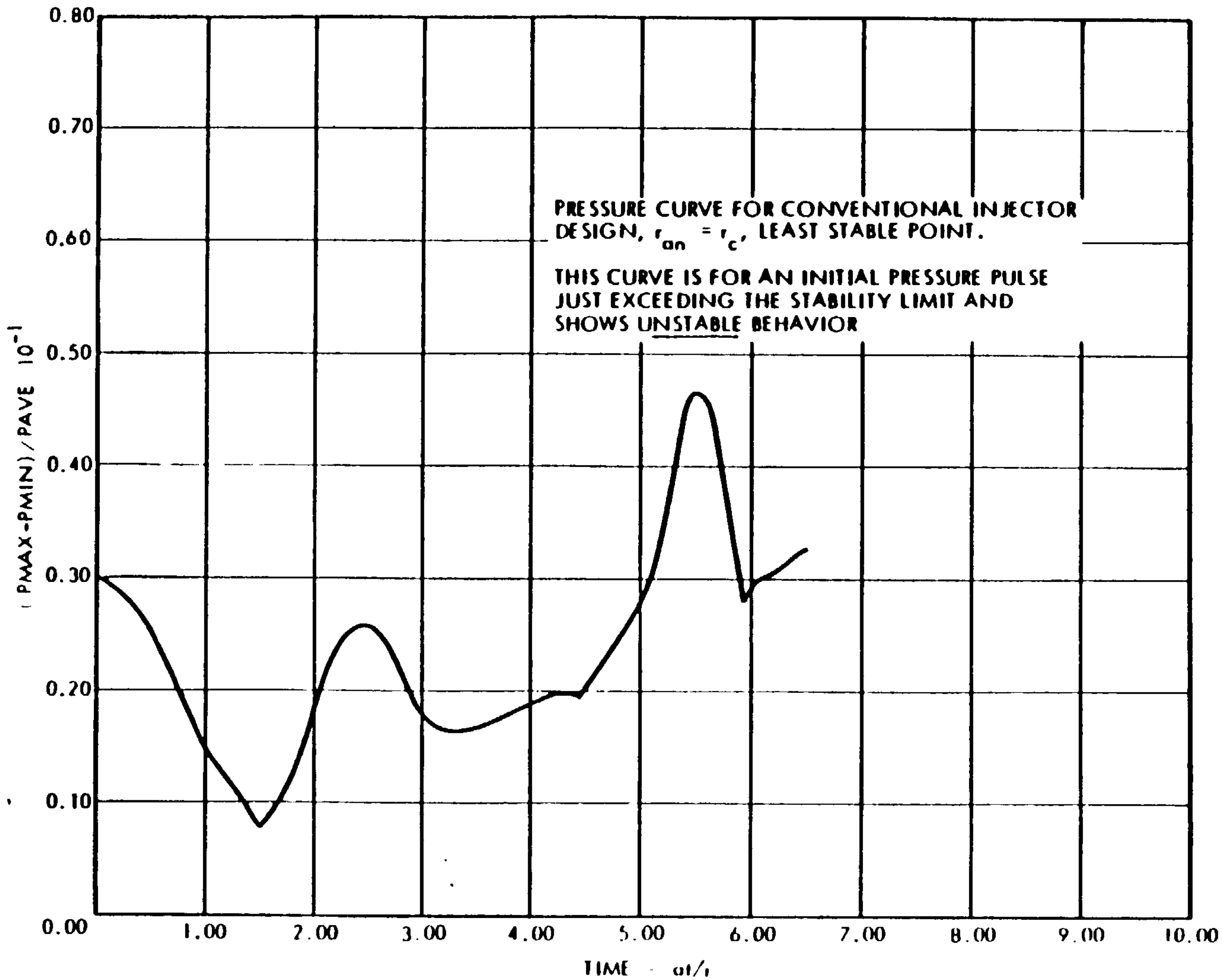
C-5

Comparison of Conventional and TRW Coaxial Injector Stability Characteristics



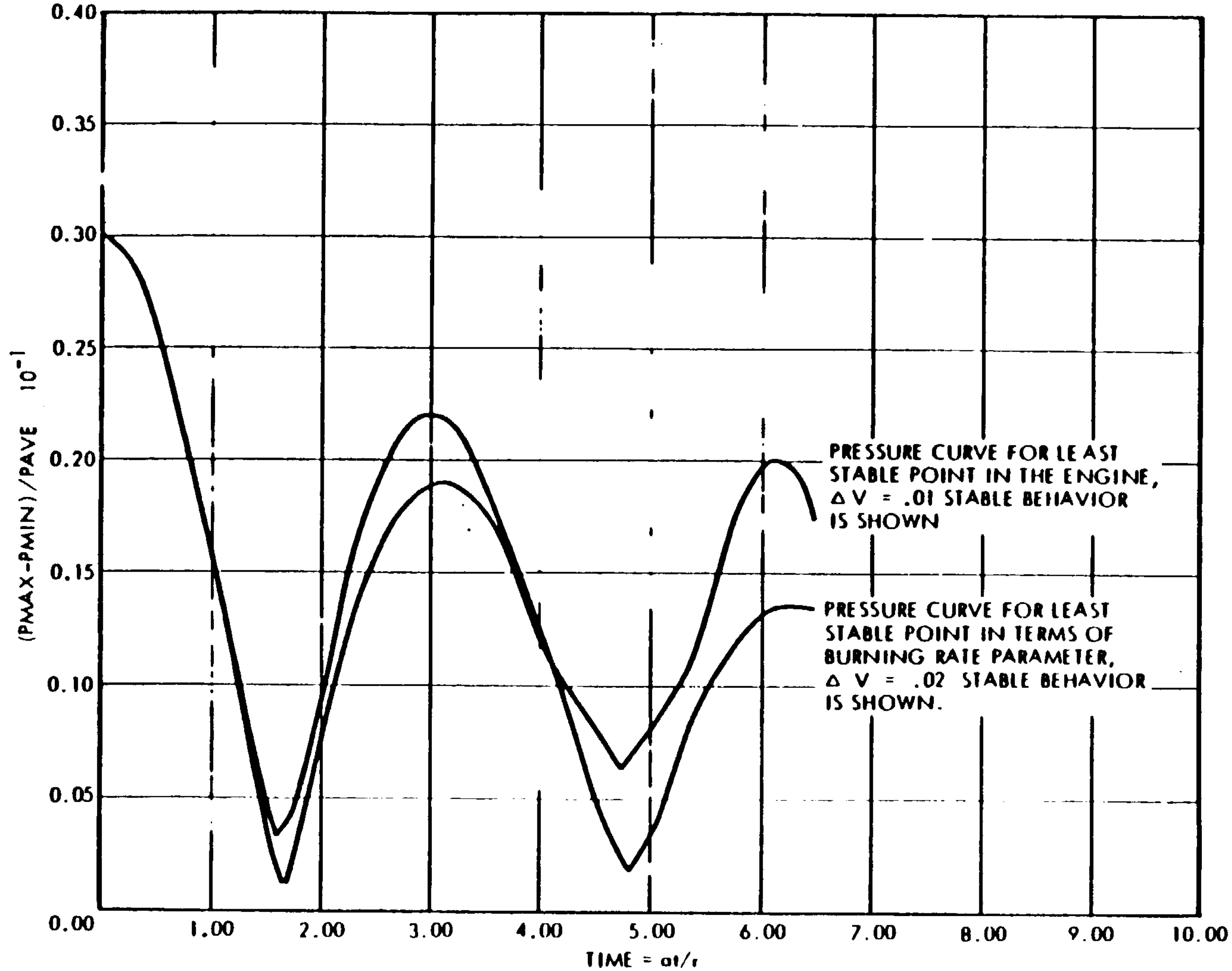
The facing curve represents the evolution of a pressure disturbance which is slightly larger than the stability limit of a 1500K engine having a conventional injector. The growth of the pressure disturbance with time is indicative of unstable behavior.

Conventional Injector Response to a Perturbation at Its Stability Limit



These curves show the response of a 1500K engine having a coaxial injector to a pressure disturbance at two positions in the combustion chamber. One of the positions selected is the least stable point in the engine. The damped pressure oscillations shown are indicative of combustion stability at both points.

TRW PFE Response to a Disturbance Indicating Strong Drive to Stability



Throttling an engine tends to produce pressure waves in the propellant supply system unless steps are taken to suppress the pressure coupling. This is easily achieved with standard devices such a cavitating venturis and, in less stringent cases, trim orifices.

Design Features Used to Suppress Pressure Coupling of Engine to Propellant Tanks

TRW

COUPLING CAN BE ELIMINATED BY USE OF CAVITATING VENTURIS

- USED FOR HIGH RATE PULSING THRUSTERS AND DEEP THROTTLING ENGINES
- DISADVANTAGE IS 15% HEAD LOSS

TRIM ORIFICES ARE SUFFICIENT FOR ISOLATION

- USED FOR LOW RATE PULSING THRUSTERS AND MILD THROTTLING ENGINES
- ADVANTAGES ARE LOW COST AND LOW HEAD LOSS

5. STS LRB PRESSURE-FED ENGINE DESIGN STUDY

This section summarizes the engine design study performed during the Phase I STS LRB Feasibility Study. The material is presented in the form of charts with explanatory facing pages. It is organized into 9 subsections. The first deals with propellant selection while the second discusses engine configuration considerations. A baseline point design concept for a high reliability, low cost engine based on TRW's coaxial injector is presented next. Performance estimates for this engine are given in subsection 4. Thrust vector control (TVC) options are described after that. Subsection 6 discusses the impact of water recovery on engine design. A preliminary reliability and safety analysis was performed for the engine concept of subsection 3 and is documented in subsection 8. The results given there predict that very high reliability levels can be achieved with the selected engine design. The final subsection presents both recurring and nonrecurring cost estimates for various engine concepts, including our baseline design concept.

5.1 PROPELLANT SELECTION

Propellant characteristics and physical properties data are presented in Section 3 and were provided to GDSS to support a propellant selection decision. Some important propulsion performance parameters for five selected propellant combinations are summarized in the facing chart.

Propellant Characteristics

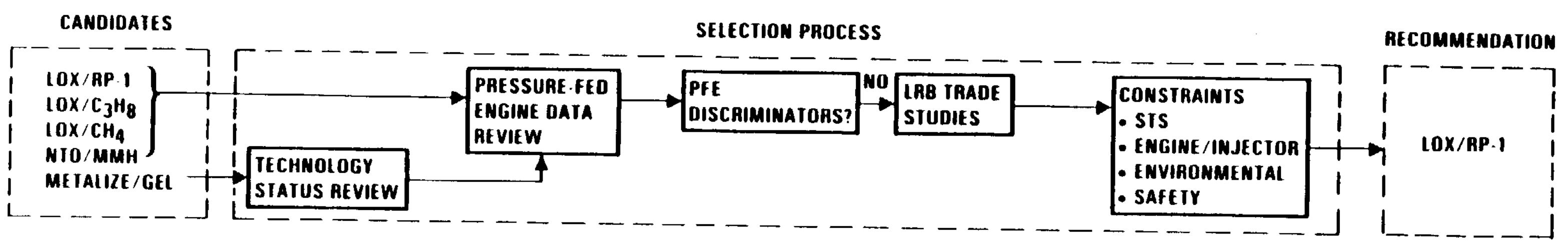
Pc = 500 psia
Aex/A* = 6.0

Propellants	Theoretical Maximum Performance					Rankings[1]				
	O/F	I_{sp} (sec)	ρI_{sp} (#sec/ft^3)	Tc (K)	L* (in)	Performance I_{sp}	Performance ρI_{sp}	Combustion Temperature	L*	Ignition System Required?
LOX/RP-1	2.43	311	19,438	3538	96	3	3	4	3	Yes
LOX/C_3H_8	2.59	316	16,558	3464	96	2	4	3	3	Yes
LOX/CH_4	2.88	321	15,954	3375	96[2]	1	5	2	3	Yes
NTO/MMH	2.00	301	22,259	3289	75[4]	4	2	1	1	No
MICOM GELS	1.00	268	90,316	3920	85[3]	5	1	5	2	No

1. Most favorable is 1.
2. Estimate, no TRW firing experience.
3. Estimate, no TRW firing experience at this low a pressure.
4. Based on LMDE.

While most TRW engine experience is with storable propellants, our engines have run successfully with LOX/HC and gelled propellants. As a result, our engines are not a discriminator with respect to propellant selection. The choice of propellant then depends on other considerations such as safety, environmental compatibility, economics, booster size constraints, etc, as noted in Section 3. Based on detailed system trade studies and other considerations, GDSS has selected LOX/RP-1 for the STS LRB. As the remainder of this subsection shows, TRW coaxial injector engines are compatible with this selection.

Propellant Selection Summary



Although our engines are not a discriminant with respect to propellants, the propellant selection does influence engine injector design as noted in this chart. None of the features noted represent a major design issue.

Effects of Propellants on TRW Coaxial Injector Design

TRW

PROPELLANTS	EFFECT
LOX/RP-1	• PROVISIONS FOR INSULATING RP-1 FROM THE LOX IS REQUIRED • EASILY IMPLEMENTED WITH MINIMUM WEIGHT AND COMPLEXITY IMPACT
LOX/C_3H_8	• LOX/FUEL INSULATION PROVISIONS LESS CRITICAL THAN FOR LOX/RP-1
LOX/CH_4	• MINIMAL LOX/FUEL INSULATION REQUIRED • NO TRW FIRING EXPERIENCE • GASIFICATION OF CH_4 IS NOT BELIEVED NECESSARY FOR TRW COAXIAL INJECTOR
NTO/MMH	• NO SPECIAL REQUIREMENTS
MICOM GELS	• INJECTOR DESIGN SIMILAR TO THAT FOR NTO/MMH BUT FACE SHUTOFF FEATURE IS MANDATORY TO PREVENT CLOGGING OF INJECTOR

CONCLUSIONS

- ALL PROPOSED PROPELLANTS ARE EASILY ACCOMMODATED WITH TRW COAXIAL INJECTOR
- INJECTOR CONFIGURATION DETAILS WILL DIFFER FOR EACH PROPELLANT COMBINATION
  - OXIDIZER INJECTOR PORT CONFIGURATION
  - FUEL ORIFICE GAP
  - INSULATION PROVISIONS
  - SHUTOFF PROVISIONS
- NO MAJOR COST, WEIGHT, OR OTHER DISCRIMINATOR IMPACT FORESEEN

Propellant selection also influences the details of the chamber/nozzle design. Again, however, there are no major design issues that influence the choice of propellant.

Effect of Propellants on Engine Chamber/Nozzle Design

TRW

PROPELLANTS	EFFECT
LOX/RP-1 LOX/C_3H_8 LOX/CH_4	• ALL LOX/HYDROCARBONS REQUIRE ABOUT THE SAME L*, AND THEREFORE CHAMBER LENGTH • COMBUSTION TEMPERATURES ARE NEARLY THE SAME AND THEREFORE ABLATIVE THICKNESS OR REGENERATIVE COOLING REQUIREMENTS WILL NOT BE SIGNIFICANTLY DIFFERENT • NOZZLE DESIGN WILL ALSO BE THE SAME
NTO/MMH	• CHAMBER LENGTH WILL BE ABOUT THREE/FOURTHS THAT FOR LOX/HYDROCARBONS • ABLATIVE THICKNESS OR REGENERATIVE COOLING REQUIREMENTS WILL BE ABOUT THE SAME AS THAT FOR LOX/HYDROCARBONS
MICOM GELS	• CHAMBER SIZE IS BETWEEN THAT FOR NTO/MMH AND LOX/HYDROCARBONS • ABLATIVE THICKNESS OR REGENERATIVE COOLING MAY HAVE TO BE INCREASED SOMEWHAT BECAUSE OF THE INCREASED COMBUSTION TEMPERATURE

No major impact of propellant selection is expected on engine weight but cost should be lower for storable propellants than for LOX/HC because the injector design is less complex. The cost difference is not a major one, however, and therefore once again propellant selection is not a determining factor.

Effect of Propellants on Engine Weight and Cost

PROPELLANT	RELATIVE WEIGHT	RELATIVE PRODUCTION COST	DEVELOPMENT COST RANKING*
LOX/RP-1	1.00	1.00	2
LOX/C_3H_8	1.00	1.00	2
LOX/CH_4	1.00	1.00	2
NTO/MMH	0.985	0.90	1
MICOM GELS	0.992	1.00	3

*1 IS LOWEST COST

As noted in Section 4, TRW engines have never exhibited combustion instabilities with any propellant combination, including LOX/HC. As a result, combustion stability is also not a propellant discriminator.

TRW

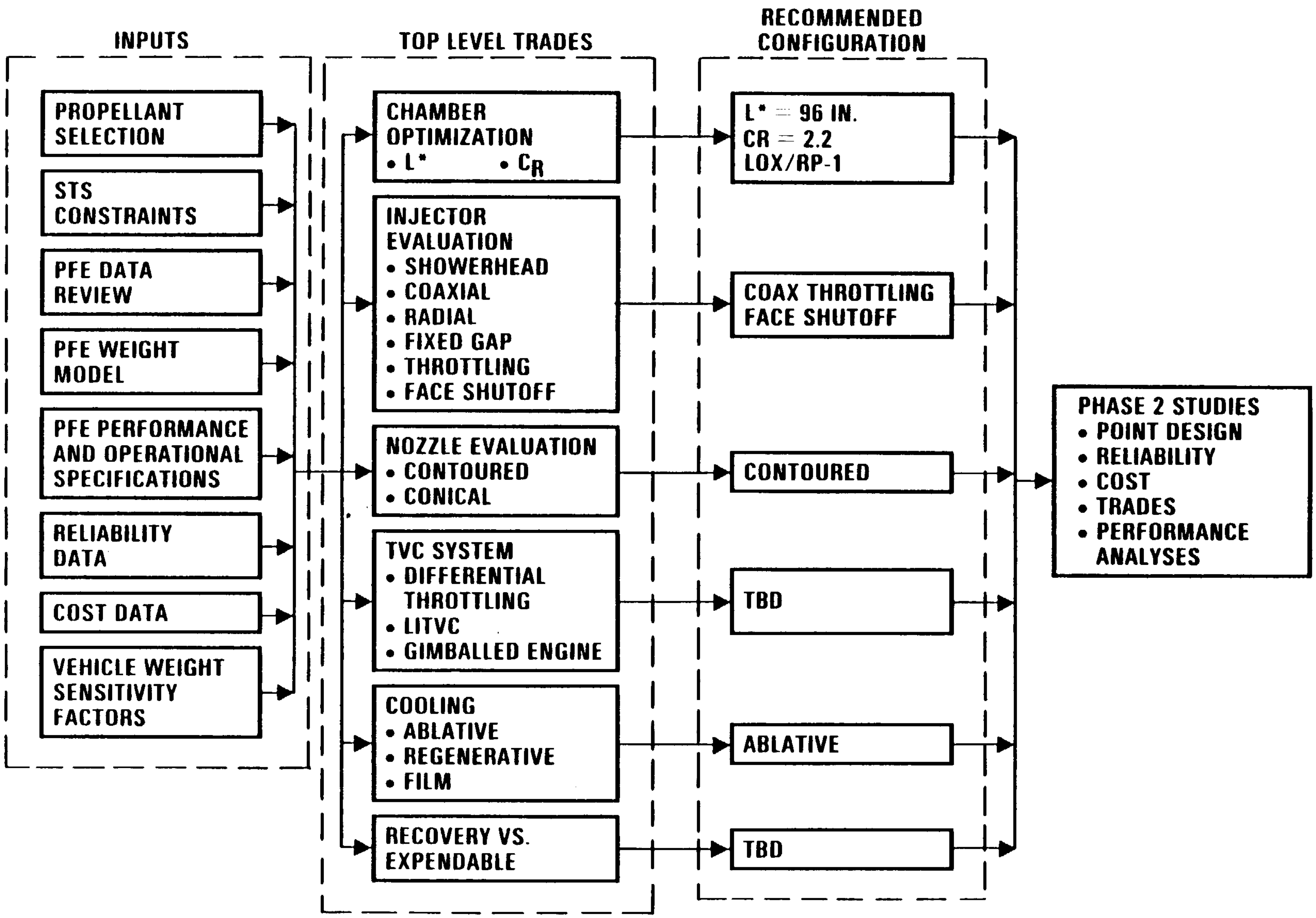
Effect of Propellant Selection on Engine Combustion Stability

- TRW's COAXIAL INJECTORS ARE INHERENTLY STABLE
- ALL CANDIDATE PROPELLANTS EXCEPT LOX/CH_4 HAVE BEEN TESTED WITH NO COMBUSTION STABILITY PROBLEMS
- LOX/CH_4 IS NOT EXPECTED TO BE DIFFERENT FROM OTHER PROPELLANTS WITH RESPECT TO STABILITY
- COMBUSTION STABILITY IS NOT A PROPELLANT DISCRIMINATOR FOR TRW's COAXIAL INJECTOR

5.2 ENGINE CONFIGURATION CONSIDERATIONS

During Phase I of the STS LRB Feasibility Study, the various engine configuration options were examined and ranked using historical data and trade studies where appropriate. Our recommended configuration was selected on the basis of high reliability and low cost considerations with maximum performance a secondary issue.

PFE Configuration Selection



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The engine combustion chamber configuration is controlled by the characteristic length, L^*, vehicle weight optimization trades and vehicle-imposed constraints. The value of L^* is determined experimentally as the minimum value consistent with good combustion performance as indicated by η_C^*. In general, it is dependent on the propellants used and their mixture ratio, as well as the injector type and geometry. Nozzle throat area is influenced by chamber pressure, combustion performance, nozzle type and performance, and the thrust specification. The contraction ratio selection is based on detailed vehicle system trade studies. Our past studies of LOX/RP-1 pressure-fed booster (Ref. 2) showed that a contraction ratio of 2.2 was optimum. As a result, this value has been selected for the preliminary design studies presented herein. The ultimate value to be used would be determined from trade studies of the STS LRB.

Chamber Configuration Definition

L^* determination ($L^* = V_c/A_t$)

- Depends on injector design and propellants
- Empirically derived
- $L^* \cong 96$ in. for LOX/RP-1 and coaxial injector, subject to experimental optimization

Nozzle throat area

- Specified by desired thrust, P_c and C_F
- C_F propellant and nozzle dependent

Contraction ratio

- Typical values
  - — 2 to 6 for low pressure, pressure-fed engines
  - — 1.3 to 2.5 for high pressure, pump-fed engines
- Selected value ultimately determined from detailed trade studies
  - — Combustion performance with selected injector
  - — Vehicle weight impact
    - Chamber pressure drop and wall cooling
    - Engine, engine-mounts and TVC
  - — Space constraints
  - — Manufacturing ease and cost

The coaxial injector is superior to other injector types in all areas except chamber length. It requires a longer and heavier chamber but compensates for it by being the lightest injector. As a result, the coaxial-injector engine is the lightest overall. It is also the lowest cost and lowest development risk because of its inherent combustion stability. All of these advantages are achieved without a performance penalty.

Injector Ranking

TRW

CANDIDATES	STABILITY	THROTTLING CAPABILITY	FACE SHUTOFF CAPABILITY	MANUFACTURING COST	DEVELOPMENT COST	COMBUSTION PERFORMANCE	CHAMBER SIZE	ENGINE WEIGHT	RELIABILITY	COMPATIBILITY WITH RECOVERY
COAXIAL	1	1	YES	1	1	1	3	1	1	1
SHOWER HEAD	2	2	NO	3	3	1	1	3	2	3
RADIAL	3	2	NO	2	2	1	2	2	3	2

NOTE 1 IS THE MOST FAVORABLE RANKING, 3 IS WORST

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Differential throttling, LITVC and gimballed engines are all viable TVC options for use with TRW's high reliability, low cost engine concepts. The gimballed-engine TVC has been baselined by GDSS.

TRW's coaxial injector is unique in that it can provide deep throttling with very little performance loss, no combustion instabilities and very rapid throttle response. Past TRW experience with throttling engines is that throttling response times are dominated by the time it takes to move the throttling sleeve. The sleeve actuator can be sized such that this time is very short. As a result, throttle response times of a small fraction of a second (less than 0.2 second) has been routinely achieved. This is believed to be well within the response times required for the LRB mission. Past TRW studies of similar vehicles indicate that a throttle response time of about 0.7 second should suffice. This throttling performance makes TVC by differential throttling an option not possible with other types of injectors. Its advantages are that differential throttling can be implemented with no additional hardware complexity or weight.

TVC System Ranking

TRW

Candidates	Weight	Cost	Reliability	Effectiveness	Response	Compatibility With Recovery	Comments
Differential Throttling*	1	1	1	1	2	1	Requires no extra hardware, just software
LITVC	3	2	2	1	1	2	Requires significant weight increase
Gimbaled Engine	2	3	3	1	3	3	Requires hydraulics and flex lines Recovery loads impact design Heavy

*Coaxial injectors

Note: 1 is the most favorable ranking, 3 is worst

Ablative cooling is superior to other candidates in terms of reliability, cost and weight. Because of its simplicity, ablative cooling is also much more readily adapted to water recovery and reliable reuse.

Cooling System Ranking

Candidates	Reliability	Cost	Weight	Effectiveness	Geometrical Stability	Compatibility With Recovery
Ablative	1	1	1	1	2	1
Regenerative	2	3	3	1	1	3
Film	2	2	2	1	1	2

Note: 1 is the most favorable ranking, 3 is worst

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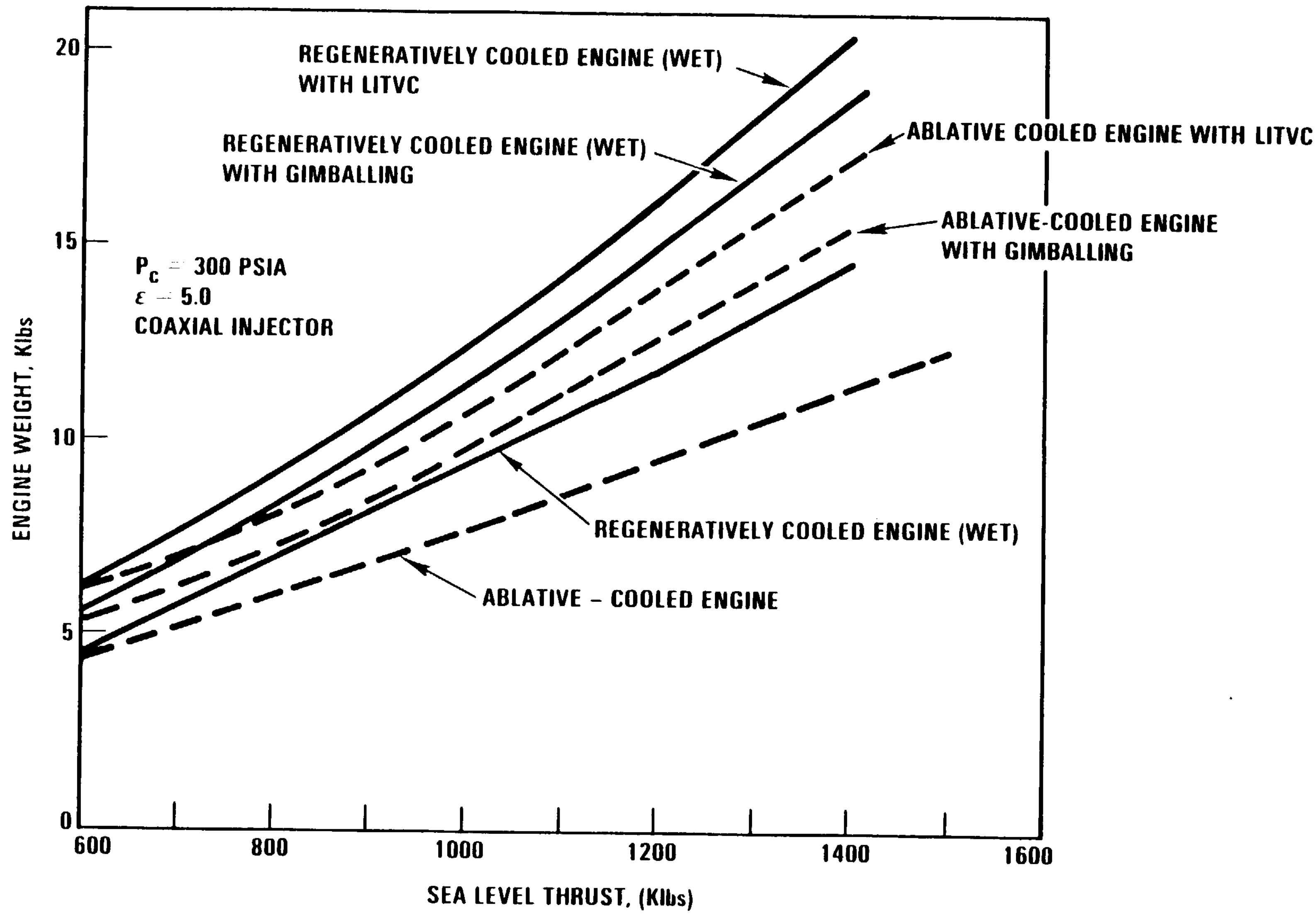
Engine weight is influenced by thrust level, TVC type, and cooling method. The curves shown here for a 300 psia chamber pressure engine confirm that LITVC is the heaviest TVC candidate considered. Differential throttling, of course, is the lightest. The lightest cooling candidate is the ablative material, although its advantage is not great at the lower thrust levels.

Effect of Engine Cooling Method, TVC Type and Thrust on Engine Weight



Recovery has little impact on the design of a throttling coaxial-injector engine with ablative cooling. In this case, injector face shutoff is easily provided to prevent sea water from penetrating the injector and propellant supply lines. This cannot be done with other types of injectors.

An ablative liner would be replaced after every flight and so could be used as a sacrificial nozzle extension to reduce water-slap impact loads. This could significantly reduce engine and attachment weights. In contrast, a regeneratively cooled engine is much more difficult to make compatible with survivable water recovery because of its complex nozzle structure.

Recovery Versus Expendable Booster

TRW

Effect of recovery on engine specifications

- Increased design loads due to water impact
- Face shut-off feature to prevent salt water penetration into injector is desirable
- Refurbishment requirement puts premium on design simplicity

Low cost, ablative-cooled, pressure-fed booster engine design

- Design is easily made compatible with recovery
- Recovery requirements are not major design drivers

5.3 BASELINE CONCEPT FOR A HIGH RELIABILITY, LOW COST LRB ENGINE

A baseline concept for a high reliability, low cost LRB engine is presented in this section. The concept is based on the TRW coaxial injector and low cost engine manufacturing technology. A gimballed TVC system was selected by GDSS and so is included in the baseline concept. At this point, a clear decision on recovery has not been made. The baseline design, however, can easily be made compatible with recovery and reuse if that approach is taken.

Baseline Engine Concept

- TRW COAXIAL INJECTOR
  - THROTTLING
  - FACE SHUT-OFF
- LOW COST CHAMBER/NOZZLE
  - ROLLED AND WELDED STEEL
  - ABLATIVE COOLED
- GIMBALLED ENGINE TVC
- DESIGN IS EASILY MADE COMPATIBLE WITH RECOVERY IF THAT OPTION IS SELECTED

GDSS trade studies (Ref. 3) provide the basis for a preliminary engine specification. Our conceptual design is based on these requirements.

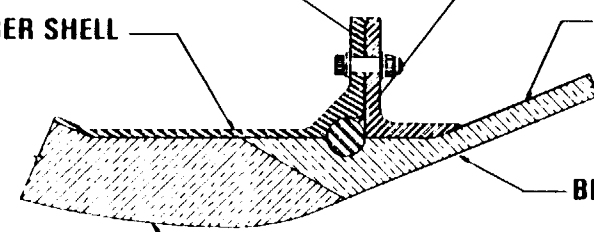
Preliminary Engine Specifications

TRW

- BASIS:
  - GDSS TRADE STUDIES
- ENGINE THRUST (SEA LEVEL)
  - 619K POUNDS
- PROPELLANTS
  - LOX/RP-1
- CHAMBER PRESSURE
  - 400 PSIA
- THROTTLING CAPABILITY
  - TBD
- TVC BASELINE
  - GIMBALLED ENGINE
- NOZZLE EXIT DIAMETER LIMIT
  - 90 INCH (IMPOSED BY FLAME DUCT COMPATABILITY REQUIREMENT)

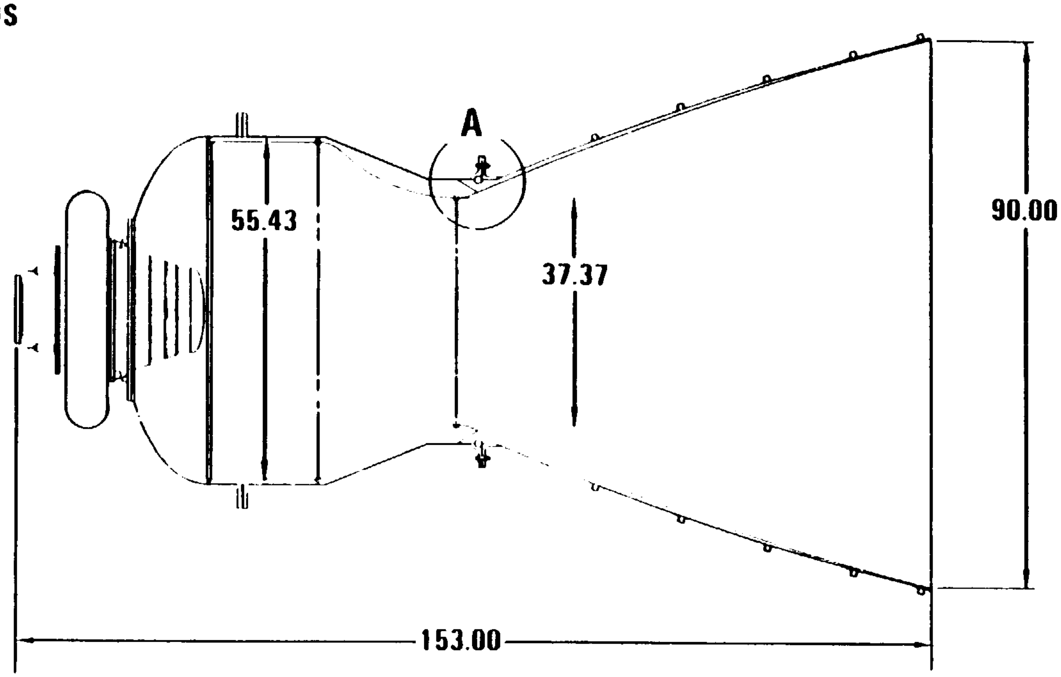
The baseline TRW engine concept for STS LRB applications is an ablative-cooled engine having a stainless steel shell, a silica-phenolic liner, an 80% bell nozzle and a coaxial face shutoff injector. The chamber contraction ratio is 2.2 and nozzle area ratio is 5.8 subject to the 90-inch exit diameter limitation imposed by compatibility with the launch stand flame duct. Sea level engine thrust is conservatively estimated as 619K pounds at 400 psia chamber pressure with LOX/RP-1 at a mixture ratio of 2.4. This engine concept utilizes well-proven, high reliability engine technology and yet can be produced using low-cost materials and processes.

Ablative Cooled 619K Engine Concept



Detail A

NOTE: ALL DIMENSIONS
ARE IN INCHES



A bell nozzle has been selected for the baseline engine because it saves 228 pounds per engine, increases thrust by 1%, and reduces the engine length by about 20 inches as compared to a 15° conical nozzle. Total gimballed engine weight of 5563 pounds is very light compared to other engine designs and is achieved using stainless steel materials and low cost fabrication techniques.

TRW

Weight Summary (lb)

Ablative-Cooled, Gimballed, Throttling Engine

COMPONENT	NOZZLE TYPE: 80% BELL	NOZZLE TYPE: 15^{0} SEMI-ANGLE CONE
CHAMBER HEAD	452	452
CHAMBER SHELL	710	710
CHAMBER ABLATIVE	718	718
NOZZLE SHELL	415	520
NOZZLE ABLATIVE	621	755
STIFFENING RINGS		
NO. 1	15	12
NO. 2	17	14
NO. 3	19	16
NO. 4	20	18
NO. 5	47	47
LOCKING RING	50	50
INJECTOR ASSEMBLY	1089	1089
GIMBAL ASSEMBLY	970	970
VALVES*	200	200
INTEGRATION HARDWARE	220	220
	----	----
TOTAL WEIGHT (LB)	5563	5791

*INCLUDES ACTUATORS

NOTE: GIMBAL ACTUATORS AND APU SYSTEMS NOT INCLUDED.

Past TRW work has considered ablative, regenerative, and film-cooled engines. TRW LMDE and Sentry engines used ablative cooling supplemented by film cooling and never experienced a cooling failure. TRW booster-type engines up to 250K lb_f thrust levels successfully used ablative cooling only without failure.

For the STS LRB engine, we recommend using a silica-phenolic ablative liner. The facing chart shows that such an engine is significantly lighter than a comparable regeneratively cooled engine. The weight difference is even greater than shown because of the effects of the fuel pressure drop that occurs across the regenerative cooling tubes. An ablative cooled engine is also more reliable, of lower cost to develop and manufacture and more compatible with water recovery and refurbishment than a regeneratively cooled engine.

TRW

Weight Summary (lb)
619K Gimballed, Throttling Engine with 80% Bell Nozzle

Ablative Cooled

Component	Weight
Chamber Head	452
Chamber Shell	710
Chamber Ablative	718
Nozzle Shell	415
Nozzle Ablative	621
Stiffening Rings	
No. 1	15
No. 2	17
No. 3	19
No. 4	20
No. 5	47
Locking Ring	50
Injector Assembly	1089
Gimbal Assembly	970
Valves*	200
Integration Hardware	220
Total Weight (lb)	5563

*Includes actuators.

Regenerative Cooled

Component	Weight
Heat End Shell	309
Head Tubes	232
Combustion Chamber Shell	721
Combustion Chamber Tubes	726
Nozzle Band	331
Nozzle Tubes	577
Injector Element	290
Fuel Manifold and Duct	250
Gimbal Assembly	970
Shut-off Valves	200
Integration Hardware	220
Total Dry Weight (lb)	4826
Residual Propellant	1250
Total Wet Weight	6076

Note: Gimbal actuators and APU systems not included.

A detailed optimization of the ablative liner has not yet been performed. A calculation of the maximum bond line temperature, however, was performed to verify that the ablative thickness is adequate. This result indicates that it is.

Engine Maximum Ablative Temperature at Bond Line

TRW

PARAMETERS

ABLATIVE THICKNESS	0.75 INCH
MISSION BURN TIME	112 SEC
MATERIAL	MX-2600 SILICA PHENOLIC
PROPELLANTS	LOX/RP-1
CHAMBER PRESSURE	400 PSIA

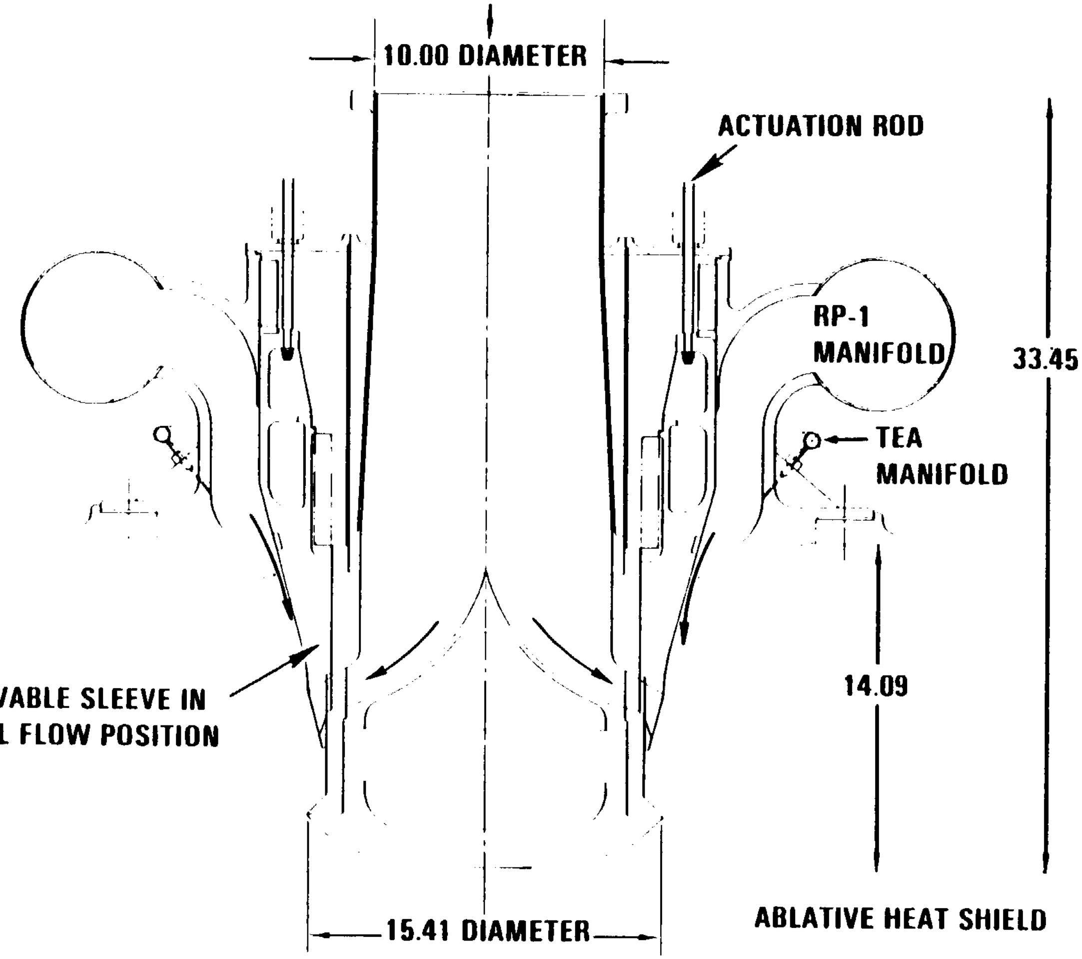
RESULTS

BOND LINE TEMP AT 112 SEC = 284^{o}F

Two related injector concepts have been considered. The first is basically a scaled-up version of the LMDE injector modified for LOX/RP-1 use. LOX flows in a central tube and is injected into the chamber radially through a series of slots through the cylindrical wall of the tube near its end. RP-1 is admitted through an annular slot about the LOX tube and forms an essentially cylindrical sheet that mixes with the radially flowing LOX "spokes." Throttling is provided by a slotted movable sleeve that fits over the LOX supply tube. Both the RP-1 and LOX are throttled simultaneously, maintaining the desired mixture ratio, as the sleeve is pushed forward. As its most forward position, the sleeve completely closes the LOX and RP-1 ports, achieving face shutoff. This is a desirable feature for water recovery or GEL propellants. Sleeve actuation is provided by an external servo valve and linkage arrangement. Thermal isolation between LOX and RP-1 is provided by vented chambers in the sleeve. Mixing of LOX and RP-1 in sleeve passages is prevented through use of bellows seals.

The baseline ignition system relies on injection of triethylaluminum (TEA). TEA is injected behind the RP-1 annular slot so that it is also sealed off from the chamber at the face shutoff limit. A spark ignition system can also be used.

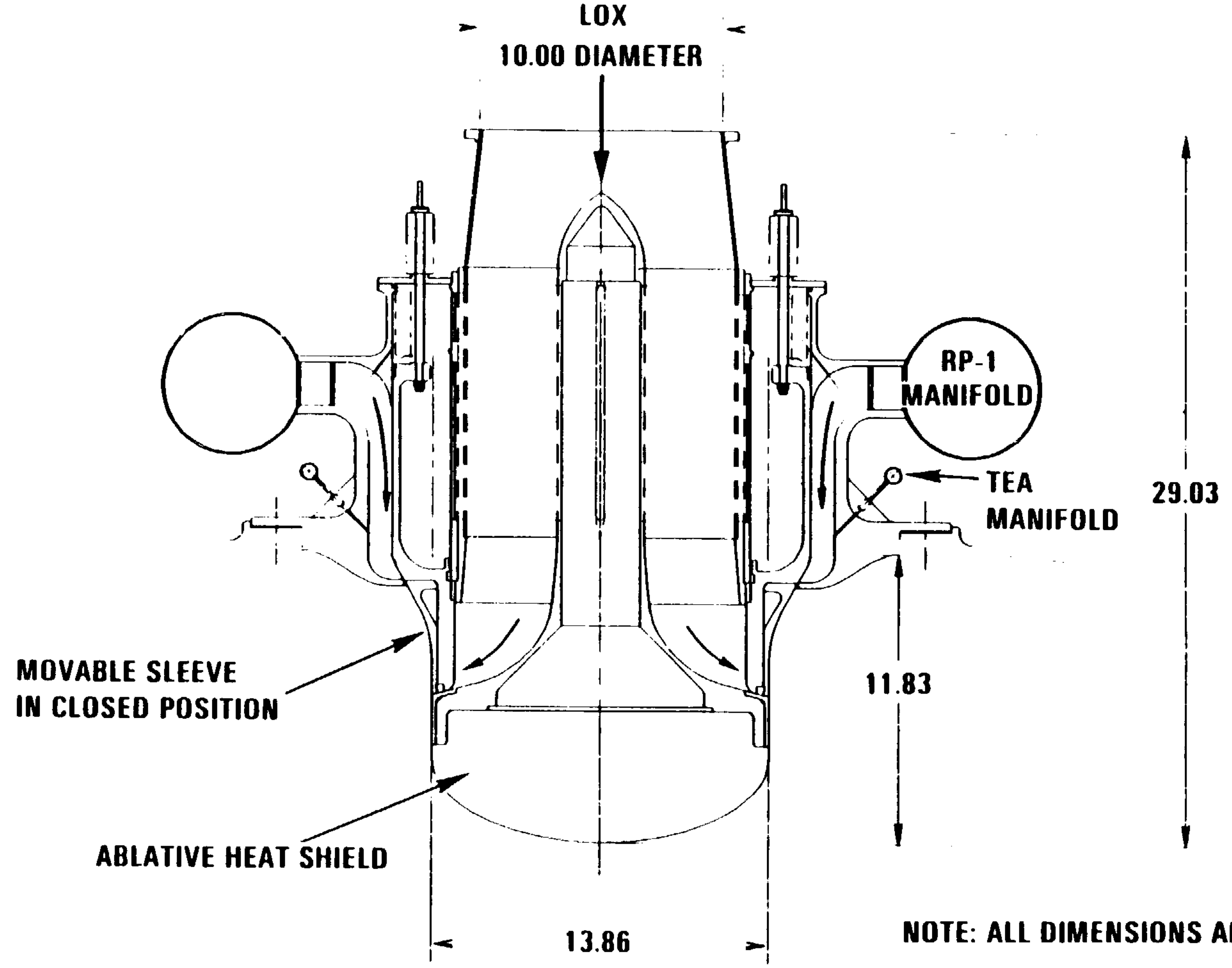
Face Shutoff Throttling Injector Concept Based On LMDE Technology



NOTE: ALL DIMENSIONS ARE IN INCHES

The second injector concept considered is a modified scaled-up version of the SENTRY engine injector. It differs from the LMDE injector in that the end of the LOX tube is supported by a spider arrangement that allows the slots of the LMDE oxidizer pintle to be replaced by a cylindrical orifice. Thus the SENTRY oxidizer pintle produces a disc-like flow of LOX, rather than a series of spokes. Recent engine developments at TRW using injectors similar to this design have produced very high performance using storable hypergolic fuels.

Face Shutoff Throttling Injector Concept Based On Sentry Engine Technology



NOTE: ALL DIMENSIONS ARE IN INCHES

TEA ignition has been used successfully at TRW and other places for igniting LOX/RP-1. An augmented spark igniter has also been used. Either system is satisfactory.

Ignition Options

TRW

CANDIDATES

- HYPERGOLIC IGNITER (TRIETHYLALUMINUM, TEA)
- SPARK IGNITER

PROS AND CONS

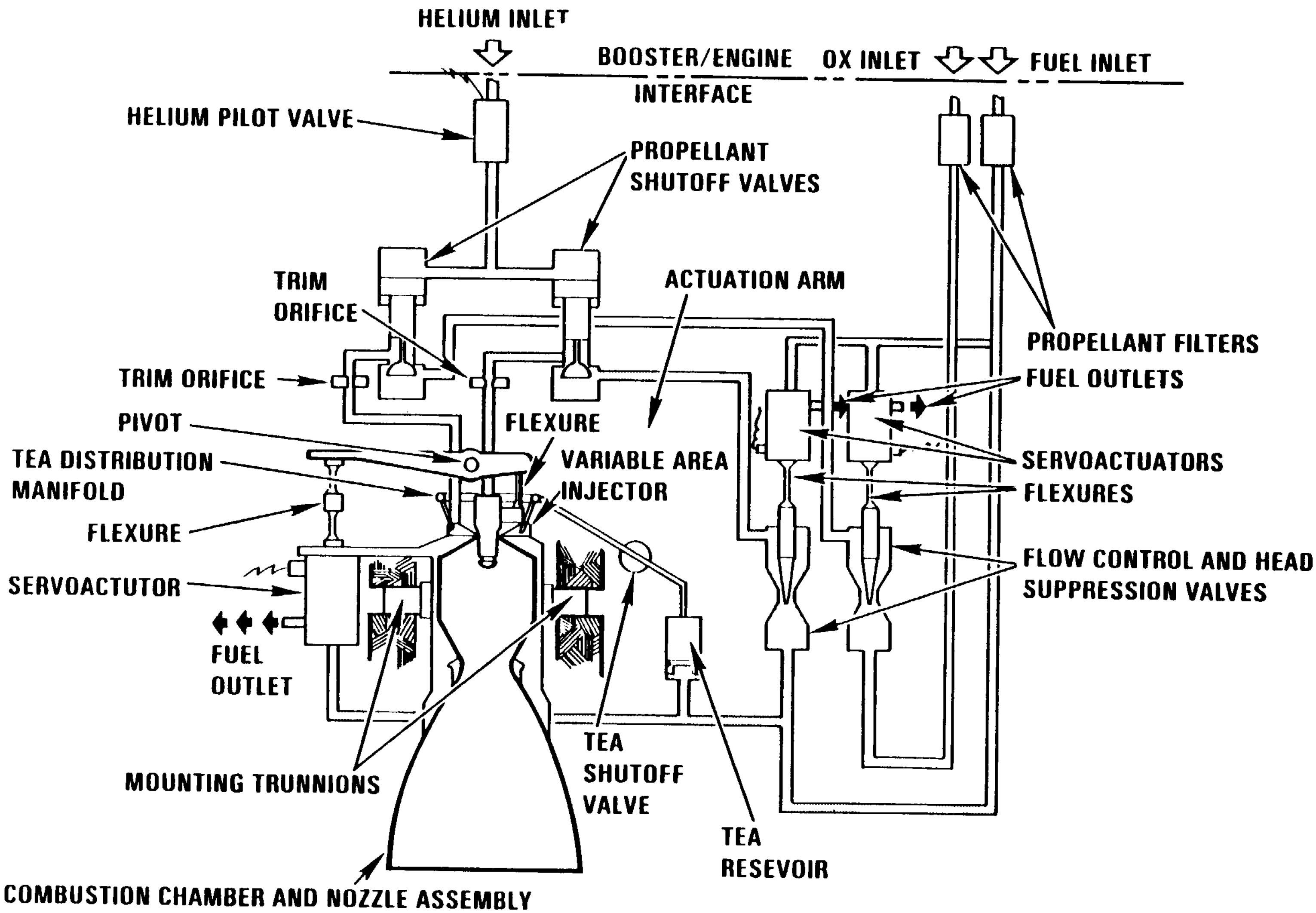
- BOTH SYSTEMS WORK WITH LOX/RP-1
- BOTH ARE COMPATIBLE WITH MULTIPLE RESTART CAPABILITY ALTHOUGH THE TEA SYSTEM IS MORE LIMITED
- THE TEA SYSTEM IS EASIER TO SEAL OFF TO MINIMIZE REFURBISHMENT CLEANUP AFTER WATER RECOVERY

CONCLUSION

- EITHER SYSTEM CAN BE USED
- WE HAVE BASELINED THE TEA SYSTEM FOR CONCEPTUAL DESIGN PURPOSES

This schematic shows the booster/engine interface upstream of the flow control head suppression valves. These valves are shown individually controlled but they could be readily linked mechanically to the injector linkage, as done on LMDE. For increased reliability, redundant valves can be added to this schematic.

Gimballed Engine Concept Schematic



The development risk of this engine concept is low. Its mechanical complexity is minimal and no new basic technology is required. The inherent combustion stability and demonstrated scalability of the TRW coaxial injector are major benefits in this regard.

Engine Technology Level and Risk

TRW's PROPOSED BASELINE ENGINE CONCEPT INVOLVES MINIMAL TECHNOLOGICAL RISK

- PROVEN DESIGN CONCEPT
- LOW COST/LOW RISK APPROACH
- DEMONSTRATED SCALABILITY

CAPABLE OF DEVELOPMENT ON SHORT TIME SCHEDULE

REPRESENTS SCALE-UP OF DEMONSTRATED DESIGNS

- HIGHER THRUST LEVELS
  - 250K DEMONSTRATED USING NTO/UDMH
  - 50K DEMONSTRATED USING LOX/RP-1 AND IRFNA/UDMH
  - 2K DEMONSTRATED USING LOX/C_3H_8 AND LOX/RP-1
  - 10K DEMONSTRATED USING NTO/MMH AND NTO/A-50
  - 19:1 THROTTLING DEMONSTRATED AT 10K USING NTO/A-50
  - 10:1 THROTTLING DEMONSTRATED AT 10K USING NTO/MMH

NO APPARENT PROPELLANT DISCRIMINATORS

5.4 ENGINE CONTROLS

The control system required to meet operational requirements for engine start-up and shutdown, gimballed engine thrust vector control, throttling and propellant utilization operations is illustrated in the facing figure. The following design features are included:

(1) Shutoff valves including high pressure RP-1 actuators

(2) Two-stage, solenoid piloted fuel igniter valve

(3) Two-stage, solenoid piloted throttle actuator for injector controlled throttling

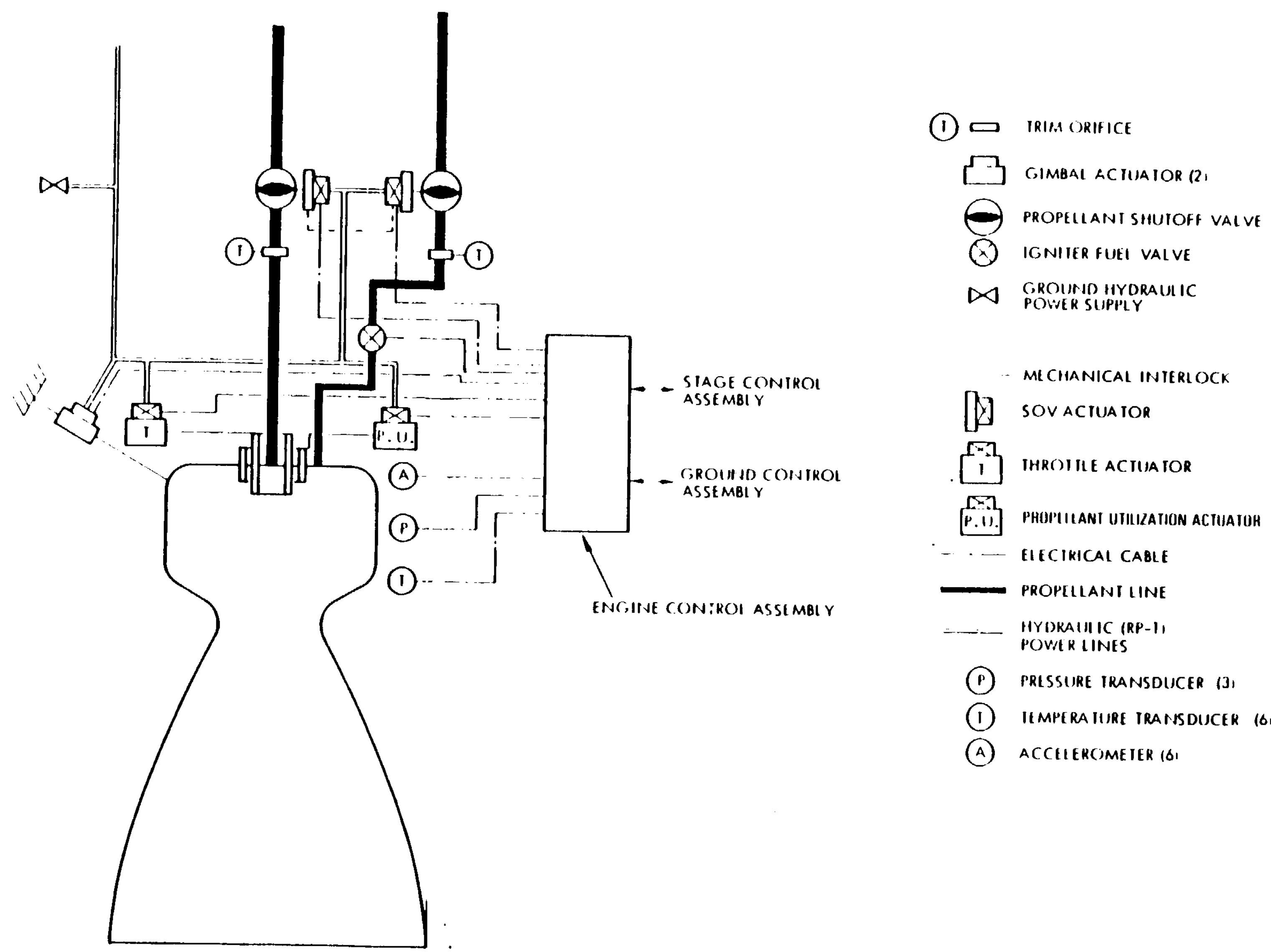
(4) Two stage, solenoid piloted throttle actuator for injector trim mixture ratio control

(5) Pitch and yaw thrust vector control actuators with high pressure RP-1 gimbal actuators

(6) Separate engine control assemblies (ECAs) for each engine plus a stage control assembly (SCA) for vehicle electrical interfacing

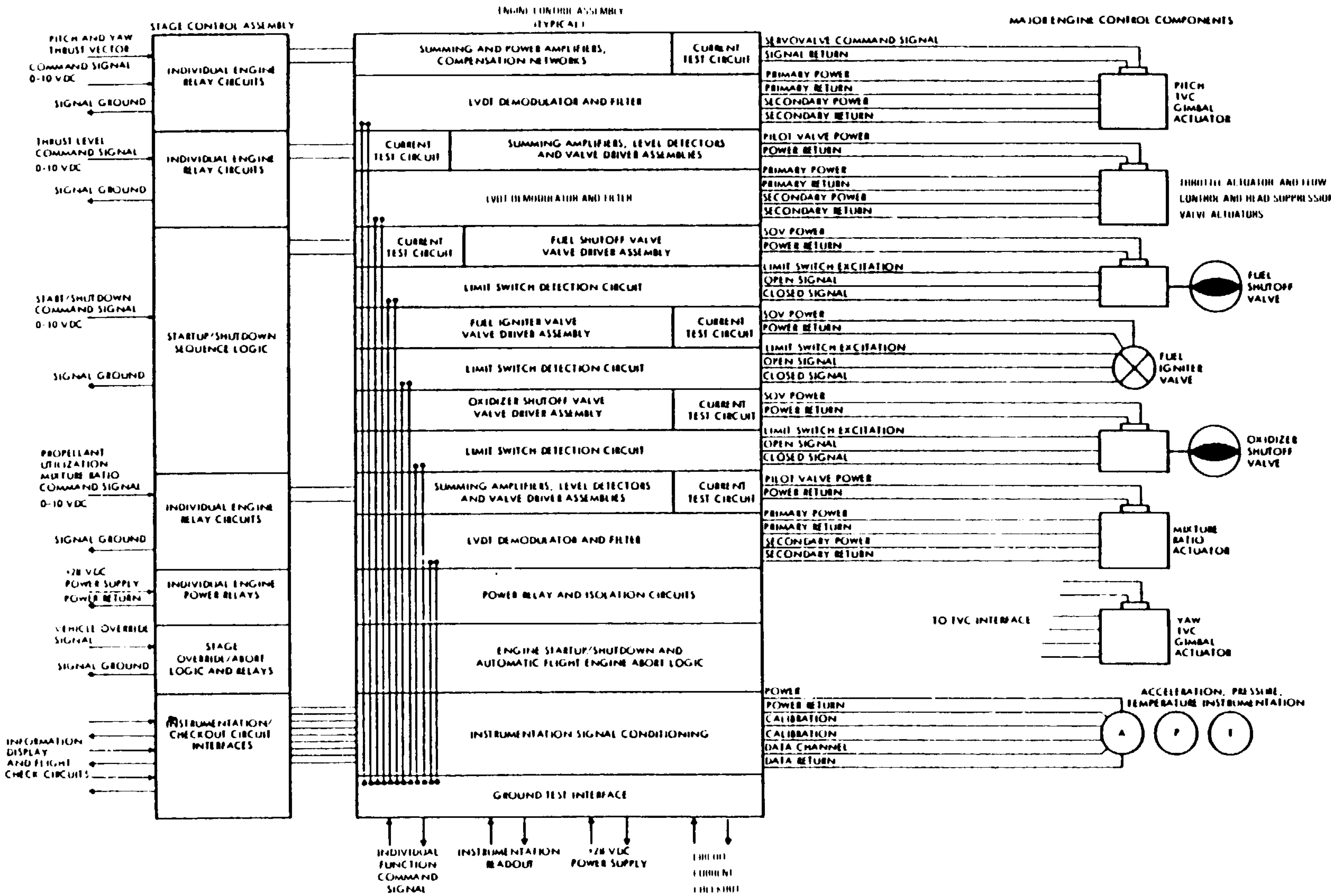
(7) Pressure, temperature and accelerometer instrumentation for ground checkout, flight telemetry data, engine start-up/shutdown sequencing and abort override functions.

Pressure Fed Engine Functional Schematic



The facing figure shows the basic electrical system concept indicating the major vehicle and ground support modules. The stage control assembly (SCA) provides the vehicle interface for all eight engines in the two strap-on boosters. It also relays the power, command signals and instrumentation data to the individual engine control assemblies (ECAs) located at each engine. Each ECA also includes a ground interface connector for engine acceptance and integration test purposes as well as preflight checkout. The electrical continuity of each ECA circuit is also checked in this manner.

Electrical System Schematic



Engine peak power requirements are shown in the facing table. The gimbal actuator, throttle, propellant utilization actuator, and fuel igniter valve actuator power draw is intermittent, however, so that a simple sum of the peak power levels shown yields a very conservative peak power estimate. A 28 Vdc power supply is required. Voltage regulation for transducers is performed within the ECA.

Peak Electrical Power Requirements per Engine

TRW

ITEM	PEAK POWER (WATTS)
FUEL SHUTOFF PILOT VALVE	60
OXIDIZER SHUTOFF PILOT VALVE	60
FUEL IGNITER PILOT VALVE	28
TVC GIMBAL ACTUATORS (2)	5
THROTTLE ACTUATOR	30
PROPELLANT UTILIZATION ACTUATOR	30
INSTRUMENTATION	20
ECA	10

ENGINE INSTRUMENTATION

Engine instrumentation will be required to provide reliable data during engine start-up and flight operations. We recommend that chamber pressure and chamber and nozzle temperatures each be measured by three transducers such that multiple voting schemes can be used. It is desirable to use chamber pressure as a start-up sequencing parameter. It is doubtful, however, whether it can be used as a requirement to open the main fuel valve due to relatively low ignition pressures (based on the use of 1% of total fuel flow at ignition). Therefore, both a fusible wire and chamber pressure transducer could be used for this purpose. An engine shutdown override criterion could be based on a requirement that the chamber pressure reach a specified fraction of the nominal start-up chamber pressure, e.g., 50%, after 1 second. Temperature transducers would be included for determining potential burn-throughs; detailed thermal analyses are required to determine the number sufficient to accomplish this. Redundant accelerometers would be located on the engine head end to measure structural loads. Both the temperature and acceleration data would be monitored for pilot/ground decision making and would be red lined for automatic engine shutdown.

In addition to the above instrumentation, position indicators are required on each main valve and actuator. Linear variable differential transformers (LVDTs) would be used on the throttle propellant utilization and gimbal actuation. Continuous position measurements would be used for closed loop control purposes and data. Limit switches would be used on the flight main shutoff and igniter valves to show the open/closed position status.

5.6 TVC OPTIONS

Differential throttling, LITVC and gimballed-engine TVC systems are all compatible with TRW engine designs and all have been considered in various past systems studies. Gimballed TRW engines have been flown and a gimballed TVC has been baselined by GDSS for this study. While the gimballed system is certainly viable, it has certain disadvantages. It is heavy, especially if high TVC rates are required or if water recovery and reuse is intended. It also requires flex lines to feed the engines and the engine spacing may have to be increased to accommodate the engine motion required to achieve thrust vectoring.

An LITVC system overcomes some of these disadvantages in that it permits use of rigidly mounted fixed engines and therefore does away with the need for flex lines. It also can be made to respond faster than the gimballed system. It is, however, still quite heavy given that it consumes propellant in a relatively inefficient manner. It also is quite complex and expensive, requiring the addition of considerable amounts of plumbing and valves. This additional complexity makes the LITVC less attractive as an option for a recoverable, refurbishable booster.

A particularly attractive option is the differential throttling TVC. The STS LRB will require throttling engines to perform its mission anyway. Therefore, differential throttling capability is there to use at the cost of additional control software; no additional mechanical hardware is required. This system is potentially the lighest weight, lowest cost, most reliable and most compatible with recovery of the three systems considered.

TRW's coaxial injector engine is unique in that it can provide deep throttling with very little performance loss, no combustion instabilities and very rapid throttle response. Past TRW experience with throttling engines proved that throttle response time is dominated by the time it takes to move the throttling sleeve. The sleeve actuator can be readily sized so that this time is very short even in a large engine. As a result, throttle response times of a small fraction of a second, less than 0.2 second from minimum to maximum thrust, has been routinely achieved. This should be well within the TVC response times required for STS flights and certainly faster than can be achieved with a gimballed TVC system. Recent TRW simulations of a launch vehicle similar to STS has indicated that a throttle response time of about 0.7 second with suffice even when absolute, worst-case flight conditions are considered.

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5.7 RECOVERY AND REFURBISHMENT ISSUES

Recovery and reuse has little design impact on the baseline TRW engine with a face shutoff throttling injector and ablative cooling. In general, there may be some items for which reuse may influence design and weight by making fatigue life a dominant consideration in their design. The main design impact of recovery, however, will be the changes necessary to cope with water impact loads. This influences the engine mountings and nozzle design. Past studies (Reference 2) showed that touch down water-slap loads dominated gimbal design loads. This may result in significant added weight, of the order of several thousand pounds/engine. Also, water-slap loads will be a particularly difficult issue for reusable regeneratively cooled engines. It will be difficult to make the nozzle, with its cooling tubes and plumbing, survive water impact and immersion and still meet reliability goals after refurbishment, which could be quite expensive.

An ablative-cooled engine, such that described herein, is much more easily made compatible with water recovery and refurbishment. After recovery, refurbishment consists of removing the old ablatives, cleaning the engine, installing new ablatives and acceptance testing the refurbished engine. Cleanup after recovery is facilitated by use of our face shutoff injector design which prevents saltwater penetration of the injector and propellant lines. Because the ablatives are not reused, the nozzle can be designed so that the ablatives are sacrificed during water slap so that the resulting engine-mount design loads are much reduced. This is easily done by terminating the metal external nozzle shell a short distance downstream of the throat, making the remainder of the nozzle entirely from the ablative material. Because this material would be replaced anyway, it is not necessary to design it to survive water impact. By letting it break off on impact, the nozzle and engine mounting weight are greatly reduced without increasing the danger of damage to the reusable parts of the engine.

An alternative approach to protecting the engines from water-slap loads is to surround the booster base with a protective skirt. Past study (Reference 1), however, indicated that this is a heavy alternative.

5.8 RELIABILITY AND SAFETY

5.8.1 Baseline-Engine Reliability Estimates

A preliminary assessment of the baseline-engine reliability and safety was performed to support Phase I of the STS LRB Feasibility Study. The analysis relies on past TRW and industry test experience. It starts with a failure modes and effects analysis (FMEA) of the major engine components. The probability of occurrence of each failure mode was assessed as being high, moderate, low or very low. Then subsystem reliability estimates are obtained based on actual test experience and relative reliability rankings. Data are presented for competing candidate systems to show how reliability considerations favor the baseline engine design we have selected. Overall reliability for our recommended engine design and a cluster of four engines is presented to conclude this section.

Engine Components/Subsystems Considered in Preliminary FMEA

- COMBUSTION CHAMBER (INCLUDING NOZZLE)
- IGNITION SYSTEM
- PROPELLANT SHUTOFF AND HEAD SUPPRESSION VALVES
- INJECTOR AND ACTUATOR SUBSYSTEM

The ablative cooled chamber has significantly fewer failure modes than other chamber cooling designs and is therefore the preferred approach for high reliability. The regeneratively cooled chamber design has the greatest reliability uncertainty for reuse after saltwater exposure if water recovery is used. The transpiration cooled chamber represents an unknown reliability risk since this design concept has not been characterized as well as the other types of chambers.

Combustion Chamber Failure Modes

Regeratively Cooled Chamber		Ablative Chamber		Duct Cooled Chamber		Transpiration Cooled Chamber	
Failure Mode	Risk	Failure Mode	Risk	Failure Mode	Risk	Failure Mode	Risk
1. Fatigue Stress Failure of Tubes via Thermal Cycles	Low	1. Delamination of Silica Wrap and Steel Shell Burn-through	Low	1. Tube burn-throughs due to Plugging of Fuel Orifices	Very low	1. Local Plugging of Porous Material Due to Fuel Coking	Moderate
2. Galvanic corrosion in Tube Weld Pinholes due to salt-water exposure	Moderate	2. Loss of Bond of Ablative Cone to Steel Shell	Low	2. Thermal Distortion of Metal Exit Cone Due to Unsymmetrical Injector Flow	Low	2. Plugging of Porous Mesh via Saltwater Corrosion Products	Low
3. Tube Burn-throughs due to Unsymmetrical Injector Flow	Low	3. Streaking of Ablative Cone Due to Unsymmetrical Injector Flow	Low	3. Burn-through of Metal Exit Cone due to Velocity Differential Between Ducted Fuel and Hot Gas Flow	Low	3. Corrosion of Porous Mesh at Points of Attachment to Shell	Low
4. Tube Hot Spots or Burn-Through due to Fuel Coking	Low	4. Chunking or Bubbling of Ablative Cone due to Trapped Volatiles	Low	4. Galvanic Corrosion in Tube Weld Pinholes due to Salt-water Exposure	Moderate	4. Other Unknown Failure Modes Resulting from lack of Test History	Moderate
5. Tube Hot Spots due to Fuel Shunting Between Tubes	Very low			5. Tube Burn-throughs or Thin Areas due to repeated missions	Moderate	5. Damage to Mesh on Impact	Moderate
6. Tube Burn-Throughs or Thin Areas due to Repeated Missions	Moderate			6. Tube Damage on Impact	Low		
7. Tube Damage on Impact	Moderate						
8. Inability to detect tube braze defects							

A spark-augmented stand-mounted igniter has significantly fewer failure modes than the hypergolic slug igniter and should therefore be more reliable. It is believed that a spark igniter would perform adequately although it has never been tested in a TRW coaxial injector engine. The hypergolic igniter has proved very reliable in TRW tests, is easy to implement and has been chosen as a tentative baseline for this study.

Ignition Subsystem Failure Modes

Hypergolic Slug Ignition		Stand Mounted Augmented Spark Ignition	
Failure Modes	Risk	Failure Modes	Risk
1. Failure of fuel igniter sequence valve to open on command	Low	1. Leakage of cooling water into fuel/ oxidizer injector pipes	Very low
2. Leakage of downstream burst disk and burning of hypergolic slug	Very low	2. Failure of ground stand LOX/GOX and/or fuel valves to open upon command	Very low
3. Leakage of upstream burst disk and contamination jamming of sequence valve	Very low	3. Leakage of ground stand LOX/GOX and/or fuel valves	Very low
4. Failure of burst disks to rupture	Very low	4. Burn off of the injector pipes and subsequent damage to the combustion chamber/exit cone assembly	Low
5. Plugging of ignition injection ports due to burst disk fragments	Very low	5. Low frequency combustion chamber pressure oscillations (chugging) initiated by ignition transients	Low
6. Leakage of fuel past sequence valve and premature injection of hypergolic slug	Low/ Moderate	6. Plugging of 1st or 2nd stage igniter injection ports	Very low
7. Saltwater corrosion of sequence valve after water impact	Low		
8. Degradation of hypergolic slug during storage	Very low		
9. Low frequency combustion chamber pressure oscillations (chugging) initiated by ignition transients	Low		
10. Rupture of igniter slug injection tubes and leakage of combustion chamber gases	Low		

The head suppression valves have fewer failure modes than the shutoff valves primarily because the head suppression valves cannot fail closed due to built-in position stops and valve internal leakage does not constitute a failure as it does for the shutoff valves.

Propellant Shutoff Valve and Head Suppression Valve Failure Modes

Shutoff Valve (2)		Propellant Utilization Trim Valve (2) (Head Suppression Valve)	
Failure Modes	Risk	Failure Modes	Risk
1. Fail closed or open of the rotary actuator	Low	1. Fail open or closed of the two solenoid pilot valves	Moderate
2. Fail short or open of limit switch	Low	2. Open or short circuit of the two solenoid pilot valves	Very low
3. Fail open or closed of the solenoid pilot valve or failure to vent	Moderate	3. Full open failure of the rotary actuator	Very low
4. Leakage of pressurant gas from pilot valve stage	Low/ Moderate	4. Failure of LVDT position transducer	Low
5. Open or short circuit failure of solenoid pilot valve	Very low	5. Leakage of pressurant gas from pilot valve stages (two per trim valve)	Moderate
6. Fuel or oxidizer leakage past valve butterfly	Low/ Moderate	6. Saltwater corrosion of solenoid pilot valves	Low
7. Saltwater corrosion of solenoid pilot valves	Low		

Hydraulic actuation has fewer failure modes than either pneumatic or electromechanical actuation. Hydraulic actuation has been selected for the purpose of this study.

Injector and Actuator Subsystem Failure Modes

TRW

Electro-Mechanical		Hydraulic		Pneumatic	
Failure Mode	Risk	Failure Mode	Risk	Failure Mode	Risk
1. Leakage of actuator rod dynamic seal	Moderate	1. Leakage of actuator fluid	Low	1. Leakage of pressurant	Low/ Moderate
2. Short or open circuit failure of torque motor or binding of rotor	Low	2. Leakage of static seals	Very low	2. Leakage of static seals	Very Low
3. Binding of ball screw (or rack and pinion)	Low	3. Spring/bellows failure	Very Low	3. Spring/bellows failure	Very Low
4. Failure of LVDT position transducer	Low	4. Failure short or open of solenoid pilot valve	Very Low	4. Failure short or open of solenoid pilot valve	Very Low
5. Misalignment of injector actuator drive causing erratic control response	Low	5. Jamming of solenoid pilot valve in open or closed position	Low	5. Jamming of solenoid pilot valve in open or closed position	Low
6. Binding of the injector sleeve with the pintle due to thermal distortion or breakage of injector slots	Moderate	6. Failure of LVDT position transducer	Low	6. Failure of LVDT position transducer	Low
7. Fuel/oxidizer leaks past injector dynamic seals	Low/ Moderate	7. Binding of the injector sleeve with the pintle due to thermal distortion or breakage of injector slots	Moderate	7. Binding of the injector sleeve with the pintle due to thermal distortion or breakage of injector slots	Moderate
8. Saltwater corrosion	Low	8. Fuel/oxidizer leaks past injector dynamic seals	Low/ Moderate	8. Fuel/oxidizer leaks past injector dynamic seals	Low/ Moderate
		9. Saltwater corrosion	Low	9. Saltwater corrosion	Low

Calculation of the failure mode risk ignores very low risk modes and weights moderate failure risk modes at 2.5 times the unity weighting factor of the low risk modes. No high risk modes have been identified for any of the candidate subsystems and therefore such modes are not included in the reliability calculation.

Subsystem Reliability Prediction Methodology

TRW

- IDENTIFY AND USE DEMONSTRATED RELIABILITY FOR AT LEAST ONE DESIGN CONCEPT OF EACH SUBSYSTEM OR SUBSYSTEM BUILDING BLOCKS TO ESTABLISH RELIABILITY BASELINE
  - USE DEMONSTRATED RELIABILITY DATA FROM AS SIMILAR A UNIT AS POSSIBLE
- QUANTIFY CANDIDATE AND RELIABILITY BASELINE FAILURE MODE RISKS BY CALCULATING Σ (NO. OF LOW FAILURE MODES + 2.5 X NO. OF MODERATE FAILURE MODES)
- MULTIPLY BASELINE SUBSYSTEM FAILURE PROBABILITY BY (Σ SUBSYSTEM FAILURE MODE RISKS FOR CANDIDATE SUBSYSTEM) $\div$ (Σ SUBSYSTEM FAILURE MODE RISKS FOR BASELINE SUBSYSTEM) TO GET CANDIDATE SUBSYSTEM FAILURE PROBABILITY

The regeneratively cooled chamber design is used as a baseline from which the other candidate chamber design reliability estimates are derived. The estimates are based on flight test data reported in Reference 2.

The ablative chamber reliability is estimated from the regenerative chamber test data and the weighted ratio of the number of moderate and low risk failure modes identified for both concepts. However, it should be noted that the observed reliability for the LMDE ablative chamber was 1.0 with no failures in 55 tests. Thus, the estimated failure rate shown is believed to be a conservative estimate. Even so, its superior reliability compared to that of the other candidates is clear.

Reliability Comparisons of Candidate Combustion Chamber Design Concepts

	Susceptibility and Risk of Failure					
	Risk	Quantity	Weighting Factors	Score	Prob. of Failure	Reliability Prediction*
Regenerative Cooled Chamber	High: Moderate: Low: Very Low:	0 3 3 1	 2.5 1 0	 7.5 3.0 10.5	0.0065 (demon-strated)	Rel., Demonstrated: 0.9935 (1 failure out of 153 flights)
Ablative Cooled Chamber	High: Moderate: Low: Very Low:	0 0 4 0	 1 0	 4.0 4.0	0.0025 (estimated)	Rel., Best estimate: 0.9975
Duct Cooled Chamber	High: Moderate: Low: Very Low:	0 2 3 1	 2.5 1 0	 5.0 3.0 8.0	0.0050 (estimated)	Rel., Best estimate: 0.9950
Transpiration Cooled Chamber	High: Moderate: Low: Very Low:	0 3 2 0	 2.5 1 0	 7.5 2.0 9.5	0.0059 (estimated)	Rel., Best estimate: 0.9941

*Reliability predictions are based on the comparative susceptibility of the design candidates to their principal failure modes using the demonstrated reliability of the regenerative design as a baseline.

The hypergolic slug igniter reliability is based on 7150 hot firings of the Saturn H-1 engine. Titan I, Thor and Atlas used a hypergolic slug igniter that was stand-mounted and had a combined flight test history of one ignition failure in 398 flights.

Reliability Comparison of Candidate Ignition Subsystems

	Susceptibility and Risk of Failure					
	Risk	Quantity	Weighting Factors	Score	Prob. of Failure	Reliability Prediction*
Hypergolic Slug Igniter	High: Moderate: Low: Very Low:	0 1/2 4-1/2 5	 2.5 1	 2.5 4.5 7.0	0.00014 (demon-strated)	Rel., Demonstrated = 0.99986 (One failure out of 715 tests for the Saturn H1 Engine using hypergolic slug igniters)
Stand Mounted Augmented Spark Igniter	High: Moderate: Low: Very Low:	0 0 2 4	 2.5 1 0	 0.0 2.0 2.0	0.00004 (estimated)	Rel., Best estimate = 0.99996

*Reliability predictions are based on the comparative susceptibility of the design candidates to their principal failure modes using the demonstrated reliability of the hypergolic slug igniter as a baseline.

The Minuteman III post-boost vehicle propellant shutoff valve had demonstrated a reliability of 198 successes out of 199 tests as of the date of publication of Reference 2. It was observed there, however, that 20% of all shutoff valve failures are associated with extensive valve cycling whereas in this application the shutoff valves experience only one operational cycle per flight. As a result, the shutoff valve reliability has been adjusted to remove a 20% failure rate associated with repeated valve cycling.

Reliability Predictions for Propellant Shutoff Valve and Propellant Head Suppression Valve

	Susceptibility and Risk of Failure					
	Risk	Quantity	Weighting Factors	Score	Prob. of Failure	Reliability Prediction*
Propellant Shutoff Valve	High: Moderate: Low: Very Low:	0 2 4 1	 2.5 1 0	 5.0 4 9.0	0.0050 (demon- strated) 0.0040 (adjusted for PFE applica- tion)	Rel., PFE Applica- tion = 0.9960 Rel., Demonstrated: 0.9950 (one failure out of 199 equivalent missions for the Minuteman Post Boost Vehicle propellant shutoff valve)
Propellant Head Sup- pression (trim) Valve	High: Moderate: Low: Very Low:	0 2 2 2	 2.5 1	 5.0 2.0 7.0	0.0031 (estimated)	Rel., Best estimate: 0.9969

*Reliability predictions are based on the comparative susceptibility of the design candidates to their principal failure modes using the demonstrated reliability of the propellant shutoff valve as the baseline.

The LMDE electromechanical actuator is used as the baseline. It experienced one failure out of 231 equivalent LMDE missions yielding a demonstrated reliability of 0.9957.

Reliability Comparison of Candidate Injector and Actuator Design Concepts

TRW

	Susceptibility and Risk of Failure					
	Risk	Quantity	Weighting Factors	Score	Prob. of Failure	Reliability Prediction*
Electro-Mechanical Actuation	High: Moderate: Low: Very Low:	0 2.5 5.5 0	 2.5 1	 6.25 5.5 11.75	0.0043 (demon-strated)	Actuator Rel., Demonstrated: 0.9957 (one failure out of 231 equivalent LMDE missions)
Hydraulic Actuation	High: Moderate: Low: Very Low:	0 1.5 4.5 3	 2.5 1 0	 3.75 4.50 8.25	0.0030 (estimated)	Rel., Best estimate: 0.9970
Pneumatic Actuation	High: Moderate: Low: Very Low:	0 2 4 3	 2.5 1	 5.0 4.0 9.0	0.0033 (estimated)	Rel., Best estimate: 0.9967

*Reliability predictions are based on the comparative susceptibility of the design candidates to their principal failure modes using the demonstrated reliability of the electro-mechanical actuator as the baseline.

In addition to the mechanical component reliability, one must consider the reliability of electronic controls and engine instrumentation in estimating overall engine reliability. For a preliminary estimate of the reliability of these items, the design and reliability methodology established in Reference 2 is applicable.

In-Flight Controls and Instrumentation Reliability Estimation Methodology

$$R = E^{-\Sigma K_1 \lambda T_1} \times E^{-\Sigma K_2 \lambda T_2}$$

WHERE

R = RELIABILITY

K_I = ENVIRONMENTAL APPLICATION FACTOR

- K_1 = 10 FOR LAUNCH ENVIRONMENT
- K_2 = 1 FOR GROUND ENVIRONMENT

λ = FAILURE RATE OF COMPONENTS IN ORBIT ENVIRONMENT

T_I = OPERATING TIME, HR

- T_1 = 150 SEC = 0.04167 HR FOR LAUNCH ENVIRONMENT
- T_2 = 64 HR FOR GROUND ENVIRONMENT

In-Flight Controls and Operational Instrumentation Reliability

SUBSYSTEM AND ELECTRONIC COMPONENTS		FAILURE RATE (10^{-9}/HR)	RELIABILITY (R)
1. POWER CONVERTER FOR SUBASSEMBLY CONTROLS			0.9_4650*
TRANSFORMER	2	2 X 56 = 112	
TRANSISTORS	6	6 X 50 = 300	
CAPACITORS	11	11 X 3 = 33	
LINEAR ICs	2	2 X 20 = 40	
DIODES	19	19 X 0.8 = 15	
ZENOR DIODES	1	1 X 15 = 15	
RESISTORS	14	14 X 2 = 28	
		$\Sigma\lambda$ = 543	
2. PROPELLANT VALVE CONTROL ELECTRONICS			0.9_4791
DIGITAL ICs	4	4 X 20 = 80	
TRANSISTORS	4	4 X 50 = 200	
RESISTORS	12	12 X 2 = 24	
DIODES	4	4 X 0.8 = 3	
CAPACITORS	6	6 X 3 = 18	
		$\Sigma\lambda$ = 325	

*0.9_4650 = 0.9999650

In-Flight Controls and Operational Instrumentation Reliability (Continued)

SUBSYSTEM AND ELECTRONIC COMPONENTS		FAILURE RATE (10^{-9}/HR)	RELIABILITY (R)
3. INJECTOR THROTTLE CONTROL ELECTRONICS			0.9_4643
LINEAR ICs	6	6 X 20 = 120	
RESISTORS	46	46 X 2 = 92	
DIODES	8	8 X 0.8 = 6	
TRANSISTORS	4	4 X 50 = 200	
CAPACITORS	8	8 X 3 = 24	
TRANSFORMERS	2	2 X 56 = 112	
		$\Sigma\lambda$ = 554	
4. PRESSURE TRANSDUCER ELECTRONICS			0.9_4630
LINEAR ICs	2	2 X 20 = 40	
RESISTORS	14	14 X 2 = 28	
DIODES	4	4 X 0.8 = 3	
CAPACITORS	8	8 X 3 = 24	
TRANSDUCERS	6	6 X 95 = 480	
		$\Sigma\lambda$ = 575	

TRW

In-Flight Controls and Operational Instrumentation Reliability (Continued)

SUBSYSTEM AND ELECTRONIC COMPONENTS		FAILURE RATE (10^{-9}/HR)	RELIABILITY (R)
5. DATA CONVERSION UNIT ELECTRONICS			0.9_38178
DIGITAL ICs	60	60 X 20 = 1200	
LINEAR ICs	4	4 X 20 = 80	
FETs	15	15 X 100 = 1500	
RESISTORS	12	12 X 2 = 24	
CAPACITORS	8	8 X 3 = 24	
		$\Sigma\lambda$ = 2828	

Overall single engine reliability is the product of the component reliabilities. For no valve redundancy, this figure is 0.9800. Use of quadruply redundant valves and actuators raises this value to 0.9968. Use of a booster with a 4-engine cluster designed for safe operation or mission survivability with 3 of the 4 engines functioning raises the combined engine probability to very high levels, especially when redundant components are used (e.g., $R_{3 \text{ of } 4} = (0.9968)^4 + 4\,(0.9968)^3\,(1 - 0.9968) = 0.99994$).

The PFE reliability predictions presented herein are pessimistic mission <u>flight</u> predictions for two reasons. First, the predictions are based on the probability of the engines operating successfully from the initiation of engine ignition while still restrained on the launch pad to the completion of powered flight. The engine reliability predictions are presented in this manner to include the effects of the ignition and propellant valve building block reliabilities on engine countdown reliability. If an engine in the cluster fails to ignite or a propellant fuel valve fails to open, the countdown would be aborted since the cluster would fail to reach the required level of rated thrust, and the cluster restraint would not release to permit launch. Thus <u>no flight abort would occur</u>. Since the launch vehicle will be restrained on the launch pad, it is more realistic to predict flight reliability for the time period from the required level of full rated thrust to completion of powered flight and thus delete the ignition and propellant fuel valve building blocks from flight reliability considerations.

Second, the flight reliability predictions are pessimistic since they were derived primarily from defense missile systems which do not incorporate the higher design margins associated with man-rated systems. Since engine reliability decreases with decreasing design margins, the PFE reliability predictions based on defense missiles reliability data are conservative.

Overall Baseline Engine Concept Reliability

TRW

COMPONENT	COMPONENT RELIABILITY	
	NO VALVE REDUNDANCY	QUADRUPLY REDUNDANT VALVES AND ACTUATORS
1. COMBUSTION CHAMBER (ABLATIVE-COOLED)	0.9975	0.9975
2. IGNITION SYSTEM (HYPERGOLIC SLUG)	$0.9_{3}86$	$0.9_{3}86$
3. PROPELLANT SHUTOFF VALVES (2)	$(0.9960)^2 = 0.99202$	$0.9_{3}8747$
4. PROPELLANT HEAD SUPPRESSION VALVES (2)	$(0.9969)^2 = 0.99381$	$0.9_{4}243$
5. INJECTOR AND ACTUATOR (HYDRAULIC)	0.9970	$0.9_{4}462$
6. POWER CONVERTER ELECTRONICS	$0.9_{4}650$	$0.9_{4}650$
7. PROPELLANT VALVE CONTROL ELECTRONICS	$0.9_{4}791$	$0.9_{4}791$
8. INJECTOR THROTTLE CONTROL ELECTRONICS	$0.9_{4}643$	$0.9_{4}643$
9. PRESSURE TRANSDUCER ELECTRONICS	$0.9_{4}630$	$0.9_{4}630$
10. DATA CONVERSION UNIT ELECTRONICS	$0.9_{3}8178$	$0.9_{3}8178$
OVERALL SINGLE ENGINE RELIABILITY = πR_i =	0.9800	0.9968
RELIABILITY OF 3 OF 4 ENGINES FUNCTIONAL THROUGHOUT FLIGHT	0.99766	0.99994

5.8.2 Historical Reliability Data

Historically, launch vehicles with redundant subsystems have proved much more reliable than those without redundancy.

Historical Reliability Launch Vehicles With Redundancy

TRW

	RELIABILITY	FAILURES
BOOST VEHICLES (ACTUAL)		
SATURN	1.00	(0/APPROX. 25)
STS	1.00	(0/24)*
LIQUID UPPER STAGES (PROJECTED)		
CENTAUR	.96	(3/66)
TRANSTAGE	1.00	(0/35)

*51-L NOT INCLUDED (NONREDUNDANT SRM)

For launch vehicles without redundancy, the engine and airframe (pressurization systems and hydraulic components) have been more failure-prone than the avionics.

C-6

TRW

Historical Reliability Launch Vehicle Without Redundancy

	ENGINE & AIRFRAME	GUIDANCE & CONTROL	TOTALS	FAILURES
BOOST VEHICLES				
ATLAS SLV	0.974	0.983	0.96	(5/117)
TITAN III	0.96	0.984	0.94	(7/124)
DELTA	0.975	0.988	0.96	(3/80)
TOTAL	0.969	0.984	0.953	
UPPER STAGES				
CENTAUR	0.92	0.985	0.91	(6/66)
TRANSTAGE	0.91	0.94	0.86	(5/35)
TOTAL	0.93	0.97	0.89	

Engine-out capability has historically been the key to an excellent record for launch vehicles with redundant systems.

Rocketdyne Flight History Turbo-Pump Engines

	ENGINE TYPE	FLIGHT ENGINES	FAILURES	LOSS OF VEHICLE
	THOR	316	3	YES
NO ENGINE-OUT CAPABILITY	DELTA	76	0	NO*
	ATLAS	839	14	YES
ENGINE-OUT CAPABILITY WITH REDLINES	SATURN (M-1, F-1, J-2)	303	3	NO
	SPACE SHUTTLE (SSME)	75	1	NO

*SOME DELTA FAILURES ATTRIBUTED TO NON-ROCKETDYNE ENGINES

This chart compares the engine reliability data that exists for the three booster engine candidates: pressure- and pump-fed liquid-propellant engines and solid-propellant motors. The best demonstrated reliability has been achieved with pressure-fed liquid-propellant rocket engines with no known in-flight failures. Solid-propellant rocket motors have the next best apparent reliability, although the data shown do not include Minuteman which has had many failures. Inclusion of the Minuteman data would make the solid propellant reliability look less favorable. According to the data shown, turbo-pumped engines have demonstrated the poorest reliability overall.

Engine Flight Reliability Experience

ENGINE TYPE	FLIGHT ENGINES OR EQUIVALENT FLIGHTS	FAILURES	SUCCESS RATE
PRESSURE-FED LIQUIDS			
TRW DELTA STAGE 2	67	NONE	100%
AEROJET DELTA STAGE 2	MANY	NONE KNOWN	
TRW LMDE	APPROX. 10	NONE	
TRANSTAGE	APPROX. 33	NONE KNOWN	
TURBO-PUMPED LIQUIDS			
SSME			
100%	486	1	0.9979
100-104%	337	3	0.9881
104-109%	168	9	0.9464
ALL OTHER ENGINES	1534	20	0.9870
TOTALS	2525	33	0.9869
SOLID PROPELLANTS			
DELTA	500	1	0.994*
TITAN, SHUTTLE	200	3	

*MINUTEMAN NOT INCLUDED - MANY FAILURES.

5.8.3 Preliminary Hazard and Safety Analysis

5.8.3.1 Hazard Analysis

The first step in safety assessment is the establishment of the hazardous events which could occur for each major subsystem of the engine and then the development of a list of hazardous events which might occur. Listing of the events does not imply that they will occur in the designs as proposed. Rather it is a type of safety checklist to enable the subsequent concept development to include provision for these safety items. After establishment of the possible hazardous event list, a list of design features which can reduce risk is made. The hazards analysis that follows includes some of the failure modes information presented in the Failure Modes and Effects Analysis (FMEA) in subsection 5.8.1.

Engine Subsystem and Hazard Condition	Means of Eliminating or Controlling Hazard
A. Propellant Manifolds	
1. Prelaunch icing of LOX bellows followed by rupture or cracking of the bellows during gimbaling of the engine.	1. Protective lightweight bag or jacket placed over the bellows to prevent formation of ice on the bellows.
2. Excessive propellant flow pressure drop across bellows convolutions.	2. Design may require lightweight flex hose internal to the bellows (to eliminate flow disturbances caused by bellows convolutions) with pressurized gas between bellows and flex hose to prevent rupture of the flex hose.
3. Failure of the transducers such that full manifold pressure is indicated when in fact insufficient pressure exists, or no pressure is indicated when in fact full pressure exists. Either failure, while not affecting flight reliability, does affect engine ignition sequencing.	3. Majority voting circuits for the chamber pressure transducers are recommended for the propellant manifold pressure transducers in order to reliably prevent ignition start-up when pressure actually exists. Majority voting circuitry designed such that any one of the transducers may fail yet the true state of manifold pressure is obtained from the remaining transducers. For example, if two out of three transducers indicate abnormally low manifold pressure, the engine shutdown sequence is initiated.
B. Propellant Valves and Control Electronics	
1. External leakage of the LOX valve resulting in freezing of the nozzle gimbal actuators, injector actuator, etc. External leakage of the fuel represents a potential fire hazard.	1. Welded bellows seals may be required on valve shaft seals to prevent external leakage of LOX or fuel.

Hazard Analysis (Continued)

TRW

Engine Subsystem and Hazard Condition	Means of Eliminating or Controlling Hazard
B. Propellant Valves and Control Electronics (Continued)	
2. Valve or electronic circuit fails such that propellant valves fail closed and engine ignition cannot be initiated.	2. Use of parallel propellant valves for the booster engines would protect against the fail closed mode. In the case of such a failure, however, no flight abort occurs since the engine cluster will not achieve the required fraction of rated thrust and mission launch is thus prevented.
3. LOX or fuel valve internal leakage such that LOX leak may freeze the fuel or cause excessive ignition pressure. Leakage of both valves could result in the formation of an explosive jell.	3. Adverse effects of propellant shutoff valve internal leaks may be minimized by having a minimum time delay between opening of propellant pre-valves and start of engine ignition. In this way the probability of developing either a fuel-rich ignition start and possible excessive chamber pressure or a freezing of the fuel is minimized. Pressure transducer in LOX manifold may detect leakage past shutoff valve and prevent ignition sequence.
4. Incorrect opening rates or sequencing of the LOX, TEA or fuel valves resulting in engine chugging if fuel and LOX rates are too slow, or excessive vibration and thrust level buildup if opening rates are too high.	4. Chamber pressure and propellant manifold transducers together with timing circuits will be required to initiate engine shutdown sequence if any of the valve sequences or opening rates are incorrect. If insufficient pressure or excessive pressure decay is noted during TEA injection, engine shutdown is initiated. If chamber or manifold pressure rise rate is too fast, engine shutdown should be initiated.

Hazard Analysis (Continued)

TRW

Engine Subsystem and Hazard Condition	Means of Eliminating or Controlling Hazard
B. Propellant Valves and Control Electronics (Continued)	
5. Valve or electronic circuits fail such that either propellant valve fails open and engine shutdown sequence is jeopardized.	5. Throttleable injector acts as a redundant backup to propellant shutoff valves. If either propellant valve (fuel or oxidizer) fails in the open position, the injector is also closed for engine shutdown with no penalty to the mission.
C. Throttleable Injector and Control Electronics	
1. Electronic circuit failure such that inadvertant minimum thrust level or engine shutdown occurs.	1. Electronic fault detection circuit designed such that circuit failure will result in injector positioning to the wide open position. Remaining engines can be signaled to a slight thrust cutback to balance wide open injector. For thrust termination, the propellant valves are programmed shut without a need for shutoff of the injector. As an alternative to a fault detection circuit, majority voting for two out of three control circuits can be utilized such that any one of the three control circuits could fail yet the true position of the injector would be known and a true actuation signal fed to the injector actuator.
2. Electronic circuit failure such that maximum thrust level is erroneously signaled to an engine in the cluster.	2. The remaining engines in the cluster are throttled back to compensate for the wide open injector. For thrust termination the propellant valves are programmed shut without a need for shutoff of the injector.

Hazard Analysis (Continued)

TRW

Engine Subsystem and Hazard Condition	Means of Eliminating or Controlling Hazard
D. Gimballed TVC Actuator and Drive Circuits	
1. Electronic circuits failure such that TVC deflection occurs when not required or no deflection occurs when a deflection is required.	1. Electronic circuits to be designed incorporating majority voting from three separate control circuits. This means that any one of the three circuits can fail open or short and yet the correct TVC signal is applied to the actuator servovalve.
E. Combustion Chamber Transducers	
1. Failure of engine chamber pressure transducers such that given a fail closed mode of a propellant valve, the transducer indicates full chamber pressure when in fact no chamber pressure exists or conversely a transducer indicates less than full chamber pressure when in fact full pressure exists.	1. Multiple pressure transducers are incorporated in each engine's combustion chamber. Majority voting circuitry to be designed such that any one of the transducers may fail yet the true state of chamber pressure is obtained from the remaining transducers and used for engine control purposes. For example, if two out of the three transducers indicate abnormally low chamber pressure, then engine propellant valves and injector are programmed closed to prevent loss of propellant.

Hazard Analysis (Continued)

Engine Subsystem and Hazard Condition	Means of Eliminating or Controlling Hazard
F. Chamber Nozzle and Head End Temperature Sensors	
1. These sensors would be used as malfunction detectors to permit rapid engine shutdown in the event that hot spots were detected. However, since temperature sensors, like pressure transducers, can give false readings, it is important that the temperature sensors do not give false indications of hot spots.	1. Redundant temperature sensors where both sensors must indicate a hot spot or majority voting temperature sensors are recommended. The redundant temperature sensors would significantly reduce the electronic parts count as compared to majority voting circuitry and therefore may represent the more practical design approach to employ.

5.8.3.2 Safety Analysis

LOX-RP-1 properties important for safety are shown on the facing chart. The relatively high flash point and low vapor pressure for RP-1 means that the probability of reaching a minimum flammability concentration of 1.3% is low. Also RP-1 vapors are lighter than air and will therefore disperse, reducing the possibility that an explosive mixture with air will be formed. Cryogenic LOX in contact with RP-1 forms a pressure-sensitive explosive jell which becomes more sensitive with time until a spontaneous explosion can occur. For this safety hazard, however, both LOX and RP-1 leakage must occur.

Safety Data for LOX/RP-1 Propellant System

	RP-1
FLASH POINT	+134°F
AUTO IGNITION TEMPERATURE	450°F
VAPOR PRESSURE	0.01 PSIA AT 100°F
BOILING POINT	422°F
FLAMMABILITY RANGE	1.3% - 8% WEIGHT RATIO
TNT EQUIVALENTS	1 LB LOX/RP-1 IN A 2.25/1 WEIGHT RATIO = 1.23 LB OF TNT
EXPLOSIVE YIELD, MEAN	APPROX 4% OF THEORETICAL MAX.

The facing table presents safety and launch reliability considerations of the hypergolic-slug igniter and the spark augmented stand-mounted igniter. The spark-augmented stand-mounted igniter is the preferred candidate from a safety standpoint since the safety hazard associated with a pyrophoric (burns spontaneously in air) hypergolic slug does not exist.

The spark-augmented igniter is also preferred for high launch reliability. Ground check of the igniter can be performed as often as required to verify performance. In addition, engine complexity is reduced since all ignition components are launch facility rather than booster mounted.

TRW

Safety and Launch Reliability Comparison of the TEA Hypergolic Slug Igniter Versus the Spark Augmented Stand Mounted Igniter

Triethylene Aluminum (TEA) Hypergolic Slug	Spark Augmented Stand Mounted Igniter
Safety:	Safety:
Triethylene Aluminum is pyrophoric - leaking burst disk in TEA slug is potential fire hazard.	Multiple checkouts of stand-mounted igniter possible to verify that propellant valves do not leak and cause safety hazard. Neither fuel nor oxidizer is pyrophoric.
Launch Reliability:	Launch Reliability:
1) Additional engine complexity introduced using burst disks and sequencing valve.	1) Engine complexity reduced by removing sequencing valves and spark source electronics to launch facility.
2) Prelaunch checkout ignition impossible.	2) Prelaunch checkout of ignition possible.

5.9 COST SUMMARY

In support of this effort, recurring and nonrecurring cost estimates were developed for various engine cooling/TVC options. All costs are based on TRW low-cost coaxial injector engine concepts. The bases of the cost estimates were a combination of vendor quotes and inflation-corrected historical data from Reference 4. Details of the cost models are given in Appendix B.

The results are summarized in the facing tables. It should be noted that the recurring costs of the TRW coaxial injector engines are much lower than for engines with other injector types. The nonrecurring costs are also low, reflecting the reduced development testing required to solve the combustion instability problems of other injector engines.

PFE Recurring Cost Summary ($K)

TRW

	BASELINE EXPENDABLE WITH ABLATIVES AND DIFFERENTIAL THROTTLING	BASELINE RECOVERABLE WITH ABLATIVES AND DIFFERENTIAL THROTTLING	BASELINE EXPENDABLE WITH ABLATIVES AND LITVC	BASELINE RECOVERABLE WITH ABLATIVES AND LITVC	REGENERATIVE COOLED EXPENDABLE AND DIFFERENTIAL THROTTLING	REGENERATIVE COOLED RECOVERABLE AND DIFFERENTIAL THROTTLING	REGENERATIVE COOLED EXPENDABLE WITH LITVC	REGENERATIVE COOLED RECOVERABLE WITH LITVC
C_{ENG}	481	505	481	505	1188	1425	1188	1425
$C_{COOLING}$	161	161	161	161				
C_{TVC}	0	0	135*	135*	0	0	135*	135*
C_{RFC} AND REFURB / FLIGHT	0	336	0	386	0	225	0	275
AV C_{ENG} FOR 10 FLIGHTS	642	369	777	428	1188	345	1323	404
AV C_{ENG} FOR 20 FLIGHTS	642	353	777	407	1188	285	1323	339

*DOES NOT ACCOUNT FOR COST OF INCREASED PROPELLANT LOAD

PFE Nonrecurring Cost Summary ($K)

TRW

ENGINE TYPE	BASELINE EXPENDABLE WITH ABLATIVES AND DIFFERENTIAL THROTTLING	BASELINE RECOVERABLE WITH ABLATIVES AND DIFFERENTIAL THROTTLING	BASELINE EXPENDABLE WITH ABLATIVES AND LITVC	BASELINE RECOVERABLE WITH ABLATIVES AND LITVC	REGENERATIVE COOLED EXPENDABLE AND DIFFERENTIAL THROTTLING	REGENERATIVE COOLED RECOVERABLE AND DIFFERENTIAL THROTTLING	REGENERATIVE COOLED EXPENDABLE WITH LITVC	REGENERATIVE COOLED RECOVERABLE WITH LITVC
NRC FOR ENGINE	194 6	204 3	243 3	255 4	360 0	437 2	414 2	497 5
NRC FOR FACILITIES	61 1	61 1	61 1	61 1	61 1	61 1	61 1	61 1
TOTAL NRC	255 7	265 4	304 4	316 5	421 1	498 3	475 3	558 6

6. PRESSURIZATION SYSTEM STUDIES

PRESSURIZATION SYSTEMS

TRW analyzed eight (8) pressurization system operational concepts defined by GDSS. Analysis summaries of each option are presented in chart form and a weight summary chart compares all options. The candidates were evaluated using a constant set of ground rules provided by GDSS (flows, temperatures, pressures, pressure drops, etc.) which allows relative ranking to be done for the selection process.

All options involve heating the pressurant gas (helium) by some form of heat exchanger. A propellant tank volume of 17,610 cu ft represents the combined volume of oxidizer (LOX) and fuel (RP-1) tanks for a reference booster design. To minimize system weight it is desirable to start with cold high-density helium for efficient packaging and end the mission with warm low density helium residuals. In all options a propellant tank ullage regulated pressure of 700 psia and temperatures of 800°R was defined.

The H_2/O_2/catalytic heat exchanger options resulted in the lightest weight and most simple/reliable designs. On this basis TRW concurs with the GDSS tentative selection of this concept as their baseline. Likewise, a cascaded multipressurant tank design is very attractive compared with single tank options. These are discussed in greater detail below.

HELIUM TANK WEIGHT

All helium tank weights presented in the pressurization system option studies were scaled from a point design supplied by Structural Composites Industries (SCI). All tanks were assumed to be spheres and weight-scaled to the reference by the products of pressure and volume as follows:

$$W = \frac{PV}{P_o V_o} W_o$$

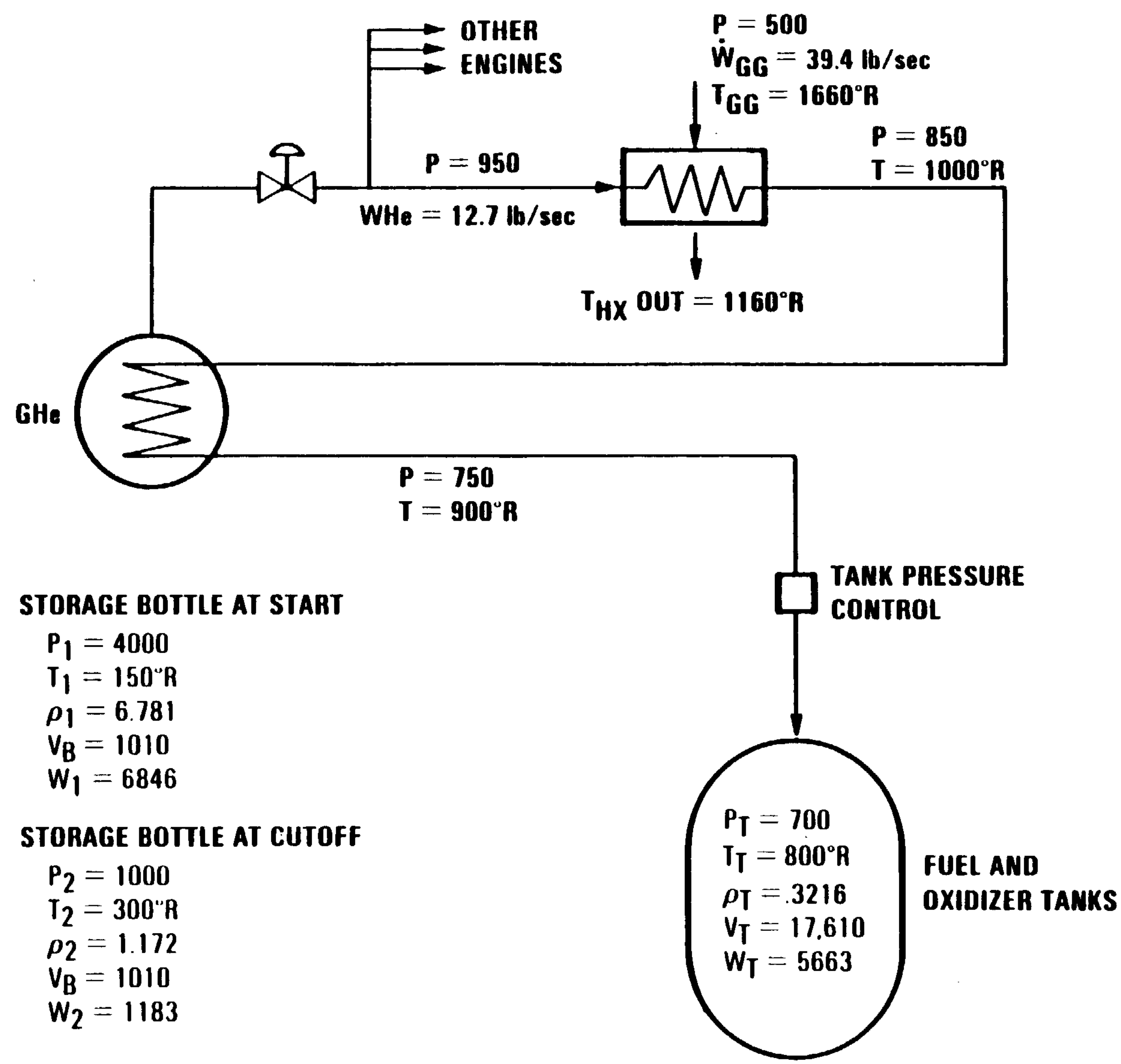
Details of the SCI reference design are as follows:

- Shape: sphere
- Pressures: operating/proof/burst = 6000/7500/9000 psia
- Volume: 1767.5 cu ft (15 ft dia)
- Type: fiber overwrapped metal liner
- Liner: Ti-6Al-4V, 0.140-in thick wall
- Composite: graphite epoxy, σ = 900 ksi delivered
- Weight: 12,355 lb

PRESSURIZATION SYSTEM - OPTION 1

This option features a gas generator (GG) heat exchanger using LOX/RP-1 propellants from the main booster tanks to heat the helium flow. Prior to flowing into the propellant tanks, the heated helium passes through a heat exchanger coil in the storage bottle to lower the density (i.e., mass) of the residual helium via natural convection heat transfer during blowdown. The GG is arbitrarily shown as heating one-fourth of the total required helium flow (i.e., four GGs of this size are required). The GG produces 17,341 lb_f thrust which reduces the booster main LOX/RP-1 propellant load by 6831 pounds. Details of the coil in the bottle and the GG heat exchanger are presented later in this section.

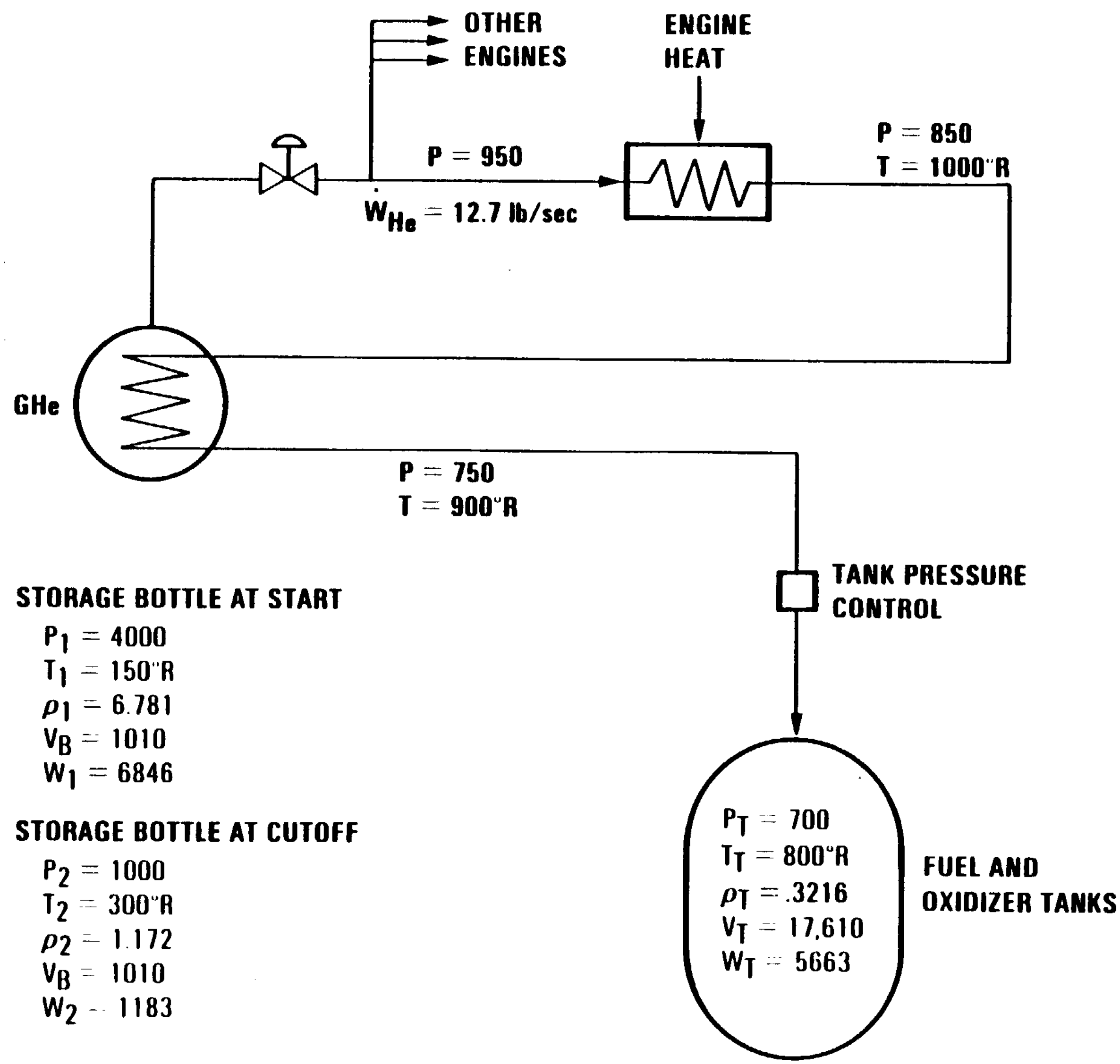
Pressurization System – Option 1



PRESSURIZATION SYSTEM - OPTION 2

This option features heat exchangers using the booster main engines as heat sources to heat the helium flow. Prior to flowing into the propellant tanks, the heated helium passes through a heat exchanger coil in the storage bottle to lower the density (hence mass) of the residual helium via natural convection heat transfer during blowdown. The engine heat exchanger extracts heat energy from the exhaust gases which reduces performance through a drop in I_{sp} of about 1 second. This results in an increase in the booster LOX/RP-1 propellant load of 3965 pounds. Details of the coil in the bottle and engine heat exchangers are presented later in this section.

Pressurization System – Option 2

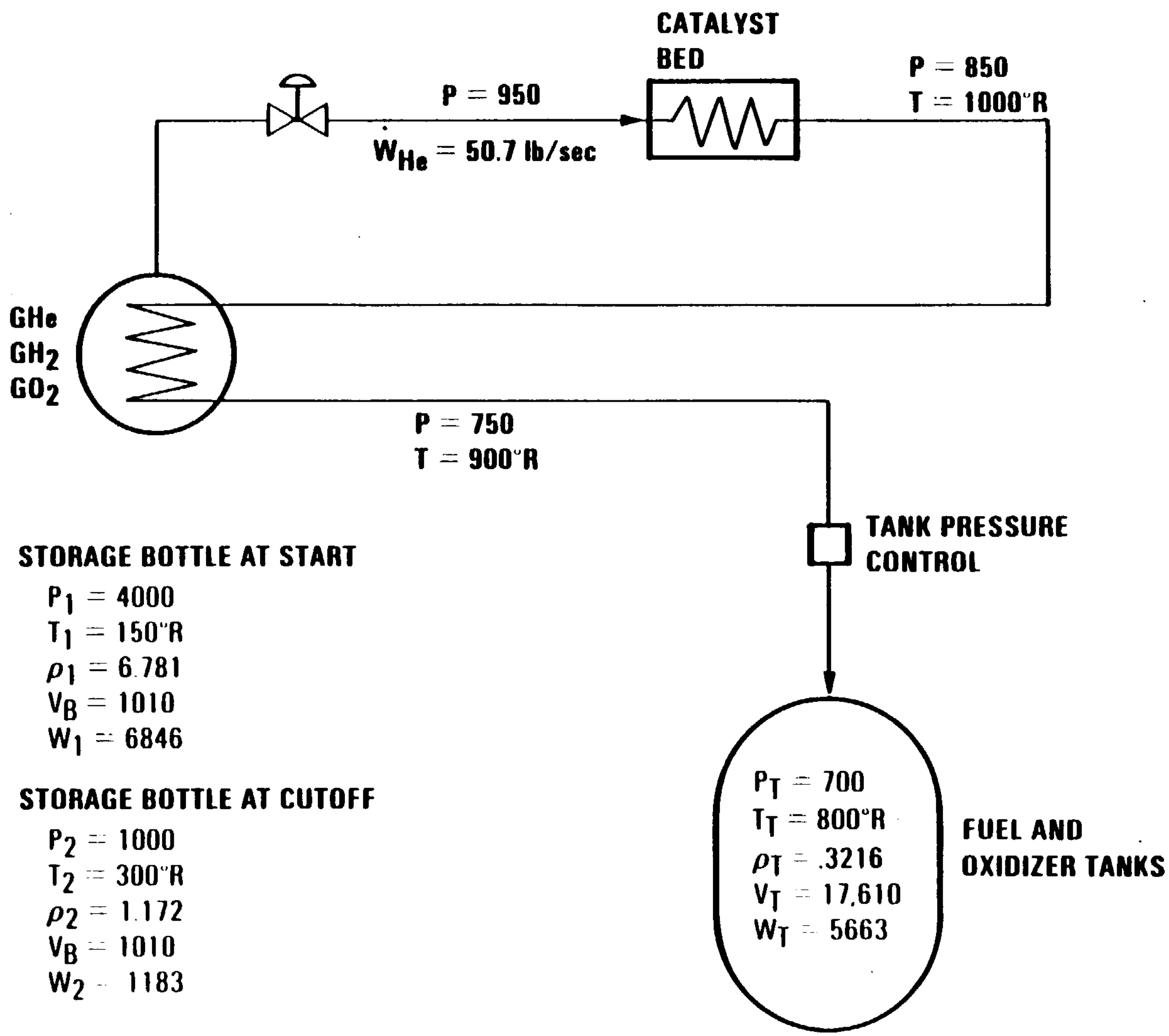


PRESSURIZATION SYSTEM - OPTION 3

This option features a catalytic converter system to heat the helium flow. In this system small amounts of hydrogen and oxygen (of the order of a few percent by volume) are mixed with the helium in the storage bottle. According to a preliminary literature survey, these small quantities of H_2 and O_2 are below the flammability or explosive limits for the mixture and impose no safety hazard. When this mixture is passed through an appropriate catalytic converter, the exothermic reaction converting H_2 and O_2 into water (steam) raises the bulk helium temperature. The outlet temperature is determined by the original fractional amounts of H_2 and O_2 in the helium and the converter design.

Prior to flowing into the propellant tanks, the heated helium passes through a heat exchanger coil in the storage bottle to lower the density (hence mass) of the residual helium via natural convection heat transfer during blowdown. Details of the coil in the bottle and catalytic heat exchanger are presented later in this section.

Pressurization System – Option 3



STORAGE BOTTLE AT START

$P_1 = 4000$
$T_1 = 150°R$
$\rho_1 = 6.781$
$V_B = 1010$
$W_1 = 6846$

STORAGE BOTTLE AT CUTOFF

$P_2 = 1000$
$T_2 = 300°R$
$\rho_2 = 1.172$
$V_B = 1010$
$W_2 = 1183$

CALCULATIONS FOR PRESSURIZATION OPTIONS 1 THROUGH 3

<u>Heat Exchanger (helium side)</u>

$$\dot{W}_{He} = \frac{5663 \text{ lb}}{(111.7 \text{ sec})(4 \text{ eng})} = 12.7 \text{ lb/sec helium flow rate per engine}$$

$$T_{in} = 150 \left(\frac{950}{4000}\right)^{0.4} = 84^{o}R \text{ at start}$$

$$T_{in} = 300^{o}R \text{ at cutoff (assumed - see chart on coil in bottle)}$$

$$T_{in} = \frac{84+300}{2} = 192^{o}R \text{ average}$$

$$T_{out} = 1000^{o}R$$

$$\Delta T = 1000-192 = 808^{o}R$$

$$Q = W_{He}\ Cp\ \Delta T = (5663)\ (1.24)\ (808) = 5{,}673,\ 873 \text{ Btu}$$

Gas Generator (Option 1 only)

LOX/RP-1 propellants

$T_{in} = 1660^{o}R$

$T_{out} = 1160^{o}R$

$MR = 0.35$

$Cp = 0.645$ Btu/lb^{o}R

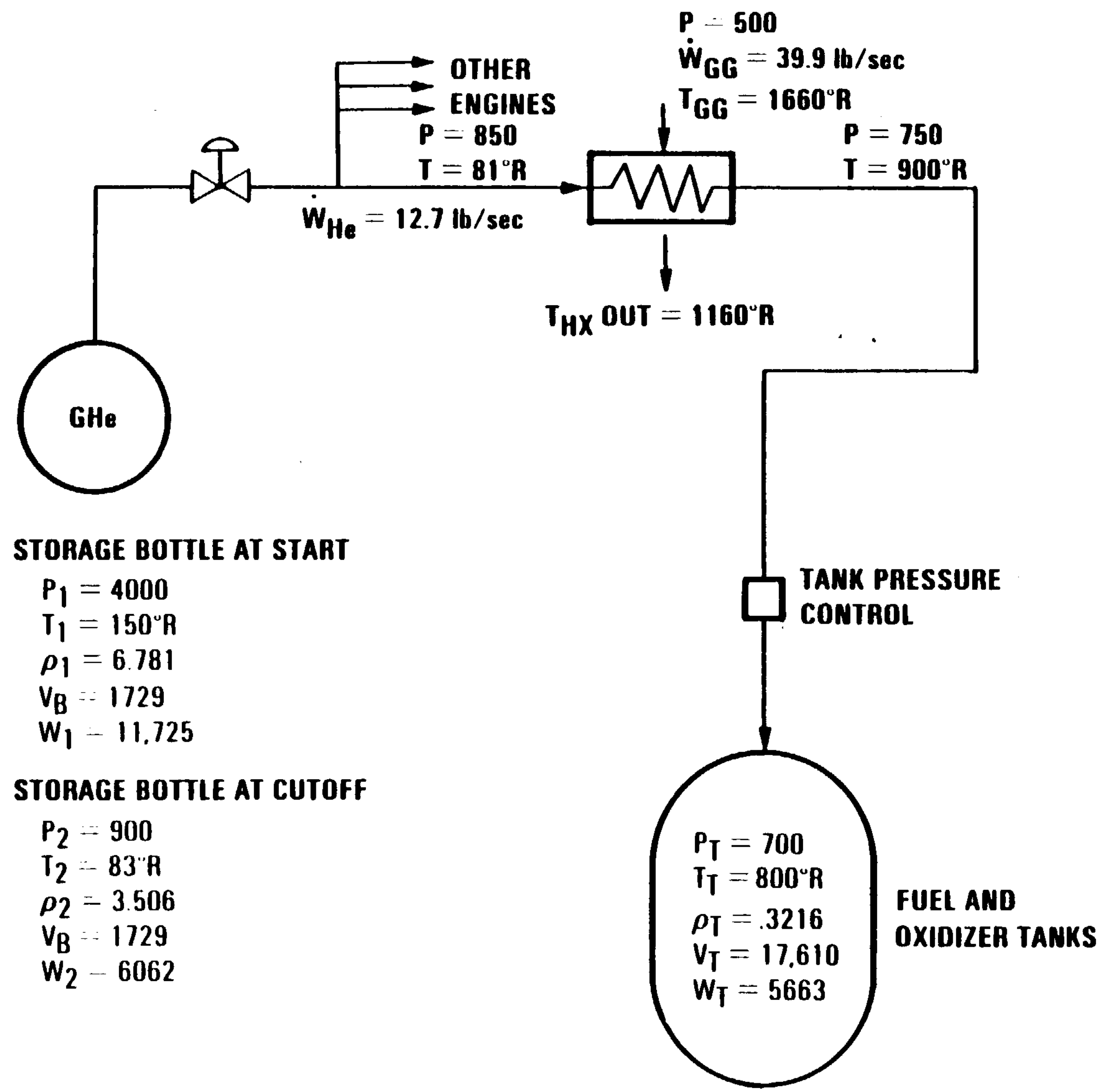
$$W_{GG} = \frac{Q}{\Delta T\ Cp} = \frac{5,673,873}{500\ (0.645)} = 17,593 \text{ lb propellant}$$

$$\dot{W}_{GG} = \frac{17,593 \text{ lb}}{(111.7 \text{ sec})(4 \text{ eng})} = 39.4 \text{ lb/sec GG flow rate per engine}$$

PRESSURIZATION SYSTEM - OPTION 4

This option features a gas generator (GG) heat exchanger using LOX/RP-1 propellants from the main booster tanks to heat the helium flow. The GG is arbitrarily shown as heating one-fourth of the total required helium flow (i.e., four GGs of this size are required). The GG produces 17,580 lb_f thrust which reduces the booster main LOX/RP-1 propellant load by 6925 pounds. Details of the GG heat exchanger are presented later in this section.

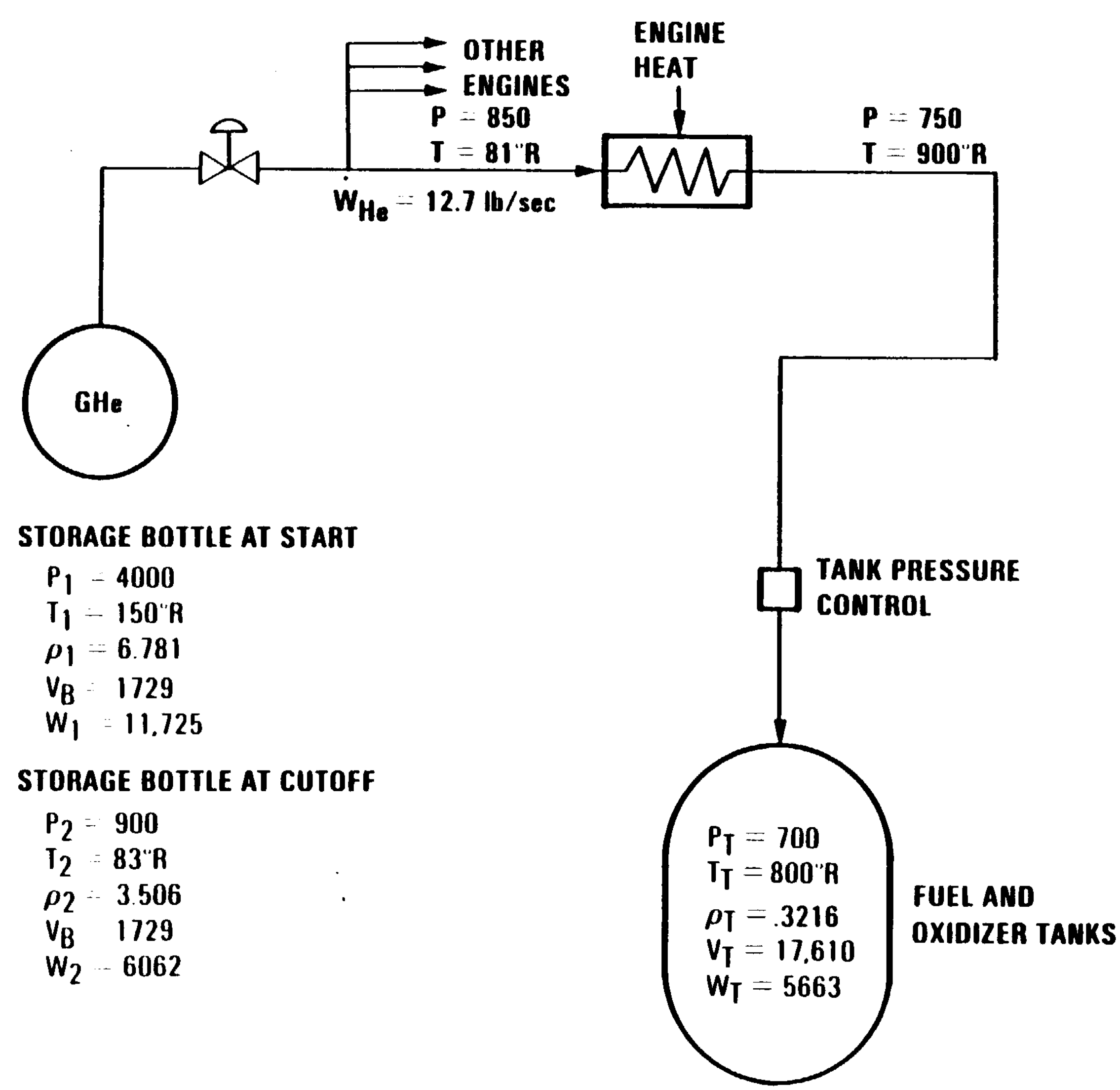
Pressurization System – Option 4



PRESSURIZATION SYSTEM - OPTION 5

This option features heat exchangers using the booster main engines as heat sources to heat the helium flow. The engine heat exchangers extract heat energy from the exhaust gases which reduces performance through a drop in I_{sp} of about 1 second. This results in an increase in the booster LOX/RP-1 propellant load of 4019 pounds. Details of the engine heat exchangers are presented later in this section.

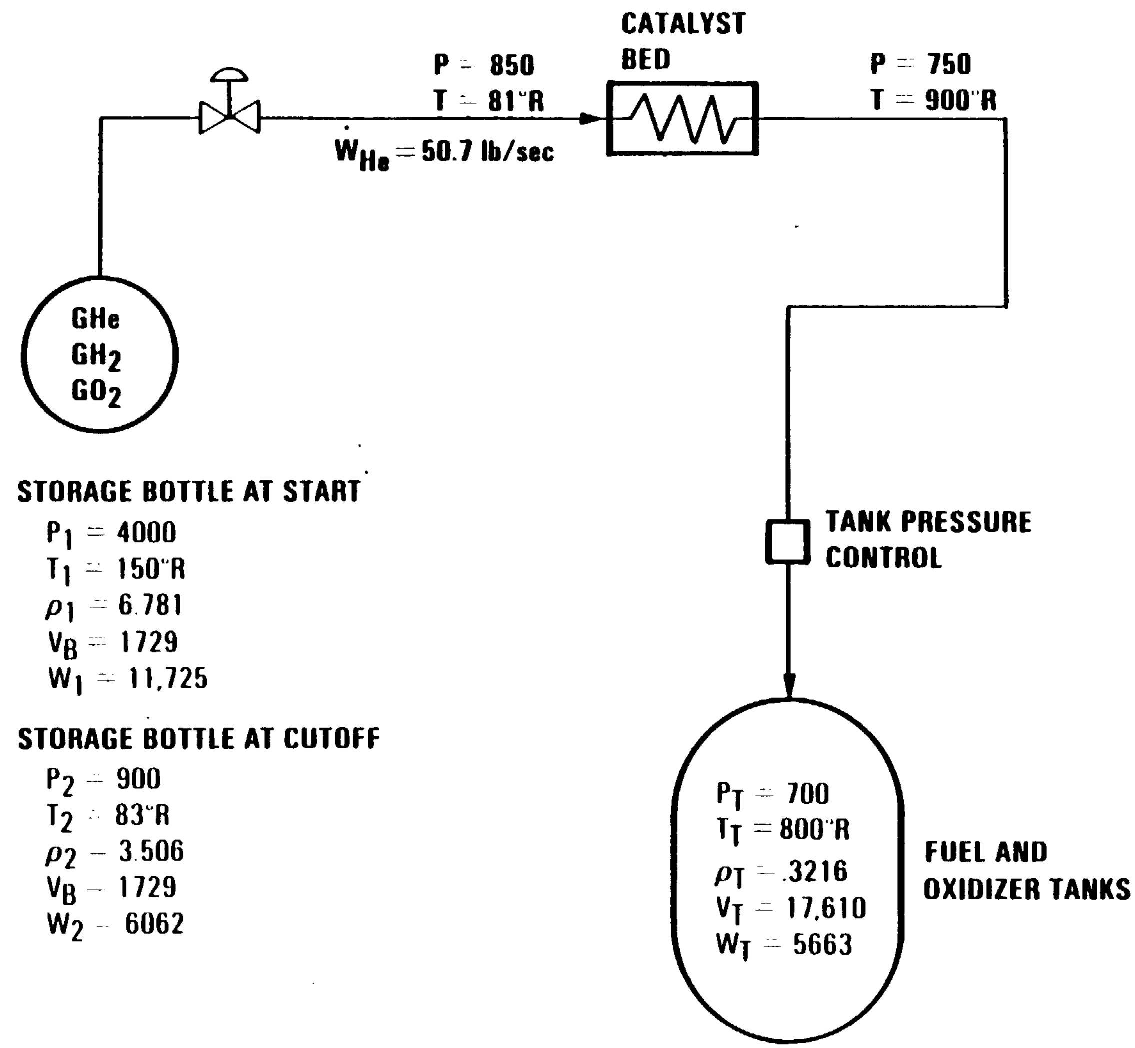
Pressurization System – Option 5



PRESSURIZATION SYSTEM - OPTION 6

This option features a catalytic converter system to heat the helium flow. In this system small amounts of hydrogen and oxygen (of the order of a few percent by volume) are mixed with the helium in the storage bottle. These small quantities of H_2 and O_2 are below the flammability or explosive limits for the mixture and impose no safety hazard. When this mixture is passed through an appropriate catalytic converter the exothermic reaction converting H_2 and O_2 into water (steam) raises the bulk helium temperature. The outlet temperature is determined by the original fractional amounts of H_2 and O_2 in the helium and the converter design. Details of the catalytic heat exchanger are presented later in this section.

Pressurization System – Option 6



CALCULATIONS FOR PRESSURIZATION OPTIONS 4 THROUGH 6

Heat Exchanger (helium side)

$$\dot{W}_{He} = \frac{5663 \text{ lb}}{(111.7 \text{ sec})(4 \text{ eng})} = 12.7 \text{ lb/sec helium flow rate per engine}$$

$$T_{in} = 150 \left(\frac{850}{4000}\right)^{0.4} = 81^{o}R$$

$$T_{out} = 900^{o}R$$

$$\Delta T = 900-81 = 819^{o}R$$

$$Q = W_{He}\ Cp\ \Delta T = (5663)\ (1.24)\ (819) = 5{,}751{,}116 \text{ Btu}$$

Gas Generator (Option 4 only)

LOX/RP-1 propellants

$$T_{in} = 1660^{o}R$$

$$T_{out} = 1160^{o}R$$

$$MR = 0.35$$

$$Cp = 0.645 \text{ Btu/lb}^{o}R$$

$$W_{gg} = \frac{Q}{\Delta T\ Cp} = \frac{5,751,116}{500\ (0.645)} = 17,833 \text{ lb propellant}$$

$$\dot{W}_{GG} = \frac{17,593 \text{ lb}}{(111.7 \text{ sec})(4 \text{ eng})} = 39.9 \text{ lb/sec GG flow rate per engine}$$

PRESSURIZATION SYSTEM - OPTION 7

This option features a hydrazine (N_2H_4) gas generator (GG) heat exchanger to heat the helium flow. The heated helium pressurizes both the oxidizer (LOX) tank and the N_2H_4 tank. The GG combustion products pressurize the fuel (RP-1) tank. No overboard dumping of GG exhaust is required. Details of the GG heat exchanger are presented later in this section.

Pressurization System—Option 7

TRW

N_2H_4 STORAGE BOTTLE AT CUTOFF

$P_H = 750$

$T_H = 800°R$

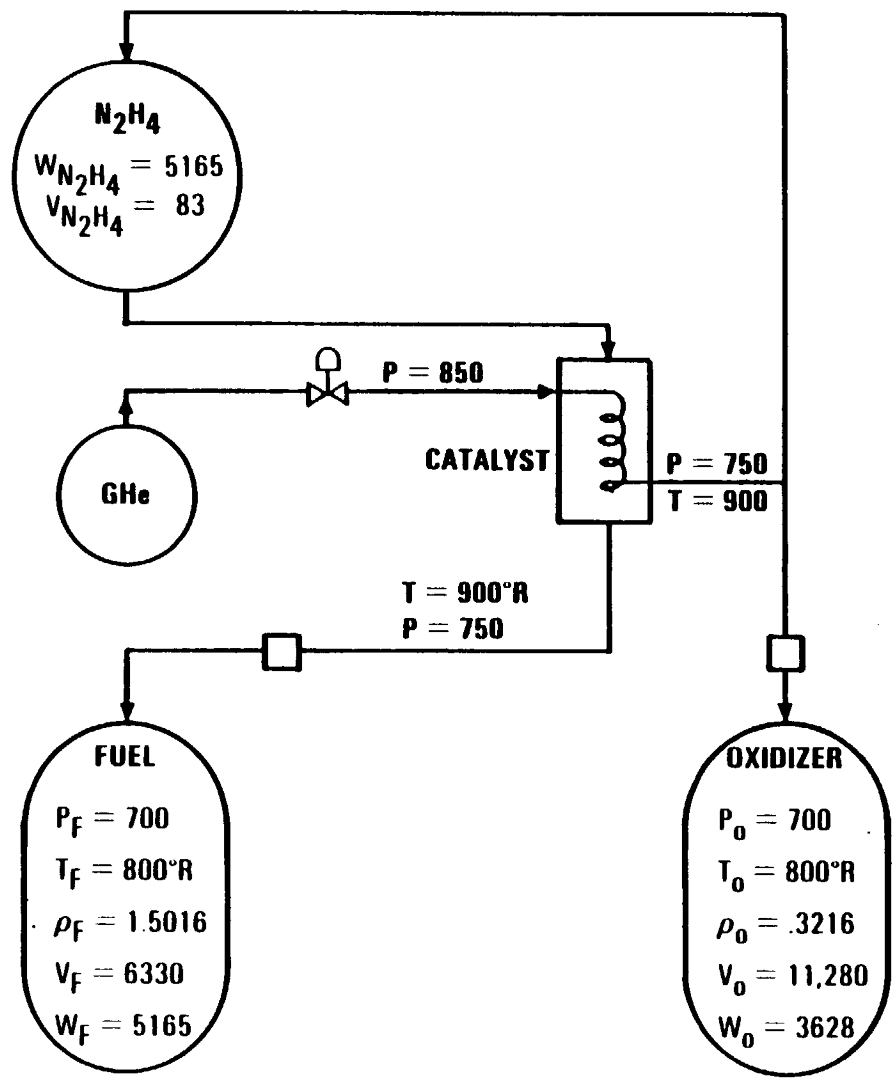
$$\rho_H = \frac{.3216 + .3668}{2} = .3442$$

$V_H = 83$ (5.4 FT. DIAMETER)

$W_h = 83 \times .3442 = 29$

HELIUM STORAGE BOTTLE

START	**CUTOFF**
$P_1 = 4000$	$P_2 = 900$
$T_1 = 150°R$	$T_2 = 150\left(\frac{900}{4000}\right)^{0.4} = 83°R$
$\rho_1 = 6.781$	$\rho_2 = 3.506$
$V_B = 1117$	$V_B = 1117$
$W_1 = 7574$	$W_2 = 3916$



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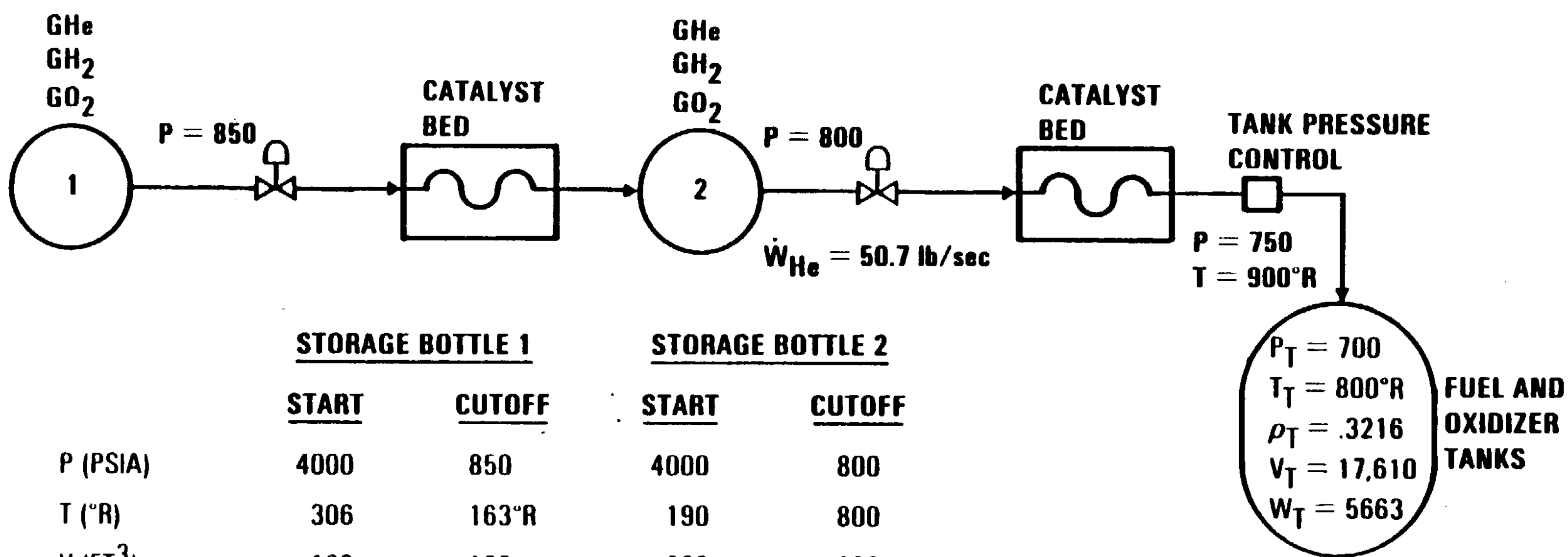
PRESSURIZATION SYSTEM - OPTION 8

This option features a cascaded helium storage bottle concept with catalytic converters to heat the helium flow. In this system, small amounts of hydrogen and oxygen (of the order of a few percent by volume) are mixed with the helium in the storage bottles. According to a preliminary literature survey, these small quantities of H_2 and O_2 are below the flammability or explosive limits for the mixture and impose no safety hazard. When these mixtures are passed through an appropriate catalytic converter, the exothermic reaction converting H_2 and O_2 into water (steam) raises the bulk helium temperature. The outlet temperature is determined by the original fractional amounts of H_2 and O_2 in the helium and the converter design.

The two-bottle cascaded concept is beneficial in that at the end of the mission the amount of cold (high density) residual helium is reduced to that trapped in bottle 1, which is small in comparison to bottle 2. The larger helium residual in bottle 2 has been heated to low density to minimize its weight. Details of the catalytic converter heaters are presented later in this section.

Pressurization System—Option 8



	STORAGE BOTTLE 1		STORAGE BOTTLE 2	
	START	CUTOFF	START	CUTOFF
P (PSIA)	4000	850	4000	800
T (°R)	306	163°R	190	800
V (FT^3)	163	163	990	990
W_{He} (LB)	655	291	5663	363
W_{H_2} (LB)	9	4	53	5
W_{O_2} (LB)	68	30	427	38

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PRESSURIZATION SYSTEM OPTION SUMMARY

The tank volumes and system weights for the pressurization system options which were studied during Phase I of the contract are summarized on the facing chart. Details of each option have been discussed earlier in this section. The following observations are made:

- On the basis of minimum weight and complexity, Option 8 is certainly attractive. This option features a cascaded two-tank system and catalytic heat exchangers.
- The cascaded tank concept (two or more tanks) in Option 8 could be applied to other options as well to reduce their weight.
- It can be seen by comparing Options 1 through 3 with Options 4 through 6 that the value of heating the helium residual with a coil-in-the-bottle concept results in about a 20% weight savings in each case. Cost and reliability issues would have to determine if this weight savings is worthwhile.
- The catalytic heater options are lowest weight but may have safety issues requiring additional study and development.
- Other options not in this list (e.g., LO_2/LH_2 burner substitute for catalytic heat exchangers) should be evaluated.

Pressurization System Option Summary

TRW

GDSS OPTION	VOLUME (ft^3) He TANK He	O_2	H_2	COIL	TOTAL	WEIGHT (lb) He	O_2	H_2	He TANK COIL	He TANK	CAT HX	GG HX	GG HX PROP	LRB THRUST Δ PROP	LRB Δ ISP Δ PROP	4 ENG HX	TANK DIA (ft)	TOTAL WEIGHT (lb)
1	1010			30	1040	6846			3592	4848		15526	17585	-6831			12.6	41566
2	1010			30	1040	6846			3592	4848					3965	397	12.6	19648
3	1010	11	22	30	1073	6846	873	109	3592	4999	150						12.7	16569
4	1729				1729	11725				8052		15740	17827	6925			14.9	46419
5	1729				1729	11725				8052					4019	402	14.9	24198
6	1729	11	22		1762	11725	884	110		8237	150						15.0	21106
7	1117				1117	7574				5205		7542	5165				12.9	25486
8	1153	11	22		1186	6318	495	62		5527	300						6.8/12.4	12702

CASCADED TWO TANK SYSTEM

INCLUDES 387 lb N_2H_4 TANK

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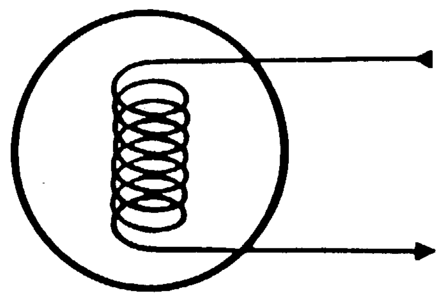
HELIUM TANK HEAT EXCHANGER (COIL IN BOTTLE)

Pressurization system Options 1 through 3 include a heat exchanger within the helium storage tank to warm the helium residual thus reducing system weight. An analysis of this concept was made to evaluate its feasibility and determine a preliminary size and weight. Since the heat transfer can only be done via natural convection in the stagnant helium on the outside of the tube coil, it was initially thought that this concept would not be practical. However, the heat transfer coefficient turns out to be surprisingly high for helium at these high pressures and densities.

The conclusion is that the concept is practical and that the size and weight of the tube coil is reasonable. In Options 1 through 3 the tube coil is only about 3% of the total tank volume and weighs 3592 pounds. There is enough reduction of helium residuals and tank weight for the net system weight to be reduced by about 4000 pounds in each option. Engineering considerations such as cost and reliability would have to be evaluated before deciding on this concept.

Helium Tank Heat Exchanger (Coil in Bottle)

TRW



- Materials - 3-inch diameter INCONEL - 718 tubing
- Tube coil length - 616 ft
- Tube coil weight - 3592 lb
- Helium conditions in tank

  Initial - 150°R at 4000 psia
  Final - 300°R at 1000 psia

- Tank volume - 1010 cu ft
- Convection force field - 1.5 Gs
- Helium heat transfer coefficient - 84 Btu/ft^2 hr °R

LOX/RP-1 GAS GENERATOR/HEAT EXCHANGER (OPTIONS 1 AND 4)

- Configuration - pure counterflow, coiled tubes in shell
- Materials - 2-inch diameter INCONEL - 718 tubes and shell
- Helium pressure drop - 100 psid
- Helium flow rate - 50.7 lb_m/sec total
- $\dot{Q}$ - 51,487 Btu/sec total
- Dimensions - 42 inch dia x 18 ft long, per engine (4 required)
- Weight - 3935 x 4 eng = 15,740 lb total
- GG flow rate - 39.9 lb_m/sec (LOX/RP-1), per engine (4 required)
- GG net vacuum thrust - 4395 x 4 eng = 17,580 lb_f total
- GG chamber pressure - 500 psia
- GG I_{sp} - 110 sec
- LOX/RP-1 fuel rich mixture ratio - 0.35
- LOX/RP-1 C_p - 0.645

NOTE: Option 1 thermal design is slightly smaller than Option 4 (shown) due to the lower helium ΔT requirement (808°R vs. 819°R).

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ENGINE HEAT EXCHANGER (OPTIONS 2 AND 5)

- Configuration - axial tube array on booster engine nozzles
- Materials - 2-inch diameter INCONEL - 718 tubes
- Helium pressure drop - 100 psid
- Helium flow rate - 12.7 lb_m/sec (per engine)
- $\dot{Q}$ - 50,796 Btu/sec total
- Dimensions - from area ratio 2 to 6 on engine nozzles
- Weight - tube weight added minus ablative weight removed, W = 2175-1778 = 397 lb net (4 engines)
- Maximum tube temperature - 1200°F

NOTE: Option 2 (shown) thermal design is slightly smaller than Option 5 due to the lower helium ΔT requirement (808°R vs 819°R).

HYDRAZINE GAS GENERATOR/HEAT EXCHANGER (OPTION 7)

- Configuration - pure counterflow, coiled tubes in shell
- Materials - 2-inch diameter INCONEL - 718 tubes and shell
- Helium pressure drop - 100 psid
- Helium flow rate - 32.7 lb_m/sec
- $\dot{Q}$ - 33,041 Btu/sec
- Dimensions - 3 ft dia x 11 ft long
- Weight - 7155 lb
- GG flow rate - 46.2 lb_m/sec N_2H_4
- GG chamber pressure - 500 psia
- GG adiabatic flame temp - 1353^{o}F
- Ammonia dissociation - 82% (in N_2H_4 combustion products)
- N_2H_4 decomposition - 100%

CATALYTIC HEAT EXCHANGER (OPTIONS 3, 6, AND 8)

The concept for the catalytic converter heat exchanger is quite simple. Small amounts of hydrogen and oxygen (of the order of a few percent by volume) are mixed with the helium in the storage bottle. According to a preliminary literature survey, these small quantities of H_2 and O_2 are below the flammability or explosive limits for the mixture and impose no safety hazard. When the mixture is passed through an appropriate catalyst bed, the exothermic reaction converting H_2 and O_2 into water (steam) raises the bulk helium temperature. The outlet temperature is determined by the original fractional amounts of H_2 and O_2 in the helium and the converter design.

An important converter design/performance parameter is space velocity, which is defined as gas volumetric flow rate divided by catalyst volume (units are reciprical time). This parameter is a measure of "residence time" of the gas in contact with the catalyst. TRW has recently conducted parametric laboratory experiments with dilute H_2/O_2 mixtures in helium with different catalysts to characterize space velocity thresholds. These tests were done over a range of mixtures and temperatures. These data indicate that a space velocity of 500,000 1/hr is a reasonable value to use for the pressurization system options studied herein.

Catalytic Converter Preliminary Design

- Catalyst Volume

$$V = \frac{\text{flow rate}}{\text{space velocity}} = \frac{5.69 \times 10^5 \ ft^3/hr}{5 \times 10^5 \ 1/hr} \quad 1.14 \text{ cu ft}$$

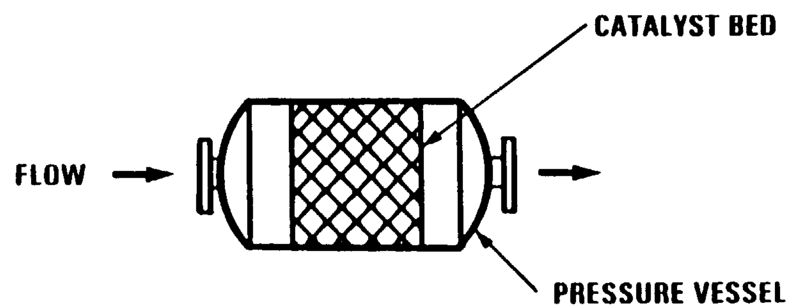
- Catalyst Weight

$$W = (\text{volume})\ (\text{density}) = (1.14)\ (70) = 80 \text{ lb}$$

- Weight Summary

  80 lb catalyst
  70 lb pressure vessel/flanges
  ---
  150 lb Total Weight

- Simplified design concept



CATALYTIC CONVERTER ISSUES

Safety

Flammability and explosion concerns have been very extensively reported upon in the literature. While a survey has confirmed, for the low concentrations of hydrogen and oxygen noted herein, that the homogeneous mixture of interest is not within the limits of flammability, there remain possible concerns about safety hazards associated with filling the vessel, and with catalytic reactions induced by impurities in the wetted surfaces. An alternative system that avoids these potential problems stores the H_2 and O_2 in separate tanks and blends these gases with pure helium at the converter inlet.

Design

It is likely that the converter design will not be a packed-bed type due to the pressure drop penalty. A better approach is to use a ceramic or other honeycomb as the catalyst support to minimize pressure drop across the catalyst, and to minimize the possibility of channeling, which may occur with a pelletized form of catalyst due to launch/flight vibration. An additional benefit of a monolithic design is more efficient use of catalyst material.

Another area of design which should be examined in more detail is the issue of cold inlet gases (approaching liquid O_2 temperatures in the designs herein). This may require incorporation of a gas/gas heat exchanger for the reactor downstream of the gas tank. This would use the catalyst outlet gas to preheat the inlet gas. Warm inlet gas may be necessary to achieve the rapid reaction rates needed to keep the catalyst inventory small.

CATALYTIC HEAT EXCHANGER - WATER IN LOX TANK ISSUE

Issue

The catalytic heat exchanger pressurization options presented result in water (steam) in the hot ullage gas of the fuel (RP-1) and oxidizer (LOX) tanks. Concern has been raised regarding water condensing in the form of ice in the LOX tank at the cold LOX surface.

Discussion

One approach to addressing this issue is to estimate the largest possible amount of water condensation in the LOX and compare this with accepted NASA standards for Shuttle flights. This analysis was done and is summarized below.

- Total lb moles of LOX in tank = 25,200
- Total lb moles of H_2O (steam) in ullage = 17
- Mole fraction of H_2O in LOX if all ullage H_2O condenses:

$$M_{H_2O} = \frac{17}{25,217} = 0.00067 = 0.067\% \text{ (volume percent)}$$

- Since only the LOX/ullage interface surface is involved (very small compared with ullage volume) the amount of ullage H_2O which will actually condense is expected to be an extremely small fraction of the above 0.067%. Most H_2O will remain as steam in the ullage.

Now compare this with Shuttle LOX contamination requirements:

- NASA uses MIL-P-25508E for the procurement of LOX. It defines two propellant grades - A=99.6% pure and B-99.5% pure. Water contamination is less than 0.0003% for grade A and 0.0026% for grade B (by volume).

- For <u>usage</u> a different standard is used - EG&G Propellant and Sampling Plan Rev. 3. This <u>was</u> not available for review but other sources indicate that a contamination factor of 30 might be reasonable. Thus the loaded LOX might have as much as 0.078% water content (by volume).

The nature of the interface during expulsion remains in doubt. The cryogenic fluid system experts at NBS have not encountered a similar situation in over 25 years of work. The matter is being referred to the thermophysical properties section which has recently been measuring properties of supercritical air and CO_2. Any condensation is expected to be in the form of microscopic ice crystals or snow. Large pieces of ice are not expected.

This condensation is not expected to have any performance, reliability, or other adverse affects on the booster engines. In particular, TRW's coaxial injector design is inherently resistant to propellant contamination effects due to its relatively large orifice dimensions.

<u>Conclusion</u>

Even if all pressurization ullage H_2O condenses in the LOX (a gross overestimation), the contamination percentage is of the same order of magnitude of that which NASA currently accepts on Shuttle. Furthermore, the condensation is expected to be in very benign form (microscopic ice crystals or snow) and not have any adverse affects on booster engine performance or reliability.

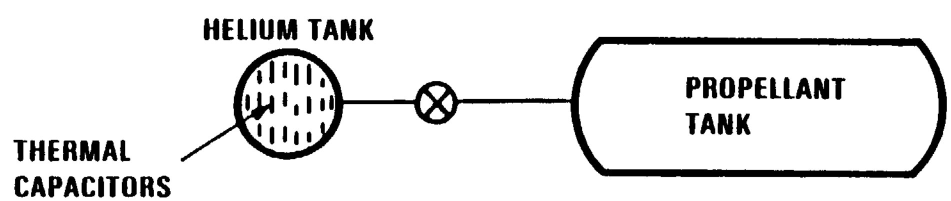
THERMAL CAPACITOR HEAT EXCHANGER PERFORMANCE

In any pressurization system option it is desirable to reduce helium tank residuals at the end of the blowdown to minimize system weight. Heating the helium to lower its density is one way. A passive method of transferring heat to the helium is through thermal capacitors. This involves providing a distributed thermal mass inside the tank which gives up its thermal energy to the helium as it cools during blowdown. Metal strips, fins, mesh, etc., can be used. This concept was briefly examined for this study and found to have merit.

To evaluate the concept a thermal capacitor system was compared with ideal isothermal (best case) and isentropic (worst case) blowdowns. The thermal mass used was derived from experimental data found in the literature which showed that blowdown performance could be achieved which was within about 80% of isothermal predictions by filling about 1% of the tank volume with aluminum strips. The facing chart compares the thermal capacitor design with isothermal and isentropic systems. The isothermal (ideal) system represents the lowest possible weight system since it achieves heating with no weight penalty. A system weight savings of about 17% was achieved with the thermal capacitor design over the unheated (isentropic) system. This completely passive concept with its inherent high reliability deserves consideration.

Thermal Capacitor Heat Exchanger Performance

System	Weight (lb)			
	Helium	Helium Tank	Capacitor	Total
Isothermal (ideal)	11,395	21,101	-	32,496
Thermal Capacitor	12,900	23,887	8,858	45,645
Isentropic (ideal)	19,242	35,631	-	54,873



	INITIAL	FINAL	FINAL
PRESSURE (PSIA)	4000	900	700
TEMPERATURE (^{o}R)			
ISOTHERMAL	527	527	527
CAPACITOR	527	477	477
ISENTROPIC	527	291	291

BOOSTER PRESSURE BUDGET

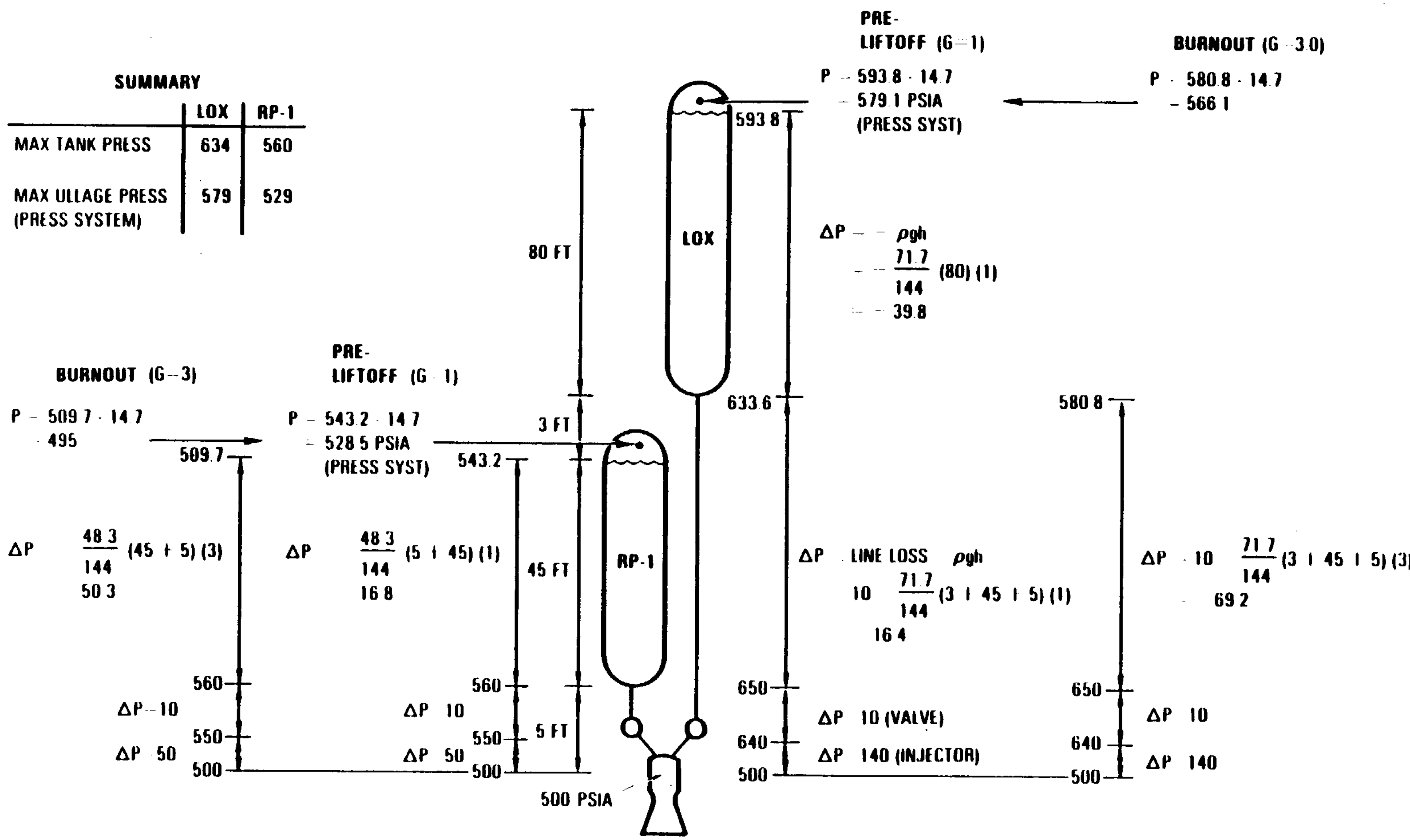
The pressurization systems analyzed during the study assumed a chamber pressure of 500 psia in the booster main engines. Using this as a basis, GDSS assumed a propellant tank pressure of 700 psia to compare pressurization system options. Although this is adequate for relative system comparisons, actual tank pressure requirements will likely be lower than 700 psia as the facing chart shows. In addition, it was assumed that full tank pressure was maintained through booster burnout. For throttling-back at max-G conditions near burnout it would be very advantageous to transfer from the constant pressure regulated mode to a blowdown mode to reduce thrust. Since the weight of the pressurization system is proportional to the final tank pressure, a weight and size reduction could be made. Both of the above factors could result in a significant improvement in booster mass fraction.

The facing chart shows a preliminary pressure budget for a P_C = 500 psia system including dynamic head pressure effects and recommended pressure drops in a TRW coaxial injector. Pre-liftoff and burnout (G=3) conditions were compared. For each tank (LOX and RP-1) the maximum ullage and tank wall pressures were driven by pre-liftoff conditions rather than burnout. Maximum-ullage pressures (which size the pressurization system) are 579 and 529 psia for the LOX and RP-1 tanks, respectively. Maximum bottom-of-tank wall pressures (which drive tank weight) are 634 and 560 psia for the LOX and RP-1 tanks, respectively.

Booster Pressure Budget
(P_C = 500 PSIA, LOX/RP-1)

SUMMARY

	LOX	RP-1
MAX TANK PRESS	634	560
MAX ULLAGE PRESS (PRESS SYSTEM)	579	529



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7. COMPOSITE TANK TECHNOLOGY STATUS

Tanks Summary

- Graphite/epoxy filament-wound composite tanks with load-sharing aluminum liners make pressure-fed LOX boosters feasible
  - High reliability/safety
  - Light weight
  - Reasonable cost
  - Low risk
- Fabrication of large diameter filament-wound tanks with load-sharing liners is demonstrated state-of-the-art technology
- Optimum fabrication approach would be to set up large-diameter winding machines at liner fabrication site

Summary: Assessment of Filament Winding Manufacturing Feasibility

TRW

Materials:	State-of-the-art materials are acceptable
Nondestructive inspection:	Technology exists, need scale-up of equipment
Facilities:	Minor extension of current filament winding machines
Concept:	Composite overwrap of load sharing liner is lowest risk/cost approach

Manufacture of pressure vessels is within state-of-the-art. Fabrication of 15-foot diameter vessels requires new winding machines preferably located at liner fabrication site.

State-of-the-Art Materials/Inspection Technology for Filament Wound Structures

TRW

Materials

- Currently available ultra high strength graphite fibers, filament winding epoxy resins and adhesives are suitable
- Manufacturing technology for fabrication of metallic liners exists and can be readily applied. Close interaction of liner fabricator and winder is required. Liner fabrication is simpler than that of STS external tank and Saturn S-II tanks

Nondestructive Inspection (NDI)

- Current ultrasonic and computed tomography inspection techniques are applicable. 15-foot diameter is beyond capability of existing equipment, i.e., 12-foot diameter limit for ultrasonic inspection and 8-foot diameter limit for CT inspection

Existing materials and nondestructive inspection techniques for filament winding composite overwrapped/metal lined pressure vessels are suitable

Composite Overwrap Concepts Discussed with Filament Winders

Concept	Comments
A. Hoop-only wrap (only at SCI)	Liner not inspectable
B. Load sharing metal liner	Lowest technical risk Lowest cost
C. Thin metal liner	Lightest weight Highest technical risk Highest cost
D. Metallized polymeric liner	Not technically feasible

State-of-the-art composite overwrapped load-sharing liner concept selected

State-of-the-Art Filament Winding Capabilities

TRW

- Large diameter structures have been filament wound

  12 feet diameter × 28-32 feet length SRM segment for Shuttle

  22 feet diameter × 56 feet length silo

  15 feet diameter × 54 feet length railroad tank car

- 15-foot diameter is beyond the current capacity of any winding facility in the U.S. Maximum size capability (14 feet diameter × 90 feet length) exists at Hercules. Aerojet capability is 12 feet diameter × 30 feet length. SCI has no current large diameter capability, but has wound very large (15 feet diameter) structures in the past.

- Design/fabrication of a filament winding machine with a 15-foot diameter capability is a simple extension of current FW machines having a 12- to 14-foot diameter capability. Optimum manufacturing concept would be to set up winding machine at liner fabrication site.

15-foot diameter filament-wound composite overwrapped/metal lined pressure vessels can be fabricated

Overwrapped Tank Technology Status

TRW

- 23,000 PSIA BURST PRESSURE TANK FABRICATED BY SCI
  - TESTED TO 20,000 PSIA WITHOUT BURSTING
  - 4-INCH O.D., 9-INCH LONG CYLINDER/ELLIPSOID
  - 10 MIL AL LINER
  - GRAPHITE/EPOXY OVERWRAP
  - 50,000 - 60,000 PSIA REGARDED AS MAXIMUM PROBABLE PRESSURE LIMIT OF OVERWRAPPED TANKS
- HERCULES GRAPHITE/EPOXY SRB MOTOR CASE
  - 12 FEET O.D.
  - P_C = 1000 PSI
  - SUCCESSFULLY TESTED
- SHUTTLE PRESSURANT TANK
  - 5000 PSIA, 40-INCH DIAMETER SPHERE
  - KEVLAR OVERWRAP
- NAVY SUBMARINE APPLICATION CRYOGENIC STORAGE TANK
  - CYCLED FROM AMBIENT PRESSURE AND TEMPERATURE TO 3000 PSIA LIQUID HELIUM TEMPERATURE 25 TIMES WITHOUT FAILURE
  - GRAPHITE/EPOXY OVERWRAPPED ALUMINUM LINER
  - 20.0-INCH DIAMETER, 100-INCH LONG
- 10.2 KKV PROPULSION TECHNOLOGY PRESSURANT TANK
  - IN FINAL FLIGHT WEIGHT FABRICATION, TO BE TESTED SOON
  - 5.2-INCH O.D., 11-INCHES LONG
  - GRAPHITE/EPOXY OVERWRAP ON 0.020 TITANIUM LINER
  - 8700 PSIA MAXIMUM OPERATING PRESSURE

8. REFERENCES

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2. "Feasibility Study of a Pressure-Fed Engine for a Water Recoverable Space Shuttle Booster," Vol. II, Technical Report, NASA MSFC Report No. SE-019-008-2H-B, 15 March 1972.

3. 2nd Project Review, Liquid Rocket Booster System Study, Presentation at NASA/Marshall Space Flight Center, 16 December 1987.

4. J. B. Kendrick, "Final Technical Report on Low Cost Launch-Vehicle Study." NASA Headquarters Report No. CR-106662, 23 May 1969.

APPENDIX A. WEIGHT SCALING MODEL

The weight scaling model presented herein allows engine weight to be computed as a function of varying engine parameters once the weight of a reference configuration is known. It is restricted to an ablative-cooled coaxial-injector engine and is derived by breaking the engine down into its major components and examining how the weight of each one varies with engine thrust, chamber pressure, contraction ratio, etc.

Assumptions

- TRW COAXIAL INJECTOR
- ROLLED AND WELDED STEEL CHAMBER
- SILICA PHENOLIC ABLATIVE LINER
- PROPELLANT LINE FLANGES INCLUDED
- PROPELLANT SHUTOFF VALVES NOT INCLUDED
- 130-SECOND FIRING TIME

The engine has been divided into nine basic components:

Item	Component
1	Fuel manifold
2	Oxidizer pintle
3	Chamber cap
4	Chamber wall
5	Chamber ablative
6	Nozzle wall
7	Nozzle ablative
8	Exit ring
9	Miscellaneous hardware

Engine Weight Computation Model

$$W_{PFE} = \sum_{I=1}^{9} w_I F_I$$

WHERE

W_{PFE} = ABLATIVE COOLED PRESSURE-FED ENGINE WEIGHT, LB

w_I = WEIGHT COEFFICIENT OF ITH COMPONENT

F_I = WEIGHT SCALING FACTOR OF ITH COMPONENT

These weight coefficients are based on a 50K sea level thrust engine design. The weight scaling model can use any appropriate reference configuration. Clearly the closer the reference configuration matches a particular design, the more representative the model will be.

50K Engine Reference Configuration Weight Coefficient Summary (pounds)

$W_1 = 20.97$
$W_2 = 53.8$
$W_3 = 4.17$
$W_4 = 18.76$
$W_5 = 130.46$
$W_6 = 13.09$
$W_7 = 109.98$
$W_8 = 23.82$
$W_9 = 81.59$

Definition of terms:

F = sea level thrust

P_c = Chamber pressure

CF = sea level thrust coefficient

CR = contraction ratio

L^* = Characteristic chamber length

The subscript, o, refers to reference engine values. For the 50K reference engine:

F_o = 50,000 lb_f

P_{c_o} = 300 psia

CF_o = 1.37

CR_o = 3.37

L^*_o = 83 in.

TRW

Weight Scaling Factor Summary

$$F_1 = \left(\frac{F}{F_0}\right)^{1.5}$$

$$F_2 = \left(F/F_0\right)^{1.266}$$

$$F_3 = \left(F/F_0\right)^{1.5} \left(\frac{P_c}{P_{c_0}}\right)^{-0.5} \left(\frac{CF}{CF_0}\right)^{-1.5} \left(\frac{CR}{CR_0}\right)^{1.5}$$

$$F_4 = \left(F/F_0\right) \left(\frac{CF}{CF_0}\right)^{-1} \left(\frac{L^*}{L^*_0}\right)$$

$$F_5 = \left(F/F_0\right) \quad \left(P_c/P_{c_0}\right)^{0.5} \left(\frac{CF}{CF_0}\right)^{-0.5} \left(\frac{CF}{CR_0}\right)^{-0.5} \left(\frac{L^*}{L^*_0}\right)$$

$$F_6 = \left(F/F_0\right) \left(P_c/P_{c_0}\right)^{-1} \left(\frac{CF}{CF_0}\right)^{-1} \left(\frac{\epsilon-1}{\epsilon_0-1}\right)$$

$$F_7 = \left(F/F_0\right) \left(P_c/P_{c_0}\right)^{-1} \left(\frac{CF}{CF_0}\right)^{-1} \left(\frac{\epsilon-1}{\epsilon_0-1}\right)$$

$$F_8 = \left(F/F_0\right)^{0.5} \left(P_c/P_{c_0}\right)^{-0.5} \left(\frac{CF}{CF_0}\right)^{-0.5} \left(\frac{\epsilon}{\epsilon_0}\right)^{0.5}$$

$$F_9 = \left(F/F_0\right)$$