FR-19691-5
1 DECEMBER 1989

SPACE TRANSPORTATION BOOSTER ENGINE CONFIGURATION STUDY

ADDENDUM FINAL REPORT (DR4) INCLUDES DESIGN DEFINITION DOCUMENT (DR8)

Prepared for
Procurement Office
George C. Marshall Space Flight Center
National Aeronautics and Space Administration
Marshall Space Flight Center, AL 35812

Prepared Under
CONTRACT NAS8-36857

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UNITED TECHNOLOGIES
PRATT&WHITNEY

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Printed in the United States of America

FOREWORD

This study was conducted by the Pratt & Whitney Government Engine Business of the United Technologies Corporation (UTC) under NASA/MSFC contract NAS8-36857. The NASA/MSFC program manager was Mr. J. Thomson. The Pratt & Whitney program manager was Mr. W. Visek.

The technical effort started in May 1986 and was completed in July 1989. The study results for the period April 1989 to July 1989 are presented in this report.

Special thanks go to the numerous individuals at NASA, UTC, and the major vehicle contractors who contributed to this study effort.

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SECTION 1.0
INTRODUCTION

The final two extensions of the Space Transportation Booster Engine (STBE) Phase A program covered the time period as shown in Figure 1.0-1. The fifth extension covered April and May of 1989 while the sixth extension covered June and July of 1989. The STBE Final Report, FR-19691-4, issued in October 1989, discusses all work conducted on the STBE Contract NAS8-36857 up through the fourth extension, ending 31 March 1989. This Addendum Final Report, FR-19691-5, includes the description and results of the fifth and sixth extensions ending 31 July 1989.

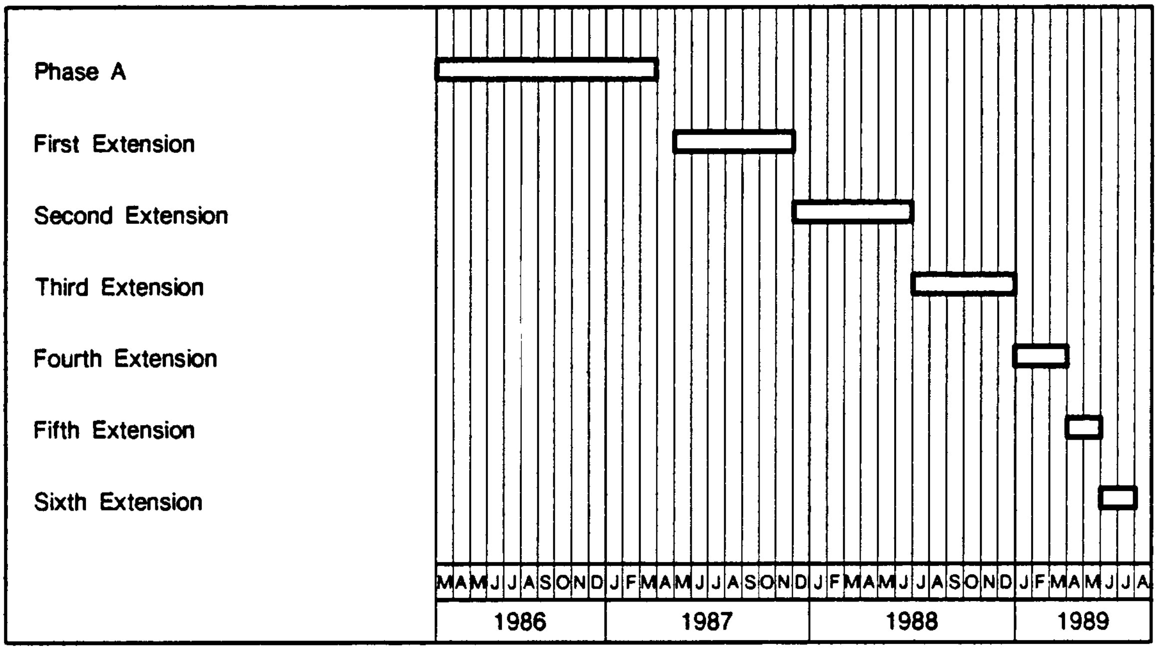


Figure 1.0-1. Space Transportation Booster Engine Phase A and Extension Time Periods

Primary activity during the fifth and sixth extensions consisted of the engine integration and ocean recovery tasks. In addition, updates of component designs and refinement of the overall engine configuration were completed.

Section 2.0 of this report discusses gas generator engine characteristics and results of engine configuration refinements for the fifth and sixth extensions. Section 3.0 provides updated component mechanical design, performance, and manufacturing information for the fifth and sixth extensions.

Section 4.0 provides the results of ocean recovery studies and various engine integration tasks.

Section 5.0 provides details of the maintenance plan for the STBE.

SECTION 2.0
STBE GAS GENERATOR ENGINE CHARACTERISTICS

During the Phase A fifth and sixth extensions that occurred from April to July 1989, Pratt & Whitney (P&W) defined a design concept for the Space Transportation Booster Engine (STBE), a derivative of the Space Transportation Main Engine (STME), which incorporates all integrated system requirements as defined by the Contract End Item (CEI) and Interface Control Document (ICD) specifications. System characteristics such as low recurring cost, high-reliability, reusability, and ease of maintainability were emphasized throughout the Phase A conceptual design.

The significant differences between the engine system concept that was produced during this period and the engine system discussed in the final report focused on changes in the main combustion chamber cooling system.

The main combustion chamber cooling passages have been redesigned so that an aspect ratio of 1.5 is achieved, resulting in a significant improvement in chamber life and minimizing the life limiting effects of cyclic strain ratcheting phenomena. This change prompted several component level and system level design changes, primarily the following:

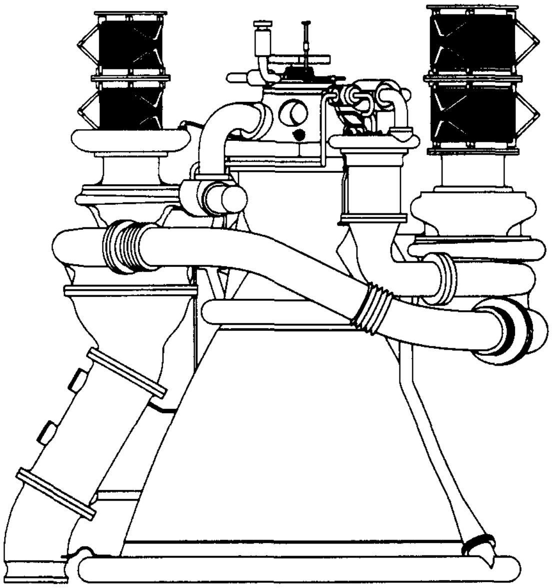
- A chamber bypass line was added to allow fuel pump discharge to flow directly to the main injector.
- A fuel manifold was added to the main injector to accommodate the chamber bypass flow.
- The smaller chamber passages prompted an increase in the pressure drop requirements of the coolant flow in order to maintain adequate cooling of the chamber (hot wall temp < 1425 °R). This resulted in an increase in the fuel pump discharge pressure, which results in higher tip speeds and thus higher stresses in the pump rotors.

The specific changes at the component level are described in Section 3.0. The remainder of Section 2.0 describes the engine system concept produced during the Phase A fifth and sixth extensions.

2.1 SPACE TRANSPORTATION BOOSTER ENGINE CONCEPT DESCRIPTION

Pratt & Whitney's proposed CH_4/O_2 gas generator cycle engine design concept is a derivative of the STME design for booster applications. The engine is designed at 644,900-pound sea-level thrust with a chamber pressure of 2250 psia and an inlet mixture ratio of 2.7. Nominal engine performance, weight, and dimensions are shown in Figure 2.1-1.

The STBE component placement was chosen to permit easy access to facilitate routine maintenance and component removal and replacement. This engine configuration incorporates vertically mounted turbopumps located 180 degrees apart, with scissor bellows as propellant inlets, mounted to the pump inlets to permit engine thrust vectoring. Engine thrust vectoring/gimballing capability is configured for ±6 degree square pattern. Common STME hardware is used in the STBE and the basic engine configuration is maintained similar to the STME. A listing of identical and modified hardware between the STME and STBE is shown in Tables 2.1-1 and 2.1-2.



Propellants		H_2/LO_2	CH_4/LO_2
Mixture Ratio		6.0	2.70
Chamber Pressure		2250 psia	2250 psia
Thrust	Vacuum	580,000 lb	711,823 lb
	Sea Level	461,446 lb	644,898 lb
Specific Impulse	Vacuum	440.3 sec	328.4 sec
	Sea Level	350.3 sec	297.5 sec
Nozzle Area Ratio		62	29
Exit Plane Diameter		108 in.	91 in.
Overall Length		175 in.	99 in.
Weight		7981 lb	6960 lb

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Figure 2.1-1. Derivative STBE Gas Generator Cycle Design Operating Conditions

Table 2.1-1. *Identical Hardware Components of STME and STBE Gas Generator Engines*

Turbomachinery
- Fuel Pump Housing Flowpaths
- Fuel Pump Impeller Flowpath
- Ball and Roller Bearings
- Turbine Outer Seals
- Tiebolt Shaft and Disks, Modified Blade Attachments
- Internal Labyrinth Seals
- Major Flange Seals
- Bolts, Nuts, Studs, Washers, and Pins
- 1st- and 2nd-Stage Impeller Castings
- Uniform Cross-Section Static Housing Seals
- Inducer Retaining Bolts
- Blade Retaining Rings, Tip Seals
- Spacers, Bearings Sleeves, and Wave Washers Made From Same Forging or Identical Hardware

Engine Controls
- Engine Controller
- Engine and Component Instrumentation

Combustion Devices
- Gas Generator Injector Interpropellant Plate
- Gas Generator Injector Housing
- Gas Generator Combustion Chamber
- Gas Generator Combustion Chamber Liner
- Tubular Nozzle
- Nozzle Inlet Manifold
- Nozzle Discharge Manifold
- Main Injector Interpropellant Plate
- Main Injector Housing
- Main Injector Faceplate
- Igniter Assembly — Main Injector
- Igniter Assembly — Gas Generator
- Main Chamber to Injector Flange, Seals, and Fasteners

Engine Assembly
- Ducting
  - 80% Small Lines
  - 80% Large Lines
- Engine/Vehicle Interface Points
- GO_2 Hex
- POGO Suppressor
- Fuel Inlet Flex Joints
- Fasteners and Seals

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Table 2.1-2. *Partial Commonality and Modified Design Components of STME and Derivative STBE Gas Generator Engines*

Components That Use Same Internal Flowpath Geometry But Operate at a Higher Pressure

- Fuel Pump Impeller and Housings
- Fuel Shutoff Valve

Components That Will Be a Modified Design

- Main Combustion Chamber
- Oxidizer Pump Impeller and Housings
- Oxidizer Turbine Blading
- Fuel Turbine Blading
- GG Oxidizer Valve
- GG Fuel Valve
- Main Oxidizer Valve
- Gimbal

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2.2 ENGINE OPERATION

2.2.1 Main Stage Engine Operating Conditions

LO_2 and CH_4 enter the engine at net positive suction head (NPSH) levels, supplied by the vehicle, sufficient for the high-speed, high-pressure pumps to operate without boost pumps. At the design power level, the methane pump operates at 10,717 rpm to provide the fuel pressure of 4925 psia required by the cycle. From the pump exit, the methane flows through the fuel shutoff valve and down to the chamber nozzle cooling passages. Prior to entering the common manifold at the chamber nozzle interface, 47 percent of the methane flow is routed directly to the injector manifold. Of the remaining fuel flow, 138 lbm/sec is used to cool the milled channel chamber, mixing with the bypassed methane in the injector manifold. The methane flow which is required for tank pressurization and the gas generator, cools the tubular nozzle down to an area ratio of 29 to 1. From the nozzle coolant exit manifold, this flow proceeds to the gas generator after the tank pressurant is bled off. After flowing through the fuel gas generator valve, this methane is injected into the gas generator to combust with the oxygen to provide power to the turbines.

On the oxidizer side, the one-stage oxygen pump operates at 8181 rpm to provide the oxygen pressure of 3902 psia required by the cycle at the design point. From the pump exit, approximately 98 percent of the total oxidizer flow is routed through the main oxidizer control valve and is injected into the chamber after the tank pressurant, required by the vehicle, is extracted. The remainder of the oxygen flows through the oxidizer gas generator control valve before being injected into the gas generator. The gas generator flow powers the two propellant pumps and is exhausted to ambient through a 5 to 1 area ratio nozzle. The detailed STBE power balance model is shown in Table 2.2-1 and the cycle schematic in Figure 2.2-1.

2.2.2 Engine Start and Shutdown Operation

A representative engine transient thrust characteristic is depicted in Figure 2.2.2-1 along with the valve schedule.

Table 2.2-1. *Gas Generator Cycle Off-Design Deck STBE (CH4/02) Engine Study*

Engine Performance		*Engine Heat Transfer*	
Vacuum Thrust	696,728	Chamber Coolant DP	1,944
Sea-Level Thrust	627,373	Chamber Coolant DT	573
Vacuum Impulse	332.03	Chamber Q	65,607
Sea-Level Impulse	298.98	Nozzle Coolant DP	533
Total Engine Inlet Flowrate	2,105.3	Nozzle Coolant DT	430
Overall Engine Mixture Ratio	2.70	Nozzle Q	40,600

Chamber Performance		*Gas Generator Performance*	
Pressure	2,244.8	Pressure	2,310.9
Temperature	6,558.2	Temperature	1,800.0
Thrust	672,064	Thrust	24,664
Impulse	343.45	Impulse	174.20
Flowrate	1,956.8	Flowrate	141.6
Throat Area	162.18	Mixture Ratio	0.273
Nozzle Area Ratio	29	Nozzle Efficiency	0.980
Mixture Ratio	3.30	Nozzle Gas Constant	95.1
Nozzle Efficiency	0.965	Nozzle Gamma	1.093
CSTAR Efficiency	0.980	Nozzle Area	74.0

Engine Station Conditions

Station	*Pressure*	*Temp*	*Flow*	*Enthalpy*	*Density*
		Fuel System Conditions			
Main Pump Inlet	47.0	201.0	569.0	123.1	26.40
1st-Stage Exit	2,475.6	217.3	569.0	147.2	26.54
Main Pump Exit	4,925.4	232.9	569.0	170.7	26.74
FSOV Inlet	4,809.5	233.6	569.0	170.7	26.67
FSOV Exit	4,634.1	234.6	569.0	170.7	26.57
Cham/Cool Inlet	4,524.0	235.3	138.1	170.7	26.51
Cham/Cool Exit	2,580.2	807.9	138.1	645.8	4.76
Cham BP Inlet	4,524.0	235.3	317.0	170.7	26.51
Cham BP Exit	2,580.2	245.8	317.0	170.7	25.31
Ch Inj Inlet	2,538.8	405.8	455.1	314.9	15.95
Noz/Cool Inlet	4,524.0	235.3	113.9	170.7	26.51
Noz/Cool Exit	3,991.5	665.7	113.9	527.0	9.12
Tank Press. In	3,832.1	644.2	2.7	527.0	8.85
Tank Press. In	47.0	562.1	2.7	527.0	0.13
FGCV Inlet	3,832.1	664.2	111.2	527.0	8.85
FGCV Exit	2,985.1	652.7	111.2	527.0	7.27
GG Inj Inlet	2,933.2	651.9	111.2	527.0	7.17
		Oxidizer System Conditions			
Main Pump Inlet	47.0	164.0	1,536.3	61.6	70.98
Main Pump Exit	3,902.3	181.9	1,536.3	75.0	71.72
MOV Inlet	3,778.3	182.4	1,506.0	75.0	71.53
MOV Exit	2,722.2	186.6	1,506.0	75.0	69.94
GO_2 Hex In	2,722.2	186.6	4.2	75.0	69.94
Tank Press. In	47.0	720.0	4.2	275.4	0.19
CH Inj Inlet	2,628.9	187.0	1,501.8	75.0	68.80
OGCV Inlet	3,622.6	183.0	30.3	75.0	71.30
OGCV Exit	3,389.4	183.9	30.3	75.0	70.95
GG Inj Inlet	3,323.5	184.2	30.3	75.0	70.86
		Gas Generator System Conditions			
Fuel Turb Inlet	2,220.1	1,797.0	141.6		
Fuel Turb Exit	737.4	1,614.5	141.6		
LO_2 Turb Inlet	682.0	1,609.4	141.6		
LO_2 Turb Exit	257.7	1,468.9	141.6		
Noz/Cool Inlet	199.8	1,450.6	141.6		
Noz/Cool Exit	199.8	1,450.6	141.6		

Table 2.2-1. *Gas Generator Cycle Off-Design Deck STBE (CH4/02) Engine Study (Continued)*

Turbomachinery Performance Data

Fuel Turbine

	Stage One	Stage Two
Efficiency, T/T	0.799	0.854
Horsepower	23,687	14,664
Speed, rpm	10,717	10,717
S Speed	20.6	37.6
S Diameter	2.36	1.73
Mean Diameter, in.	17.90	17.90
Vel Ratio, Actual	0.3440	0.4372
Max Tip Speed	866	880
Blade Height, in.	0.61	0.90
An^2	39.6	58.1
Effective Area	7.43	14.62
Press. Ratio, T/T	1.99	1.51
Gas Constant	96.33	
Gamma	1.1073	

Fuel Pump

	Stage One	Stage Two
Efficiency	0.703	0.715
Horsepower	19,398	18,953
Speed, rpm	10,717	10,717
NPSH, ft	178	13,272
SS Speed	21,389	850
S Speed	855	856
Head, ft	13,173	13,098
Diameter, in.	18.80	18.80
Tip Speed, ft/sec	880	880
Volume Flow	9,622	9,553
Head Coefficient	0.5476	0.5445
Flow Coefficient, Exit	0.1042	0.1051

LO_2 Turbine

	Stage One	Stage Two
Efficiency, T/T	0.815	0.754
Horsepower	13,835	15,137
Speed, rpm	8,181	8,181
S Speed	36.3	41.1
S Diameter	1.43	1.19
Mean Diameter, in.	18.18	18.18
Vel Ratio, Actual	0.3490	0.3337
Max Tip Speed	699	715
Blade Height, in.	1.38	1.84
An^2	52.8	70.4
Effective Area	23.08	34.81
Press Ratio, T/T	1.55	1.71
Gas Constant	95.63	
Gamma	1.0999	

LO_2 Pump

	Stage One
Efficiency	0.746
Horsepower	28,972
Speed, rpm	8,181
NPSH, ft	63
SS Speed	32,436
S Speed	972
Head, ft	7,740
Diameter, in.	18.86
Tip Speed, ft/sec	674
Volume Flow	9,615
Head Coefficient	0.5486
Flow Coefficient, Exit	0.1061

Valve Data

Station	Delp	Area	Flow	%Delp/P
Fuel Shutoff Valve	175.4	12.45	569.0	3.65
Fuel Bypass	1,411.2	2.030	317.0	35.36
Fuel GG Valve	847.1	2.195	111.2	22.10
Main Oxidizer Valve	1,056.1	8.20	1,501.8	27.95
LO_2 GG Valve	233.2	0.352	30.3	6.44

Injector Data

Station	Delp	Area	Flow	%Delp/P
Fuel GG Injector	622.2	2.886	111.2	21.21
Fuel CH Injector	294.0	13.49	455.1	11.58
LO_2 GG Injector	1,012.6	0.169	30.3	30.47
LO_2 CH Injector	384.1	13.72	1,501.8	14.61

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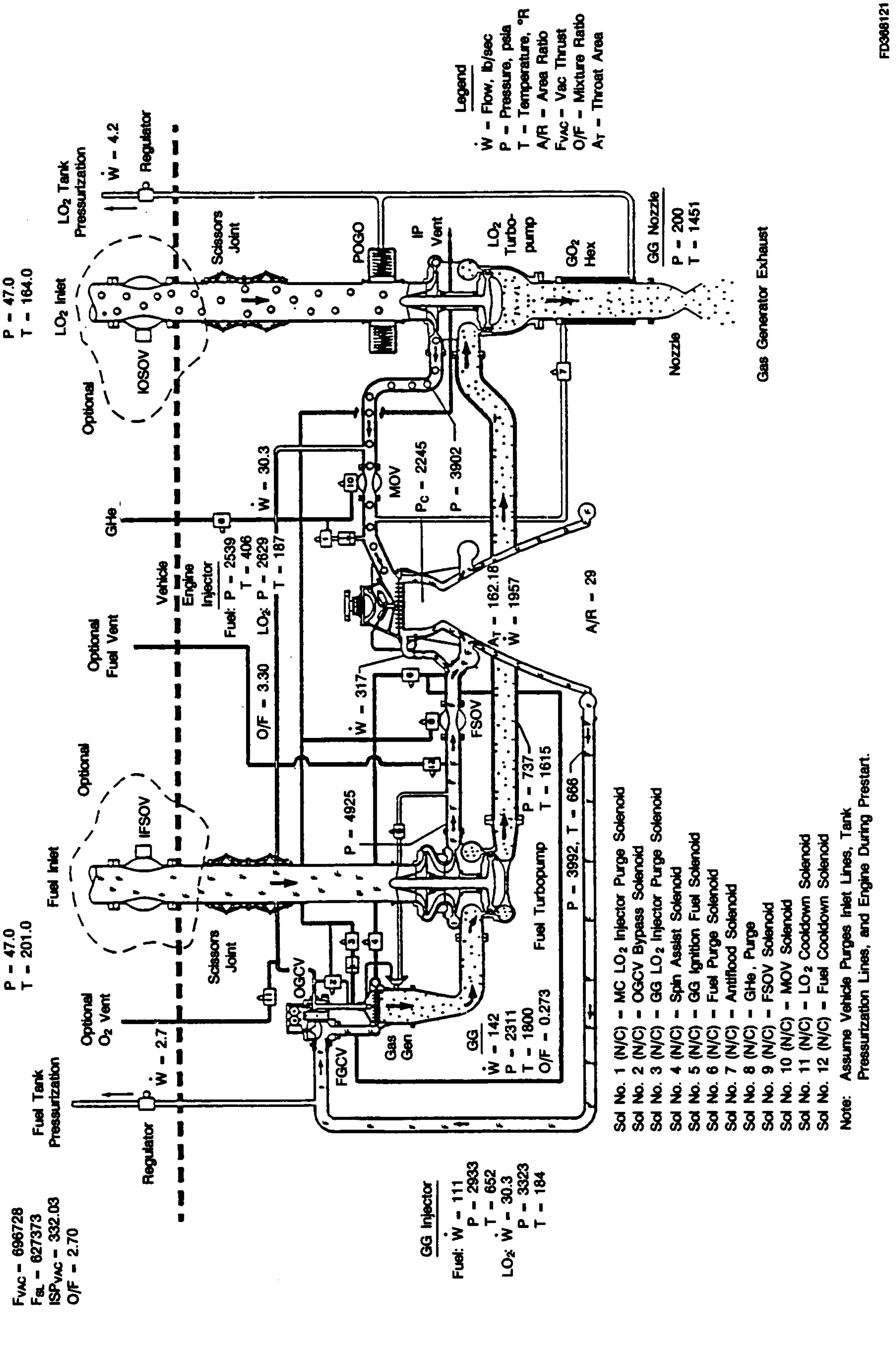

Figure 2.2-1. Derivative STBE Gas Generator Cycle, Design Conditions

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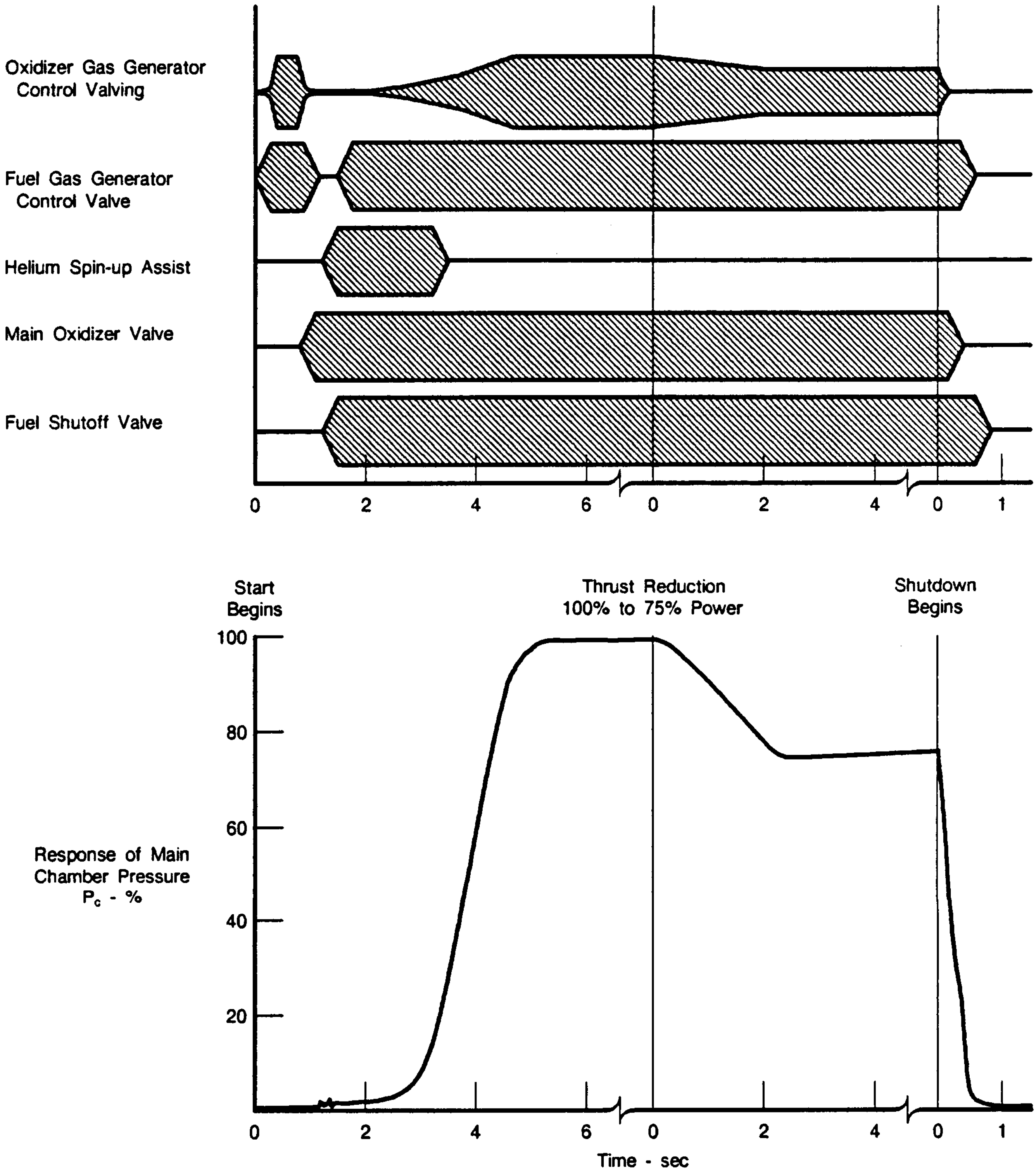

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Figure 2.2.2-1. Pratt & Whitney's STBE Design Concept Uses a Simple Open-Loop Control System to Satisfy System Transient Requirements

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In this engine concept, pumps will be preconditioned by cold soaking in liquid propellants supplied through the vehicle inlet lines. Any vehicle prevalves would be opened allowing propellants from the tanks to flow into the turbopumps, and let any vapors that form percolate back up to the tank to be vented. The turbopumps are mounted vertically to facilitate percolation cooldown. The STBE design does not require a bleed system.

The engine start uses an oxidizer lead for reliable soft propellant ignition. With the oxidizer lead, this transition from GO_2 to LO_2 occurs prior to fuel injection and the fuel is consumed immediately upon injection. The propellant mixture ratio makes a single rapid controlled excursion through stoichiometric and avoids the uncontrolled multiple excursions through stoichiometric which is often experienced with a fuel lead because the gas-to-liquid oxygen flow transition has to occur after ignition. Pratt & Whitney has had extensive successful experience with oxidizer leads with the RL10, XLR129, and more recently, the Alternate Turbopump Development (ATD) preburner igniters.

Using a timed sequence process, the gas generator and main oxidizer injectors are primed with LO_2 prior to opening the fuel shutoff valve (FSOV). Once the oxidizer injectors are primed with LO_2, a helium spin assist is activated to begin turbopump rotation. As soon as the turbopumps have begun rotation, the FSOV is opened, allowing methane into both the gas generator and main chamber. Dual electrical spark excited oxygen/methane torch igniters are used to provide ignition in both the gas generator and main combustion chamber. Once ignition has occurred, the oxidizer gas generator control valve (OGCV) is gradually opened to cause the engine to smoothly accelerate up to full thrust.

The helium acts as a diluent and lessens the effects on the turbine hardware of any short term temperature spike. During the start and shutdown, a small helium purge is used in the gas generator and main chamber injectors to eliminate the danger of hot gas flow reversals during the transient operation.

Main stage engine operation uses open-loop control. The OGCV and the main oxidizer valve (MOV) set engine thrust and mixture ratio, respectively. Gas generator mixture ratio and turbine drive gas temperature are set by trimming the fuel gas generator control valve (FGCV).

Engine shutdown is also achieved through time-phased scheduling of the propellant valves. The OGCV is closed first to terminate power to the turbopumps, then the MOV closes, followed by closing the methane system (FGCV and FSOV).

2.2.3 Controls

The STBE control system, a derivative of the STME system, consists of sensors, interconnects, controller, actuators, propellant valves, ancillary valves, and a health monitor. The functional layout of the STBE controls components is shown in Figure 2.2.3-1. The controller time sequences the valves for engine control and maintains engine safety by sensing hazards and taking corrective action. A single electromechanical actuator (EMA) drives both the gas generator fuel and oxidizer valves. The main chamber oxidizer and fuel shutoff valves are helium actuated. The gas generator fuel and oxidizer valves use similar sleeve valves, and the main chamber oxidizer and fuel shutoff valve use similar poppett valves. The health monitor is integrated with the controller but electrically isolated to prevent health monitor faults from propagating into the controller and jeopardizing engine safety.

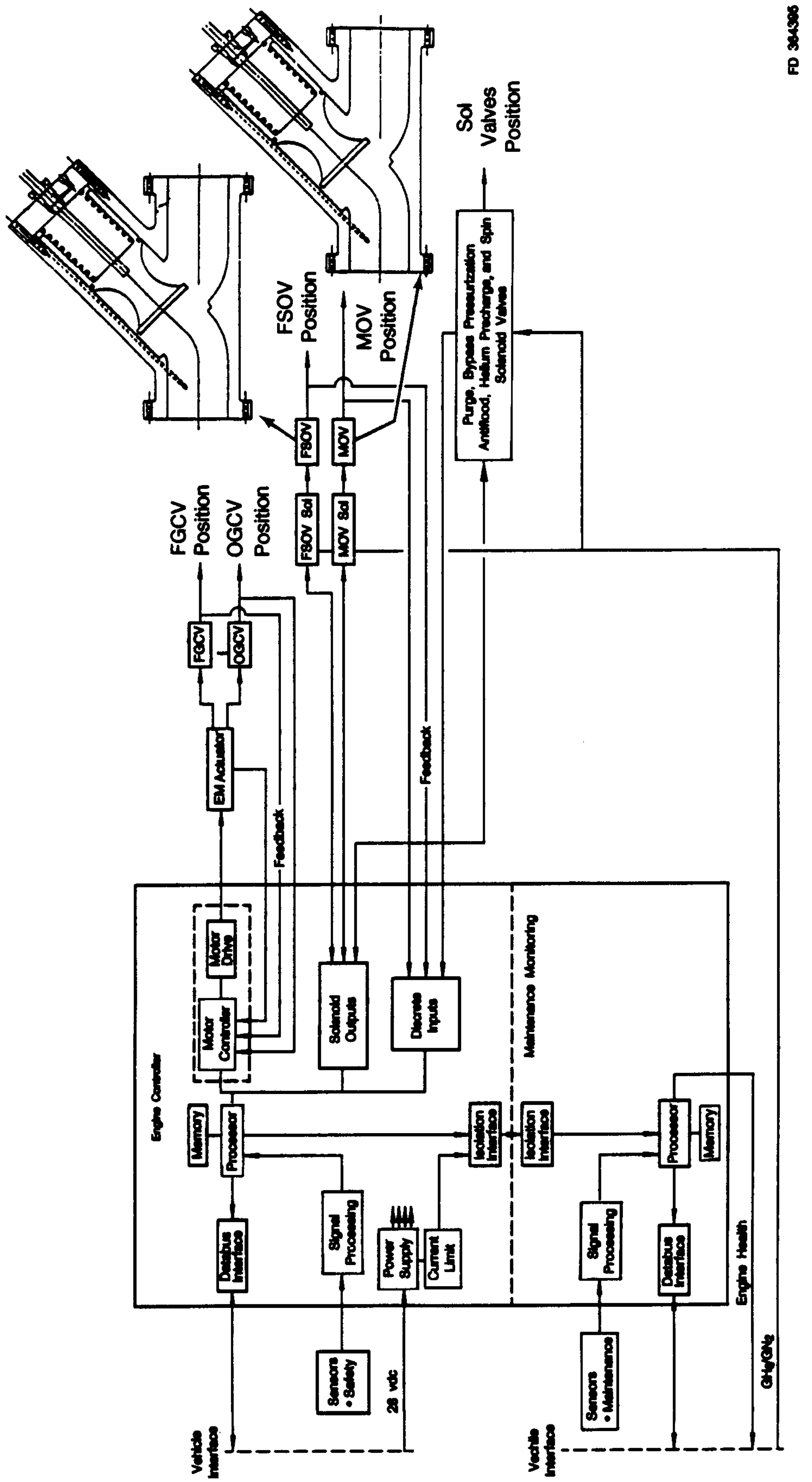

Figure 2.2.3-1. Engine Controller Health Monitor System Functional Concept Simple Actuator Valves Provides High Reliability

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Engine thrust is regulated by trimming the gas generator oxidizer valve while engine mixture ratio is regulated by trimming the MOV. Oxidizer flow shutoff is provided by the gas generator oxidizer valve and the MOV while positive fuel flow shutoff is provided by the main fuel shutoff valve.

2.2.3.1 Control/Health Monitor Conceptual Architecture

Conceptually the controller/health monitor is comprised of two functions: (1) control and safety monitoring and (2) maintenance monitoring. Control functions are those required to start, maintain normal operating conditions, and shutdown the engine. Safety monitoring consists of real time engine evaluation to determine if an emergency shutdown is required. Maintenance monitoring consists of functions which are not critical for flight, but necessary to determine maintenance action.

The STBE engine uses a simplex full authority digital electronic engine control with dual channel input/output (I/O). A single channel control with an effector system designed to provide fail-safe conditions upon loss of controller function meets the fail-safe design requirement. Controller reliability and fail-safe requirements are met with dual I/O interfaces. Under normal operating conditions, the controller I/O interface receives inputs from dual sensors and the information is processed by a single microprocessor.

The output interface supports solenoids with dual windings and a dual channel EMA interface. One of the two solenoid windings in each device has the capacity for solenoid operation in the event that one winding fails opens. Shorted solenoid switches are accommodated by switching both high and low sides of the solenoid. The EMA interface is a dual active effector system with single processor control. Under normal conditions, each output interface provides one half the drive signal necessary for actuator control. If one of the EMA interfaces fails, the current drivers in the failed interface are depowered and the gain in the remaining interface is doubled to provide full control capability. The dual active interface provides smooth transfers to single channel upon failure.

Actuator loop failure detection is provided by current wrap around, feedback failure detection, and open loop detection. Current wrap around is provided by measuring actuator winding current and comparing the result to the requested value. Feedback failure detection is accomplished by comparing the feedback to request. In the event that an actuator failure cannot be isolated to a given interface, the logic transfers to fail-safe.

An initiated built-in-test (IBIT) mode is provided by the controller to detect latent faults during prestart. In the IBIT mode, the controller sequences solenoid valves and EMAs throughout their operating range. This feature enhances mission reliability by providing a low-cost method for testing the system prior to launch.

The health monitoring system works as an interface between the electronic control, engine sensors, and the vehicle avionics while transmitting real time data to the vehicle health monitoring system (VHMS). Safety monitoring is performed by the electronic control with any performance or anomaly information passed to the maintenance monitoring unit through an isolation interface. Instrumentation not critical to flight operation is processed by maintenance monitoring electronics. Maintenance monitoring information is transmitted to the vehicle independently of the control.

2.2.3.2 Controller Hardware Concept

Highlights of the control/health monitoring system architecture include modular design of the engine control functional requirements. The system level design includes control of discrete inputs and outputs (solenoids and switches), actuator positioning, sensor signal processing, and control law processing. This system design is implemented using state-of-the-art hardware which provides a low-risk, low-cost flexible control.

Actuators/Valves

An extensive trade study was conducted in Phase A to select valve and actuator types based upon an assessment of reliability, cost, risk, and hardware commonality. The study considered pneumatic, hydraulic, and EMAs as well as sleeve, poppett, ball, and butterfly valves.

Ganged Gas Generator Valves/Actuation

The ganged gas generator valve system consists of an OGCV, a FGCV, and an EMA. These valves are ganged together to provide a fuel-rich, safe shutdown capability. A linear EMA sequences the fuel and oxidizer valves to achieve proper engine start, throttling, and shutdown. Additionally, an oxidizer gas generator bypass valve supplies 5 percent of oxidizer gas generator flow necessary for starting.

Oxidizer Gas Generator Control Valve

The OGCV is a modulating control valve that is located downstream of the oxidizer pump and upstream of the gas generator injector. The OGCV has a right angle inlet to outlet translating sleeve that is contoured to provide an area versus stroke relationship to meet the 3 percent accuracy requirement at all engine conditions. To meet the fail-safe safety requirements and to minimize required actuator force, the OGCV is pressure balanced and spring-loaded in the closed direction.

Fuel Gas Generator Control Valve

The FGCV is an on/off valve located downstream of the primary nozzle coolant exit and upstream of the gas generator injector. The FGCV is pressure balanced, spring-loaded closed and uses the same sleeve as the OGCV, but has been contoured specifically to provide fast opening.

Electromechanical Actuator Module

The EMA module consists of a dual channel actuator controller and a linear ballscrew actuator. The actuator module consists of dual switched reluctance motors directly coupled to a ballscrew device.

Main Oxidizer Valve

The MOV is a helium actuated poppett valve which provides oxygen to the thrust chamber. The valve provides ± 10 percent trimmability at the open position for engine mixture ratio trimming during the engine acceptance testing.

Fuel Shutoff Valve

A helium actuated poppett type valve identical to the MOV provides fuel shutoff capability at low cost.

Ancillary Valves

Solenoid actuated poppett type ancillary valves will provide propellant purging upon engine shutdown, tank pressurization during engine operation, pump interstage dam pressurization, and oxidizer gas generator valve bypass. Where required, check valves are located between the poppett and the propellant line to insure that the propellant is isolated from the helium system. The main chamber and gas generator LO_2 injector is purged with GN_2 during prestart and with helium during all other purging operations. All other ancillary valves will use gaseous helium (GHe) for actuation supply pressure. Limit switches provide information on the ancillary valves position.

2.2.3.3 Operation

Valve/solenoid/ignition sequencing during prestart, start, mainstage, shutdown, and post shutdown (in-flight) are shown in Figure 2.2.3-2.

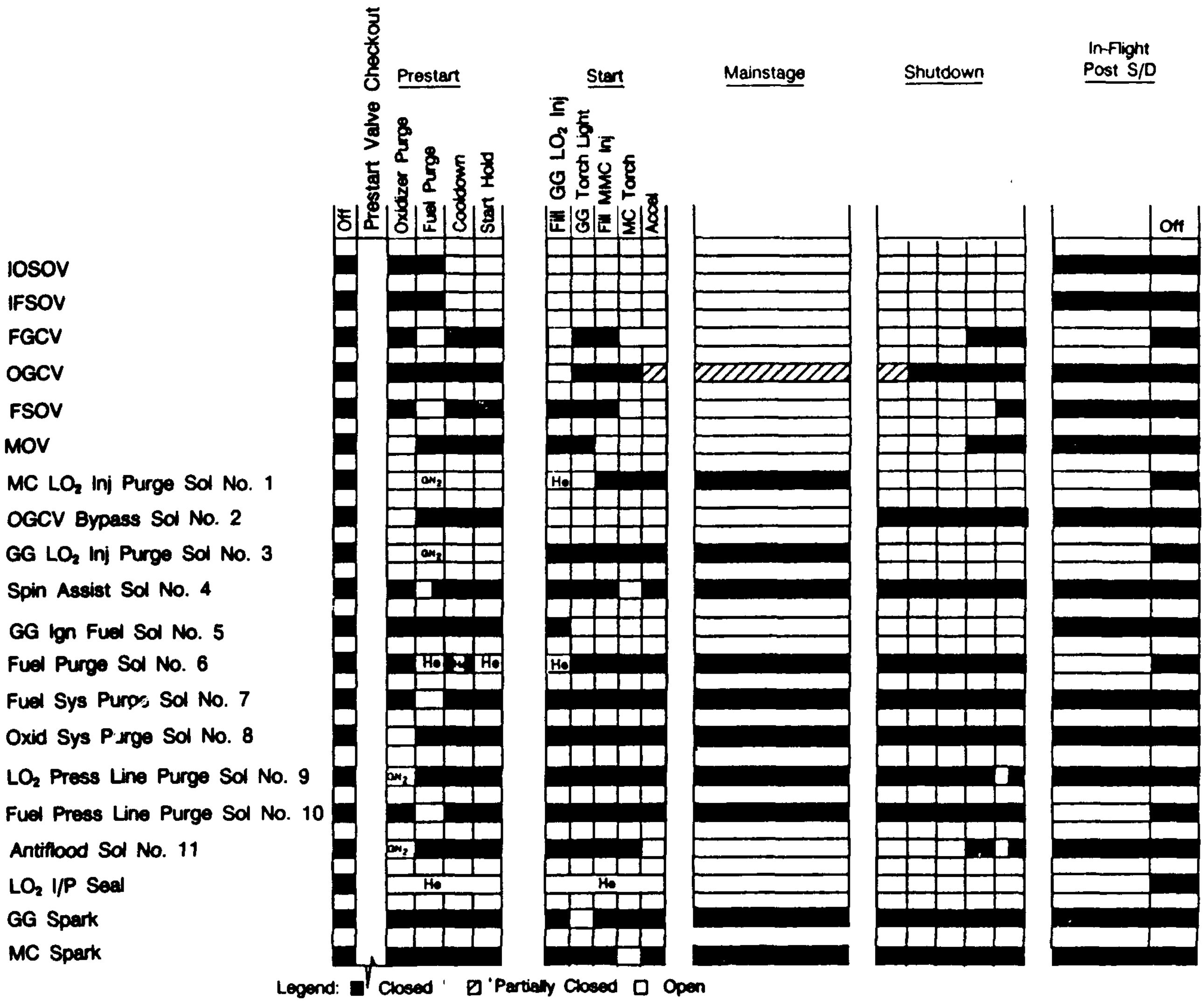

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Figure 2.2.3-2 Valve Sequencing Accomplished via Timed Logic

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Prelaunch Checkout

All valves are stroked from full closed to full open to full closed. Valve slew times provide verification that the valves are operational.

Pumps Cooldown

The turbopumps are cooled to cryogenic temperatures by CH_4 and LO_2 supplied through the vehicle inlet lines. Other than activating purge flows, no control valve sequencing is required by the engine.

Start

The engine start is a timed sequence process using a LO_2 lead for both the gas generator (GG) and main chamber (MC). In the LO_2 lead concept GG and MC fuel is delayed until the injector volumes are filled and LO_2 flow is established. This results in a smooth start and eliminates the potential temperature spikes and combustion instability associated with two-phase LO_2 injector flow.

Helium is introduced to the GG via the GG fuel injector simultaneously purging any oxygen from the fuel injector and providing helium spin-up assist to improve start repeatability and help in achieving the 5 second start requirement. Figure 2.2.3-3 shows the valve scheduling and thrust building characteristic during the start. Thrust buildup rates can be tailored to meet start requirement by modifying the GG valve start schedule. The oxidizer gas generator control valve bypass (OGCVBP) is used to provide LO_2 starting flow prior to opening the GG valves. Fuel rich torches are used for ignition of both the GG and MC. The use of a fuel rich torch is compatible with safe, fast, and reliable ignition when an LO_2 lead start is used.

Mainstage

Mainstage engine operation is an open-loop process. Analysis has shown that an open-loop control concept can be used to meet the ± 3.0 percent thrust and mixture ratio requirement, at constant inlet pressure, once the engine is trimmed at the 645K thrust point during the acceptance test. Engine mixture ratio and gas generator mixture ratio are remotely trimmed during engine acceptance testing by trimming the full open position of the MOV and FGCV, respectively.

Shutdown

Shutdown is performed by scheduling the propellant valves closed. The OGCV and the OGCVBP are closed first to power down the turbopumps. The MOV and the FGCV are then closed. The FSOV, which shuts off all methane flow to the engine, is closed last thus completing the shutdown sequence.

The GG and MC LO_2 injector purge solenoid valves are opened when the shutdown signal is received from the vehicle. Check valves are included to prevent backflow into the purge lines. When LO_2 injector pressure drops below the checked helium supply pressure the helium purge flow will commence. This flow purges any LO_2 trapped downstream of the OGCV and MOV after they are closed.

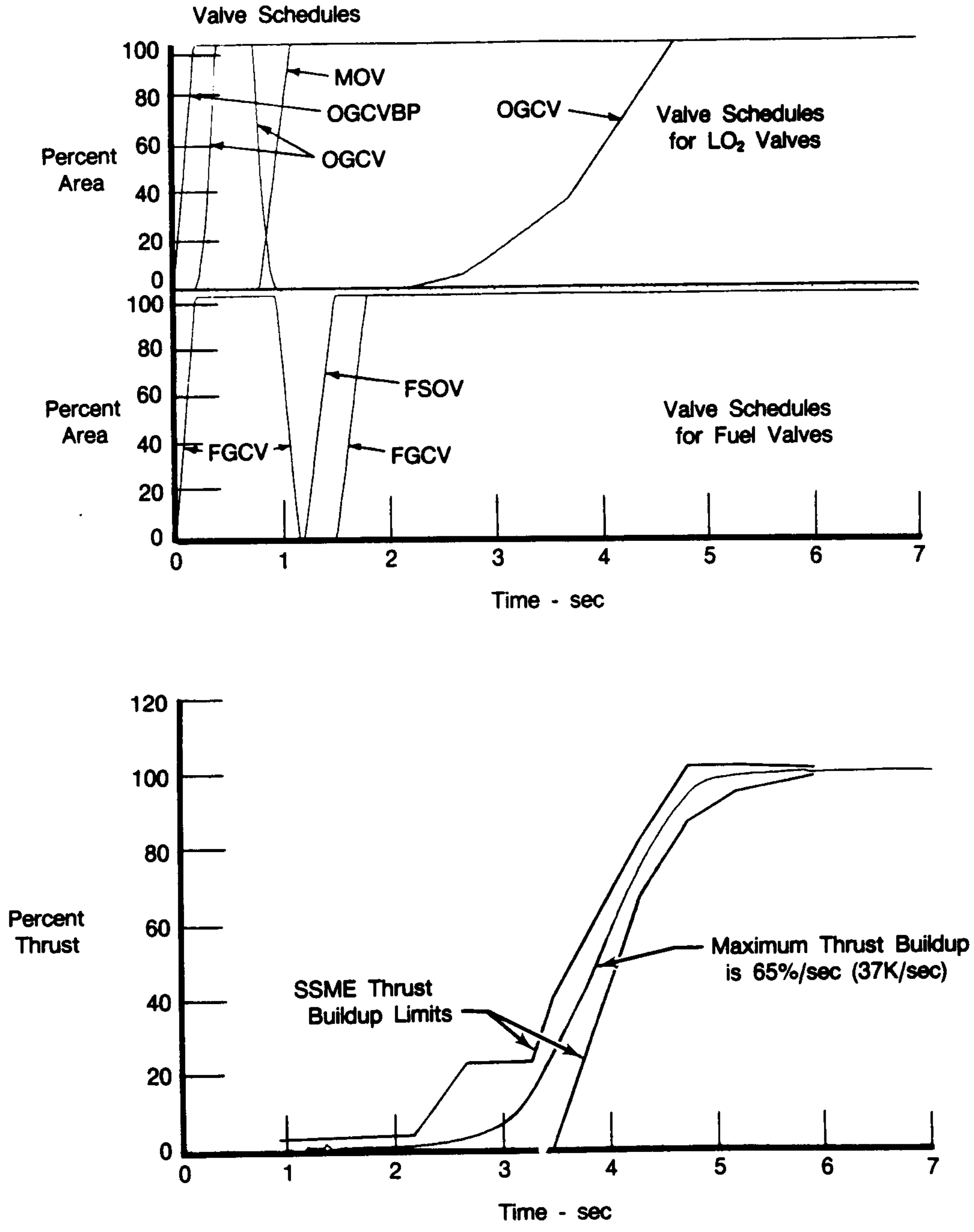

Figure 2.2.3-3. Engine Start Transients to 645K LO_2 Lead Minimizes Turbine Inlet Temperature Spike

Predicted characteristics of an engine shutdown from 645K thrust level are shown in Figure 2.2.3-4.

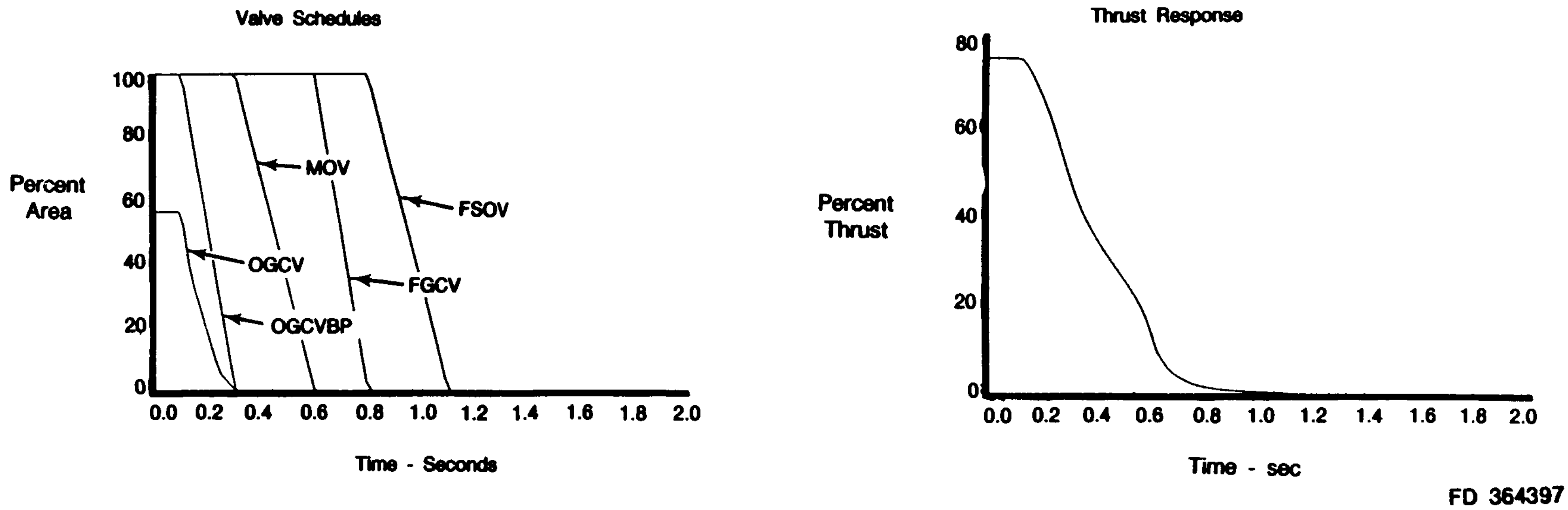

Figure 2.2.3-4. Shutdown Transient from 75% Thrust Reduced to Less Than One Percent in 1.0 Second

Post Shutdown

Methane downstream of the FSOV is purged out through the MC and FGCV. Methane upstream of the FSOV and oxygen upstream of the MOV, OGCV and OGCV Bypass is allowed to percolate back to the propellant tanks.

2.2.4 Support Devices

2.2.4.1 Engine Gimbal Mount

The gimbal assembly is based on a ball and socket spherical bearing which transmits axial thrust loads and a central retaining bar which restrains torsional movement. A Teflon impregnated fiberglass fiber woven fabric is applied to the ball surface to provide a dry, low-friction bearing surface which requires no maintenance and reduces the gimbal load requirements.

Cross or universal type gimbals were rejected due to the thrust level of the STBE. The gimbal bearing pressure is 30,600 psi, which is consistent with demonstrated values for this type of gimbal. Provisions for lateral adjustment to provide thrust alignment will be included between the gimbal lower bearing block and the thrust chamber mount.

2.2.4.2 GO_2 Heat Exchanger

The GO_2 heat exchanger is designed to provide gaseous oxygen to the oxygen tank for tank pressurization. The GO_2 heat exchanger uses the gas generator duct flow as the heat source to vaporize the LO_2. The heat exchanger surface is provided by three Haynes 214 stainless steel tubes wrapped in parallel around the exhaust duct. The exhaust duct wall is made of beryllium copper with trip roughened internal walls to enhance the heat transfer. The tubes are packed in powdered copper to structurally isolate the tubes from the duct wall, while providing a good heat transfer medium. This design eliminates the possibility of accidental mixing of the oxygen and the gas generator exhaust flow, thereby eliminating single event failures.

2.2.4.3 POGO Flight System Concept

POGO is an instability caused by the coupling of engine, vehicle structure, and feedsystem dynamics. Since large amplitude accelerations were observed on some early vehicles, POGO is recognized as a serious problem which must be considered during the design phase.

The preliminary POGO system design consists of a toroidal vessel around the LO_2 inlet line just upstream of the main LO_2 pump. Baffled passages around the circumference of the LO_2 inlet duct allow fluid communication with the POGO accumulator vessel. Helium is used to charge the POGO systems prior to engine start. Gaseous oxygen POGO pressurizing fluid is used during engine operation.

The POGO system design requires knowledge of vehicle longitudinal modes, propellant feed system dynamics, and engine dynamics, and requires working closely with the Advanced Launch System (ALS) vehicle contractors in Phase B to finalize the POGO system design parameters as this data becomes available.

2.2.5 Engine Integration

2.2.5.1 Engine Assembly Integration and Configuration

Component location and duct routing were affected by maintainability and weight considerations to achieve high reliability and low life cycle cost. The criteria used in layout of the engine assembly are listed in Table 2.2.5-1.

Table 2.2.5-1. Engine Assembly Layout Criteria

- LRU's Placed One Level Deep to Eliminate Unnecessary Removal of Adjacent Hardware During LRU Maintenance or Removal.
- Flanges With Adequate Clearance for Easy Access to Fasteners or For Leak Checks.
- Standard Hardware Used Where Possible to Reduce for Special Tooling and Initial Procurement Cost.
- Valves and Smaller Components are Mounted Close Coupled to the Major Components to Minimize the Use of Brackets for Their Support.
- Ducts Eliminate Delaminating of Major Components for Assembly or Seal Replacement and are Removable Without Affecting Major Components.
- Unimpeded Line of Sight is Provided for Inspection Displays and Position Indicators.
- Blind Assemblies or Installations are Avoided.
- Foolproofing of Fasteners and Seals Will be Provided.
- Component Positioning and Routing Minimizes Impact on Adjacent Areas During Changeout or Maintenance.

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2.2.5.2 Propellant Feed System

Propellant ducting design work performed during Phase A has focused on trade analysis to identify potential candidate designs for flange seals, gimballed flex joints, compression bellows, and stabilized inlet bellows ducts. Preliminary duct routing was performed to allow preliminary fabrication and maintainability assessment. Ducts have been sized to deliver flow velocities consistent with low pressure drop and separation characteristics. The limiting gimbal parameter is the pump inlet to gimbal centerline spacing which directly affects the stretch and twist induced in the scissor duct during gimbaling. The current spacing is limited by the pump turbine volute size and the manifold at the top of the regenerative nozzle section.

The inlet duct bellows using the current inlet spacings is limited based on buckling criteria. As bellows length, number of convolutions, and convolution height are iterated to provide adequate stress margin at a given gimbal angle, the critical buckling pressure is reduced. As a result, the stabilized scissor ducts are limited to approximately 10 degrees by buckling criteria.

Investigation will continue in Phase B to address concerns with the scissor ducts. Chiefly the possible thrust oscillation which could result as the engine is gimballed due to the volume change in the duct section and the bellows vibration driven by vortex shedding. Pressure volume compensating and wrap-around ducts will continue to be studied as alternate concepts.

2.2.5.3 Thermal Protection

To this point, the environment near a STBE derivative has been estimated for a vehicle design using three STME core engines and seven derivative STBE CH_4 engines on the booster. This analysis, with maximum heat flux values of approximately 23 Btu/ft^{2}sec, indicates that some form of thermal protection would be required for selected engine components, particularly electrical harnesses, the controller, and valve actuators and solenoids.

It is recognized that base heating effects occurring at altitude are strongly vehicle dependent. Extensive collaboration with the base heating community at NASA and the vehicle and engine contractors is required to arrive at a consensus on methods for analysis to provide base heating estimates.

2.2.5.4 Vehicle/Engine Interfaces

The STBE configuration meets all identified ICD physical interface requirements.

The physical location of the six fluid interfaces are shown in Figure 2.2.5-1. The pump inlet to centerline spacing has been limited by the diameter of the pump turbine volutes and the manifold on the base of the combustion chamber. The main pumps have been mounted vertically to maximize the potential for complete percolation of gas bubbles from the pumps during chilldown to eliminate the need for recirculation.

Two ancillary fluids have been identified as required for ground and flight purges, valve actuation, and turbopump spin assist. Nitrogen has been selected for ground purges of LO_2 lines and ground supplied helium has been chosen for ground purges of methane lines. Helium is also used for turbopump spin assist as well as all flight purges and some valve actuation. Location of the two ancillary fluid interface flanges, as well as the interfaces for fuel and LO_2 tank pressurization and the electrical panel have been located based on Space Shuttle Main Engine (SSME) locations.

2.3 ENGINE COST AND WEIGHT CHARACTERISTICS

As part of the STBE Phase A studies, updated program cost estimates were made for the program. These updated estimates reflect both configuration changes and refined engine cost estimates. All engine related design and development, operational production, operations, and product improvement and support program costs were addressed in the new estimates.

The methodologies and ground rules used to generate the updated cost estimates are the same as reported in Volume III of FR-19691-4. That document presents detailed cost data for the STBE program and should be referenced for these cost estimates.

Reduced system pressures and temperatures in the gas generator cycle compared to the SSME coupled with ample design margins allow lower cost materials and processes to be used and permit higher reliability, reduced maintenance and inspection requirements, and reduced development costs. Pratt & Whitney's STBE concept has at least a 75 percent reduction in the number of individual parts as compared to the SSME, and increased use of castings and other low-cost fabrication techniques.

The STBE has been designed to reduce the number of the parts, processes, operations, materials, assemblies, and use of standard parts wherever practical. Preliminary results of commonality studies conducted during Phase A are shown in Table 2.3-1.

One ALS scenario (Scenario 2) designated by NASA for the methane booster was evaluated for the STBE estimates. The Scenario 2 vehicle consists of a H2/O2 core stage powered by three reusable STMEs, and a CH_4/O_2 booster stage powered by seven reusable derivative STBEs. Nominal, maximum, and minimum flight schedules and production engine quantities were evaluated for this scenario. The STME used on the core stage is the baseline STME with the nozzle skirt, defined in the STME configuration study, FR-19830-3. The derivative STBE is the final CH_4 derivative configuration of the STME which has 73 percent cost commonality with the STME. The number of missions and quantities of engines assumed for each of the three missions schedules are summarized in Table 2.3-2.

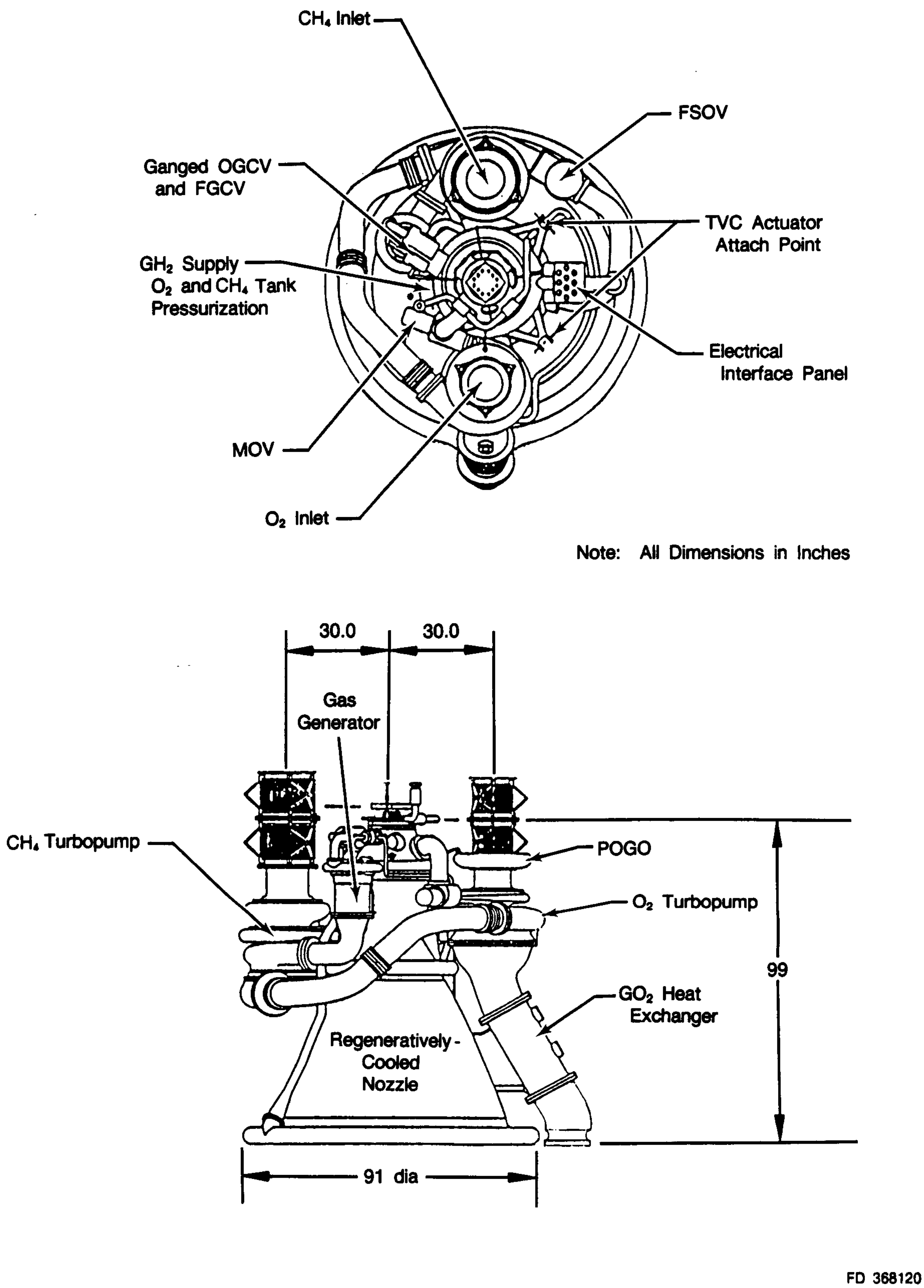

Figure 2.2.5-1. Derivative STBE Gas Generator Cycle Engine Assembly

Table 2.3-1. *Use of Common Hardware and Simplified Designs Reduces Part Count and Operations Cost*

STBE Components	Common Hardware and Simplified Design Features
Turbomachinery	
Impellers	Common Castings
Inducer Retaining Bolts	Identical Hardware
Bearings	Identical Ball and Roller Bearings
Rotor Seals	Uniform Labyrinth Seals/Identical Outer Turbine Seals Per Stage
Static Seals	Uniform Cross-Section
Turbine Disk and Shaft	Integral
Turbine Blades	Common Attachment Per Stages
Housings	Cast To Eliminate Welded Subassemblies
Washers,Sleeves and Spacers	Common Forgings
Fasteners	One Bolt Size — Two Lengths
Combustion Devices	
Interpropellant Plates/Elements (GG and Main Injector)	Common Casting Process
Ignitor (GG and Main Injector)	Identical Hardware
Manifolds	Cast To Eliminate Welded Subassemblies
Housings	Cast To Eliminate Welded Subassemblies
Regeneratively-Cooled Nozzle	SPIF Panels Construction Reduces Part Count
Engine Assembly	
Ducting	Uniform Size
Seals	Uniform Cross-Section
Fasteners	Standard Parts

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Table 2.3-2. *Advanced Launch System Scenarios For STME/Derivative STBE Program Cost Estimates*

	Scenario					
	Core Stage			*Booster Stage*		
	Nominal	*Maximum*	*Minimum*	*Nominal*	*Maximum*	*Minimum*
Total Number of Missions	300	625	250	300	625	250
Maximum Number of Missions/Year	14	33	12	14	33	12
Total Number of Operational Production Engines	175	350	100	425	850	275
Maximum Number of Production Engines/Year	30	30	30	70	70	70
Average Number of Reuses/Engine	5	5	7	5	5	6
Operational Production Period, Yrs	24	23	9	24	23	12

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Program cost estimates for the STBE are summarized in Table 2.3-3. Total program cost for the nominal mission schedule is \$5.7 billion, down from a previous estimate of \$6.6 billion. Costs for the other two mission schedules are also lower. These reductions are driven largely by lower engine unit costs and their impact on various phases of the program, particularly operational production.

Table 2.3-3. Space Transportation Main Engine/Derivative STBE Program Cost Summary

	Scenario 2 Mission Schedule		
	Nominal	*Maximum*	*Minimum*
Design and Development	1,593	1,593	1,593
Non-Recurring Operational Production	329	617	317
Core Engines	106	203	100
Booster Engines	223	414	217
Recurring Operational Production	2,716	4,832	1,729
Core Engines	892	1,587	521
Booster Engines	1,824	3,245	1,208
Operations	466	716	425
Core Engines	136	208	124
Booster Engines	330	508	301
Product Improvement and Support Program	642	642	642
Total Program Cost	5,746	8,400	4,706

Note: All costs in millions of constant FY87 dollars.

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Design and development program cost summaries for the STME and STBE are shown in Table 2.3-4. The updated development cost for the STME is $1043 million, down from $1183 million. The new cost for the STBE is $550 million which is a reduction of $108 million. These reductions result in a new total development cost of $1593 million for the STME/STBE program. These lower development costs are the result of lower engine unit costs and revised estimates of development rebuild hardware requirements.

Table 2.3-4. Space Transportation Main Engine/Derivative STBE Program — Design and Development Program Cost Summary

	STME Portion	*STBE Portion*	*Total*
Program Management	66	13	79
System Engineering and Integration	42	24	66
Engine Design and Development	171	63	234
Engine Test			
Test Hardware	291	154	445
Test Operations and Support	194	67	261
Flight Test Hardware	60	123	183
MPTA Test Hardware	31	59	90
Facilities			
Production	8	0	8
Launch	4	0	4
Test	22	2	24
Software Engineering	12	3	15
GSE	19	9	28
Tooling	68	10	78
Special Test Equipment (STE)	25	5	30
Operations and Support	30	18	48
Total DDT&E Program Cost	1043	550	1593

Note: All costs in millions of FY87 dollars.

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Theoretical first unit (TFU) costs are shown in Table 2.3-5 for the STME and the STBE. Also included are the percentages of TFU cost commonality for the STBE. The STME TFU cost is $9.4 million, down from the previous TFU cost of $11.3 million. The TFU cost for the STBE has decreased from $9.8 million to $8.6 million. These reductions result from changes in the engine configurations and more detailed cost estimates for the unchanged components.

Table 2.3-5. *Space Transportation Main Engine and Derivative STBE — Recurring Production Theoretical First Unit Costs*

System	*Core STME TFU*	*Booster Derivative STBE TFU*	*Derivative STBE Cost Commonality % STME TFU*
Engine Hardware	9457	8629	73*
Turbomachinery	2052	2095	59
HPOTP	950	993	35
HPFTP	1102	1102	80
Combustion Devices	3552	2558	76*
Main Injector	339	339	100
Thrust Chamber	604	676	0
Nozzle	1076	1008	100
Nozzle Skirt	998	—	—
Gas Generator	267	267	100
Igniters	268	268	100
Controls	1544	1644	68
Controllers/Monitors/Software	506	506	95
Sensors	285	285	100
Valves/Actuators	670	770	30
Interconnects	83	83	100
Propellant Feed	1155	1155	83
Ducts	759	759	80
Miscellaneous (System Hardware)	396	396	90
Support Devices	611	634	75
Gimbal	152	175	0
Tank Repressurization	292	292	100
Start System	17	17	100
POGP Flight System	150	150	100
Integration, Assembly and Test	143	143	100
Acceptance Test	400	400	100

*Reflects % of applicable STME hardware costs.
Notes:
1. All costs in thousands of FY87 dollars.
2. Lot size = 100.

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Also during the STBE Phase A studies, a cost estimate for the conversion of a CH_4/O_2 gas generator STBE to a H_2/O_2 gas generator STME was conducted. This estimate was made because the STBE is a design derivative of the STME, incorporating many of the same parts and fabrication techniques. Table 2.3-6 shows that the parts unique to the STBE will cost $1.64 million, while the total cost of a booster to main engine conversion is $2.19 million.

Table 2.3-6 *Cost Estimate for STBE Conversion to STME*

Item	*Additional Cost 87M$*
Parts Cost	$1.64
High Pressure Oxidizer Turbopump, 65% New	
High Pressure Fuel Turbopump, 20% New	
Thrust Chamber	
Nozzle Skirt	
Controller Software, 5% New	
Valves/Actuators, 70% New	
Ducts, 20% New	
System Hardware, 10% New	
Gimbal System	
Labor Cost	0.13
Acceptance Test Cost	0.40
Shipping Cost	0.02
Total Cost	$2.19

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SECTION 3.0
ENGINE COMPONENT DESIGN

The following component design and analytical descriptions represent work completed during the fifth and sixth extensions of the Space Transportation Booster Engine (STBE) Phase A contract.

3.1 GAS GENERATOR ASSEMBLY

3.1.1 Mechanical Design Description and Supporting Analyses

The gas generator assembly consists of three major elements; the injector, the combustor, and the igniter. The physical arrangement and key features are shown in Figure 3.1-1.

The injector employs 199 tangential-entry LO_2 swirler elements for improved fuel/oxidizer mixing. A flat injector face selected for reduced cost is made of a porous material that uses approximately 5 percent fuel flow to provide cooling for improved durability and reliability. The number of elements was chosen based upon trade study results that compared benefits of higher number of elements (increased combustion efficiency and reduced exhaust gas turbine inlet temperature profile) against manufacturing cost and chamber weight. The cylindrical combustor uses a transpiration-cooled liner in the combustion zone near the injector to provide improved durability in the highest temperature region and an actively cooled scrub liner further downstream of the combustion zone for design margin and reliability.

The torch igniter design features include; fuel-rich combustion products for easy, early ignition of the oxidizer lead propellant, dual spark plugs for redundancy, a coaxial injection element, and a cooled copper alloy liner.

The five engine interfaces of the gas generator assembly module employ standard bolts so that no special tooling is required. Reliability features include redundant seals, proven materials, and an integral interpropellant plate. The elements integrally cast with the flow divider plate eliminate potential leak paths of brazed assemblies. Structural analysis in Phase A included a generalized shell model to determine stress levels and required wall thicknesses.

3.1.2 Fabrication Processes and Substantiation

Two alternate fabrication methods were studied in Phase A to select the one chosen. The studies compared casting parts to net shape, forging, and machining the parts. Several advantages weigh in favor of casting.

Cast parts require minimal final machining as most surfaces are cast to net shape; only close-toleranced surfaces require machining. In contrast, forged parts require extensive machining, and must be individually fabricated, then welded or brazed together. Casting eliminates many welds and braze joints to increase reliability and decrease cost. The integrally/cast injector elements eliminate brazed joints that are susceptible to cracking and leakage from processing or contamination, and thus avoid labor intensive close tolerance machining, brazing, and inspection of mating parts.

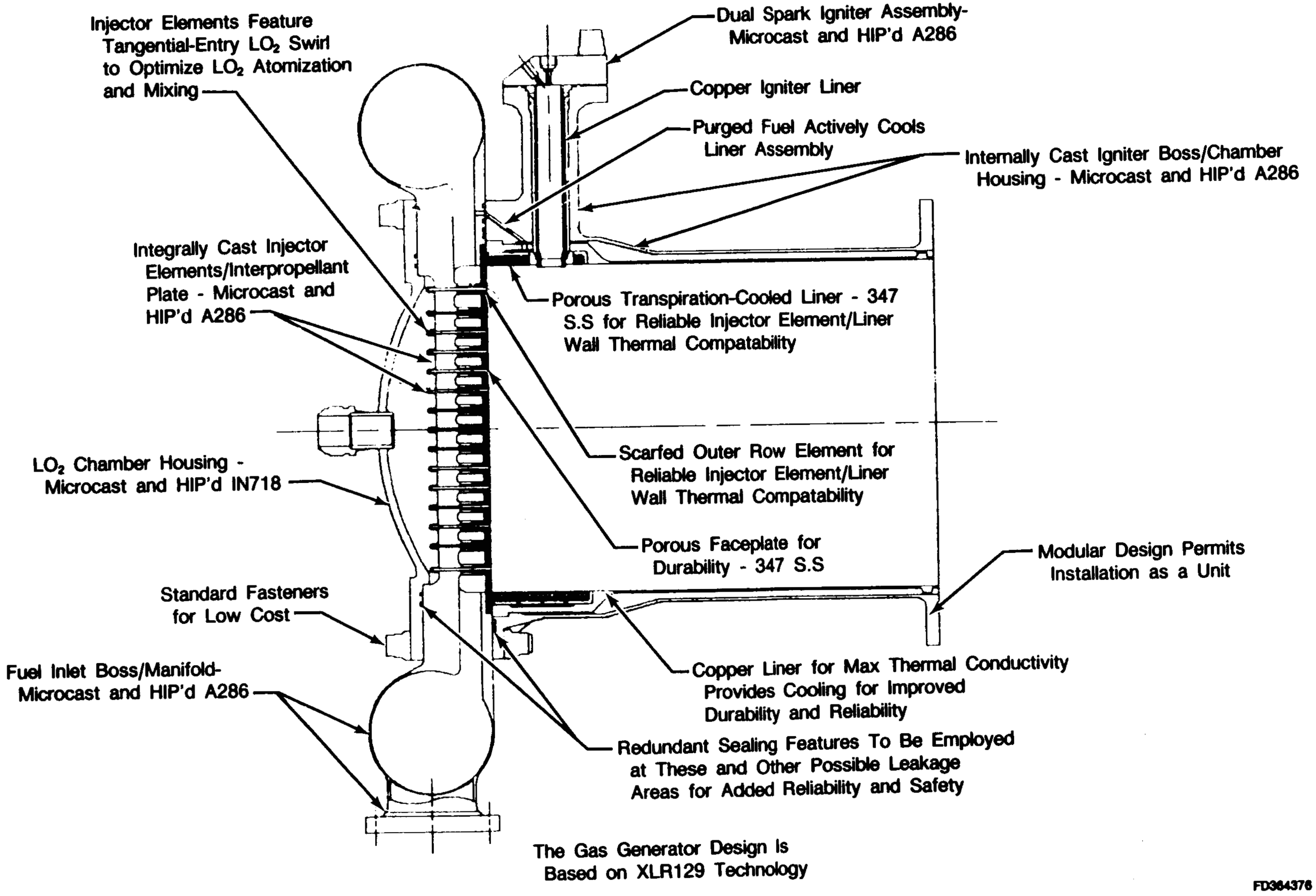

Figure 3.1-1. Reliability and Low-Cost Features of the Gas Generator Assembly

3.2 THRUST CHAMBER ASSEMBLY

The thrust chamber assembly consists of the main injector, the main combustion chamber, and the nozzle.

3.2.1 Main Injector Assembly

Mechanical Design Description and Supporting Analyses

The main injector shown in Figure 3.2-1 is the same injector used for the Space Transportation Main Engine (STME) and features 804 self-atomizing LO_2 injection elements. A vortex flow of LO_2 is induced in the tubes by entering through three tangential slots that are electrodischarge machined (EDM) tangent to the element tube inner diameter. The fuel enters the chamber through slots in a concentric annulus integral with the oxygen element. The number of elements was determined from empirical equations anchored by past experimental results (RL10, XLR129, SSME, etc.) and trade studies. These trade studies reviewed the impact of the number of elements on component weight, manufacturing cost, combustion performance, and stability.

Fuel enters the distribution manifold area from an annular mixing chamber. This mixing chamber combines cold fuel from the pump with heated fuel from the combustor coolant passages. Turbulence in the mixing chamber created by different fluid velocities provides a uniform fuel injection temperature and mass distribution. Chamber length is optimized to save weight and cost and allows rapid mixing and combustion of propellants. The injector faceplate is made of a porous material cooled with approximately 8 percent of the fuel flow. This faceplate is brazed to the element fuel sleeve outer diameter. Other features include: (1) bolted flanges for easy module replacement and maintenance, (2) proven torch igniter design, (3) integrally cast injection elements to reduce cost and to improve reliability, and (4) the need for only three fully inspectable welds.

Combustion Analysis

Thrust chamber combustion analysis is performed by using the Pratt & Whitney (P&W) supercritical combustion model shown in Figure 3.2-2. The primary features considered in this model are droplet formation, droplet heating, ignition delay, and burning rate. Atomization of propellants is a strong influence on the combustion performance. Pratt & Whitney has conducted extensive tests of the spray characteristics of tangential entry coaxial injection elements for the Alternate Turbopumt Development (ATD) program (element ID = 0.124 inch), National Aero-Space Plane (NASP) program (element ID = 0.136 inch) and in Independent Research & Development (IR&D) testing. The flow features of these elements are similar the STME design (element ID = 0.272 inch). These tests were done with fluids whose properties closely approximate the STME/STBE propellants. Figure 3.2-3 gives the data which were used to experimentally derive the equation that describes the spray for a STME/STBE type injection element. The spray correlation is a combined Reynolds number and Weber number, and at the STME/STBE main injector conditions, predicts an extremely small Sauter mean diameter (SMD) of 50 microns for the STME/STBE injection element. A vivid comparison of the advantages of swirling liquid flow is shown in Figure 3.2-4, which compares a tangential entry coaxial element with a simple non-swirl coaxial element.

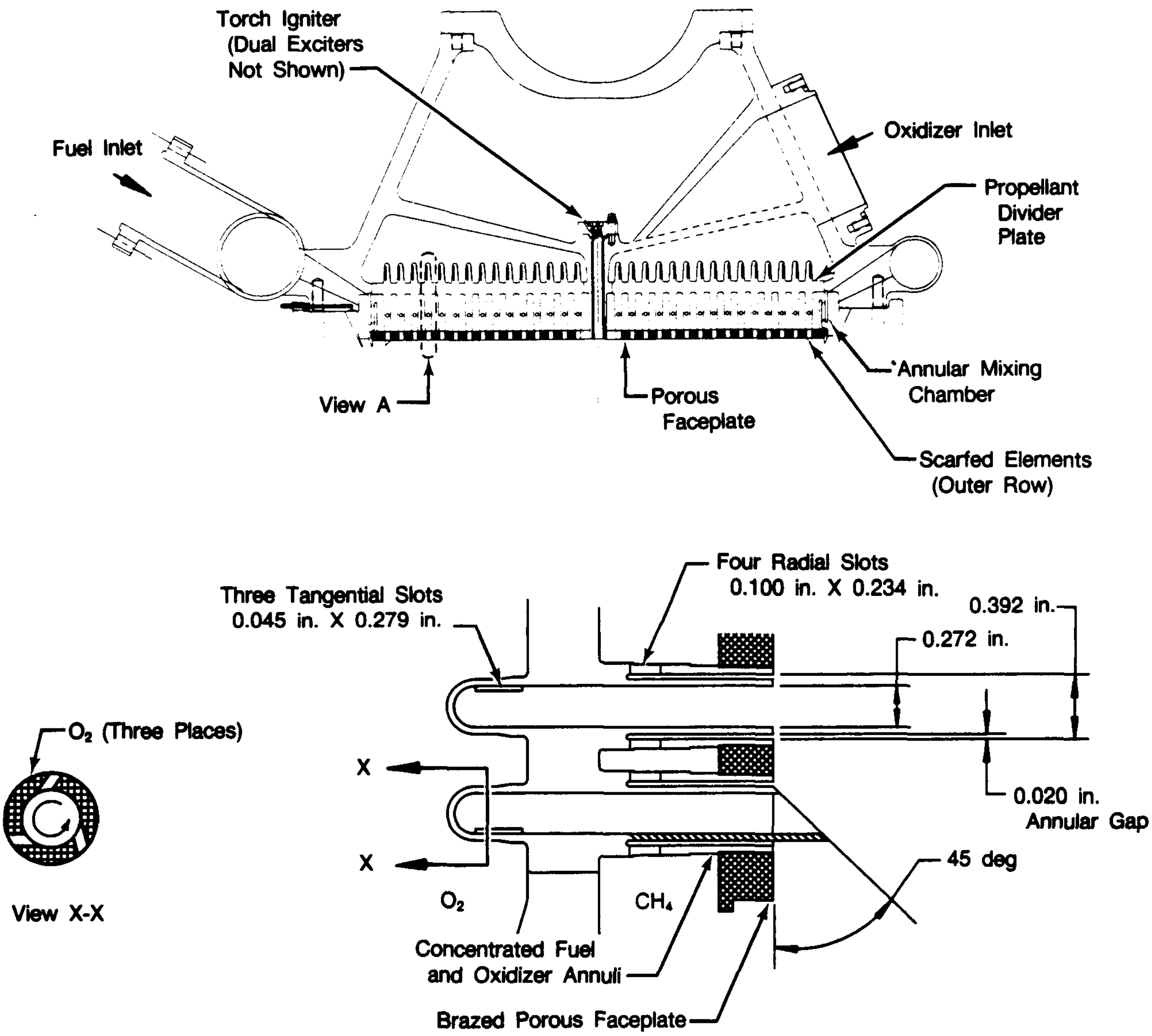

Figure 3.2-1. Main Injector Cross Section Shows Primary Design Features

Fabrication Process

The selected main injector design concept uses an integrally cast element and interpropellant plate, cast housing, and cast manifolds, thereby eliminating many structural welds. As in the gas generator interpropellant plate design, elimination of the braze joints reduces associated costs of close tolerance machining and eliminates potential contamination and leak problems and therefore enhances reliability. Further verification of casting process capabilities are being produced under IR&D programs.

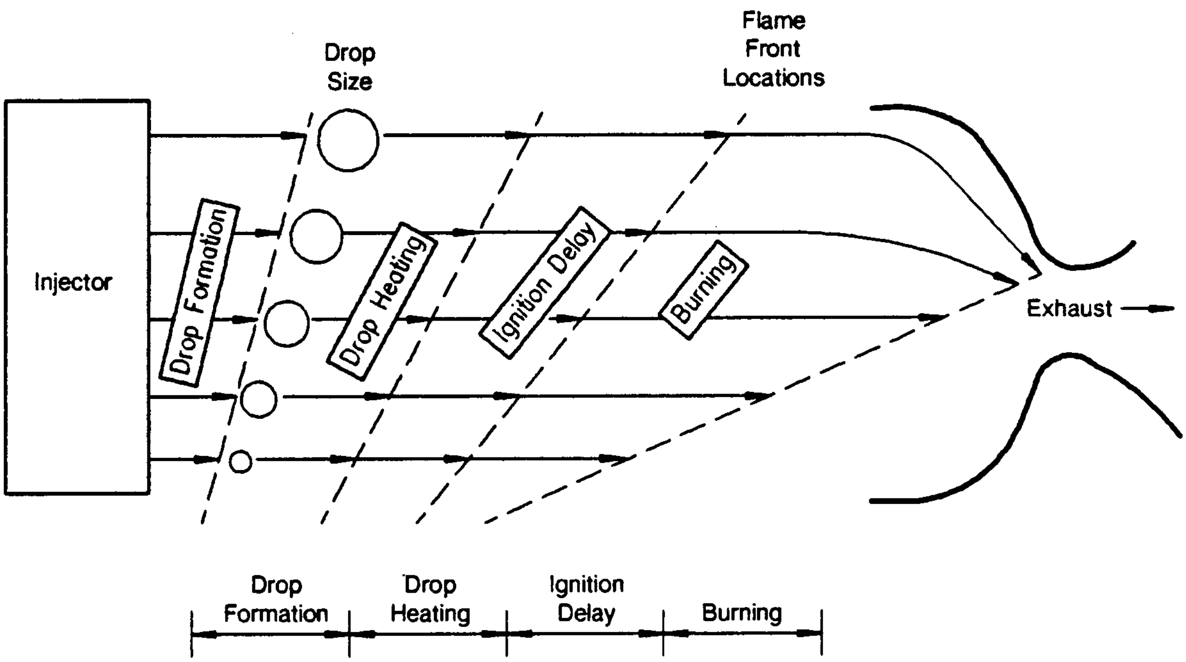

Figure 3.2-2. Space Transportation Booster Engine Main Combustor Supercritical Combustion Model

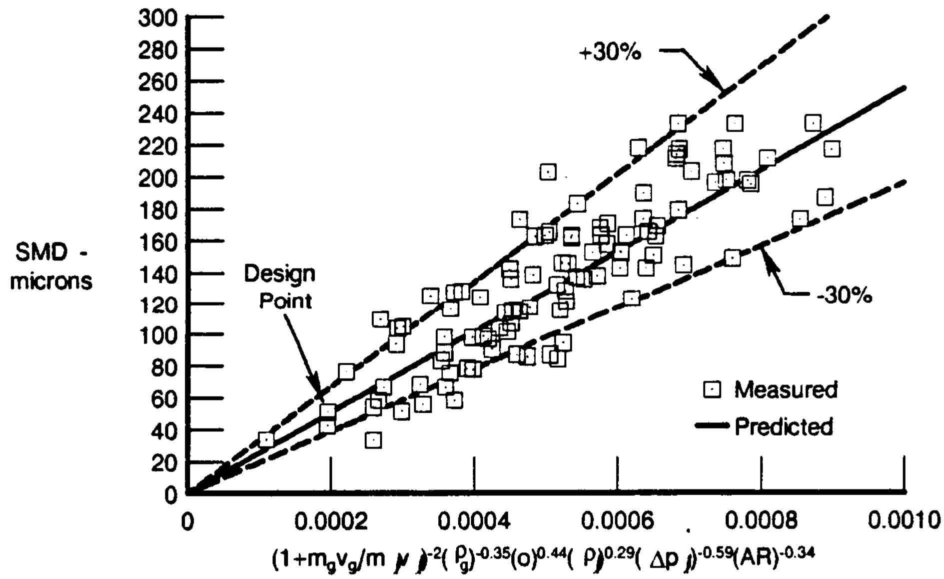

Figure 3.2-3. Experimentally Derived Spray Characteristics

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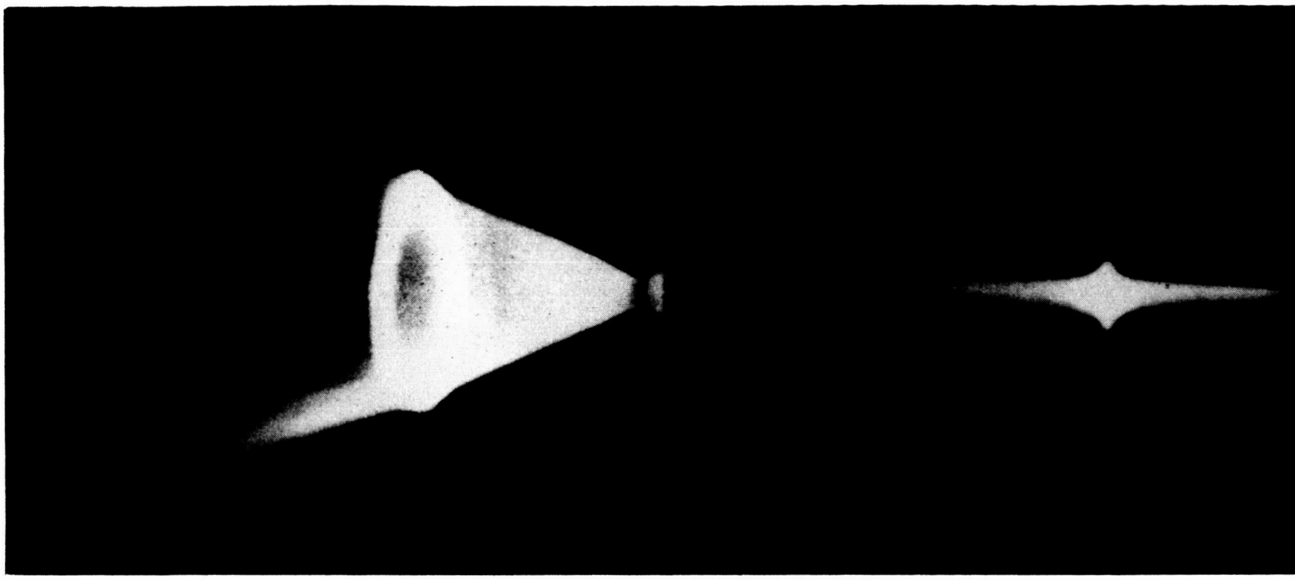


P&W Baseline Element

Concentric Axial Flow Element

Note: Time Lapse Photo With Laser Sheet Illumination

FD 358335

Figure 3.2-4. Injector Element Flow Characterization Tests — Simulated 109 Percent RPL, Matched O/F Velocities Shows Significant Improvement in Spray Distribution

3.2.2 Main Combustion Chamber

Mechanical Design Description and Supporting Analyses

The main combustion chamber module, shown in Figure 3.2-5, is a regeneratively-cooled, milled channel design. The STBE combustion chamber is a new component, although sharing many design features with the STME chamber. The inner liner is a forged NASA-Z copper alloy and the cooling passage geometry has been proven on the Space Shuttle Main Engine (SSME). The passages are closed out with electrodeposited copper and nickel and the structural jacket is made of a bicast aluminum alloy. The structural jacket is simplified by elimination of the coolant inlet and exit manifolds. The chamber and nozzle possess a common inlet manifold to allow coolant into both components. Redundant seals are used to improve reliability and provide a vent to a safe location. Preliminary thermal and load deflection calculations show that proper sealing can be achieved. The coolant exits the chamber and discharges directly into the injector thus eliminating the need for a discharge manifold and an external mixer.

The bicast structural jacket carried the chamber hoop pressure loads and the thrust loads through axial webs cast integral to the jacket. Since the design requires no structural welds except for bolt flanges, significant improvements in reliability and cost result.

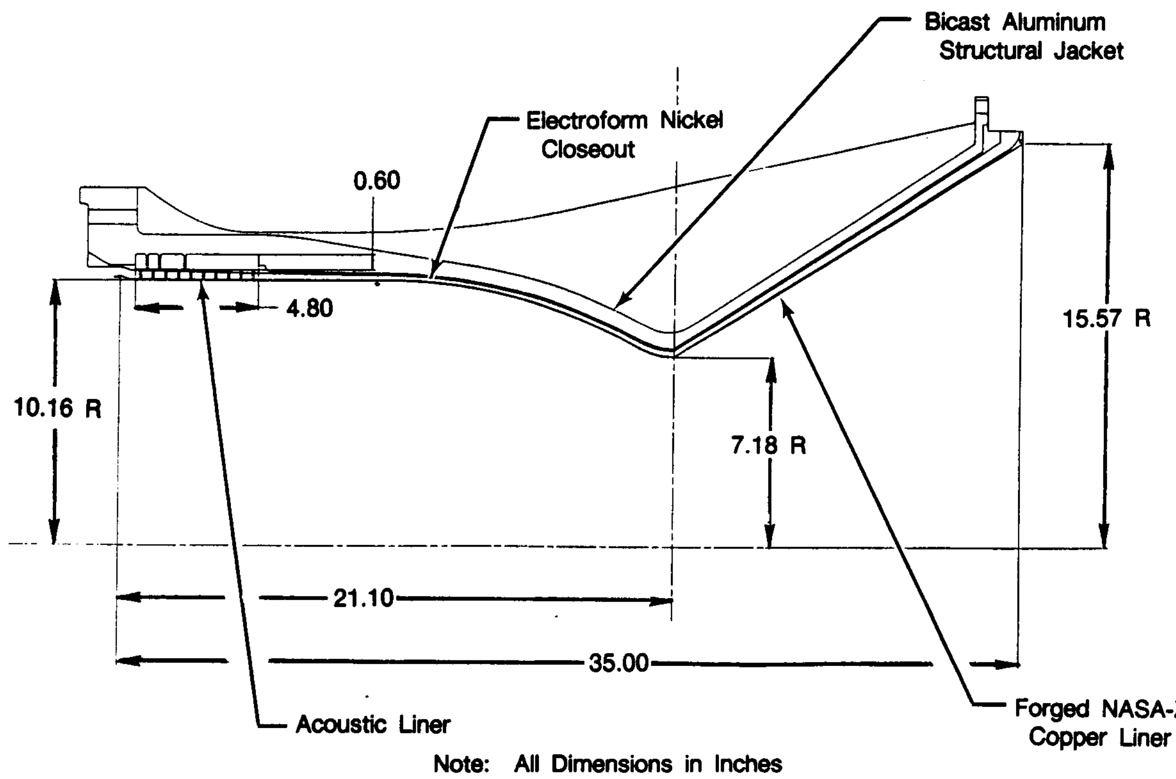


Figure 3.2-5. Derivative STBE Main Combustion Chamber Incorporates Proven SSME Concepts

Combustion Analysis

The required chamber volume is determined using the high-pressure combustion model described in Section 3.2.1. The important physical processes in this model are ignition delay and burning rate. Ignition delay is the time required for the propellants to reach combustible conditions and results in a non-combusting region between the injector and flamefront. Burning rate determines the time required for the propellants to reach the desired level of reaction completion. The combination of these two times, together with the combustor velocity, determines the chamber length requirement.

In addition to providing high performance, the combustor must also provide stable combustion. In Phase B, combustion stability analyses will be performed using the sensitive time lag theory. This theory has been widely used and accepted as one of the few valid design tools available. However, predicting combustion stability margin at the fuel temperatures envisioned is uncertain with today's tools. Until the new prediction tools being developed by the NASA and Air Force are developed it is prudent to provide a risk-mitigating design that incorporates a combustion stability device in the main chamber and main injector. An acoustic liner has been incorporated in the derivative STBE design. A similar acoustic liner has been successfully designed and tested at P&W to enhance combustion stability.

Heat Transfer Analysis

The STBE has a chamber pressure of 2250 psia, O/F of 3.3 and throat radius of 7.185 inch. The chamber liner has been designed with an acoustic liner and maximum wall temperature of

1425°R. The wall temperature limit of 1425°R has resulted in a coolant pressure drop of 2775 psid which is higher than the cycles 1944 psid. The chamber coolant heat pick-up is 63,094 Btu/sec. The SPIF nozzle, which is common to the STBE has a coolant pressure drop and heat pick-up of 381 psid and 36,086 Btu/sec, respectively. Figure 3.2-6 presents the geometry summary of the STBE chamber cooling system. Table 3.2-1 presents the coolant performance summary.

Recent NASA-Marshall Space Flight Center (MSFC) SSME Technology Test Bed engine experience indicates that designing the liner coolant passages to achieve wall temperatures below 1460°R significantly retards blanching which has proven to be a strong driver in liner crack initiation and premature wearout of SSME main combustion chamber (MCC) liners. Therefore, STBE nominal hot wall temperature has been set at 1425°R with the nominal wall thickness of 0.030 inch to provide margin for expected manufacturing variations. Enhanced cooling methods listed on Table 3.2-2 can be used to hold lower wall temperatures if manufacturing tolerances exceed the projected level.

Main Combustion Chamber Structural Analysis

The structural analysis for the main combustion chamber used advanced analytical methods such as MARC elastic/plastic finite element analysis models to ensure a robust design. The resulting configuration minimizes the alternating tensile-compressive cyclic plastic strain on the liner walls caused by the thermal fight between the liner wall and coolant passage closeout/structural jacket during transient and steady-state operation. Mid-channel wall thinning and the initiation of surface cracks due to cyclic plastic strain ratcheting is the life limited mode for the combustion chamber. Coolant passage geometry has been optimized to limit liner hot wall stresses from pressure loads.

Fabrication Processes

The fabrication techniques selected during the Phase A design studies are as follows: main chamber liner — near net shape casting spun to final shape; machined OD, ID and coolant passages; closeout — copper flash followed by nickel alloy plasma spray; main chamber structural jacket — bicast aluminum, cast and HIP.

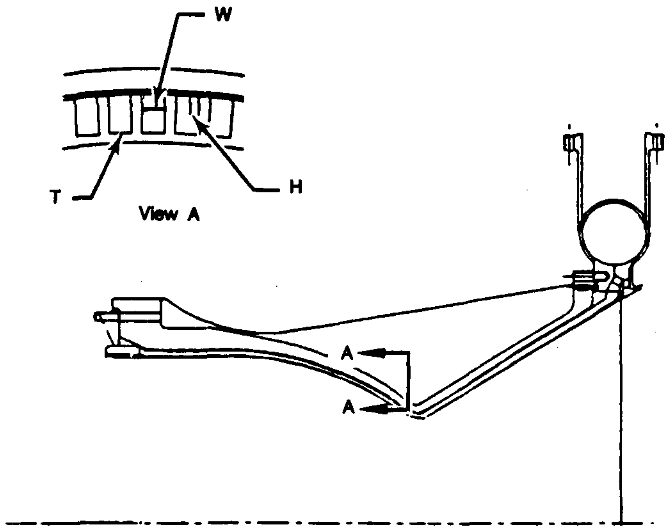
The bicast structural jacket is made simple and robust by avoiding welds and using no manifolds. These features greatly enhance structural integrity and reliability and reduce machining, welding, and inspection costs. Bicasting aluminum is simple, predictable, and adaptable to the Quality Management tool of process control. Alternate methods of forming the structural jacket also avoid welding (except bolt flanges) and manifolds, and include electroforming of nickel or higher strength nickel-cobalt or plasma spraying of nickel or stainless steel alloys.

3.2.3 Nozzle

Nozzle Mechanical Design Description and Supporting Analyses

Regeneratively-Cooled Nozzle

The regeneratively-cooled nozzle expands the main chamber gases from an area ratio of 5.8 to 1 to an area ratio of 29 to 1 and gasifies the CH_4 coolant prior to introduction into the gas generator combustor. Figure 3.2-7 shows the nozzle conceptual baseline design for the STBE. This nozzle is identical the the STME regeneratively-cooled nozzle.



STBE Chamber Coolant Liner Design Criteria

- Throat w/t Ratio </= 1.5
- Chamber w/t Ratio </= 2.0
- Passage Aspect Ratio, ph/pw </= 5.0
- Passage Land Width >/= 0.05 in.
- Wall Thickness, tht >/= 0.03 in.
- Passage Wall Roughness = 20 Microinches
- Hot Wall Temperature </= 1425°R

STBE Chamber Coolant Jacket Liner Geometry Summary

Chamber Coolant Liner Material:	NASA-Z
Chamber Construction:	Milled Channel
Number of Milled Passages:	427
Chamber Length:	20 in.
Divergent Nozzle Length:	15 in.
Throat Diameter:	12.834 in.
Chamber Diameter:	20.292 in.
Chamber Contraction Ratio:	2.5
Chamber L*:	42.5 in.
η_c(Throat):	99.3%

STBE Chamber Liner Coolant Passage Summary

Axial Length (in.)	Wall Radius (in.)	Passage Width (in.)	Passage Heigth (in.)	Passage Aspect Ratio	Passage Radius (in.)	Wall Thick. (in.)	Weld Width (in.)
13.900	15.566	0.090	0.450	5.000	0.001	0.100	0.142
12.894	14.979	0.090	0.450	4.996	0.001	0.095	0.133
11.889	14.391	0.090	0.439	4.877	0.001	0.090	0.124
10.883	13.760	0.090	0.428	4.758	0.001	0.086	0.115
9.877	13.127	0.090	0.418	4.643	0.001	0.081	0.105
8.871	12.493	0.090	0.409	4.543	0.001	0.076	0.096
7.866	11.860	0.090	0.400	4.442	0.001	0.072	0.087
6.860	11.226	0.090	0.366	4.066	0.001	0.067	0.077
5.854	10.582	0.090	0.331	3.676	0.001	0.063	0.068
4.848	9.935	0.090	0.292	3.247	0.001	0.058	0.058
3.842	9.289	0.067	0.285	4.273	0.001	0.053	0.072
2.837	8.642	0.057	0.285	5.000	0.001	0.048	0.072
1.831	8.006	0.057	0.230	4.032	0.001	0.044	0.062
0.825	7.395	0.057	0.165	2.898	0.001	0.040	0.053
0.000	7.185	0.057	0.120	2.105	0.001	0.038	0.050
-1.006	7.411	0.057	0.120	2.102	0.001	0.038	0.053
-2.012	7.859	0.057	0.123	2.160	0.001	0.038	0.060
-3.017	8.306	0.057	0.134	2.349	0.001	0.038	0.067
-4.023	8.730	0.074	0.110	1.495	0.001	0.038	0.056
-5.029	9.153	0.080	0.119	1.484	0.001	0.038	0.056
-6.035	9.539	0.080	0.127	1.587	0.001	0.038	0.062
-7.040	9.806	0.080	0.135	1.682	0.001	0.038	0.066
-8.046	10.020	0.080	0.140	1.745	0.001	0.038	0.069
-9.052	10.116	0.080	0.145	1.806	0.001	0.038	0.071
-10.057	10.161	0.080	0.149	1.868	0.001	0.038	0.071
-11.063	10.161	0.080	0.154	1.922	0.001	0.038	0.071
-12.069	10.161	0.080	0.158	1.974	0.001	0.038	0.071
-13.075	10.161	0.080	0.162	2.024	0.001	0.038	0.071
-14.080	10.161	0.050	0.250	5.000	0.001	0.038	0.101
-15.086	10.161	0.050	0.250	5.000	0.001	0.038	0.101
-16.092	10.161	0.050	0.261	5.212	0.001	0.038	0.101
-17.098	10.161	0.050	0.273	5.460	0.001	0.038	0.101
-18.103	10.161	0.050	0.287	5.732	0.001	0.038	0.101
-19.109	10.161	0.050	0.300	6.004	0.001	0.038	0.101
-20.115	10.161	0.050	0.325	6.508	0.001	0.038	0.101
-21.100	10.161	0.050	0.350	7.000	0.001	0.038	0.101

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Figure 3.2-6. Space Transportation Booster Engine Thrust Chamber Coolant Liner Geometry

Table 3.2-1. *Space Transportation Booster Engine Derivative (CH_4/LO_2) Gas Generator Engine and Coolant Jacket Performance*

Engine Performance		
Engine Thrust, lbf	627,373	
Chamber Total Pressure, psia	2,250	
O/F Ratio	3.3	
Chamber Flowrate, lbm/sec	1,956.8	
Coolant Performance		
Component	*Chamber*	*Nozzle*
Coolant Flowrate, lbm/sec	296	114
Inlet Temperature, °R	235	235
Exit Temperature, °R	482	615
Coolant Temperature Rise, °R	247	380
Coolant Heat Pickup, Btu/sec	63,094	36,086
Inlet Pressure, psia	5,354	4,524
Exit Pressure, psia	2,575	4,143
Coolant Pressure Drop, psid	2,778	381

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Table 3.2-2. *Chamber Liner Enhanced Cooling Methods*

Cooling Method	*Advantages*	*Disadvantages*
Increased Coolant Velocity	Lower Wall Temperature	High Coolant ΔP
Reduced Wall Thickness	Lower Wall Temperature	Slightly Higher Coolant ΔP Smaller Passages Increases Manufacturing Difficulty
O/F Biasing of Outer Row Injection Elements	Lower Wall Temperature Lower Coolant ΔP	Lower Impulse
Finned Coolant Sidewall	Lower Wall Temperature	High Cost Increases Manufacturing Difficulty

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The regeneratively-cooled nozzle is constructed from four AISI 347 stainless steel panels containing 1080 Super Plastic Inflation Formed (SPIF) tubular passages. The panels are welded together and are surrounded by a structural shell of closed cell foam with a filament wound graphite/epoxy composite overwrap. This shell carries all thrust generated hoop loads, and provides exterior nozzle ground handling protection.

The SPIF nozzle is welded to the INCO 718 inlet manifold and the Haynes 230 exit manifolds. The nozzle coolant inlet manifold supplies coolant to both the nozzle and the combustion chamber, eliminating a separate manifold. The manifold is made of INCO 718 which provides the necessary strength and stiffness for deflection and sealing requirements.

The AISI 347 stainless steel panels and Haynes 230 exit manifold were selected based on P&W experience with these materials.

STBE Nozzle Heat Transfer and Structural Analysis

The regeneratively-cooled nozzle is designed with 1080 passages. The number of coolant passages and their dimensions were set to meet the heat transfer requirements and the following structural criteria for high-reliability and low-risk: maximum stress < 90 percent of 0.2 percent yield strength, coolant Mn < 0.5, ultimate passage wall margin > 375°R, wall thickness > 0.013 inch, and wall temperature < 2260°R. This criteria is based on the demonstrated high reliability of the RL10.

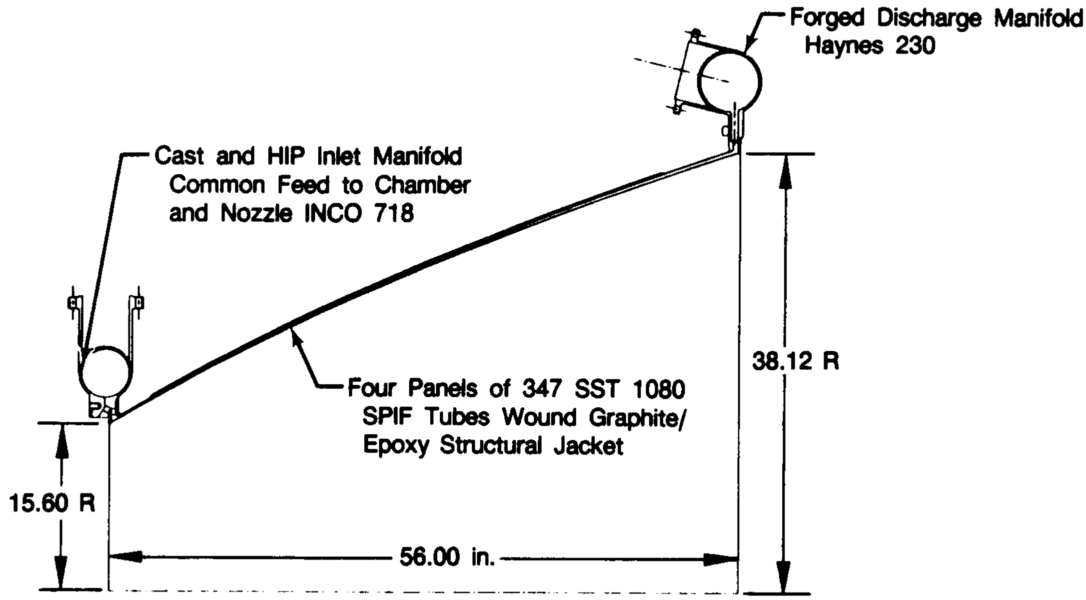

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Figure 3.2-7. Space Transportation Booster Engine Gas Generator Cycle Regeneratively-Cooled Nozzle

In Phase A, a trade study was conducted by varying the number of coolant passages to determine the relationship between stress margin, coolant pressure loss, and cost. Figures 3.2-8 and 3.2-9 present the variations in yield strength to tube stress and coolant pressure drop versus number of passages, respectively. As a result, the number of coolant passages was selected at 1080.

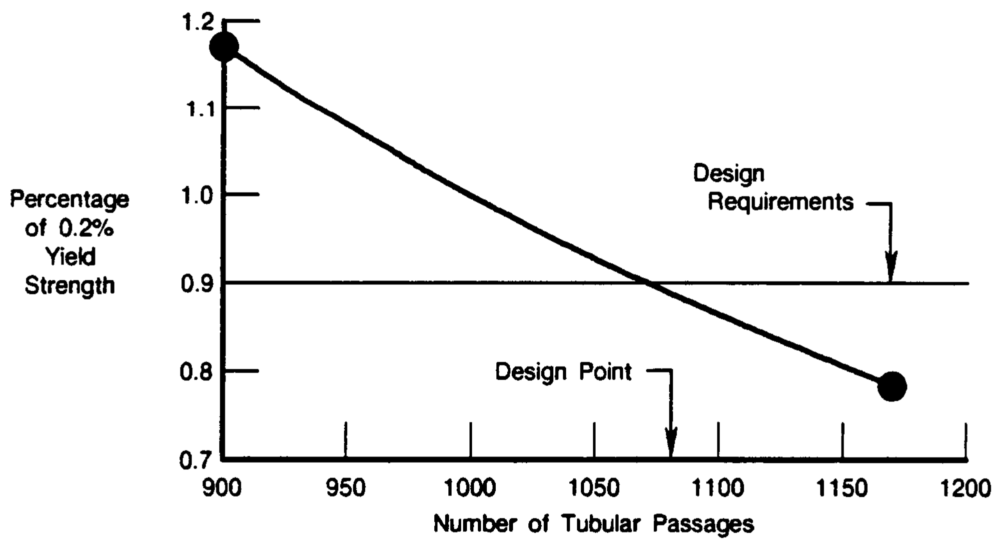

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Figure 3.2-8. More Smaller Tubes Reduce Stress

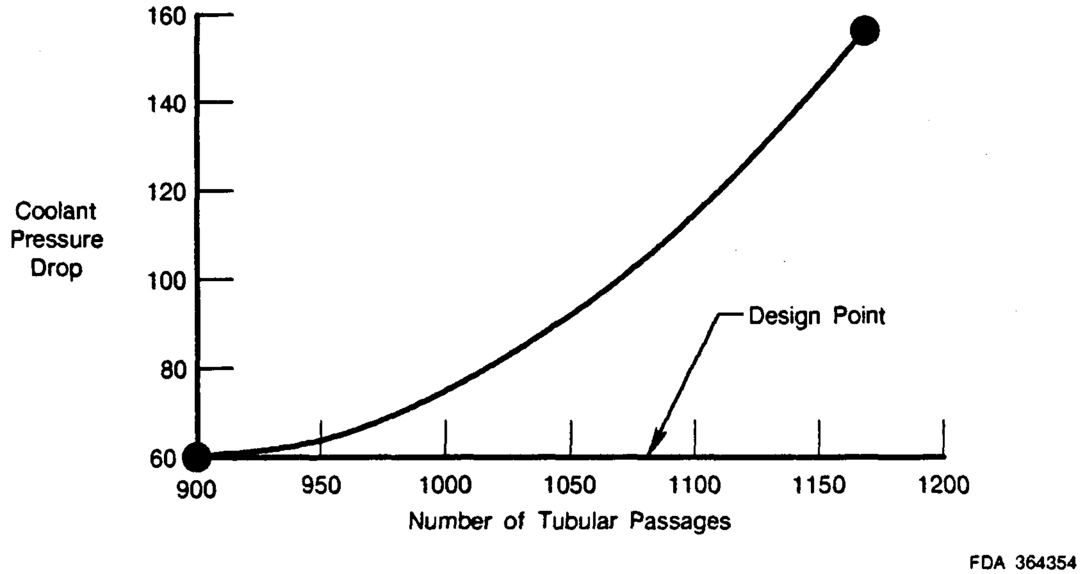

Figure 3.2-9. Coolant Pressure Drop is Acceptable With Design Tube Number

The nozzle is cooled with 114 lbm/sec of fuel at 235°R and 4524 psia and exits at 666°R and 3992 psia. Table 3.2-3 shows the coolant passage geometry summary.

Fabrication Process and Substantiation

The SPIF construction of the regenerative nozzle is achieved by diffusion bonding two sheets of metal together at the land area: the tapered passages are created by a maskant which leaves the masked area unbonded. The sheet pair is then rolled into a nozzle shape and the ends welded for sealing. This is then put into a die, heated to temperatures where superplasticity occurs, and the passages inflated to the final shape by inert gas pressure introduced into the dies. The nozzle is trimmed and the manifolds are added by brazing or welding and then final machined. This procedure is followed to produce the nozzle in four segments or panels which are joined by welding, brazing, or diffusion bonding. A demonstration sample of the SPIF nozzle concept has been made and displayed.

The fabrication/application of the structural jacket uses an automated composite wrap technique to provide structural support with a compliant closed-cell foam layer between the wrap and the SPIF nozzle. Differences in thermal coefficient of expansion for the wrap and tubular wall is accounted for by adjusting the wrap ply angles. Automated application techniques now exist which use variable winding to produce consistent thermal coefficient of expansion at any location on a surface. A type of closed-cell foam will be injected into the cavities between the nozzle and the wrap to provide a compliant layer and a seal to prevent cryo-pumping of moisture laden air and ice formation in these cavities. Several foam materials are currently under evaluation for this application.

Table 3.2-3. Space Transportation Booster Engine Derivative (CH_4/LO_2) Gas Generator SPIF Nozzle Tube Geometry Summary (No. of Tubes = 1080)

Axial Length (in.)	Wall Radius (in.)	Tube OD Width (in.)	Tube OD Height (in.)	Tube Aspect Ratio	Wall Thick. (in.)	Tube Spacing (in.)	Tube Flow Area (in.^{2})
13.90	15.57	0.081	0.120	1.49	0.018	0.010	0.0034
15.95	16.77	0.088	0.116	1.33	0.018	0.010	0.0036
18.01	17.95	0.095	0.112	1.19	0.018	0.010	0.0038
20.07	19.04	0.101	0.115	1.14	0.018	0.010	0.0043
22.12	20.12	0.107	0.121	1.13	0.018	0.010	0.0050
24.18	21.20	0.114	0.126	1.11	0.018	0.010	0.0057
26.23	22.19	0.119	0.132	1.11	0.018	0.010	0.0066
28.29	23.16	0.125	0.139	1.11	0.018	0.010	0.0074
30.34	24.13	0.131	0.145	1.11	0.018	0.010	0.0084
32.40	25.08	0.136	0.151	1.11	0.018	0.010	0.0094
34.45	25.95	0.141	0.156	1.11	0.018	0.010	0.0103
36.51	26.81	0.146	0.162	1.11	0.018	0.010	0.0113
38.56	27.67	0.151	0.168	1.11	0.018	0.010	0.0124
40.62	28.53	0.156	0.173	1.10	0.018	0.010	0.0134
42.67	29.30	0.161	0.178	1.11	0.018	0.010	0.0144
44.72	30.06	0.165	0.183	1.11	0.018	0.010	0.0155
46.78	30.83	0.170	0.188	1.11	0.018	0.010	0.0165
48.83	31.60	0.174	0.193	1.10	0.018	0.010	0.0176
50.89	32.32	0.179	0.197	1.10	0.018	0.010	0.0186
52.94	33.00	0.182	0.202	1.11	0.018	0.010	0.0197
55.00	33.68	0.186	0.207	1.11	0.018	0.010	0.0208
57.05	34.36	0.190	0.211	1.11	0.018	0.010	0.0219
59.11	35.04	0.194	0.215	1.11	0.018	0.010	0.0229
61.16	35.66	0.198	0.219	1.11	0.018	0.010	0.0240
63.22	36.26	0.202	0.223	1.11	0.018	0.010	0.0251
65.27	36.87	0.205	0.228	1.11	0.018	0.010	0.0262
67.33	37.47	0.209	0.231	1.11	0.018	0.010	0.0273
69.38	38.08	0.212	0.235	1.11	0.018	0.010	0.0285
69.90	38.23	0.213	0.236	1.11	0.018	0.010	0.0287

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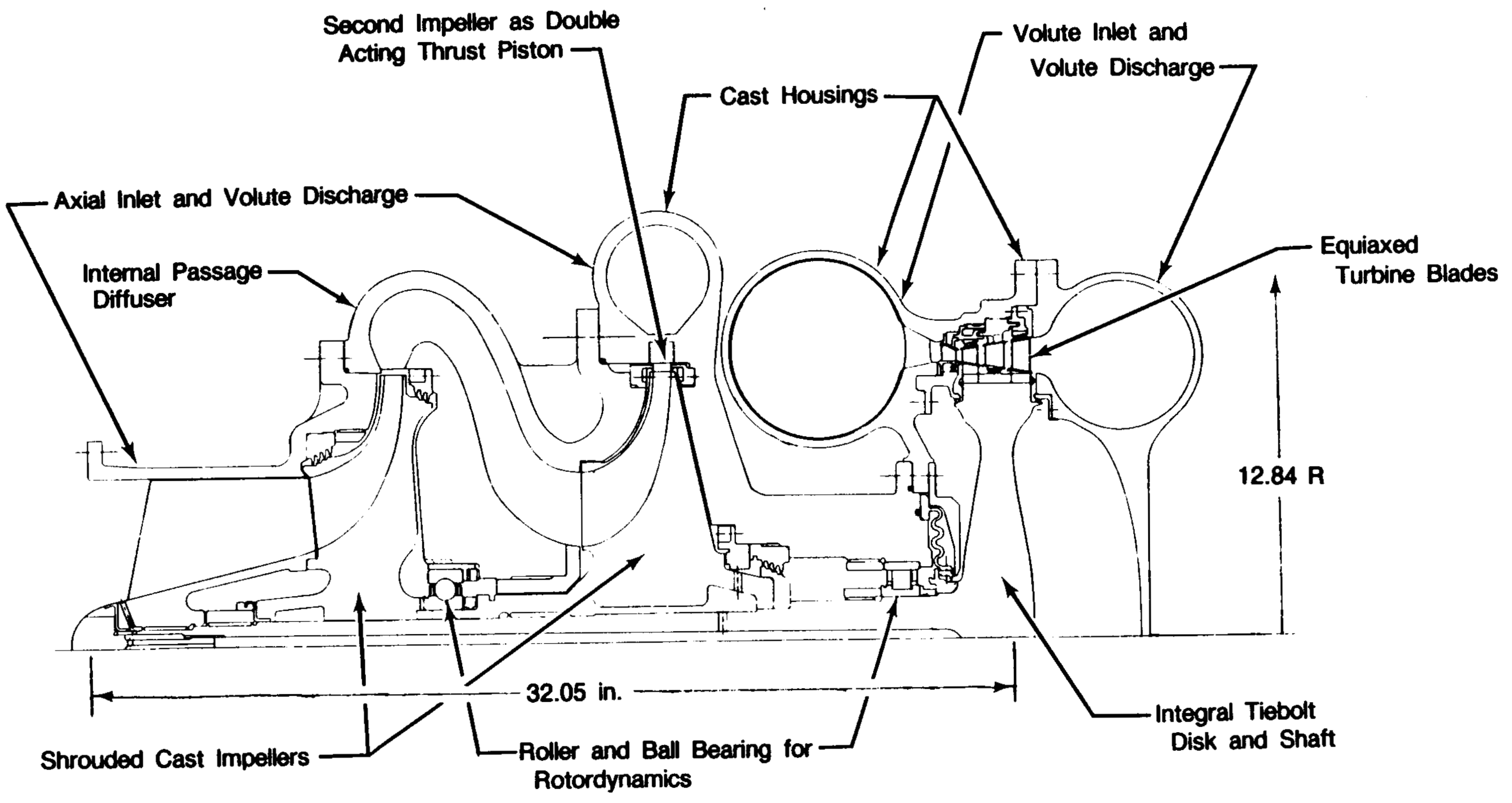
3.3 TURBOMACHINERY

3.3.1 Turbopump Design Description and Supporting Analyses

Methane Turbopump: As shown in Figure 3.3-1, a two-stage centrifugal pump with an inducer driven by a two-stage turbine are the major rotor components. The STBE turbopump is identical to the STME turbopump except the housing wall thicknesses are slightly increased to support internal pressures. Two common fine grain cast A110 extra low interstitial titanium shrouded impellers provide the required head rise. Casting and commonality permit low-cost parts to be produced uniformly. This STBE derivative fuel pump uses identical impellers and housings, but has a modified turbine blade and vane configuration compared to the STME.

The blades on the two-stage turbine have a common airfoil, blade platform, and firtree shape with only the blade height shortened for the first-stage blade to provide commonality and low-cost production for these components. Gas turbine proven hollow equiaxed blades of cast Mar-M-247 material provides reliability.



<u>Materials</u>

Pump Impeller	Titanium
Turbine Disk/Shaft	Super A-286
Turbine Blades	MAR-M-247
Turbine Vanes	MAR-M-247
Housings	INCO 718, MAR-M-247 and Haynes 230

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Figure 3.3-1. Space Transportation Booster Engine Fuel Turbopump

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An integral disk and tiebolt/shaft which is supported by a ball bearing and a roller bearing drives the pump impellers and inducer.

A simplex ball bearing at the pump end of the shaft combined with a stiff roller bearing at the turbine end of the shaft provides increased load capacity and longer bearing life for the rotor when compared to duplex ball bearings and eliminates the uncertainty of calculating load sharing.

A damper seal located between the impellers provides sealing between the pump stages and additional support damping for the rotor for safe reliable operation above the critical speed range.

The pump inlet housing, diffuser housing, and discharge housing bolt together to provide a housing and support for the rotor, a conduit and collector for the fluid flow, and a support for interstage, impeller, and shaft seals. Microcast and HIP INCO 718 material provides strong reliable housings that are dimensionally uniform, that approach forging strength, and are economical to produce.

The turbine inlet housing supplies hot gas flow into the turbine and provides support for the turbine vanes. It is supported by the pump discharge housing to reduce thermal loads. Microcast and HIP Haynes 230 alloy produces a low-cost housing that can safely operate in hot hydrogen and methane and provides a supply passage voluted to provide constant velocity and volumetric flow into the turbine at all stations around its circumference.

A one-piece stator with 105 vanes cast from fine grain Mar-M-247 turns the hot gas flow into the turbine. A cast single piece 2nd-stage stator with 174 vanes directs the flow into the 2nd-stage turbine. A Mar-M-247 stator support positions the first and second stators and supports the tip shrouds for the turbine. A turbine discharge housing cast from Haynes 230 is bolted to the turbine inlet housing trapping the vane support and vanes.

Controlling the pressure on the front face and backface of the 2nd-stage impeller through the use of seals located at the ID and OD of each face and activated by the axial travel of the rotor provides axial thrust balance to increase bearing life.

The sealing during pump cooldown is provided by a convoluted diaphragm liftoff seal located between the pump housing and turbine disk. The simple design of this type seal eliminates failure modes and leakage drain requirements.

Oxidizer Turbopump: A single-stage pump with an inducer driven by a two-stage turbine comprises the working elements of the pump rotor as shown in Figure 3.3-2. The hot gases discharged from the fuel turbopump turbine are used to drive the oxidizer turbopump turbine. An interpropellant seal between the turbine and impeller prevents the turbine gases and the oxygen from mixing. A thrust piston located between the turbine and interpropellant seal provides axial thrust balance to limit the bearing axial thrust loads.

A Microcast and HIP Inconel 718 impeller and inducer with tip speeds below 1750 fps are inexpensive to manufacture and provide reliable, low-risk performance. The tiebolt shaft and disk design and the bearing support system design are the same as the methane turbopump. The shaft supports the inducer, impeller, interpropellant seal rotating elements, and the thrust piston and drives the impeller and inducer. The disk portion of the shaft contains both stages of the turbine blades, which have the same attachment on both the 86 blade 1st-stage and the 86 blade 2nd-stage. This eliminates the necessity to produce two turbine disks and provide separate firtree machining tooling for each stage, thereby reducing part count, complexity, and cost.

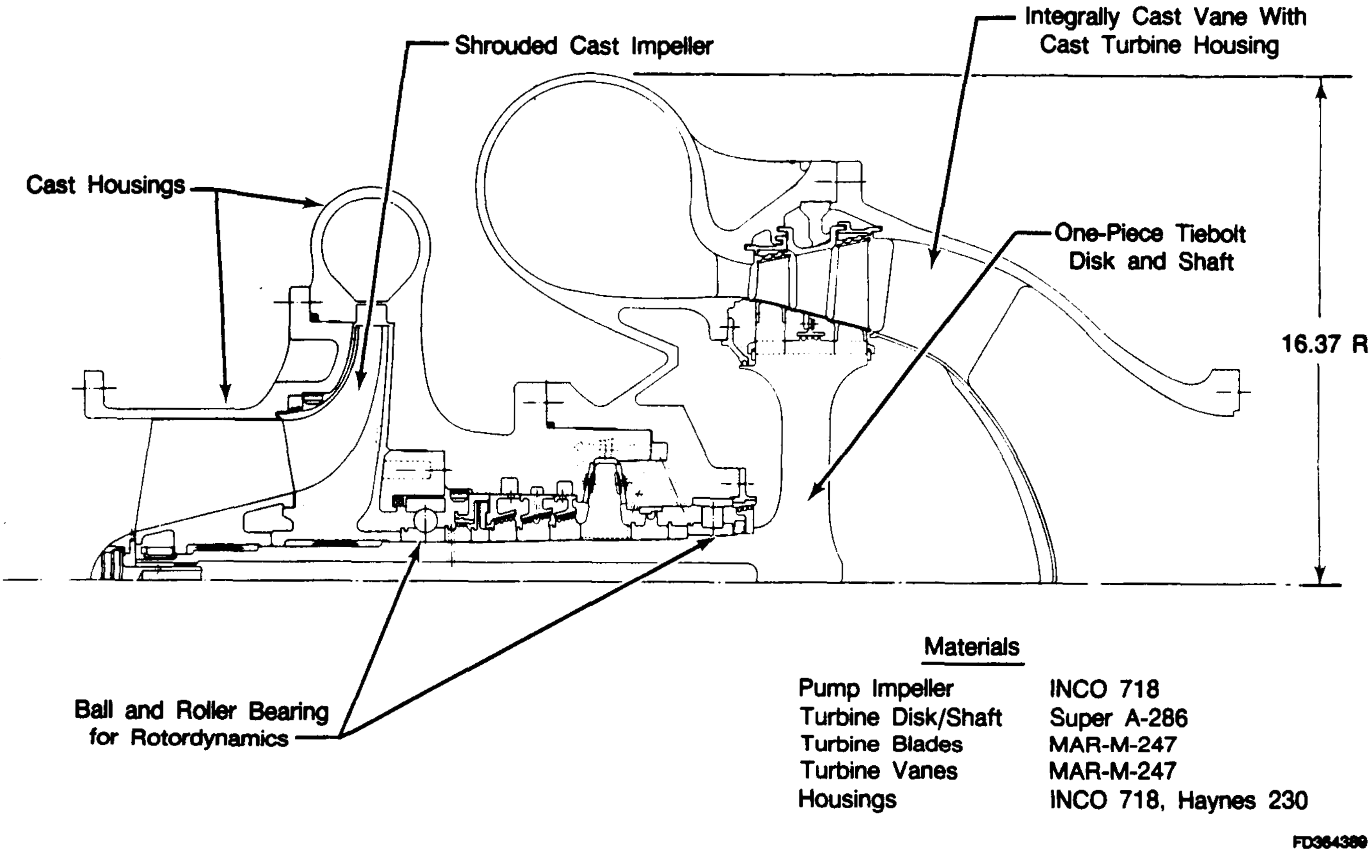

Figure 3.3-2. Space Transportation Booster Engine Oxidizer Turbopump

In the oxidizer turbopump, 1365°R turbine inlet temperatures permit the use of low-cost, low-risk, solid cast equiaxed Mar-M-247 blades. Turbine inlet vanes integrally cast into the Microcast and HIP Haynes 230 turbine inlet housing reduce part count and assembly complexity. The turbine inlet housing supports the roller bearing, the second stage stator, and the turbine discharge housing which is also made from Microcast and HIP Haynes 230. The turbine discharge housing retains the second stage and exhausts axially, thus eliminating turbine side load to produce increased bearing life as a result. The cast housings provide durable hardware with volutes optimized to provide maximum turbine performance.

An axial inlet in the pump inlet housing moderates shaft radial side loads to prolong bearing life. Casting these housings from microcast INCO 718 reduces cost, eliminates welding, and provides strength and durability equivalent to forged welded cases.

Bearing Design, Analysis, Fabrication, and Substantiation

Both of the high-pressure turbopumps employ similar rotor support systems. On the pump end, the rotors are supported by high-capability, split inner-ring ball bearings designed to accommodate both radial load and axial thrust load excursions. On the turbine end, the rotors are supported by large stiff roller bearings for stable rotor dynamics. All bearings are cooled directly by the cryogenic propellants. A solid film-lubricant obtained by rubbing contact with the bearing cage is used to lubricate the bearings. All turbopumps employ damper seals to provide supplementary support and to promote stable rotordynamics.

For the Phase A STBE bearing design, P&W has used the bearings currently being developed for the SSME-ATD Program. This could substantially reduce the level of effort required for STBE design verification. The existing ATD ball and roller bearing test rigs could be utilized for low-cost testing of the STBE bearings at a component level.

The baseline material is AMS 5618 (AISI 440C) which is used for bearing inner races and rolling elements. However, in order for the roller bearing outer race to flex over the negative internal radial clearance (IRC) rollers, a material with high fracture toughness will be used, AISI 9310. An alternate material for the inner races will be either AISI 9310 or M50Nil. Both 9310 and M50Nil are case carburized to harden the working surface for good wear resistance while the core remains soft for good fracture toughness. The additional fracture toughness provides greater margin against the effects of stress corrosion cracking. To provide protection from general corrosion effects on 9310 and M50Nil a corrosion resistant coating such as ion implantation, thin dense chrome, or gold will be considered.

Lubrication for the bearings are provided by self-lubricating cage design. The ball bearing cage consists of bronze-filled teflon segments which are riveted into a metallic shroud for structural support. The bronze-filled teflon was identified as the best performing material during the Cage Material Development Program (NAS8-11537) and has been used extensively by P&W for cryogenic ball bearings. The roller bearing cage uses a one-piece glass cloth filled teflon. Although this material has demonstrated excellent lubrication properties when used with roller bearings, high rolling element wear rates are experienced when it is used with ball bearing applications. Other low-cost materials which do not require reinforcement and therefore do not have the added manufacturing cost associated with the shrouded design for the ball bearings will be considered under the P&W Oxidizer Turbopump Advanced Development Program.

Turbine Design

Turbine Aerodynamics

Fuel Pump Turbine: A high-power density, two-stage pressure-compounded subsonic turbine was selected to provide low cost and risk. As a result of engine cycle parametric trade studies, P&W sets the gas generator propellant supply flows to create a gas generator discharge pressure level that is 75 percent of chamber pressure (Pc). This reduced pressure level enables the use of a high performance, low-risk turbine configuration with moderate stage pressure ratios. A turbine horsepower margin of 19,700 hp (33 percent) is available for development by simply increasing inlet pressure without changing turbine airfoils or raising the turbine inlet temperature.

The selected turbine incorporates low-risk aerodynamics relative to a high- pressure ratio supersonic velocity compounded turbine. Subsonic airfoil stresses are lower due to the absence of inlet and exit shock systems and reduced vibratory excitation. Supersonic turbines provide high performance for high-work, single design point operation, however, supersonic turbines have little tolerance for operation at off-design conditions.

Due to its low airfoil solidity, the subsonic turbine supports the low-cost emphasis of the program, by significantly reducing the number of turbine airfoils, approximately 200 less, when compared to the supersonic turbine.

Engine packaging also favors the pressure compounded turbine, as the axial exit from a velocity compounded turbine complicates the crossover ducting to the LO_2 pump drive turbine. Table 3.3.3-1 summarizes the benefits of the pressure compounded turbine.

Table 3.3.3-1. Pressure Compounded Turbine Benefits

	Velocity Compounded Turbine	*Pressure Compounded Turbine*
Efficiency	62	75
Inlet Pressure	100% P_c	74% P_c
Pressure Ratio	5.6	4.1
Airfoil Aerodynamic Risk	Exit Mn_{max}, = 1.4, Inlet and Exit Shocks-High Losses and Thin Leading Edges Intolerant to Incidence	Exit Mn_{max}, = 0.98 No Inlet Shocks, Lower Losses, No Blade Buffeting Loads
Development and Growth Power Margin	None. Must Raise Turbine-Inlet to Develop More High Pressure (HP)	Just Raise Inlet Pressure to 100% P_c (+30,000 HP Margin at No Temp Increase)
Turbine Exit Swirl Affects Crossover Ducting	Zero-Degree Swirl Into Vertically Down Discharge	52-Degree Swirl Into Tangential Volute Into Low Loss and Weight Horizontal Crossover Duct
Crossover Duct Fuel to LO_2 Turbine	Multi-turn Axial to Volute Longer Duct	No Crossover, Bends Volute-to-Volute Shortest Duct

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The turbine design has a mean diameter wheel speed which is compatible with allowable disk and airfoil root attachment stress criteria. The chosen wheel speed also provides a high design point wheel-to-gas velocity ratio assuring that there will not be a significant falloff at off-design point operation. The design point velocity ratio and stage loading are conservative so the aerodynamic risk is minimal.

Oxidizer Pump Turbine: A high power density, two-stage pressure compounded subsonic turbine was selected to provide low cost and risk. A single stage configuration for the same application would have had a 15 percent larger mean diameter (high cost) and an approximately 14 percent lower efficiency (high risk). Similar to the fuel turbine, the oxidizer pump drive turbine has moderate design pressure ratios per stage to provide a low-cost and low-risk aerodynamic design. The power margin available for development in the oxidizer pump turbine is estimated to be 11,400 hp.

The turbine design has a mean diameter and wheel speed which is compatible with allowable disk and airfoil root attachment stress criteria. It should be noted that the design wheel speed is set primarily by the pump hydrodynamics as was the case for the fuel pump turbine. The resulting design point velocity ratio is within the demonstrated good performance range.

Table 3.3.3-2 summarizes fuel and oxidizer pump turbine aerodynamic parameters including the predicted efficiency. The fuel and oxidizer pump flowpath elevations are shown in Figure 3.3-3 and 3.3-4.

Internal Flow Management

Turbopump internal flow management encompasses the control of all non-mainstream flows through the design and control of metering orifices and seals to provide the desired distribution of flows at required temperatures and pressures to control critical hardware metal temperatures, to minimize parasitic losses, and to provide for rotor axial thrust balance control.

Table 3.3.3-2. *Engine and Operating Point Gas Generator Cycle — Phase A STBE (CH_4/O_2) Engine*

Turbine	Fuel Stage One	Fuel Stage Two	Oxidizer Stage One	Oxidizer Stage Two
Flowrate lbm/sec	141.6		141.6	
Horsepower	38,351.0		28,792.0	
Power, Btu/sec	13,282.2	13,824.3	10,238.7	10,238.7
rpm	10,717		8181	
Gas Constant	96.33		95.63	
Gamma	1.1073		1.0999	
Inlet Stag. Temp, °R	1,797.0	1,723.6	1,609.4	1,556.0
Exit Stag. Temp, °R	1,723.6	1,647.1	1556.0	1,502.5
Inlet Stag. Press., psia	2,220.1	1,291.6	682.0	441.7
Exit Stag. Press., psia	1,291.6	710.3	441.7	276.9
Exit Static Press., psia	1,053.3	566.7	373.2	222.7
Stag. Press. Ratio	1.7189	1.8184	1.5441	1.5949
Velocity Ratio	0.3468	0.3466	0.3379	0.3413
Efficiency, T/T	0.7992	0.7877	0.8584	0.8275
Efficiency, T/S	0.5863	0.5780	0.6233	0.5697
Overall				
Gas Exit Discharge Angle	34.9		42.7	
Gas Exit Absolute Mach No.	0.6258		0.6251	
Stag. Press. Ratio	3.1257		2.4626	
Velocity Ratio	0.3467		0.3396	
Efficiency, T/T	0.7977		0.8451	
Efficiency, T/S	0.6729		0.6872	
AN^2	32.36	51.62	55.67	63.20
Blade Tip Radius	8.315	8.630	9.741	9.923
Mean Radius	8.036	8.1935	9.006	9.097
Root Radius	7.757	7.757	8.271	8.271
Blade Rim Speed, fps	725.5	725.5	590.5	590.5

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The control of critical hardware metal temperatures was based on lessons learned in the SSME ATD design and other high-pressure rocket turbopump tests. In turbopump designs with high turbine inlet temperatures, control of rim cavity gas temperatures is a critical requirement. An outflow of gases are needed at each interstage location to prevent the uncontrolled ingestion of hot flowpath gases. By controlling the temperature of the mixed gases in these cavities, blade platform to attachment temperature gradients can be controlled and minimized. The fuel turbine may require a mixed gas flow system. The oxidizer turbine can have a much simpler internal flow system because of its lower turbine inlet temperature.

The bearing coolant flows were established based on rig data from the SSME ATD Program. Methane will be used to provide cooling to the roller bearings in both turbopumps and to the ball bearing in the fuel turbopump. The ball bearing in the oxidizer turbopump will be cooled with LO_2 supplied from the rear side of the pump impeller. In Phase B additional data from these rigs will be used to confirm the coolant flows needed to ensure a reliable design. The interpropellant seal of the oxidizer turbopump is an all labyrinth seal configuration to provide maximum safety and minimum risk as tested in the SSME ATD interpropellant seal (IPS) seal rig. The seal package forms a helium dam that provides positive separation of oxygen leakage flow from the pump end and methane leakage flow from the rear bearing compartment. Additionally, this seal package restricts the propellant overboard leakage during engine operation. To minimize the overboard oxygen leakage a radially slotted rotating element reduces pressure and increases windage heat generation to vaporize the oxygen prior to exiting through the labyrinth seal. The rotor axial thrust balance control in the oxidizer turbopump uses a separate thrust balance piston based on previously successful P&W turbopump tests. The rotor axial thrust balance control of the fuel turbopump uses the 2nd-stage fuel turbopump impeller as the thrust balance piston based on P&W SSME ATD design effort. Both systems use the axial motion of the shaft to cause a reverse unbalance of pressures across the thrust balance piston to provide a restoring force to the shaft.

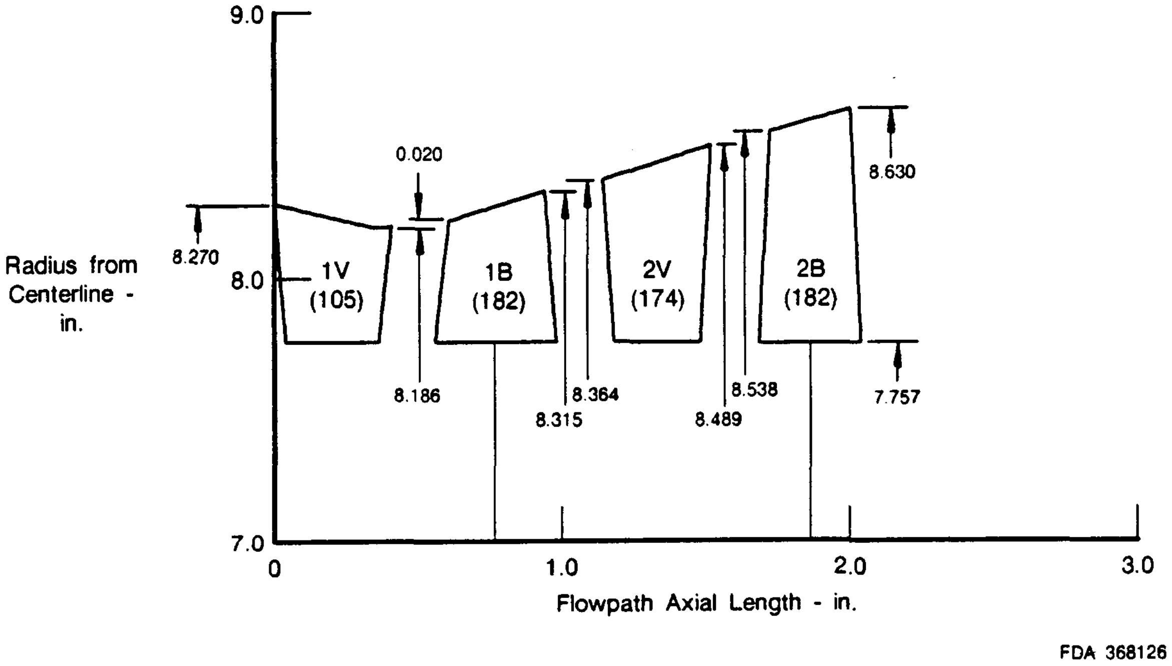


Figure 3.3-3. Space Transportation Booster Engine Gas Generator Fuel Pump Turbine Flowpath Elevation

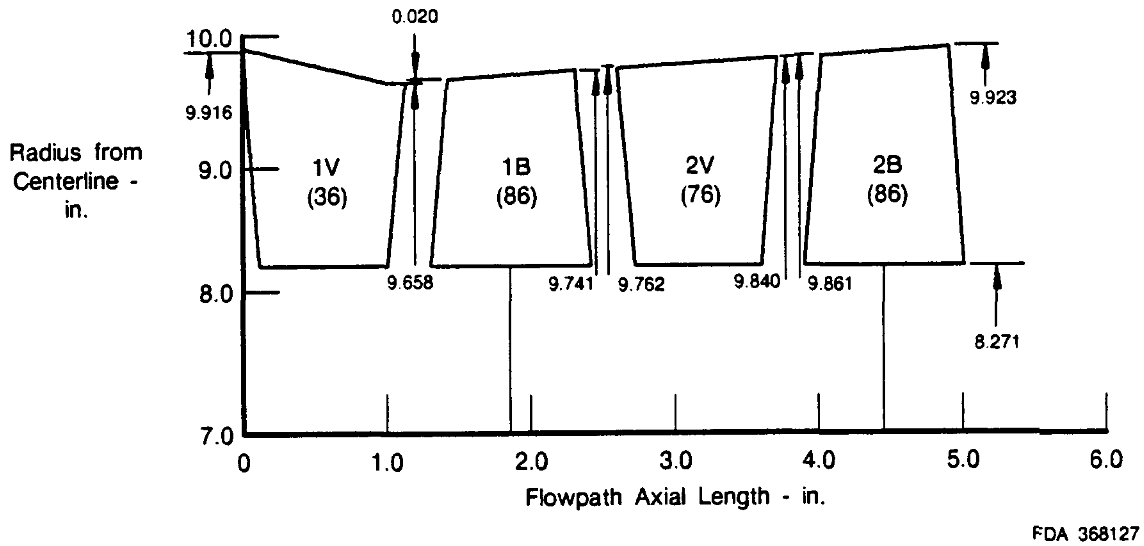


Figure 3.3-4. Space Transportation Booster Engine Gas Generator Oxidizer Pump Turbine Flowpath Evaluation

SECTION 4.0
ENGINE INTEGRATION

4.1 HEATSHIELD CONCEPTS

The purpose of the heatshield is to thermally protect the engine powerhead from engine plume base recirculation and radiant heating during ascent. Vehicles with clustered engines may experience significant convective heating in the base region because of backflow of engine exhaust gases caused by jet interaction. Other sources of base heating include radiant heating from engine exhaust gases and combustion of fuel-rich gases entrained in the base region. Base heating is also affected by the boosters.

There are three general flow regimes which occur during ascent:

- Non-interacting jets, related to low altitude operation (just after lift-off) where the dominant mode of heat transfer is radiation

- Interacting jets, where convection is significant and increases with altitude (since the jets expand due to lower ambient pressure)

- Choked flow, where convective heating reaches a near constant maximum value.

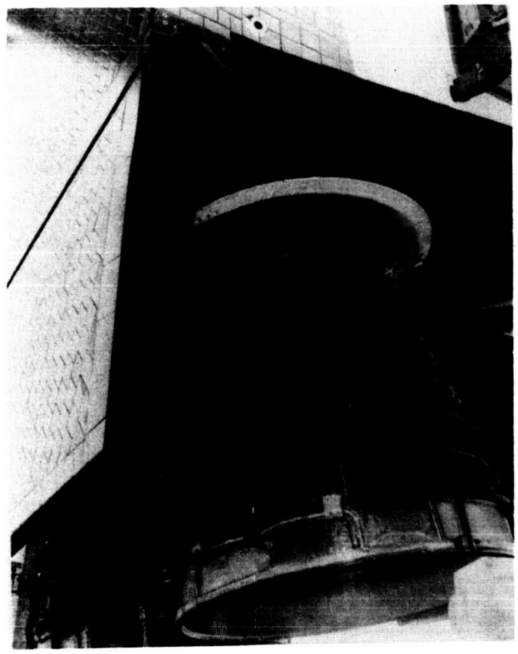
In general, radiation to the vehicle base decreases with increased altitude.

Pratt & Whitney (P&W) is offering conceptual design schemes and ideas to protect against this environment. Although the severity of the environment has not been determined, P&W is acting on the premise that thermal protection in the form of a heatshield is warranted.

The final design will depend on the determination of radiation and convection heat flux levels, view factors on the heatshields which include effects from the boosters and description of the vibratory and acoustic environment. Vibration levels and shock loads induced on the engines should be determined by evaluating the effects of fluid flow, pumping, and propulsion processes as well as vehicle responses during flight. The acoustic environment is known to be most critical at launch. Therefore, launch pad characteristics should be considered along with the number of engines and their geometry.

Beginning with the Space Shuttle Main Engine (SSME) heatshield as a baseline design, significant improvements can be identified. The SSME heatshield (Figure 4.1-1) consists of two rigid, spherical engine mounted shield halves (bolted to a flange on the engine) and two vehicle mounted shield halves (bolted to the aft compartment shield) which act as an eyelid. The vehicle mounted shield uses spring-loaded canisters to provide pressure to seal against the engine mounted shield while still allowing the engine to gimbal. Presently, removal of these shields from the three main engines requires 48 manhours while installation requires 96 manhours. For this reason, eliminating complexity, weight, and bolts as fasteners would significantly affect maintenance turnaround time since any maintenance or inspection of the powerhead requires removal of the heatshield.

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH



FC 111882

Figure 4.1-1. Space Shuttle Main Engine Heatshield Design

Pratt & Whitney's first concept involves a rigid engine mounted shield similar to the SSME (Figure 4.1-2). However, precision C-clamps are proposed rather than bolted flanges to significantly reduce removal and installation time. Another improvement results from the Advanced Launch System (ALS) powerhead being comparably larger. Due to restricted engine spacing and engine gimballing requirements, the engine must be installed external to the aft compartment. This allows for the elimination of a vehicle mounted shield and the problems inherently associated with it. Figure 4.1-2 shows a pressurized seal concept which is mounted flush with the aft compartment shield. Therefore, seal pressure is maintained without the use of spring-loaded canisters. Also, vibration concerns with this concept are diminished.

Along with a more simplified assembly, the elimination of handsewn thermal blankets used on the SSME should be mentioned. Numerous problem reports result from the fragile nature of the fiberglass blankets. It is thought that ripping and tearing occurs due to acoustic loading. Using thermal protection tiles around the base of the pressurized seal mount should provide sufficient protection without the use of thermal blankets.

A second concept (Figure 4.1-3) combines a rigid shield with a flexible shield. The flexible shield concept also uses a C-clamp for quick removal and installation with the added advantage of an ability to fold back on itself (Figure 4.1-4). This allows easy access to the engine powerhead components for inspection, final check-out, leak tests, and maintenance. Weight of the heatshield is significantly reduced making it less difficult to remove.

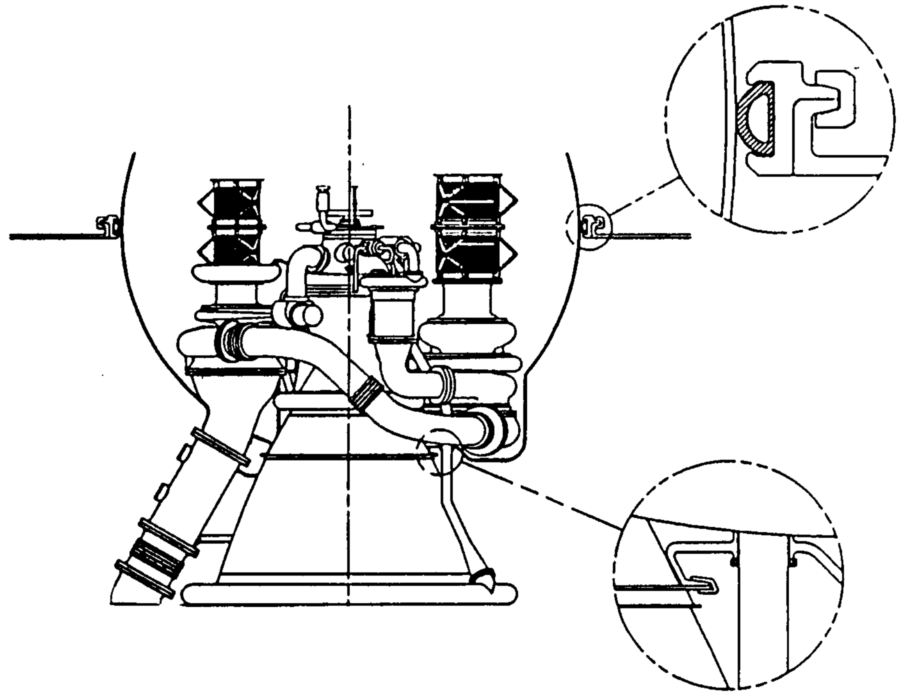


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Figure 4.1-2. Derivative STBE Heatshield Design Using Precision C-Clamps Significantly Reduces Removal and Installation Times

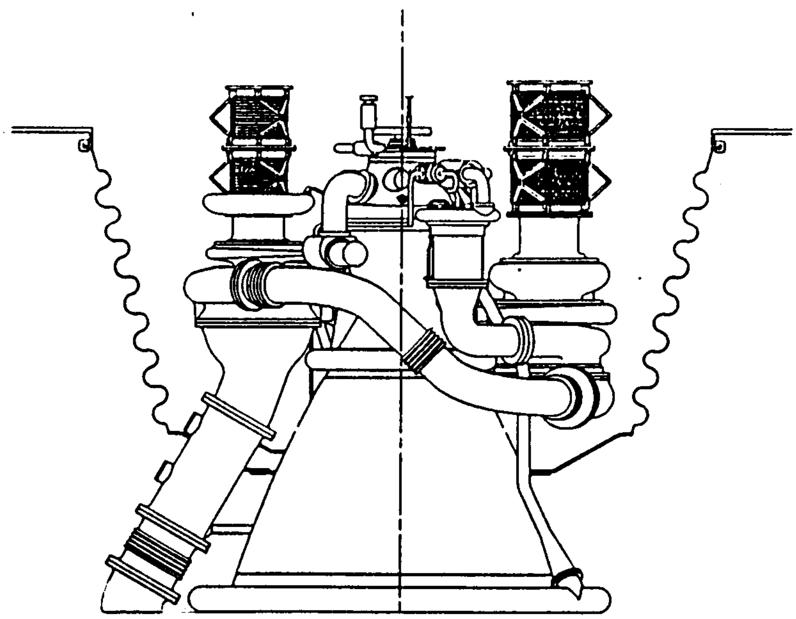

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Figure 4.1-3. Derivative STBE Flexible Heatshield Design Reduces Weight and Allows for Quick Removal and Installation

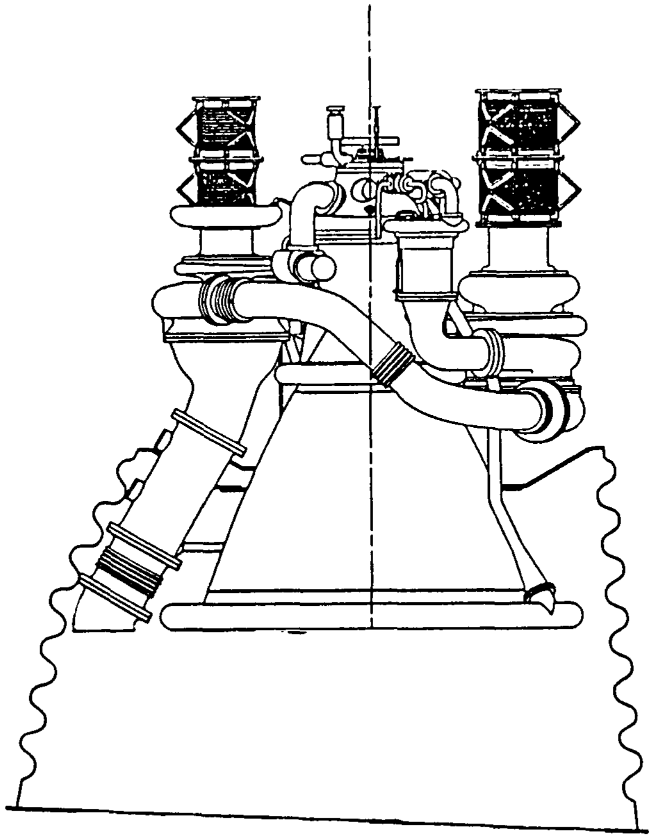

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Figure 4.1-4. Ability of Flexible Heatshield to Fold Back Allows Easy Access to the Engine Powerhead

The SSME orbiter vehicle base heatshield external surface temperature reaches a maximum of approximately 1600°F for a short duration during ascent. Materials considered for the flexible shield must be capable of surviving the vehicle base environment. Pratt & Whitney has investigated several candidate materials. One promising configuration is a thick layer of silicone foam with a reflective backing to protect against radiation. An ablative layer and cross-weaved fabric stiffening layer may be included. To add stiffness, reduce porosity, and provide tear strength, a silicone rubber backing may also be used. The foam is low-density, light-weight, flame retardant, and not subject to corrosion problems. Attachment to metal rings may be achieved by bonding and/or use of snap-locking fasteners. Fabrication costs and recurring costs once a molding and layering process has been created are believed to be significantly less.

4.2 INLET LINE CONFIGURATION

4.2.1 Scissors Inlet Ducts

Preliminary sizing of the bellows required for the STBE propellant inlet scissor ducts was completed in Phase A. Bellows design equations obtained from several bellows manufacturers and NASA documents are used to determine the elongation and buckling limits as a function of the convolution geometry. Based on this preliminary analysis it appears that the scissor type ducts may be used to gimbal angles as high as 9½ degrees when using the current inlet conditions and spacings.

The bellows are sized using equations found in several references for metal bellows design and application, specifically in NASA publications SP-125 and RSS-8507. Sizing of the bellows free length, number of convolutions, convolution height, wall thickness, and number of plies were traded to maximize the elongation and bending capability of the bellows within the materials stress limits yet provide the necessary axial spring rate to prevent buckling due to internal pressure.

An unconstrained bellows elongates under internal pressure due to the geometry of the bellows convolutions. In a rocket inlet application, this elongation is constrained by the pump housing and the fixed vehicle supply ducts. As the internal pressure increases the induced compressive load in the bellows continues to build until a rapid dislocation of the centerline of the bellows occurs, results in permanent plastic deformation or rupture. This buckling failure, commonly called squirm, is completely analogous to Euler column buckling. The scissor ducts chosen as the baseline incorporates three pinned links which creates a node at the duct center, reducing the effective column length in half.

Due to the higher pressures in the LO_2 inlet duct encountered during the mission, the elongation capability of the LO_2 inlet design sets the maximum gimbal angle allowable for the engine. The current engine specification sets a maximum LO_2 inlet pressure during the mission at 285 psi, (versus 125 psi for the fuel inlet) and 350 psi was used as a minimum allowable buckling pressure, giving a 23 percent design margin in buckling. During preliminary design, this margin should not be eroded due to the historically poor correlation between empirical and calculated axial spring rates of bellows and their corresponding buckling pressures. As the design is refined with more sophisticated analysis, perhaps some of this margin can be used to increase gimbal capability if required.

To reduce the number of variables, some assumptions were made. The number of plies was set at three. Multiple plies provide a reduction in stresses and increase axial spring rate due to ply interaction. Frictional interaction also tends to damp fluid induced bellows vibration. The selection of three plies is consistent with the bellows used on the J-2 and F-1 engine feedlines. Some manufacturing difficulty can be experienced when hydroforming bellows with greater than three plies; therefore, they were not studied in keeping with the low-cost emphasis of this program.

In general, increased bellows free length increases the axial elongation capability of the bellows; however, a reduction in buckling pressure logically accompanies the increased length. An upper bound was set for bellows free length for engine packaging reasons. The center or node of the scissor assembly must correspond to the gimbal rotation plane to reduce the excursion of the bellows to pure elongation and bending (i.e. no shear). With the pump and chamber geometries at the time of this study an upper limit on overall scissor assembly length was set at 30 inch (i.e. pump inlet 15 inch below the gimbal rotation plane). This corresponds to approximately two stacked 14 inch bellows plus the associated flanges and scissor links. The bellows free length quickly iterates to this upper bound, leaving only the number of convolutions, convolution height and wall thickness to iterate.

INCO 718 was chosen for the material due to the good formability in bellows hydroforming operations and superior strength. A-286 has very similar properties and allowable strength and is carried as a backup. Both of these alloys are considered standard aerospace bellows materials.

Three stress components were calculated; hoop stress, convolution bulging stress, and bending stress. Hoop stress was limited to 90 percent of 0.2 percent yield strength at the operating temperature. The maximum convolution bulging stress and bending stress are additive and occur at the crown and root centers of the bellows convolutions normal to the hoop stress.

The motion stress (bulging plus bending) was set equal to the stress allowable for INCO 718 corresponding to 1000 cycles fatigue life, allowing an equivalent axial deflection to be calculated. The equivalent axial deflection includes two terms which account for the bellows centerline elongation as the bellows is stretched along an arc during gimballing and the bellows angulation during bending. Using the engine packaging geometry, an angle is computed which yields the maximum equivalent axial deflection (i.e. the maximum allowable gimbal angle) for each bellows iteration. As the scissor assembly is symmetric about the gimbal rotation plane, each half of the scissor duct experiences half of the total duct deflection and can be analyzed independently.

Several bellows provided acceptable buckling pressures and significant elongation capability. These candidates, all 14 inch long, had 60 to 70, 0.55 to 0.65 high convolutions. Fewer or shorter convolutions tended to reduce the elongation capability while more or taller convolution tended to reduce buckling pressure below the 350 psi minimum criteria. Wall thickness was iterated to 0.024 inch (three 0.008 plies). Thinner walls raised the hoop stress to unacceptable levels at the 285 maximum operating pressure, while thicker walls lowers the flexibility of the bellows reducing elongation capability.

The bellows shown in Figure 4.2.1-1 was chosen for continued study to determine allowable engine gimbal angle versus inlet pressure. Using engine centerline to pump centerline spacings ranging from 20 to 34 inch in 2 inch increments, a curve of allowable gimbal angle versus centerline spacing is generated. An earlier performance study was conducted using LO_2 inlet pressures of 30, 35, 40, 47, 52, 60, 80, and 100 psi. A turbopump diameter was obtained by multiplying the baseline pump diameter by a ratio of the pump impellers from these cycle sheets and the baseline design. A corresponding pump inlet spacing could then be calculated, allowing the plot of gimbal angle versus pump inlet pressure, as in Figure 4.2.1-2, to be generated. The small range of inlet pressures did not justify optimizing a bellows for each operating pressure.

A plot of fuel inlet pressure versus gimbal angle was also generated in a similar manner using data available at 18, 20, 22, 24.5, 27, 30, 35, and 45 psi inlet pressure. In this case the bellows optimized for the LO_2 inlet was used, yielding a somewhat conservative curve as high buckling pressure margin obviously exists. However, this will be partially offset by the increased diameter of the fuel inlet. It should be noted that the inlet pressures versus gimbal angle for both LO_2 and fuel must be looked at simultaneously to ensure that both inlet have the same gimbal capability.

The fuel inlet poses a new problem in that a concentric bellows must be included around the primary flow bellows to allow either vacuum jacketing or inert gas charging for insulation of the inlet line. This bellows, even with its larger diameter, does not limit the allowable gimbal angle as it experiences very little internal or external pressure, eliminating squirm problems, allowing the axial spring rate to be reduced so that high elongation can be achieved without stress problems.

As these scissor inlet ducts change volume during gimballing, a pressure pulse is experienced at the pump inlet. This pulse immediately results in a thrust spike. A simple analysis was performed to determine the order of magnitude of this thrust oscillation. This analysis assumed that the duct acted essentially as a piston in the inlet line and that any volume change resulted in a flowrate spike. When using a gimbal rate of 18 degrees/sec, the SSME maximum gimbal rate, and a gimbal angle of 9½ degrees the corresponding flowrate spike is equal to approximately 1/20 of the nominal flowrate on the LO_2 side. Δ thrust/thrust is approximately equal to Δ flowrate/flowrate for the LO_2 side resulting in a potential thrust oscillation on the order of 25,000-pound. This is likely an upper bound as it does not reflect the influence of the corresponding pressure spike on the fuel side or any capacitance the vehicle tanks have in reducing the pressure oscillation.

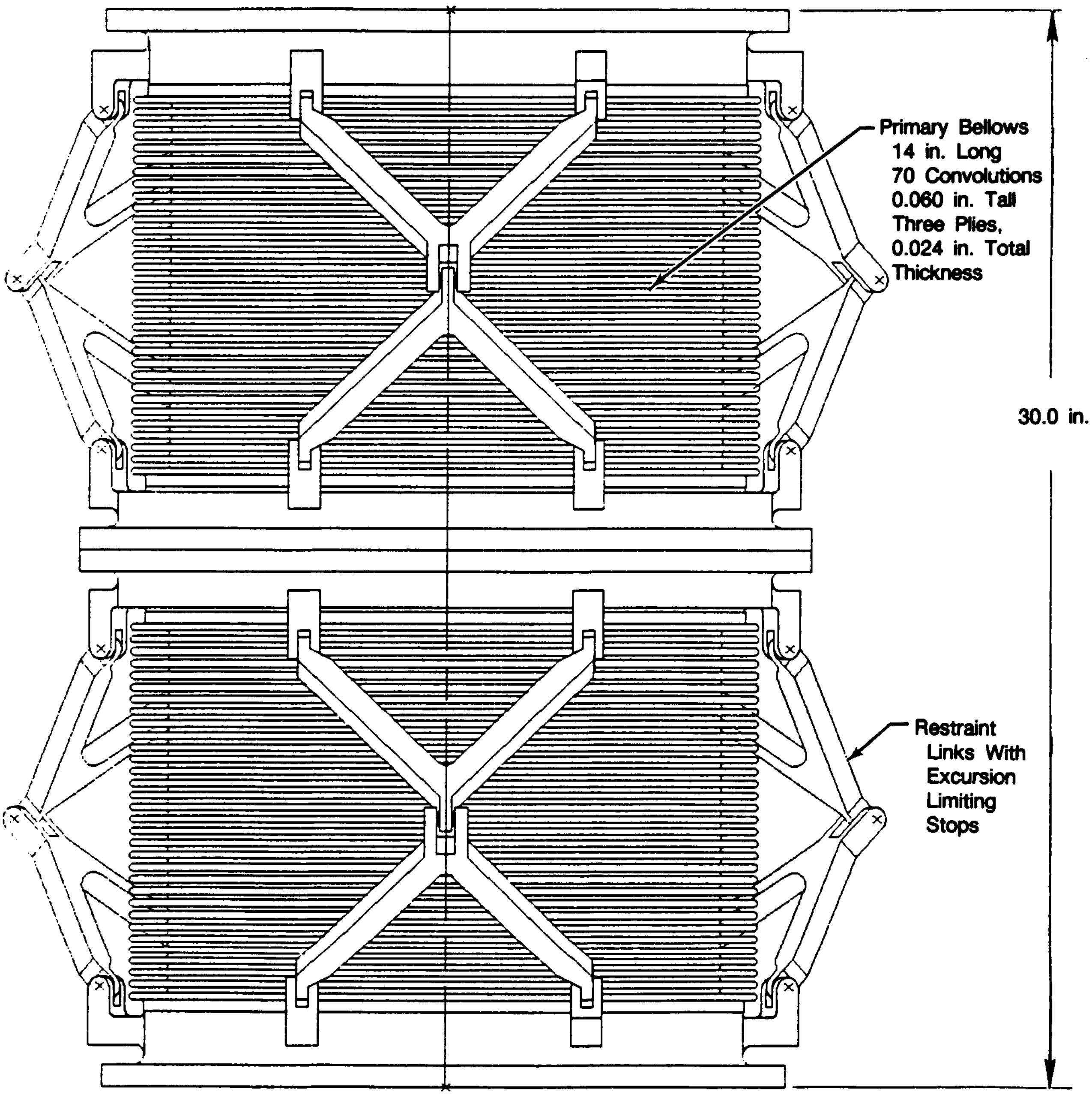

Figure 4.2.1-1. Pratt & Whitney's Scissor's Type Bellows Inlet Duct Meets STBE Requirements

Further work required during Phase B includes updating the design using current thrust chamber and pump geometries and spacings, a first cut at vibration analysis, incorporating into the design a means to isolate the bellows from induced torsion during gimballing, and a manufacturing study.

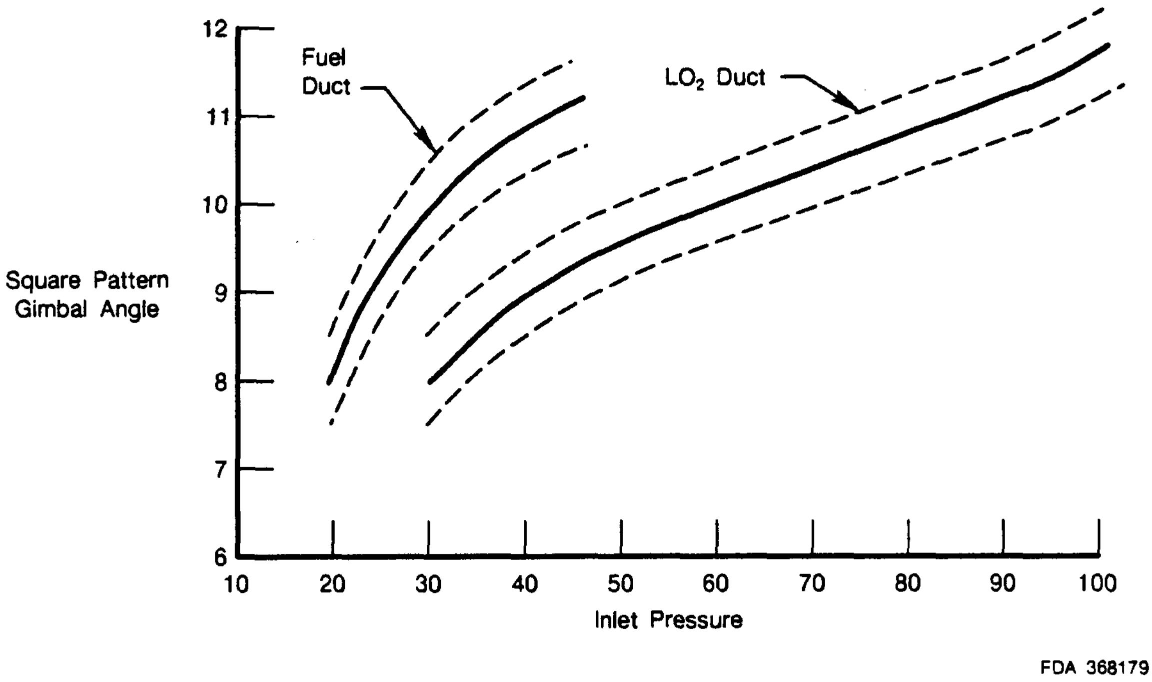

Figure 4.2.1-2. Gimbal Angle Capability as a Function of Inlet Line Pressure

Flow induced vibration of the bellows due to vortex shedding off the bellows convolutions needs to be addressed. Due to the high deflections required, internal flow liners such as that used on tied flex joints are not practical. The use of multi-plies is intended to offer some damping of these vibrations. The J-2 scissor ducts experienced several failures upon attempted restart in space after many successful ground tests. The cause was linked to the fact that during ground tests the bellows surfaces quickly frosted over during chilldown and this ice layer provided sufficient vibrational damping to mask the problem. In the vacuum of space, with the ice evaporated, the bellows quickly failed in high-cycle fatigue (HCF). In addition, the associated pressure drop and flow perturbations caused by the bellows upstream of the pump need to be investigated.

Bellows deflected about two axes, as occurs during engine gimballing, results in a torsional moment being introduced in the bellows. Due to the high gimbal angles required, this torsional moment would cause torsional bellows squirm. On the J-2, this problem was solved by placing a tightly convoluted bellows within a rotary sliding joint at the middle flange of the assembly. This was essentially a very weak torsional spring which isolates all the torsion from the large primary bellows. A design feature to address this problem needs to be incorporated into the design.

4.3 SPECIAL STUDIES

4.3.1 Gas Generator Pressure Optimization

A trade study was conducted to determine the effects on various engine components when the gas generator chamber pressure is varied. The results of the study, as shown in Figure 4.3.1-1 illustrate the effects on component cost, weight, engine performance, and resultant net effect of vehicle life cycle cost (LCC).

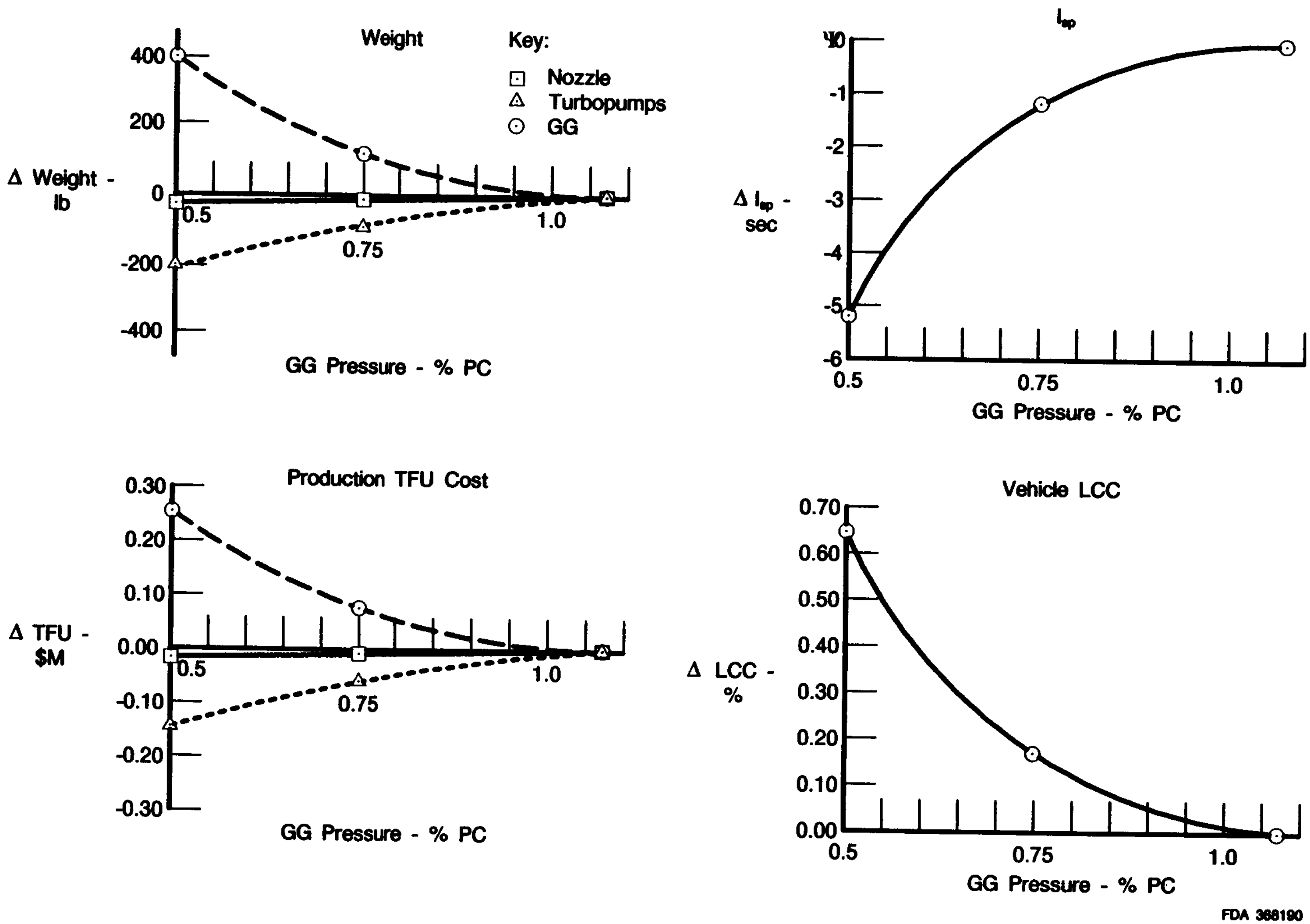

Figure 4.3.1-1. Effect of Gas Generator Chamber Pressure on Engine System

As gas generator chamber pressure is increased from 50 percent to 107 percent of main chamber pressure, the gas generator assembly decreases in overall size requirements for a given set of pump and turbine power requirements. However, turbine diameters must increase slightly to accommodate the higher inlet pressures, and the tubular nozzle costs and weight also increase slightly due to the higher fuel exit pressures (gas generator fuel supply). The summation of these cost and weight analyses result in a lower overall engine weight and theoretical first unit (TFU) cost. The higher turbine discharge pressures result in slightly improved engine performance.

The effects of engine weight, cost, and Isp were all input to vehicle contractor supplied parametric equations. The net result of these trends is that the vehicle LCC decreases as gas generator (GG) pressure increase. The benefits tend to drop off at GG pressures higher than chamber pressure, while the design complexity of the turbines starts to increase more dramatically. Therefore, a gas generator chamber pressure of 100 percent main chamber pressure was selected.

4.4 OCEAN RECOVERY STUDY

4.4.1 Background

A study to determine the feasibility of ocean recovery of a Space Transportation Main Engine (STME) used as a booster or Space Transportation Booster Engine (STBE) was initiated in May 1989 during the fifth extension of activity. The initial goals of the study were as follows:

- Determine the technical feasibility of recovering a fully immersed STME booster or STBE after complete immersion in tropical sea water following launch.

- Define the operations required and their associated costs, facilities required, inventory, schedule, and development plan impact for the ocean recovery scenario.

The following three scenarios were envisioned as potential ocean recovery concepts after discussions with the vehicle contractors.

Scenario 1 — The baseline STME/STBE (derivative of STME) is used on a booster vehicle and returned either to land (dry recovery) or recovered in the ocean but is fully protected by the vehicle and thereby encounters only an external salt spray environment.

Scenario 2 — The baseline STME/STBE is used on a booster vehicle and recovered in the ocean but encounters a severe salt spray/partial immersion or full immersion environment to a depth of 60 feet for up to 8 hours. The engine is not protected by the vehicle nor does it carry self-contained protective devices.

Scenario 3 — The STME/STBE is redesigned to accommodate partial or full immersion in sea water so that some protective devices are carried throughout the mission that will assist in preventing sea water from entering engine turbomachinery components and various line-replaceable units (LRU).

For each of the three scenarios, a complete analysis was performed as summarized in the following list of major tasks:

- Determine the effects of sea water on engine components.

- Define cleaning, inspection, and repair operations, if any, and criteria for reuse (acceptability) on a part by part basis.

- Define recovery operations on shipboard to minimize contamination and corrosion damage resulting from the recovery environment (scenario 1, 2 or 3).

- Define facilities, equipment, vesting, and inventory requirements to bring engines back to full flight ready status.

- Determine impact on development plan for each of the three scenarios.

The following is a list of ground rules and assumptions for the three scenarios:

Ground Rules and Assumptions

Scenarios 1, 2, and 3:

- STME in the booster configuration or STBE

- 10 flights/year; 7 engines/flight recovered

- Booster engine recovery only; no core engines recovered

- Labor estimates are direct hands-on time only and are typical of the 100th mission

- Cost analyses use 90 percent Crawford Learning Curve

- Engine production costs represent cumulative average of 425 engines

- Labor costs represent engine contractor costs only. Vehicle contractors must estimate labor and other costs where noted "vehicle"

- Scenario 1: Engines endure 8 hours of light salt spray on exterior only or unlimited duration of salt atmosphere

- Scenario 2 and 3: Engines endure up to 8 hours of partial or full immersion in 50° to 80°F sea water at 60 foot depth. The engine overhaul and assembly facility located at Stennis Space Center is expanded to accommodate engine teardown, cleaning, and assembly operations required

- Scenario 3: Weight penalties of approximately 200 pound associated with the protective devices are not included in any cost analyses. The vehicle contractors will include this when calculating total LCC differences.

4.4.2 Recovery and Refurbishment Operations

A complete discussion of the STBE maintenance philosophy is included in Section 5.1. The following paragraphs highlight procedures unique to ocean recovery. The recovery scenarios were broken into 6 major steps, listed as follows:

1. Shipboard Operations
2. Transportation
3. Overhaul Operations
4. Assembly
5. Test
6. Installation on Propulsion Module and Flight.

The detailed breakout for each of the three recovery scenarios is listed in Tables 4.4.2-1, 4.4.2-2, and 4.4.2-3. Note that for Scenario 1, no overhaul, assembly, or test operations are required. This results in significantly lower cost for the dry recovery scenario.

Table 4.4.2-1. *Scenario 1: Recovery and Refurbishment Operations*

Shipboard Operations:	*Overhaul and Assembly Operations:*
1. Haul BRM from Ocean.	None Required
2. Remove Heatshield.	
3. Rinse Exterior and Dry Engines.	*Engine Test Operations:*
4. Inspect Engines.	
5. Protect Inlets, Nozzle, and Other Ports.	None Required
Transportation Operations:	*Assembly on Propulsion Module Launch:*
6. Transport BRM from Docking Site to Propulsion Assembly Facility.	10. Transfer to Propulsion Module Assembly Facility.
	11. Install on BRM.
7. Remove Engines from BRM. Prepare for Transfer To Engine Base Maintenance Facility.	12. BRM Assembly to Vehicle.
	13. Transfer to Launch Pad.
8. Transfer Engines to Engine Base Maintenance Facility.	14. Preflight Checkout.
	15. Launch.
9. Conduct Post Recovery Checks.	16. Flight Data Analysis.

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The tasks referenced in the preceding paragraphs take place immediately following recovery and describe all the operations up to launch. The engine/component flow diagram shown in Figure 4.4.2-1 illustrates P&W's concept for dry or ocean recovery engine on-line and off-line maintenance.

Upon launch and propulsion module touchdown in the ocean, a recovery ship locates and retrieves the propulsion module. The vehicle contractors must provide all information regarding locating, retrieval, and positioning of the module on board the recovery ship. Once the module is positioned, the engines may be accessed for the cleaning and drying operations called out in Figures 4.4.2-1 through 4.4.2-3. These operations may take place while the recovery ship returns to port. The recovery ship will return to the docking site located at Kennedy Space Center where the module is unloaded and returned to the Propulsion Module Assembly Facility (PMAF), also located at Kennedy Space Center. At this facility the engines are removed from the module and transferred to the Engine Base Maintenance Facility (EBMF) located nearby. Depending on the scenarios, the engine is inspected and either returned to the PMAF (Scenario 1) or sent on to the Engine Overhaul Facility (EOF) located at Stennis Space Center (Scenarios 2 and 3). The following details the primary flowlines, as shown in Figure 4.4.2-1, for the various scenarios following propulsion module recovery.

Table 4.4.2-2. *Scenario 2: Recovery and Refurbishment Operations*

Shipboard Operations:

1. Haul BRM from Ocean.
2. Disconnect Engine Inlet Lines and Remove Heatshield.
3. Rinse Exterior, Flush Interior and Dry Engines.
4. Inspect Engines.
5. Protect Inlets, Nozzle, and Other Ports.

Transportation Operations:

6. Transport BRM from Docking Site to Propulsion Module Assembly Facility.
7. Remove Engines from BRM. Prepare for Transfer To Engine Base Maintenance Facility.
8. Transfer Engines to Engine Base Maintenance Facility.
9. Conduct Inspections Package for Shipment to Engine Overhaul Facility.
10. Ship to Engine Overhaul Facility.

Overhaul and Assembly Operations:

11. Engine Teardown (100% of All Hardware).
12. Part Inspections (100% of All Hardware).
13. Cleaning (80% of All Hardware).
14. Repairing (5% of All Hardware).
15. Replace New Parts (15% of All Hardware).
16. Module Assembly.
17. Engine Assembly.

Engine Test Operations:

18. Engine Test.
19. Prepare for Shipment to Engine Base Maintenance Facility.
20. Ship to Engine Base Maintenance Facility.

Assembly or Propulsion Module and Launch:

21. Acceptance Inspection at Engine Base Maintenance Facility.
22. Transfer to Propulsion Module Assembly Facility.
23. Install on BRM.
24. BRM Assembly to Vehicle.
25. Transfer to Launch Pad.
26. Preflight Checkout.
27. Launch.
28. Flight Data Analysis.

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Scenario 1 — The engines, after cursory inspections such as turbomachinery torque checks, borescope inspections, accelerometer readings, and review of flight data, will be returned to the PMAF if it is acceptable. After engines are integrated on the PMAF, the propulsion module is sent on to be integrated with the vehicle at the Vehicle Integration Facility (VIF). Once the vehicle is assembled it is transported to the launch pad via the mobile launch platform.

If post flight inspections reveal a particular problem with an engine module, such as a turbopump, the capability exits in this maintenance concept to deliver and install spare modules to the engine at the EBMF. Each module will be test fired before shipment, and each engine can accept at least one module changeout without requiring engine retrim. This additional loop between the off-line and on-line maintenance organizations is shown in Figure 4.4.2-1.

Scenario 2 — After removal from the Propulsion Module, the engines are transferred to the EBMF, wherein each engine is packaged for shipment to the Engine Maintenance Facility (EMF) located at Stennis Space Center. In this scenario, an unprotected engine was fully immersed in sea water and therefore should be completely disassembled, cleaned, and reassembled to the smallest detail level. In the EMF, as shown in Figure 4.4.2-2, new components are received in one area while used components and engines are received in a separate area.

Table 4.4.2-3. Scenario 3: Recovery and Refurbishment Operations

Shipboard Operations:	*Engine Test Operations:*
1. Haul BRM from Ocean.	18. Engine Test.
2. Remove Heatshield.	19. Prepare for Shipment to Engine Base Maintenance Facility.
3. Rinse Exterior and Interior of TCA Dry Engines.	
4. Inspect Engines.	20. Ship to Engine Base Maintenance Facility.
5. Protect Inlets, Nozzle, and Other Ports.	
Transportation Operations:	*Assembly or Propulsion Module Launch:*
6. Transport BRM from Docking Site to Propulsion Module Assembly Facility.	21. Acceptance Inspection at Engine Base Maintenance Facility.
7. Remove Engines from BRM. Prepare for Transfer to Engine Base Maintenance Facility.	22. Transfer to Propulsion Module Assembly Facility.
	23. Install on BRM.
8. Transfer Engines to Engine Base Maintenance Facility.	24. BRM Assembly to Vehicle.
	25. Transfer to Launch Pad.
9. Conduct Inspections Package for Shipment to Engine Overhaul Facility.	26. Preflight Checkout.
	27. Launch
10. Ship to Engine Overhaul Facility.	28. Flight Data Analysis.
Overhaul and Assembly Operations:	
11. Engine Teardown (50% of All Hardware).	
12. Part Inspections (50% of All Hardware).	
13. Cleaning (50% of All Hardware).	
14. Repairing (0% of All Hardware).	
15. Replace New Parts (1% of All Hardware).	
16. Component Disassembly and Assembly.	
17. Engine Assembly.	

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Scenario 3 — After removal from the Propulsion Module, the engines are transferred to the EBMF, wherein each engine is packaged for shipment to the EMF, located at Stennis Space Center. In this scenario, the turbomachinery was protected from sea water, although the components of the thrust chamber assembly were thoroughly exposed. Therefore, the engine need only be disassembled to the module level to allow cleaning of the exposed components. The turbopumps remain intact, resulting in a significant cost savings.

Scenario 2 and 3 — Once each engine is completely refurbished and reassembled, it is transferred to the Engine Test Facility (ETF) for overhaul check-out test. Upon completion of this acceptance test, the engine is packaged for shipment to the EBMF, and then is transferred to the PMAF for assembly on the vehicle and return to launch location.

Examples of the requirements definition for cleaning, inspection, and repair is provided in Figure 4.4.2-3. This Figure illustrates the level of detail that was required to generate these operations requirements. Similarly, the direct maintenance manhours were determined by detailed analysis, as illustrated in the following sample data sheets for Depot Level Assembly/Disassembly Procedures (Figure 4.4.2-4), Organization/Intermediate Level Remove and Replace Procedures (Figure 4.4.2-5), and Ocean Recovery Maintenance Procedures (Figure 4.4.2-6). The net result of these detailed analyses was used to calculate costs, and is summarized in Section 4.4.6.

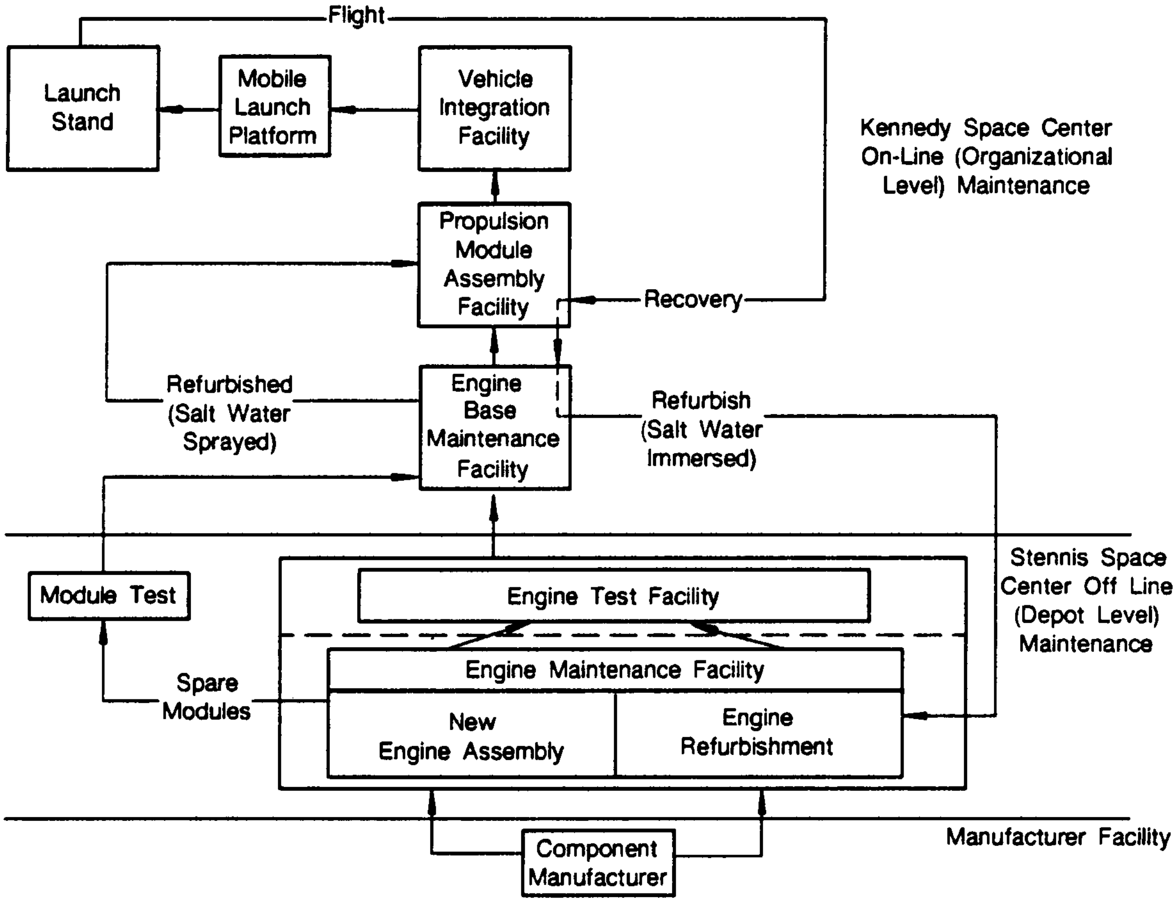

Figure 4.4.2-1. Pratt & Whitney's Concept for Dry or Ocean Recovery On-Line and Off-Line Engine Maintenance

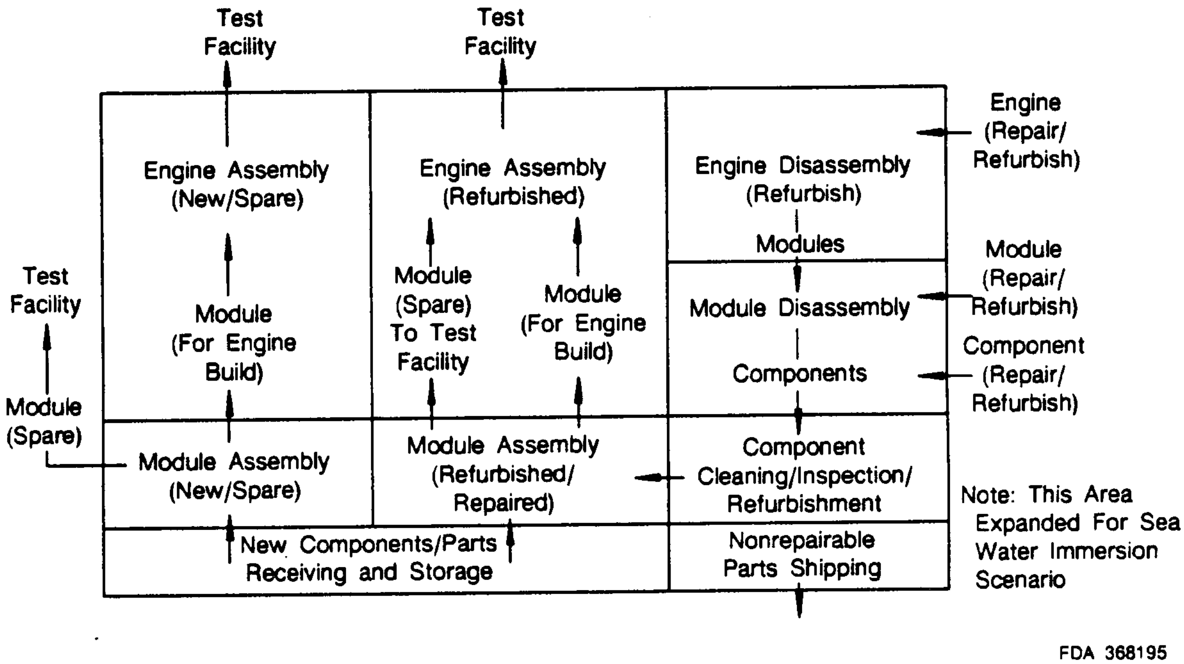

Figure 4.4.2-2. Space Transportation Booster Engine Component Flow at the EMF

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CURRENT ENGINE

1 COMPONENT	2	3 MATERIAL	4 DAMAGED BY LANDING LOADS/ SEA WATER/SAND	5 INSPECTION(S) REQUIRED	6 CLEAN	7 REPAIR	8 REPLACE
GAS GENERATOR							
LOX MANIFOLD	1	CAST INCO 718	X	L,Z,V	LOX		
FUEL MANIFOLD	2	CAST HAYNES 230	X	L,Z,V	LH_2		
GG CHAMBER	3	CAST HAYNES 230	X	V,Z	H_2		
CHAMBER LINER	4	CAST HAYNES 230	X	V,Z	H_2		
FACEPLATE	5	RIGIMESH HAYNES 188	X	V	LOX		
ELEMENT PLATE	6	CAST INCO 718	X	V	LOX		
SEALS	7	INCO-750	√	V,Z	LOX	RECOAT	
BOLTS	8	A286	√	V	LOX	RELUBE	
VENT FITTINGS	9	347 SST	X	V	H_2		
VENT LINES	10	347 SST	X	V,P	H_2		

SAMPLE

Figure 4.4.2-3. Effect of Sea Water and Refurbishment Requirements

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DEPOT LEVEL ASSEMBLY/DISASSEMBLY PROCEDURES (DETAIL PARTS)

HPOTP DISASSEMBLY

Task Procedures	# Tech.	Elapsed Time	Total MMH
6 Use sling and hoist and reposition pump in the universal stand with the turbine inlet housing interfacing with the stand.	2	0.25	0.50
a Rotate stand to position pump in an oxygen inlet housing up position	2	0.16	0.32
7 Remove tierod, item 12.			
a. Disengage tablock from tierod.	2	0.17	0.34
b. Install GSE tool and loosen tierod, item 12, and remove	2	0.25	0.50
c. Remove fairing, item xx.	1	0.08	0.08
d. Disengage tablock, item xx, from inducer spanner nut, item xx	2	0.17	0.34
e Use GSE tool and loosen inducer spanner, item xx, nut and remove	2	0.33	0.66
f Use GSE tool and remove inducer, item 5, from disk and shaft item 7	2	0.16	0.32
g Attach reparable processing tags to removed component	2	0.75	1.50
8 Remove inlet housing, item 1			
a Loosen and remove the 22 bolts at the inlet housing, item 1 to pump housing, item 2, interface flange. Bolts must be loosened in an alternating manner to preclude damage to pump.	2	0.57	1.14
b. Install jackscrew bolts at 90 degree intervals.	2	0.08	0.16
c. Tighten jackscrew bolts in an alternating sequence until inlet housing separates from pump housing.	2	0.17	0.34
d Place inlet housing in GSE holding fixture with duct flange interface attached to fixture.	2	0.25	0.50

SAMPLE

Figure 4.4.2-4. Example of Depot Level Assembly/Disassembly Procedures

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ORGANIZATION/INTERMEDIATE LEVEL REMOVE & REPLACE PROCEDURES (MODULES)

	* MMHs (R)	# Tech. Reqd.	** MMHs (I)
D) Mount Fittings Bolts	1.0	2	1.9
E) Install GSE & transition repairable pump out and serviceable pump in and remove GSE.	6.0	3	11.1
F) Remove, reposition and install misc. externals to facilitate replacement.	4.0	2	7.4
G) Perform in-process inspections of critical steps.		1	15.8
H) Conduct final quality check.		1	1.0
		Total MMHs:	65.4
4) GAS GENERATOR			
A) Discharge Duct Flange and Mount Fittings	1.5	2	2.8
B) Fuel Inlet Flange	1.0	2	1.9
C) Oxidizer Inlet Flange	1.0	2	1.9
D) Igniter	1.0	2	1.9
E) Spin Assist Tube Flange	1.0	2	1.9
F) Fuel Purge Tube Flange	1.0	2	1.9
G) Oxidizer Control Valve Tube Flange	1.0	2	1.9
H) LOX Injector Purge Tube Flange	1.0	2	1.9
I) Oxidizer Bypass Tube Flange	1.0	2	1.9
J) Install GSE & transition repairable pump out and serviceable GG in and remove GSE.	3.0	2	5.6
K) Remove, reposition, and install misc. externals to facilitate replacement.	4.0	2	7.4
L) Perform in-process inspections of critical steps.		1	15.5
M) Conduct final quality check.		1	1.0
		Total MMHs:	64.0
5) FILM COOLED NOZZLE			
A) Flange Bolts	2.9	2	5.4
B) MOP Turbine Discharge Duct Flange	1.8	2	3.3
C) Install GSE & transition repairable nozzle out & serviceable nozzle in and remove GSE.	3.0	3	5.6
D) Adjust nozzle, mating flange as necessary for adjustment of		2	1.3

* MMH (R) = Mean Maintenance Hours (Removal) .35 of Total R and I MMH
** MMH (I) = Mean Maintenance Hours (Installation) .65 of Total R and I MMH

SAMPLE

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Figure 4.4.2-5. Example of Organizational/Intermediate Level Remove and Replace Procedures

OCEAN RECOVERY MAINTENANCE REQUIREMENTS

Maintenance Operations

Page ___ of: ___
Issue Date: ___
Prepared by: ___

Engine / Component	Maintenance Requirement	Scheduled		Unscheduled	MMH	EMS	* Location and Facility
		Routine	Periodic				
	* Install LAI. (MFP)	x (TBD)					2B/E
Controller	* Function and recertification check.	x			1.0		2B/E
	* MCC closure humidity indicator (TBD humidity max) accomplished TBD hours after installation, then at TBD days interval thereafter.						
Sensors/ Transducers	* Continuity/Functional verification check.	x			1.0		2B/E
ELECTRICAL CABLES							
Vehicle to Controller Cable #1	* Disconnect and reconnect cable retention clamps, brackets, fasteners, ties and cable connectors. * Transition reparable cable out and serviceable cable in. * Conduct continuity/functional verification check.				4.7		2B/E
Vehicle to Controller Cable #2	* Disconnect and reconnect cable retention clamps, brackets, fasteners, ties and cable connectors. * Transition reparable cable out and serviceable cable in. * Conduct continuity/functional verification check.				4.7		2B/E
Vehicle to Controller Cable #3	* Disconnect and reconnect cable retention clamps, brackets, fasteners, ties and cable connectors. * Transition reparable cable out and serviceable cable in. * Conduct continuity/functional verification check.				4.7		2B/E
Controller to Solenoid Valves Cable	* Disconnect and reconnect cable retention clamps, brackets, fasteners, ties and cable connectors. * Transition reparable cable out and serviceable cable in. * Conduct continuity/functional verification check.	TBD			14.4		2B/E

* Location: 1. Stennis Space Center, 2. Kennedy Space Center, 3. Vanderberg Launch Site, 4. Marshall Space Flight Center, 5. Michoud, 6. Depot
Facility: A. Vehicle Assembly Building, B. Vehicle Processing Facility, C. Main Launch Pad, D. Test Stand, E. Shop, F. Sea Recovery Vessel

SAMPLE

Figure 4.4.2-6. Example of Ocean Recovery Maintenance Requirements

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4.4.3 Ocean Recovery Design Recommendations

The design studies conducted under the ocean recovery study assisted in establishing some general design recommendations for engines in a salt spray or immersion environment. These recommendations are summarized below:

- Eliminate silver-plated hardware
- All stainless steel (SST) welds to be stress relieved
- Reconsider use of high strength aluminum alloys
- Evaluate protective coating options (anodize/hardcoat) used on aluminum alloys
- Electrically isolate aluminum housing fasteners
- Carbon seals not to contact titanium on aluminum alloys
- Turbomachinery bearings should be designed to handle g-loads.

As a result of these studies, no aluminum turbopump housings, carbon seals, or silver-plated hardware have been included in the baseline STBE design. All other recommendations have been incorporated in P&W's baseline STBE designs.

4.4.4 Ocean Recovery Protective Design Concepts (Scenario 3)

The result of Scenario 2, an unprotected engine fully immersed in sea water, showed a large increase in costs (when complete teardown and assembly is required) when compared to the baseline (salt spray or dry recovery — Scenario 1) concept. In order to reduce these refurbishment requirements, several design concepts were generated.

As shown on the cycle schematic in Figure 4.4.4-1, the turbopumps will be protected from sea water intrusion by a hot gas valve located at the oxidizer turbopump exhaust. Although the engine schematic shown depicts an STME, this schematic is also applicable to the STBE. In addition, the turbopumps must be internally pressurized with helium to maintain a positive pressure difference across the housing seals. This pressurant will prevent water from entering the turbopump through the housing seals. Two additional solenoid valves will also be required at the interpropellant seal (IPS) vent locations to seal off these ports. In order to maintain these three normally open valves in a closed position a power source must be supplied by the vehicle throughout booster separation, landing, and recovery. Power may be disconnected upon completion of shipboard cleaning and drying operations.

Other protective concepts for the Scenario 3 engine include:

- External insulation (KEVLAR® foam) is waterproof and is not damaged by sea water and therefore does not need replacement
- LRUs are waterproof and externals are corrosion resistant. Some design modifications are warranted for these components to provide protection against immersion in sea water, such as

  - designing valves, controls, wiring harnesses, sensors, connectors, etc. to withstand higher external water pressure

  - seal modifications to noncorrosive materials

- Nozzle assemblies use stiffening bands rather than filament wound composite wrap. The carbon based composite wrap could set up a potential galvanic reaction with the stainless steel nozzle materials.

4.4.5 Schedule Impact

4.4.5.1 Development Plan Modifications for Scenarios 2 and 3

The STBE Development Plan can be modified for an ocean recovery design as shown in Figure 4.4.5.1-1. The overall time frame is unchanged, however, 24 additional engine tests will be required on an ocean recovered engine and additional Design Verification System (DVS) level tests will required. The plan is summarized as follows:

- Full-Scale Development (FSD) begins fourth quarter 1991

- DVS level testing begins first quarter 1993 for identified materials, parts, and components as required. Although DVS testing is a routine part of a full-scale engine development program, additional tests will be required to verify components and processes for use following exposure to sea water

- Component testing begins fourth quarter 1993

- Engine testing begins second quarter 1994

- Engine level validation testing begins first quarter 1995. The concept developed for validation of an ocean recovered engine is as follows:

  - Conduct initial test of engine
  - Dunk the engine in sea water
  - Refurbish engine
  - Perform six firings
  - Repeat above four times for a total of 24 engine firings
  - Evaluate results

- First flight second quarter of 1998.

4.4.5.2 Schedule Impact for Refurbishment Operations (Scenarios 2 and 3)

The timeline required to complete the refurbishment cycle of a Scenario 2 engine is estimated at 90 days from receipt of an engine at the EBMF. This estimate is based upon five days transportation time between Kennedy Space Center and the Stennis Space Center EMF and 80 days for teardown, refurbishment, assembly, and test. Estimates of the teardown, refurbishment, and assembly were based upon P&W's experience with the jet engine Air Force overhaul facilities. Engine test cost and schedule estimates were provided by NASA.

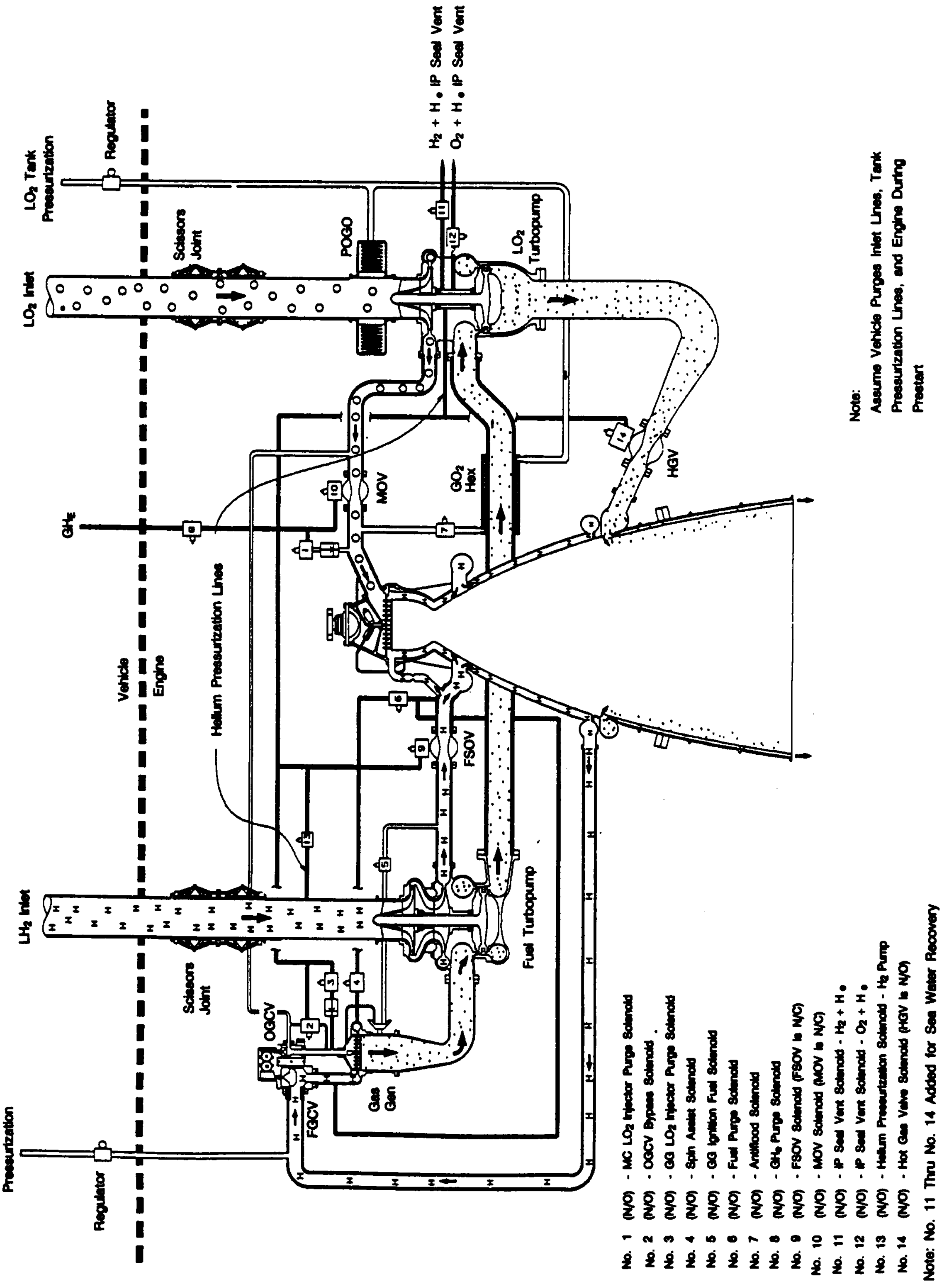

Figure 4.4.4-1. Space Transportation Booster Engine Cycle Schematic, Additional Design Features for Sea Water Immersion

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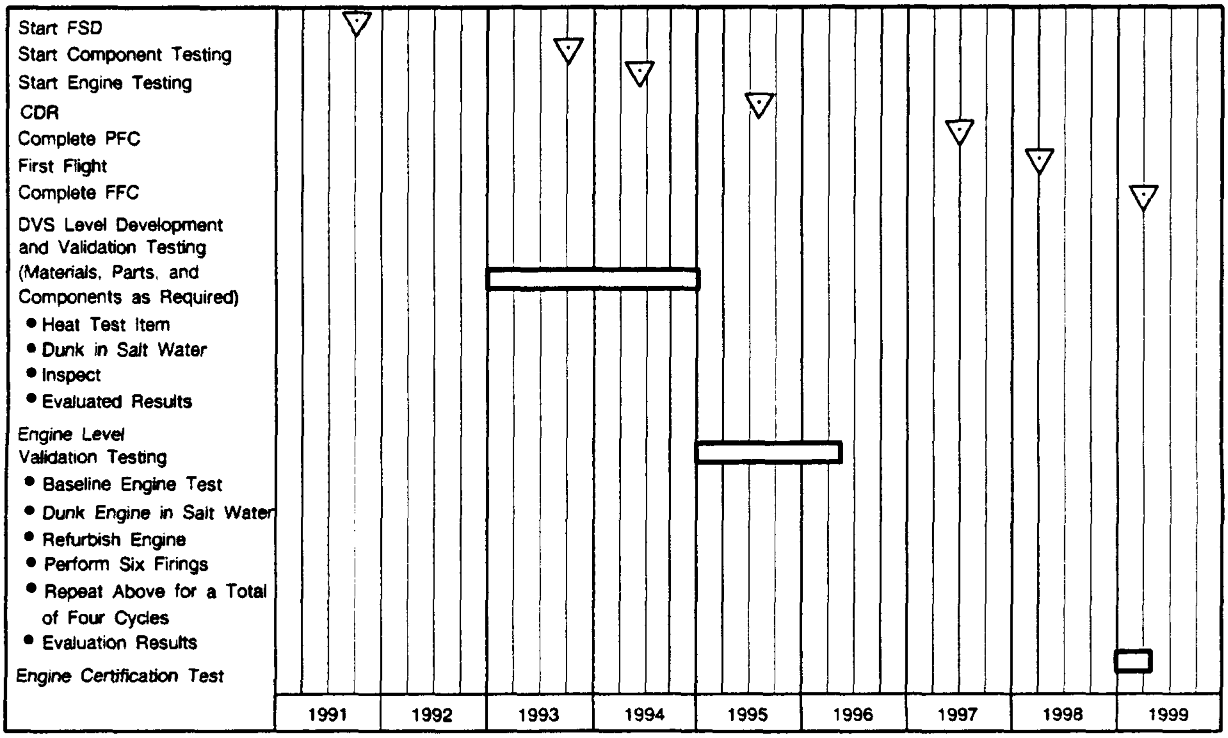

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Figure 4.4.5.1-1. Salt Water Engine Reuse Development Plan

The Scenario 3 engine schedule is significantly reduced due to the reduced refurbishment requirements. The total timeline required to complete the refurbishment cycle is estimated at 30 days. Transportation estimates are unchanged, but refurbishment estimates are significantly reduced because the turbomachinery is not impacted in this scenario.

4.4.6 Cost Summary

A cost summary for the three scenarios is provided in Table 4.4.6-1. The recurring costs are tabulated in thousands of dollars (1987) and the percentages quoted in row 3f reflects the recurring cost as a percentage of engine production costs (cumulative average cost of 425 engines). The non-recurring and Design, Development, Testing & Evaluation (DDT&E) costs are quoted in millions of dollars (1987).

The Scenario 2 option appears tq be the highest cost option for both recurring and non-recurring costs. The recurring costs are primarily labor costs, parts costs, and acceptance test costs, while the large components of non-recurring costs are the refurbishment facility and spare engine requirements. Design, Development, Testing & Evaluation costs are increased due to the 24 additional engine tests and DVS level testing requirement.

The Scenario 3 option significantly reduces both recurring and non-recurring costs. New parts requirements are no longer a significant cost item because the engine is redesigned for corrosion resistance and thus sustains little damage except minor seals, fittings, and other miscellaneous hardware. Labor and engine test costs are the large drivers; however, eliminating the requirement for engine testing after the first 2 or 3 years will bring the recurring cost percentage to approximately 6 percent of new production engine cost. Although this represents significant risk, it should be considered in future ocean recovery studies. Non-recurring costs for Scenario 3 are also reduced when compared to the Scenario 2 option due primarily to the reduced facility size and the decreased spares requirements because of shorter engine turnaround times. Design, Development, Testing & Evaluation costs increase somewhat over Scenarios 1 and 2 because of the higher initial part costs (waterproofing LRUs, additional valves) as well as DVS level and engine testing.

In summary, dry recovery or light salt spray (Scenario 1) offers the lowest cost, lowest risk method for reusing booster engines. If ocean recovery is necessary, all efforts should be made to develop concepts to keep the engines dry (vehicle supplied protection) as well as minimize damage to engine (engine supplied protection) if exposure to sea water does occur.

4.4.7 Risks and Uncertainties

The following list summarizes the risks and uncertainties apparent when evaluating the ocean recovery scenarios. These items warrant consideration in future ocean recovery studies:

Redesigned Engine System: (Scenario 3)

- Downstream side of valve internals exposed to sea water (hot gas valve, main oxidizer valve, fuel shutoff valve, fuel gas generator control valve, and oxidizer gas generator control valve)

    — Possibility of corrosion damage
    — Ability to clean intricate parts
    — Potential to irrepairably damage valve hardware upon opening

Current Engine System: (Scenario 2)

- Possible corrosion damage of copper main chamber due to galvanic reaction with structural jacket or fasteners

- Possible contamination or corrosion damage to engine gimbal

- Eliminating requirement for engine testing saves $400,000 recurring cost but represents risk and uncertainty

- Damage due to landing loads.

Table 4.4.6-1. *Cost and Schedule — All Three Scenarios*

	Scenario 1	*Scenario 2*	*Scenario 3*	*Δ(3-1)*
	Current Design		*Redesign*	
	Salt Spray	*Immersion*	*Immersion*	Δ
Engine Turnaround Time	1 Month	3 Months	1 Month	0 Month
Refurbishment Labor Hours/Engine	143	2154	1359	+1216
Recurring Costs (K87$/Engine)				
Labor	20	281	177	+157
Parts	—	649	50	+50
Acceptance Test	—	400	400	+400
Shipping	—	22	22	+22
Chemicals	5	20	20	+15
Total, K87$/Engine	25 (0.5%)	1372 (32%)	669 (15.0%)	+644 (15%)
Nonrecurring Costs (M87$)				
Refurbishment Facility	18.0	36.0	22.0	+4.0
Tooling	7.0	13.6	11.0	+4.0
Shipping Container	0.2	0.9	0.9	+0.7
Spare Engines	30	121	60	+30
Total, M87$	55.2	171.5	93.9	+38.7
DDT&E Costs	1222	1259	1266	+44

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SECTION 5.0
MAINTENANCE PLAN

5.1 MAINTENANCE PLAN

Although maintainability has always been considered in the design process, it has not always been given equal status with other design considerations by contractors and contracting agencies. Thus, optimum maintainability for air/space vehicles has not been achieved on previous programs. This has resulted in excessive task times for maintenance actions and extended system downtime culminating in increased operation and support costs.

On the Space Transportation Booster Engine (STBE) program, safety, reliability, and maintainability will receive priority over cost, weight, schedule, and performance. This will result in a design that minimizes operating and support costs through reduced recurring maintenance requirements.

Some proven maintenance concepts learned from over 30 years experience on gas turbine engines can be applied to the STBE. Gas turbine engine maintenance is normally conducted at three levels of maintenance: the organizational level ("O" level); the intermediate level ("I" level), and the depot level ("D" level) — see Table 5.1-1. The "O" and "I" levels are accomplished by the user/maintenance at the operating location. "D" level is off-site at a centralized overhaul/repair facility.

Table 5.1-1. *Maintenance Concepts*

Gas Turbine Engine

- Organizational
- Intermediate
- Depot

Space Transport Engine

- Online, Organization Level
- Offline, Depot Level

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The Advanced Launch System (ALS) Integrated Logistic Support Plan (ILSP) specifies a two-level maintenance concept for the STBE, on-equipment/off-equipment and depot level. Any on-site intermediate-level repair required will be implemented on an exception basis as indicated by repair level analysis in later phases. For simplicity we refer to on-equipment/off-equipment maintenance performed at the operating location as on-line (organizational level) maitenance, and off-line for depot level maintenance performed off-site.

Pratt & Whitney (P&W), in consideration of the ALS ILSP, has developed a preliminary maintenance concept for the STBE, as shown in Figure 5.1-1. The concept applies some of the gas turbine engine maintenance concepts and lessons learned that have proven effective in reducing operations and support costs. On-line engine maintenance activities at Kennedy Space Center begin at the Engine Base Maintenance Facility (EBMF) where serviceable engines are received from Stennis Space Center. Upon arrival, the engine is unpacked, inspected for damage during shipment, and prepared for installation. The engine is transported to the Propulsion Module Assembly Facility (PMAF) where it is mated to the Booster Recovery Module (BRM) or core module, and interface operational checks are completed. The BRM and core module are transported to the Vehicle Integration Facility where they are mated with other vehicle segments and the payload module. The assembled vehicle and payload is then transported on the mobile launch platform to the launch stand where launch preparation is completed. Following the launch, the BRM is recovered from the sea water splash-down site and brought onboard the recovery vessel. Maintenance tasks to minimize adverse effects of salt water dunking/immersion to the engine are accomplished in addition to a general inspection that provides early identification of damage repairs that may be required once the engines are returned to the appropriate engine maintenance location. If a dry recovery occurs, no maintenance tasks are required until the engines are delivered to the EBMF.

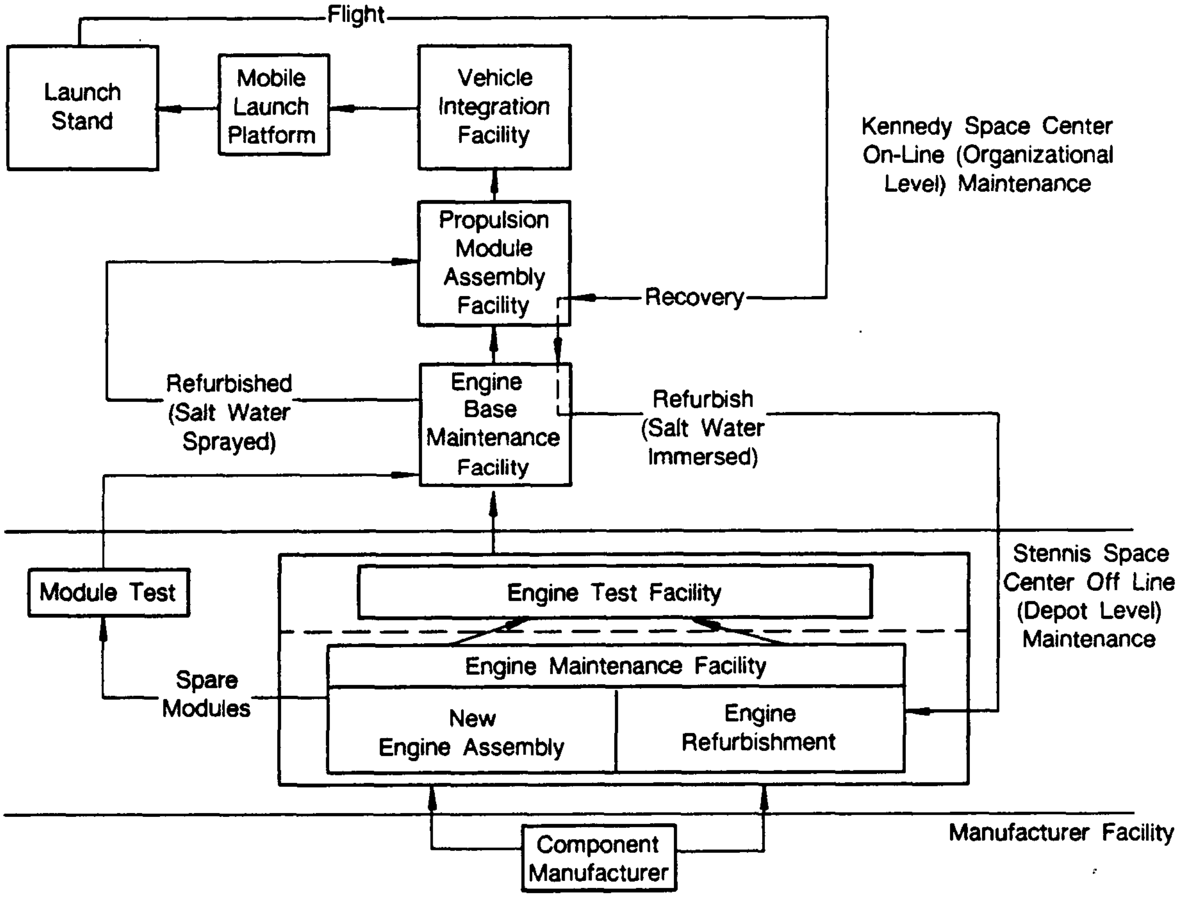

Figure 5.1-1. Space Transportation Booster Engine Maintenance Concept

From the recovery vessel the BRM is returned to the PMAF where the engines are removed and transported to the EBMF. At this phase in the program there are two possible scenarios by which engine repair/refurbishment will take place in order to ready the engines for their next flight. Which scenario takes place depends upon whether the engines are (salt water) sprayed or immersed during the sea water recovery and what protection from the sea water ingestion was afforded to the engine.

If the engine is subjected only to light salt water spray or salt atmosphere, it is anticipated that all engine refurbishment activities could be conducted at the operating location in the EBMF. A preliminary analysis was conducted by maintainability and design engineering to identify anticipated engine maintenance requirements. Maintenance requirements include some cleaning tasks, inspections, and operational checks. Upon completion of the refurbishment activities the engine would be returned to the BRM for installation. An engine readiness firing would not be required.

Should the engine be immersed in sea water, as opposed to being sprayed, it is anticipated that the engines would have to be returned to Stennis Space Center for a more thorough refurbishment activity. The EBMF at Kennedy Space Center would prepare the engine for shipment to Stennis. The Engine Maintenance Facility at Stennis would receive the engine and accomplish the refurbishment activities. A preliminary analysis of the current design indicates that maintenance actions would include disassembly of the engine into its individual component/modules and refurbishment of each component/module would be accomplished. Refurbishment activities would include cleaning, inspection, and replacement/repair of damaged parts. The module would be reassembled and the engine transported to the Engine Test Facility for a certification firing. Upon satisfactory engine testing the engine would be returned to Kennedy Space Center.

Off-Line (Depot Level) maintenance will be conducted at the Engine Maintenance Facility located at the Stennis Space Test Center. As shown in Figure 5.1-1, the facility will provide maintenance capability for new engine assembly and engine refurbishment activities. New components/parts as received from the manufacturer and used to support both new engine/module assembly and engine/module refurbishment maintenance requirements. Engine/modules are tested and forwarded to the operating location to support mission requirements.

The depot level maintenance facility at Stennis will provide maintenance capability for new engine/module assembly and engine/module refurbishment as required. As shown in Figure 5.1-2, new components/parts received from the manufacturers flow into either the new engine/module assembly line or the engine/module refurbishment assembly line. Newly assembled engine/modules are sent to the engine/module test stand for certification firing. The engine/module is prepared for shipment and transported to the operating location.

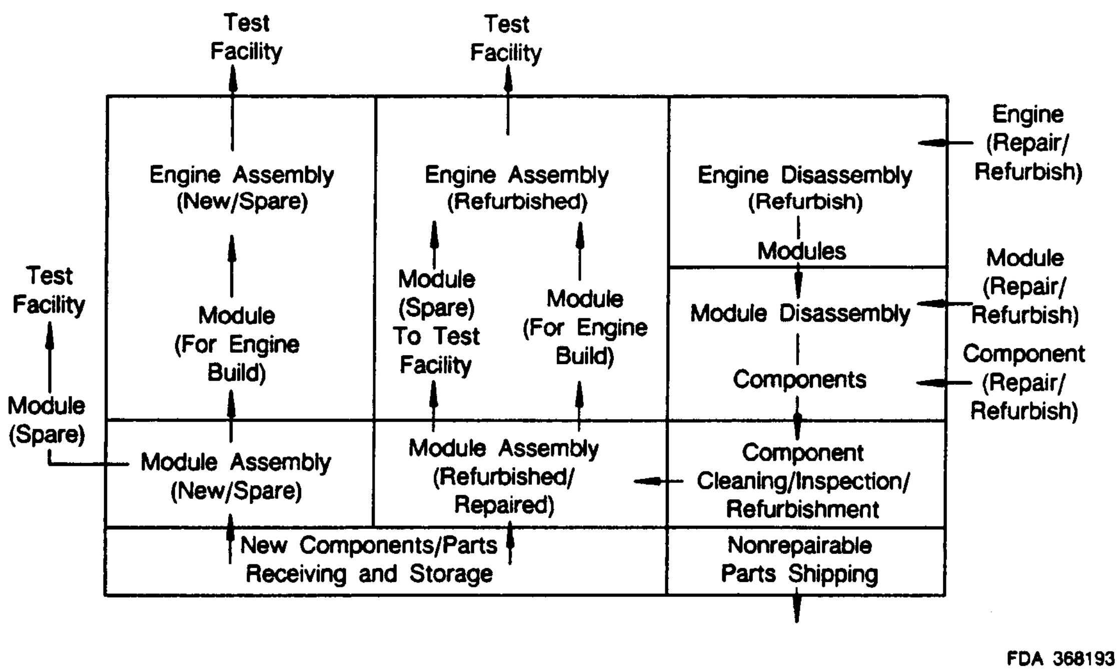

Figure 5.1-2. Space Transportation Booster Engine Off-Line Maintenance Concept

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Engines/Modules/Components that require repair or refurbishment will be received from the operating location and enter the depot level maintenance facility refurbishment disassembly line at the appropriate level. Maintenance actions (inspect, clean, repair, replace) will be performed as required. The engine/module/component will be assembled and sent to appropriate test facility for certification firing/test. Refurbishment/repaired engines/modules/components will be prepared for shipment and returned to the operating location to support mission requirements.

A preliminary estimation of direct hands-on maintenance manhours for module/component assembly/disassembly for depot level maintenance has been completed. This analysis allows maintainability and design engineers to focus on those modules/components that are labor intense and incorporate design features that enhance assembly/disassembly. These maintainability enhancement features are in the memorandum of design requirements distributed to design engineers. Maintainability engineering design reviews are conducted on an on-going basis to evaluate the design characteristics for ease of maintenance. The maintenance manhours also provide necessary data for development of timeliness and early manpower and cost estimates for operations.

Pratt & Whitney has been conducting fact finding trips to Stennis and Kennedy Space Centers to identify supportability problems associated with the current Space Shuttle Main Engine (SSME) and solicit recommendations for improving upon future designs. Pratt & Whitney has gained valuable lessons learned from knowledgeable individuals involved in ground/flight operations at Stennis and Kennedy Space Centers. Available papers/reports/studies, such as the recent Boeing Shuttle Turnaround Efficiencies/Technologies Study, have been thoroughly reviewed. Pratt & Whitney has also considered lessons learned from the RL-10 rocket engine program and other non-P&W rocket engine programs. From this analysis, P&W's Maintainability Design Requirements for the STBE have been established and distributed to design engineers in a memorandum. A section of this memorandum is given in Figure 5.1-3. Support group design reviews are conducted on an on-going basis to assess whether design goals for enhanced maintainability are being realized.

In order to minimize recurring maintenance tasks on the STBE, P&W has identified current requirements for the SSME listed in the Operations and Maintenance Requirements Specification Document (OMRSD), see Figure 5.1-4. This enables P&W to establish design goals that will simplify/eliminate similar requirements relative to the STBE, thus benefitting from the lessons learned on the SSME.

As mentioned previously, an analysis was performed by Maintainability Engineering, with support from other P&W engineering departments, to identify anticipated STBE maintenance requirements for turnaround activities. As Figure 5.1-5 shows, the analysis included most likely location/facility where the maintenance event would occur, the maintenance manhours required to perform each task, and classification of maintenance (routine, periodic, etc.). This also helps to identify those maintenance actions that may be precluded through the STBE design effort.

A preliminary estimation of direct hands-on maintenance manhours required for component removal and installation has been completed. Figure 5.1-6 gives a sample for some STBE components. This analysis allows maintainability and design engineers to focus on those components identified as labor intense and incorporate design features that enhance maintainability and reduce task times. The analysis also provides necessary data for development of timeliness and early manpower cost estimates for operations.

STE Maintainability Design Criteria

Qualitative Criteria

Assembly/Dissassembly:

- Design modules/components with anticipated Line-Removal (engine installed) capability.
- Provide adequate accessibility for technician/tools/equipments to enhance module/LRU/component inspection replacement, and checkout with the engine installed or uninstalled.
- LRUs shall be replaceable without engine removal/rollback and not require removal of other LRUs/components, tubing, harnesses, etc., for removal accessibility.
- LRUs shall be interchangeable from engine to engine. (ICD)
- Configure engine so that modules/LRUs/components with the anticipated highest removal rates or critical maintenance requirements are in most accessible positions. (Review reliability data attached).
- Design LRUs and shop replaceable units (SRUs) for replacement of integral parts with minimum disassembly and support equipment (SE).
- Configure components only one deep to en[illegible]ponent need not be removed to access another.
- Strive for modular design conc[illegible] complexity (mounting, alignment, rigging, operational checks) t[illegible]y/disassembly procedures.
- Consider interface of [illegible] to anticipate impact on engine maintainability features after engine [illegible]
- Provide electromechani[illegible]umatic operated valve actuators. Hydraulic systems are maintenance intensive w[illegible] detrimental impact on system downtime. (KSC)
- Provide for minimum purge requirements that use on-board systems when installed, in vehicle or test facility resources during engine test. There will be no requirement for ground service equipment after propellants are loaded in engine. (STME, ICD)
- No loose hardware (i.e., bolts/nuts/washers/gaskets) for component replacement. Use captured hardware, where practical.
- Provide fool-proof installation/assembly procedures and avoid seal, and/or gasket uni-directional installation requirements.
- Provide alignment guides/marks for components indexing to minimize damage and expedite installation.
- Component positioning/routing should result in minimum impact on adjacent areas during changeout/maintenance.
- Do not require use of sealants at the organizational maintenance level (engine installed) which require extended cure time (exceeding 1 hour) after application.
- Provide external component adjustment/calibration capability to prevent need for disassembly (applicable to organizational, intermediate, and depot maintenance levels).
- Minimize need for component heating and cooling requirements during assembly.

SAMPLE

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Figure 5.1-3. Sample STBE Maintainability Design Criteria

System/Task	Requirement	Current Method (SSME)	Intervals
Orbiter Main Propulsion System (MPS)/Fluid System Rework (remove and replace a fluid system component or assembly)	• Pressure leak test of all affected joints (use helium unless otherwise specified). • Radiographic inspections of all reworked brazed and welded joints. • Functional test to specified requirements.	• Soap solution leak checks (exposed joints). • Audible and mass spectrometer probe or enclosure technique (foam insulated joints, flexlines, bellows, or gimbal joints). • Flowmeter (joints with leak detection ports).	
Pneumatically Operated Valves and Valve Response Time	• Bi-Stable Valve is time from power off or valve command indication to completion of trav[illegible] • Mono-Stable [illegible] time from sw[illegible] switch and po[illegible] on/off indicatio[illegible] to completion of travel.	• App[illegible] sure to [illegible] to opposite • Manual commands using control panel switches. • Remote control using computer commands.	
V41AK0.010 - Mounted Sensor Functional Test	Perform after LRU replacement functional verification of discretes.	Controller, Monitoring Sensor??	
V41AL0.010 - SSME Gimbal Electronic Bonding Test	Verify D.C. resistance orbiter structure to aft position of SSME (after bond disturb. engine installation)	Multimeter (Ohmeter)	After bond disturb, after engine installation.
V41AL0.020 - SSME Elect. I/F Panel Bonding Test	Verify D.C. resistance orbiter structure to electrical	Multimeter (Ohmeter)	After bond disturb, after engine installation.

Figure 5.1-4. Sample STBE Requirement Summary

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Maintenance Operations

Page ___ of: ___
Issue Date: ___
Prepared by: ___

Engine / Component	Maintenance Requirement	Scheduled: Routine	Scheduled: Periodic	Unscheduled	MMH	EMS	* Location and Facility
	* Gimbal clearance check (Perform after STME installation)						
Main Combustion Chamber	* Wipe upper chamber liner with cleaner.	x			0.5		2B/E
	* Visual inspect chamber throat and wall for erosion, burning, distress.	x			1.5		2B/E
	* Visual inspect transpiration cooling liner, main injector face plate for erosion, burning, distress.	x			2.5		2B/E
	* Combustion chamber and regenerative cooled nozzle flow and pressure leak check. (part of overall engine leak check)	x			0.5		2B/E
	* MCC closure humidity indicator (TBD humidity max) accomplished TBD hours after installation, then at TBD d- interval thereafter.						
GG	* Borescope inspect face plates for erosion, burnin, tress plate for erosion, burning, distress.				2.5		2B/E
Main Fuel and Oxidizer Pumps	* Borescope inspect bearings and turbine areas. (2 MMHs per pump)	x			4.0		2B/E
	* Rotation torque and axial travel checks. (2 MMHs per pump)	x			4.0		2B/E
	* Lift off seal pressure and leak check. (MFP)	x			1.0		2B/E
	* Interpropellant seal integrity check. (MOP)	x			1.0		2B/E

SAMPLE

* Location: 1. Stennis Space Center, 2. Kennedy Space Center, 3. Vanderberg Launch Site, 4. Marshall Space Flight Center, 5. Michoud, 6. Depot Facility: A. Vehicle Assembly Building, B. Vehicle Processing Facility, C. Main Launch Pad, D. Test Stand, E. Shop, F. Sea Recovery Vessel

Figure 5.1-5. Sample STBE Maintenance Operations

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	* MMHs (R)	# Tech. Reqd.	** MMHs (I)
1) GOX HEX			
A) LOX Inlet and GOX Outlet Flanges	1.0	2	1.3
B) LOX Pump Discharge Duct Flanges	3.5	2	4.4
C) Compress & release flex joint.	1.0	2	1.3
D) Transition repairable HEX out and serviceable HEX in.	1.0	2	1.3
E) Remove, reposition, and reinstall misc. externals to facilitate replacement.	2.0	2	2.5
F) Perform in-process inspections of critical steps.		1	4.9
G) Conduct final quality check.		1	0.5
		Total MMHs:	24.7
2) LOX TURBOPUMP			
A) GG Inlet Duct Flange	1.5	2	1.9
B) LOX Inlet Duct	2.0	2	2.5
C) GG Outlet Duct Flange	1.75	2	2.2
D) LOX Outlet Duct Flange	[illegible]	2	1.3
E) Interpropellant Vent Flange		2	0.6
F) Mount Fitting Bolts		2	1.3
G) Install GSE & transition repairable pump out and servic[illegible] pump in and remove GSE.		3	7.5
H) Remove, reposition & reinstall misc. externals [illegible] replacement.	4.0	2	5.0
I) Perform in-process inspections of critical steps.		1	11.2
J) Conduct final quality check.		1	1.0
		Total MMHs:	52.3
3) FUEL TURBOPUMP			
A) GG Inlet and Outlet Flanges	3.0	2	3.75
B) Fuel Inlet Duct Flange	2.0	2	2.5
C) Fuel Outlet Duct Flange	1.0	2	1.3

* MMH (R) = Mean Maintenance Hours (Removal)
** MMH (I) = Mean Maintenance Hours (Installation)

SAMPLE

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Figure 5.1-6. Sample STBE Maintenance Manhours

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Pratt & Whitney has completed a preliminary component classification list that is divided into three categories: components that meet line-replaceable unit (LRU) criteria; potential LRU candidates, and non-LRU candidates. In order for a component to be classified as an LRU, it must meet certain qualitative and quantitative characteristics.

The purpose of establishing a preliminary list of LRUs and potential LRUs is to assist design and support group engineers. Design engineers an incorporate LRU design criteria into the applicable component design, and support group engineers can assess whether qualitative and quantitative goals are being achieved for the evolving design.

As seen in Table 5.1-2, preliminary analysis indicates the components listed under solenoid valve assemblies, electrical cables, actuators, GG igniter assembly, controller and rocket engine condition monitoring system (RECMS) are most likely to qualify as LRUs.

Table 5.1-2. *Components Meeting LRU Criteria*

- Solenoid Valve Assemblies
  - MC LO_2 Injector Purge
  - OGCV Bypass
  - GG LO_2 Injector Purge
  - Spin Assist
  - GG Ignition Fuel
  - Fuel Purge
  - FSOV
  - MOV
  - Antiflood
  - GHe Purge
  - LO_2 Cooldown
  - CH_4 Cooldown
- Electrical Cables
  - Vehicle to Controller Cable No. 1
  - Vehicle to Controller Cable No. 2
  - Vehicle to Controller Cable No. 3
  - Controller to Solenoid Valves Cables
  - Controller to Fuel/Oxidizer GG Control Valves Cable
- Actuators
  - OGCV Bypass Valve
  - FSOV Valve
  - MOV Valve
  - FGCV/OGCV Valve
- GG Igniter Assembly
  - Spark Exciter
  - Torch Igniter
  - Igniter Plug
  - Engine MCC
- Controller
- RECMS
  - Sensors
  - Cables

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Table 5.1-3 outlines the list of potential LRU candidates. Preliminary analysis indicates the components listed under ducts, fuel system, oxidizer system, the GG assembly, and regeneratively-cooled nozzle will require additional analysis as the design evolves and quantitative goals are established in order to determine their classification as LRU versus non-LRU.

Table 5.1-3. *Potential LRU Candidates*

- *Ducts*
  - FSOV to MCC Coolant Inlet — Fuel
  - Regen Nozzle Coolant Discharge to FGCV — Fuel
  - FGCV to GG Injector — Fuel
  - MOV to MCC Injector Inlet — O_2
  - Duct No. 4 to OGCV — O_2
  - OGCV to GG Injector — O_2
  - GG Discharge to Fuel Pump Turbine Inlet — Fuel
  - Fuel Pump Turbine Discharge to LO_2 Pump Turbine Inlet — Fuel
  - GO_2 Hex to Film Nozzle Coolant Inlet — Fuel
  - Duct No. 1 to MCC Injector Inlet — Fuel
  - Fuel Inlet Duct — Fuel
  - Oxidizer Inlet Duct — O_2
  - LO_2 Tank Pressurization Duct — O_2
  - MCC Torch Igniter Oxidizer Supply Duct — O_2
  - MCC Torch Igniter Fuel Supply Duct — O_2
  - Fuel Tank Pressurization Duct — Fuel
  - GO_2 Hex Supply Duct — O_2
  - POGO GO_2 Supply Duct — O_2
  - Helium System Ducts
- Fuel System
  - HPFTP
  - FSOV
  - FGCV
- Oxidizer System
  - HPOTP
  - MOV
  - OGCV
  - POGO Accumulator
  - GO_2 Hex
- GG Assembly
- Regeneratively-Cooled Nozzle

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The non-LRU components, by nature of their design and accessibility when installed in the vehicle, include the gimbal, main combustion chamber (MCC) injector assembly, the MCC, and turbopump mount structure.

Note, this is a preliminary classification and components may be included or excluded from the LRU list as requirements are added or deleted, and the design can be analyzed in more detail as it evolves.

Preliminary Quick-Engine-Change maintenance concepts have been established based on prior P&W experience with propulsion system/vehicle integration for gas turbine engines and recommendations from Kennedy Space Center operations specialists. Table 5.1-4 provides a list of preliminary quick-engine-change maintenance concepts.

Table 5.1-4. Preliminary Quick Engine Change Maintenance Concepts

- Self-Aligning Feature (Guide Cone) Integral to Gimbal Bearing Assembly.
- Heatshield Engine Mounted — Upon Engine to BRM Installation, Heatshield is Attached to Bulkhead With Simple Device, i.e., Single-Bolt C-Clamp.
- Engine Configuration Allows Installation in Any Position.
- Engines Can be Installed/Removed in Any Sequence Desired.
- Attaching Hardware Bolts Gimbal and TVC Actuators Require Standard Torques and Wrenching Device; No Tensiometers, etc.
- Captured Self-locking Nuts.
- Only Common Hand Tools Required to Connect/Disconnect Interface Hardware.
- Fluid/Electrical Connectors Self-Locking Quick-Disconnect Type; Redundant Locking/No X-ray or Special Inspection Required.
- Fluid/Electrical Connectors Will be Located for Optimum Accessibility on Installed Engines.
- Color-Coded Electrical Harnesses and Bulkhead Connectors.
- Colored Identification Banding for All Engine Plumbing Lines on Engine and Bulkhead.
- No Rigging or Critical Alignment Requirements for TVC Actuators; Engine to Vehicle.
- Engine TVC Attach Point Will be Designed For Easy Connecting/Disconnecting/Alignment.
- Vehicle Health Monitoring System Will Have Built-in-test Capability to Perform All Electrical Interface Verifications.
- Automated Leak Checks of Engine Plumbing Connections Performed With Onboard Systems; No Government Support Equipment Required.
- Automated TVC Flight Controls Test.

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Modular engine design optimizes repair capability at the operating location and reduces number of spare engines and associated pipeline time. This concept has evolved from Gas Turbine Engine designs and has proven to be successful in reducing operating and support costs while increasing system availability. The preliminary concept is to have modules assembled and tested at Stennis Space Test Center. The required number of serviceable spare modules to support the mission will be stocked at the operating location for ready access. Modules will be interchangeable from engine to engine maintaining the required performance tolerance band for the engine without an engine trim run. Module self-test capability would reduce maintenance task times and require less ground support type equipment.

Preliminary analysis has identified the following components as engine modules for the current STBE design: oxidizer turbopump, fuel turbopump, main combustion chamber, main injector, gas generator, regeneratively-cooled nozzle, and control components. (See Figure 5.1-7.)

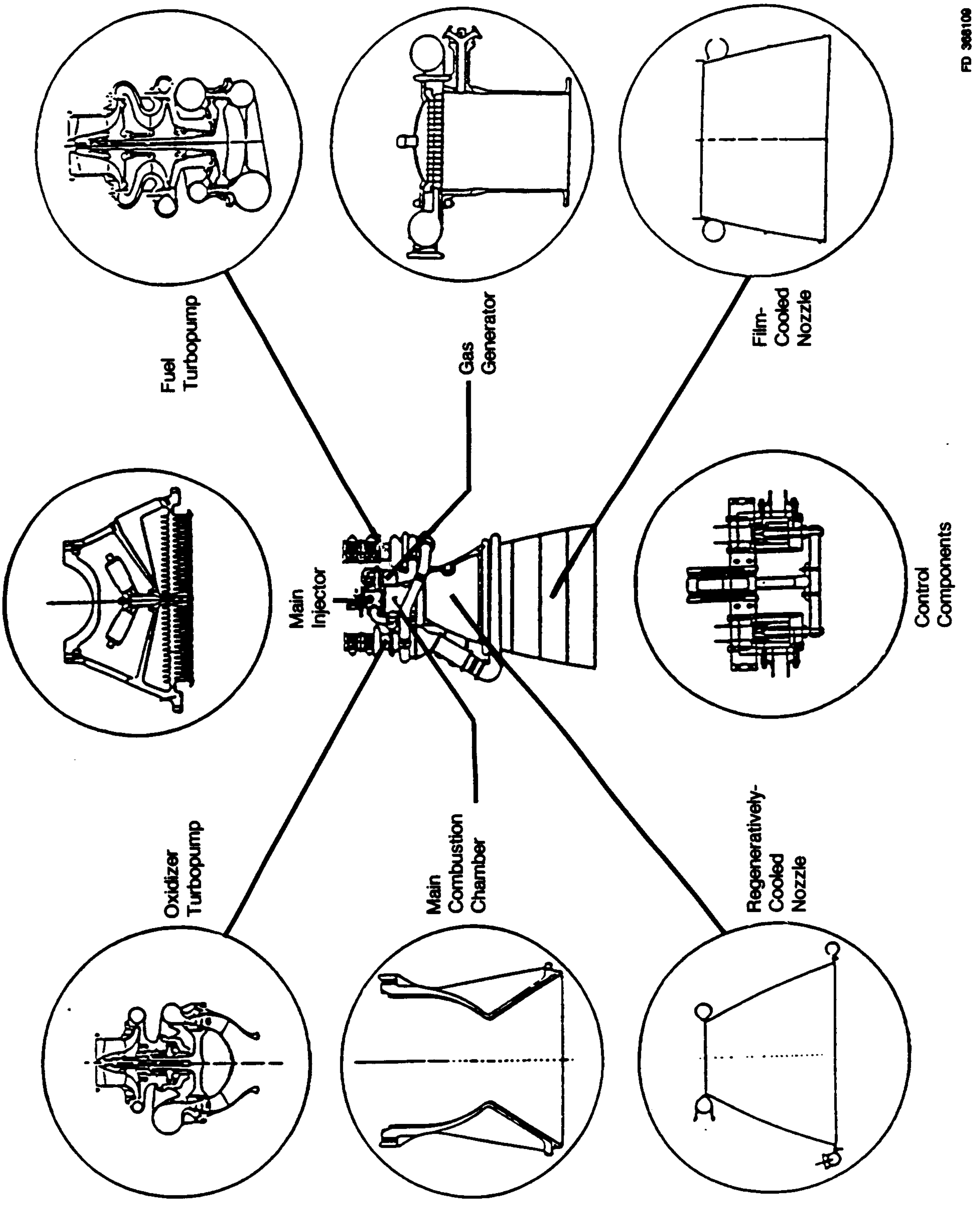

Figure 5.1-7. Space Transportation Booster Engine Gas Generator Modular Design