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C70-1091

AIR FORCE REUSABLE ROCKET ENGINE PROGRAM

XLR129-P-1

2

AD 508757

DEMONSTRATOR ENGINE DESIGN

AFRPL-TR-70-6

APRIL 1970

R. R. ATHERTON

PRATT & WHITNEY AIRCRAFT

DIVISION OF UNITED AIRCRAFT CORPORATION

FLORIDA RESEARCH AND DEVELOPMENT CENTER

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LAMINATED CHAMBER COOLING MEANS AND A SLOT TUBE INJECTOR CONCEPT

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AIR FORCE REUSABLE ROCKET ENGINE PROGRAM

XLR129-P-1

DEMONSTRATOR ENGINE DESIGN

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FOREWORD

(U) This Milestone Report is issued at the time in the program when the design of all major components as well as the design of the demonstrator engine system has been completed. It presents the design approach, mechanical description, and operating characteristics of the XLR129-P-1 engine and each major component. This report is issued as a technical report in compliance with the requirements of Contract F04611-68-C-0002. Classified information has been extracted from documents listed under References.

(U) This publication was prepared by the Pratt & Whitney Aircraft Florida Research and Development Center as PWA FR-3337.

(U) This Technical Report has been reviewed and is approved.

Robert E. Probst
Captain, USAF
Program Manager
Air Force Rocket Propulsion Laboratory

UNCLASSIFIED ABSTRACT

(U) This report describes the design of the XLR129-P-1 Demonstrator Rocket Engine and its principal components. The program is being conducted by Pratt & Whitney Aircraft under Air Force sponsorship at the Florida Research and Development Center. Design of all components, the second of five program phases, has been completed. Included in the report is the design approach, mechanical description, and operating characteristics for each component. The engine is designed to operate with liquid oxygen and liquid hydrogen propellants, uses the staged combustion cycle, incorporates variable thrust, and variable mixture ratio capability. The XLR129-P-1 engine, having a high area ratio nozzle, is designed to be reusable as in aircraft engine practice, and provides 250,000 pounds thrust in vacuum. The program started 6 November 1967, and is planned for 54 months. The major program objectives include: (1) design of the components and engine system with a series of component tests to support the design effort, (2) development of the components to qualify them for engine use and to demonstrate the life of life-limited sub-components, and (3) a series of engine tests to demonstrate operational capabilities.

PREFACE

(U) This volume contains introductory and summary material relating to the requirements of the XLR129-P-1 Demonstrator Rocket Engine Program and to the components and characteristics of the engine system. Detailed component descriptions are contained in Volume 2. Volume 3 contains data pertaining to the control system, demonstrator engine mockup, and plumbing system. Also included in Volume 3 are appendixes containing structural design criteria and data.

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LIST OF SYMBOLS

A_{cd}	Effective Flow Area, in.^{2}
A_s	Slot Area, in.^{2}
A_o	Element Area, in.^{2}
A_1	Inlet flow area excluding blockage, in.^{2}
A_2	Exit flow area excluding blockage, in.^{2}
B_x	Blade Angle Distribution
B_2	Exit blade height, in.
C	Radial clearance, in.
C_L	Coefficient of Lift
C_{M1}	Inlet meridional velocity, fps
C_{M2}	Exit meridional velocity, fps
C_r	Clearance, in.
C_{U1}	Inlet tangential velocity, fps
C_{U2}	Exit tangential velocity, fps
C_1	Inlet absolute velocity, fps
C_2	Exit absolute velocity, fps
C_3	Absolute Volute Velocity, fps

LIST OF SYMBOLS (Continued)

D_1	Diffuser Inlet Diameter, in.
D_2	Diffuser Exit Diameter, in.
D_{1h}	Inlet hub diameter, in.
D_{2h}	Exit hub diameter, in.
D_{1M}	Inlet mean diameter, in.
D_{2M}	Exit mean diameter, in.
D_{1T}	Inlet tip diameter, in.
D_{2T}	Exit tip diameter, in.
i	Leading Edge Incidence, deg
I_s	Specific Impulse (instantaneous), lb_f - sec/lb_m
L/R	Diffuser Length to Inlet Radius Ratio
ℓ_x	Mean axial length, in. tip
M	Momentum, Ft-lb/sec
N	Speed, rpm
S	Specific speed
$NPSH_{Fluid}$	Net positive suction head in $fluid_{LH_2}$ ft required
$NPSH_{H_2O}$	Net positive suction head in H_2O ft required
P	Pressure, psi
P_c	Chamber Pressure (throat total), psia

LIST OF SYMBOLS (Continued)

P_S	Static Pressure, psi
P_{VAP}	Vapor pressure, psi/ft
P_1	Inlet pressure, psi total required
Q_1	Flow rate at inlet/exit, gpm
R_c	Cutwater Radius, in.
R_{c3}	Velocity Correction Factor, C_3/C_2
r	Mixture Ratio (oxidizer to final) by weight
S	Suction specific speed
S_{Fluid}	Suction specific speed in $fluid_{LH_2}$ capability
S_{H_2O}	Suction specific speed in H_2O capability
SMD	Sauter Mean Diameter, in.
t_c	Cutwater Thickness, in.
t_h	Hub Thickness, in.
T_{H1}	Inlet hub blade thickness, in.
T_{H2}	Exit hub blade thickness, in.
t_t	Tip Thickness, in.
T_{T1}	Inlet tip blade thickness, in.
T_{T2}	Exit tip blade thickness, in.
TSH	Thermodynamic suppression head, ft
T_1	Inlet temp., °R

LIST OF SYMBOLS (Continued)

U_{1T}	Inlet blade tip speed, fps
U_{2m}	Exit mean blade speed, fps
$\dot{W}$	Flow rate, lb/sec
$\dot{W}_c$	Cooling Flowrate, lb/sec
X	Number of Diffusers
Z	Number of Blades
σ	Surface Tension
α_2	Absolute Fluid Angle, deg
β1h	Inlet hub blade angle, degrees
β2h	Exit hub blade angle, degrees
β1M	Inlet mean blade angle, degrees
β2M	Exit mean blade angle, degrees
β1T	Inlet tip blade angle, degrees
β2T	Exit tip blade angle, degrees
β3	Volute Width, in.
ΔH_{T_P}	Total head rise overall, ft
ψT_P	Total Head Coefficient
ΔP_{T_P}	Total pressure rise, overall, psi
ΔH_{S_P}	Static head rise overall, ft
ψS_P	Static Head Coefficient
ΔP_{S_P}	Static pressure rise overall, psi

LIST OF SYMBOLS (Concluded)

ΔH_{T_I}	Static head rise rotor, ft
ψ_{S_I}	Rotor Head Coefficient
ΔP_{T_I}	Total pressure rise rotor, psi
ΔH_{S_I}	Static head rise rotor, ft
ΔP_{S_I}	Static pressure rise rotor, psi
δ	Specific Gravity
ϵ	Nozzle Area Ratio
η	Efficiency, % overall
η_c^*	(Average Temperature/Ideal Temperature)$^{0.5}$ x 100
η_{IO}	Engine Impulse Efficiency, percent
θ	Camber, deg
2θ	Diffuser Included Angle, deg
λ_1	Inlet Hub-Tip Diameter Ratio
λ_2	Exit Hub-Tip Diameter Ratio
μ	Viscosity, lb-sec/ft^2
ρ	Density, slug/ft^3, lb-sec/ft^4
ρ_i	Inlet/exit density, lb/ft^3
σ	solidity
Φ_{it}	Inlet Tip Flow Coefficient
Φ_{2M}	Flow coeff. @ mean exit diameter
Φ_{1T}	Flow coeff. @ inlet tip diameter
ψ	Head Coefficient

SECTION I
INTRODUCTION

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SECTION I
INTRODUCTION

A. GENERAL

(U) The Air Force XLR129-P-1 Reusable Rocket Engine Program is an Advanced Development Program that covers a 54-month period starting 6 November 1967 and ending 6 May 1972. The overall objective of this program is to demonstrate the performance and mechanical integrity of a 250K, oxygen-hydrogen, reusable rocket engine having characteristics outlined in table I.

(U) Table I. Demonstrator Engine Characteristics

Nominal Thrust	250,000-lb vacuum thrust with area ratio of 166:1 244,000-lb vacuum thrust with area ratio of 75:1 209,000-lb sea level thrust with area ratio of 35:1
Minimum Delivered Specific Impulse Efficiency	96% of theoretical shifting I_s at nominal thrust; 94% of theoretical shifting I_s during throttling
Throttling Range	Continuous from 100 to 20% of nominal thrust over the mixture ratio range
Overall Mixture Ratio Range	Engine operation from 5.0:1 to 7.0:1
Rated Chamber Pressure	2740 psia
Engine Weight (with 75:1 nozzle)	3520 lb with flight-type actuators and engine command unit 3380 lb without flight-type actuators and engine command unit
Expansion Ratio	Two-position booster-type nozzle with area ratios of 35:1 and 75:1
Durability	10 hours time between overhauls, 100 reuses, 300 starts, 300 thermal cycles, 10,000 valve cycles
Single Continuous Run Duration	Capability from 10 seconds to 600 seconds
Engine Starts	Multiple restart at sea level or altitude
Thrust Vector Control	Amplitude: ±7 deg Rate: 30 deg/sec Acceleration: 30 rad/sec^2

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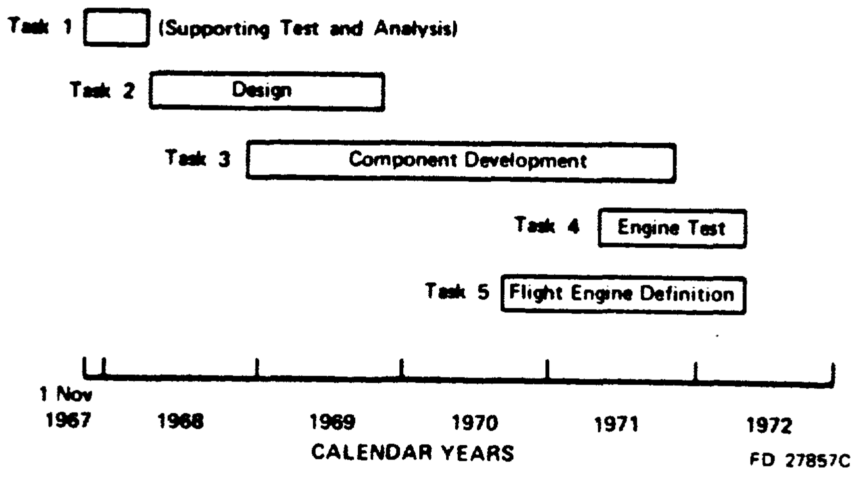
(U) Table I. Demonstrator Engine Characteristics (Continued)

Control Capability	±3% accuracy in thrust and mixture ratio at nominal thrust. Excursions from extreme to extreme in thrust and mixture ratio within 5 seconds.
Propellant Conditions	LO_2: 16 ft NPSH from 1 atmosphere boiling temperature to 180°R LH_2: 60 ft NPSH from 1 atmosphere boiling temperature to 45°R
Environmental Conditions	Sea level to vacuum conditions Combined acceleration: 10 g axial with 2 g transverse, 6.5 g axial with 3 g transverse, 3 g axial with 6 g transverse CONFIDENTIAL
Engine/Vehicle	The engine will receive no external power, with the exception of normal electrical power and 1500-psia helium from the vehicle

(U) The XLR129-P-1 demonstrator engine program schedule is shown in figure 1. The program has been divided into five tasks. Task 1 which has already been completed, generated test and analytical data to complete the necessary technology to design the engine and components. During task 2, all the components and the demonstrator engine were designed. This milestone report describes the results of work accomplished during task 2 and presents the mechanical description, operating characteristics, and design approach of each major component and the demonstrator engine. During task 3, components will be fabricated and tested to qualify them for engine use. Task 4 will be the integration of components into the demonstrator engine and testing of the demonstrator engine. A flight engine will be defined in task 5.



(U) Figure 1. XLR129-P-1 Demonstrator Engine Program Schedule

B. PROGRAM TASKS

(U) The entire program consists of five tasks and specific subtasks as follows:

Task 1 - Supporting Data and Analysis

Subtasks

1.1 - Fixed Fuel Area Preburner Injector Evaluation

1.2 - Roller Bearing Durability Tests

1.3 - Pump Inlet Evaluation

1.4 - Nozzle Fabrication Investigation

1.5 - Controls Component Test

Task 2 - Design

Subtasks

2.1 - Preburner Injector

2.2 - Main Burner Injector

2.3 - Nozzles

2.4 - Main Burner Chamber

2.5 - Transition Case

2.6 - Fuel Turbopump

2.7 - Oxidizer Turbopump

2.8 - Fuel Low-Speed Inducer

2.9 - Oxidizer Low-Speed Inducer

2.10 - Control System

Task 3 - Component Development

Subtasks

3.1 - Preburner Injector

3.2 - Main Burner Injector

3.3 - Nozzles

3.4 - Main Burner Chamber

Subtasks

3.5 - Transition Case

3.6 - Fuel Turbopump

3.7 - Oxidizer Turbopump

3.8 - Fuel Low-Speed Inducer

3.9 - Oxidizer Low-Speed Inducer

3.10- Control System

Task 4 - Engine Integration and Demonstration

Task 5 - Flight Engine Definition

SECTION II
SUMMARY

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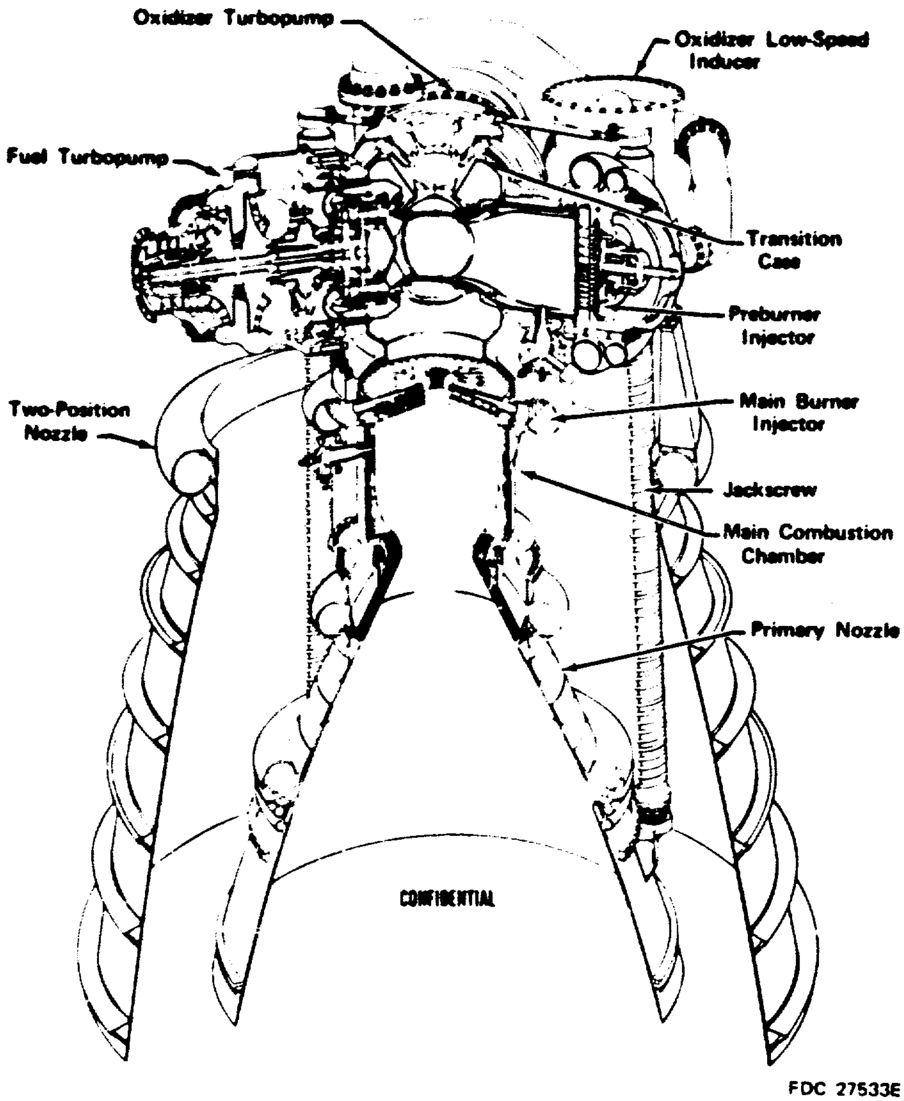
SECTION II
SUMMARY

A. DEMONSTRATOR ENGINE

1. Background

(C) Early studies of advanced liquid rocket engines by Pratt & Whitney Aircraft showed that high combustion chamber pressure could provide high performance and high thrust per unit area of the exhaust nozzle. These studies also showed the critical technology areas were cooling and turbomachinery. Air Force contracts directly related to the high-pressure rocket engine investigations are: AF04(611)-7435 High-Pressure Rocket Engine Feasibility, AF04(611)-10372 Staged Combustion Research, and AF04(611)-11401 Advanced Cryogenic Rocket Engine. NASA contracts directly related to high-pressure rocket engine investigations were: NAS8-11427 Design Study Engine System for Upper Stages of Uprated Saturn, NAS8-11714 Design, Fabricate, and Test a Breadboard Liquid Hydrogen Pump, and NAS8-20540 Liquid Oxygen Turbopump Study. The Advanced Engine Design Study, conducted under NASA contract NAS8-11427, showed that a single preburner, staged-combustion cycle, high-pressure, oxygen-hydrogen, bell-nozzle rocket engine provided flexibility, envelope, and the high performance required for future advanced vehicle applications. During this NASA contract, detailed cycle studies indicated a single-preburner, staged-combustion cycle provided the best compromise between delivered specific impulse, weight, and complexity. These cycle studies also showed that a 5 to 7 mixture ratio range capability and a 10:1 throttling range could be provided with a minimum number of control points. The exploratory development programs demonstrated the feasibility of these component concepts to permit the initiation of an Advanced Development Program. During Phase I of the Air Force Advanced Development Program for a High Performance Cryogenic Rocket Engine under Contract AF04(611)-11401, a two-position, translating bell-nozzle concept was developed that provided a more compact bell-nozzle engine. At the end of Phase I, which ended 30 September 1967, the high-pressure engine had evolved to a flexible design. By fitting nozzle skirts of different area ratios, the same engine could be optimized for a variety of missions. Today the engine design shown in figure 2 offers high performance, altitude compensation, versatility and long life in a compact engine package.



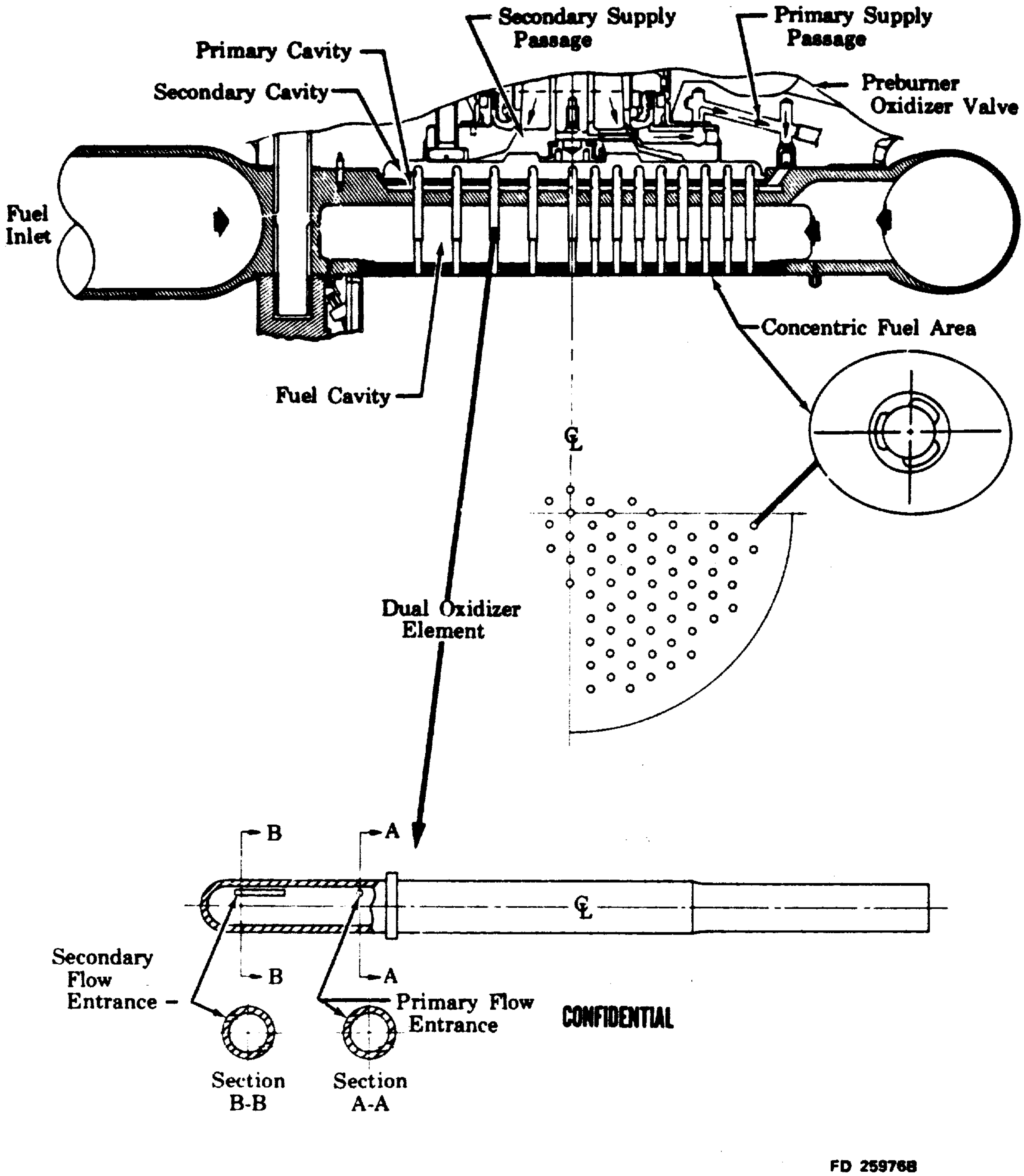
(U) Figure 2. XLR129-P-1 Demonstrator Engine

2. Design

(U) During Phase II, covering the present contract (F04(611)-68-C-0002), the design of the 250K demonstrator engine and its major components, including plumbing, has now been completed. Design of the engine and major components is based on proven test technology; namely, all the major components such as the combustion devices and turbomachinery have been tested at either the 250K or 350K thrust level, demonstrating their feasibility. Design studies were also conducted on the demonstrator engine in the areas of engine mockup and plumbing. A full-scale mockup of the demonstrator engine was used as a working tool during design. Numerous design iterations have been conducted on the mockup for component arrangement and plumbing. A satisfactory component arrangement for the engine mockup has been established. In the area of the engine plumbing, satisfactory designs and arrangements have been established for the fuel pump discharge lines, the preburner fuel inlet line, the main burner oxidizer inlet line, and associated components such as actuators, rods, and small connectors. Engine system analyses have also been conducted during the program to define component design requirements, estimate capabilities of the integrated engine system, and to include the results of component and engine tests. These analyses include: steady-state analysis, transient analysis, shutdown analysis, special design-cycle studies, and generation of performance data.

B. PREBURNER INJECTOR

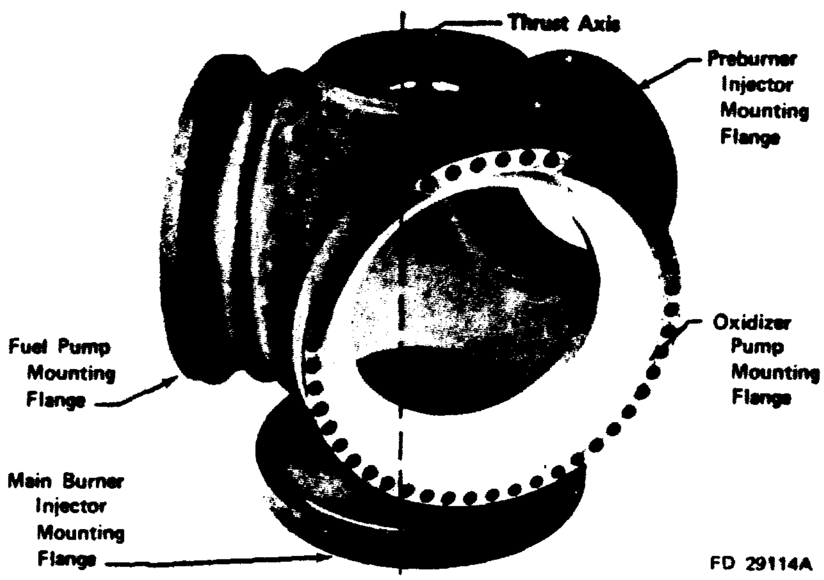
(C) The preburner is an oxygen-hydrogen combustor supplying hot gases to drive the oxidizer and fuel pump turbines. Because preburner gases are used to drive the fuel and oxidizer pump turbines, the design goal temperature profile is 150°R peak-to-average to permit operating the turbines at the maximum allowable average temperature. The design of the preburner injector consists of 254 dual-orifice, tangential-swirler oxidizer elements, with concentric fuel annuli around each oxidizer element. All are arranged in a hexagonal pattern shown in figure 3. This design has demonstrated a peak-to-average combustion temperature profile of 76°R in a radial plane at an average gas temperature of 2388°R. This fixed-area, fuel-injection design concept is feasible because density changes occurring in gaseous fuel allow throttling while simultaneously maintaining a suitable injection velocity. However, because liquid oxygen is essentially incompressible, the dual-orifice principle is applied to a slot-swirler element to provide suitable injection velocity over the throttled range. The preburner-injector housing has 28 slots to allow gaseous hydrogen to flow from the outer fuel manifold to the manifold behind the faceplate. Primary oxidizer flow enters the primary oxidizer manifold through six equally spaced ports in the preburner-injector housing. Secondary oxidizer flow arrives at the secondary oxidizer manifold directly from the preburner oxidizer valve. The porous injector faceplate is fabricated from Rigimesh. Ignition systems will be integral spark igniter-exciter units that are mounted on both the preburner and main chamber. Two systems will be provided for the preburner and two for the main chamber to provide total spark redundancy.



(U) Figure 3. Preburner Injector Cross Section

C. TRANSITION CASE

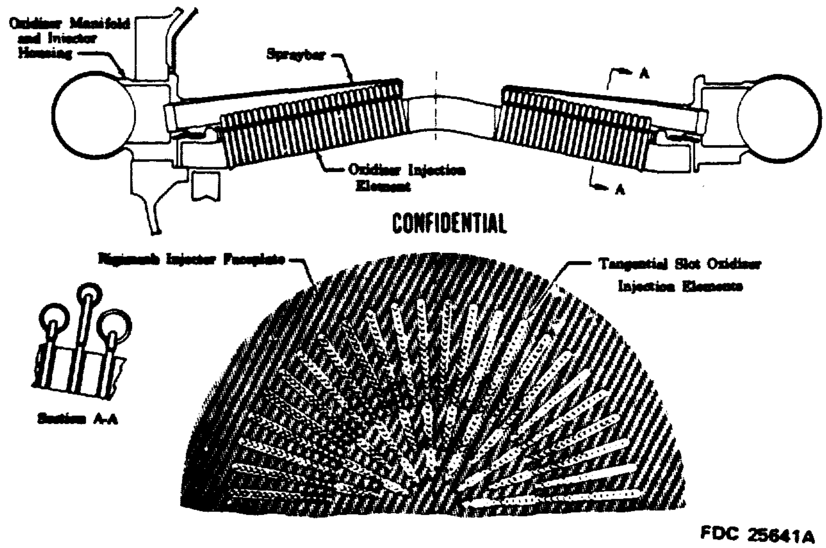
(C) The transition case consists of four spheres; one main sphere and three small spheres whose centerlines intersect the main sphere at right angles. The smaller spheres act as the attachment points for three major components; the preburner injector, the oxidizer turbopump, and the fuel turbopump. The main sphere centerline coincides with the engine thrust axis. The entire assembly is a pressure vessel. The transition case contains internal ducting that routes preburner discharge gases through the fuel and oxidizer pump turbines and to the main burner injector as well. The goal of the transition case subtask is to demonstrate the structural adequacy of the engine transition case when operating at an internal pressure of 4856 psia and with an internal gas temperature as high as 2325°R. A full-scale mockup of the primary structure of the transition case is shown in figure 4. With incorporation of the preburner injector, fuel turbopump, and oxidizer turbopump, the transition case is a self-contained powerhead supplying the main-burner thrust chamber with high-pressure propellants necessary to produce the design thrust. Moreover, it serves as the primary combustor stage for the staged-combustion cycle. Internal ducting of the transition case splits the hot fuel-rich gases from the preburner to provide adequate gas flow to each turbine. The fuel turbine requires about twice the mass flow required to drive the oxidizer turbine. Cooling liners, positioned between the outer case and the hot-gas flowpath, are included to keep the outer case temperature below 540°R. A satisfactory design of the transition case has evolved and meets all established design criteria for this important component.



(U) Figure 4. Transition Case Full-Scale Mockup

D. MAIN BURNER INJECTOR

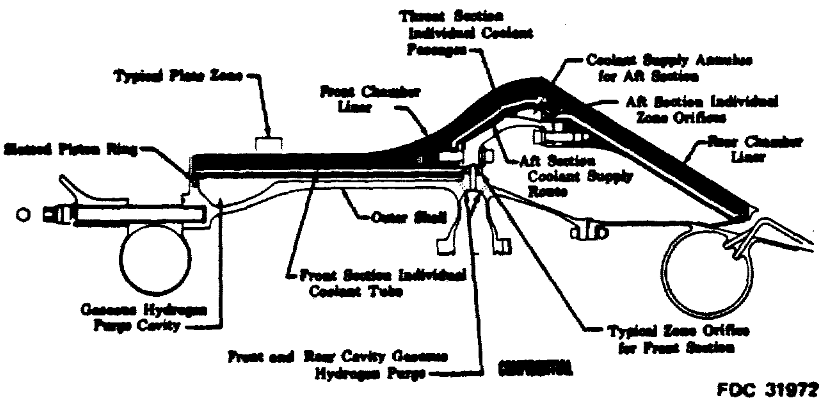
(U) The main burner injector introduces, atomizes, and mixes liquid oxygen with the hot, fuel-rich turbine discharge (preburner combustion products) so efficient and stable combustion is achieved over the full operating range of thrust and mixture ratios. The main burner injector design consists of the oxidizer manifold and housing, spraybar-type internal manifolds, oxidizer-injection elements, and the porous faceplate as shown in figure 5. The main injector housing consists of an oxidizer-inlet horn, the oxidizer manifold, and crossover passages to the spraybars. The spraybar injector body consists of 48 individually machined spraybars brazed into the oxidizer-manifold ring. The spraybars are individually supported at the outside diameter only, thus permitting free thermal growth. This approach simplifies manufacturing and provides a lightweight design. Forty-eight radial spraybars are divided into three groups; 12 long spraybars equally spaced around the circumference, 12 medium spraybars equally spaced between the long spraybars, and 24 short spraybars equally spaced between the medium and long spraybars. This arrangement yields the maximum number of spraybars consistent with mechanical considerations, and results in good oxidizer-element density and uniform radial flow distribution. Self-atomizing injection elements are spaced along the spraybars to obtain good atomization and distribution. The fuel faceplate is made of Rigimesh, which forms the support structure as well as the porous face. The faceplate directs approximately 92% of the hot, fuel-rich, preburner combustion gases through slots surrounding the oxidizer-injector elements. The remainder of the gas passes through the porous faceplate. Major components are assembled by brazing and welding techniques that simplify manufacturing the components. This main-burner injector configuration represents a minimum overall length and weight design that satisfies the demonstrator engine cycle requirements.



(U) Figure 5. Main Burner Injector Configuration

E. MAIN BURNER CHAMBER

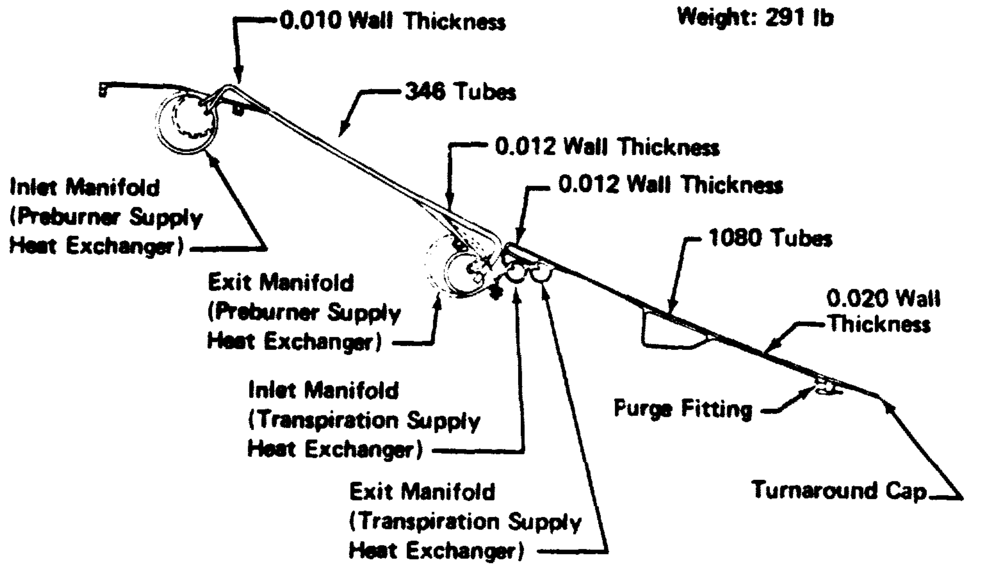
(C) The main burner chamber contains the pressure resulting from propellant combustion, serves as the structural member supporting the primary and two-position nozzles, transmits thrust, and absorbs gimbal actuator loads. The overall goals of the main burner chamber subtask are to design, build, and demonstrate through full-scale testing, performance and operational capability of a lightweight, durable, thrust chamber for use in the demonstrator engine program over the specified throttling and mixture ratio ranges. Ignition capability must also be demonstrated at both sea level and altitude conditions. The main burner chamber design consists of two main components; an outer pressure shell and a transpiration cooled, copper wafer, chamber liner shown in figure 6. The outer pressure shell also provides the coolant manifold and serves as a mount for attaching the chamber liner in two-sections. Copper cylindrical wafers forming the chamber are divided into 28 zones. Each wafer consists of front and back plates. A zone is a collection of composite wafers fed by interconnected zone coolant manifolds. The chamber liner consists of a stackup of 512 0.040-inch thick copper wafer halves brazed together. Spiral groove patterns photoetched into one side of each wafer-half provide the path from the zone coolant manifold to the inside diameter of the chamber where they open into the main burner chamber. Composite wafers are constructed of two half-plates brazed at the unetched center plane with an axial thermal relief slot in the front wafer-half. By locating the slot in this plane, the heat exchanger spiral grooves on the opposite face are not affected. The addition of axial thermal strain relief slots minimizes the wafer thermal strain level by allowing free axial growth at the hot wall of the chamber, thus producing an acceptable low cycle-fatigue life.



(U) Figure 6. Main Burner Chamber Assembly Side View

F. PRIMARY NOZZLE

(C) The function of the primary nozzle is to contain the combustion gases and allow their shock-free expansion from an area ratio of 5.3:1 to 35:1. High-pressure hydrogen from the fuel pump is supplied as coolant to two regeneratively cooled portions of the primary nozzle. A second function of the primary nozzle is to provide structural support for the two-position nozzle. A third function is to act as heat exchangers to condition the hydrogen supplied to the transpiration-cooled chamber and the preburner injector. The primary nozzle design consists of two tubular, regeneratively-cooled, heat exchangers shown in figure 7. The downstream heat exchanger is double pass, and supplies hydrogen to the hydrogen inducer turbine and transpiration-cooled main burner chamber. The upstream heat exchanger is single pass, and cools the nozzle from an area ratio of 5.3 to 18 using approximately 85 percent of the pump discharge hydrogen flow prior to delivery to the preburner injector. Both heat exchangers are shaped from tubes forming the desired nozzle contour. Based upon low cycle-fatigue data, Inconel 625 (AMS 5666B) was selected for heat exchanger tube material. This material is easily welded and may be used after welding without further heat treatment. The support for the two-position nozzle is accomplished by the rear thrust bearings for the jackscrew actuators being supported in a circumferential ring at the midspan of the transpiration heat exchanger.

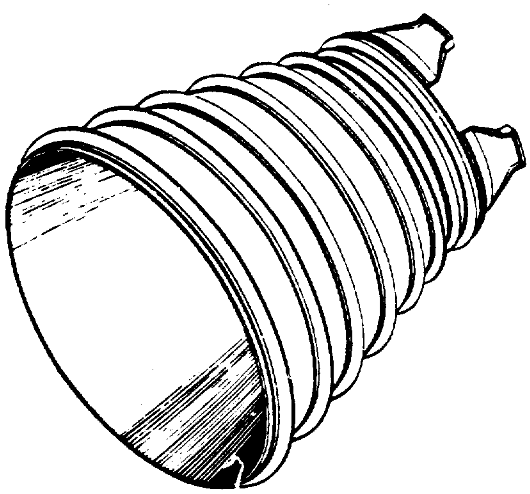


(U) Figure 7. Primary Nozzle Configuration

G. TWO-POSITION NOZZLE AND TRANSLATING MECHANISM

(C) The function of the two-position nozzle in the extended position is to contain the combustion gases and allow additional shock-free expansion from an area ratio of 35:1 to 75:1. The two-position nozzle translates to provide a compact engine package in the retracted position. The translating mechanism is designed to provide positive extending and retracting of the two-position nozzle during engine operation. The two-position nozzle design consists of a circumferential coolant distribution manifold, a smooth nozzle outer skin with circumferential stiffening bands, and a corrugated inner nozzle skin shown in figure 8. The corrugated inner skin forms longitudinal coolant passages. The two-position nozzle has a baseline contour starting at an area ratio of 35:1 and extends to a ratio of 75:1. This nozzle is designed to be dump cooled with low-pressure hydrogen taken from the fuel pump interstage. During sea level and low altitude operation with the nozzle retracted, coolant flow is not required. When the nozzle is extended, coolant is expanded through small nozzles at the ends of the corrugated coolant passages. Expansion of this warm hydrogen gas produces a specific impulse comparable to the main stream specific impulse. The nozzle is designed to withstand the maximum thrust pressure load plus a 10 g axial maneuver load. The translating mechanism provides precision positioning in the extended and retracted positions. Positive locking devices maintain the nozzle position when the engine is not operating.

Material	Inconel 625
Outer Sheet	Smooth
Outer Sheet Thickness	0.016
Inner Sheet	Corrugated
Inner Sheet Thickness	0.010 in.
Number Corrugations	360
Inlet Dia	46 in.
Exit Dia	66 in.
Length	50 in.
Weight (Sea Level)	227 lb
Weight (Altitude)	186 lb



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(U) Figure 8. Two-Position Nozzle Design

H. LOW SPEED INDUCERS

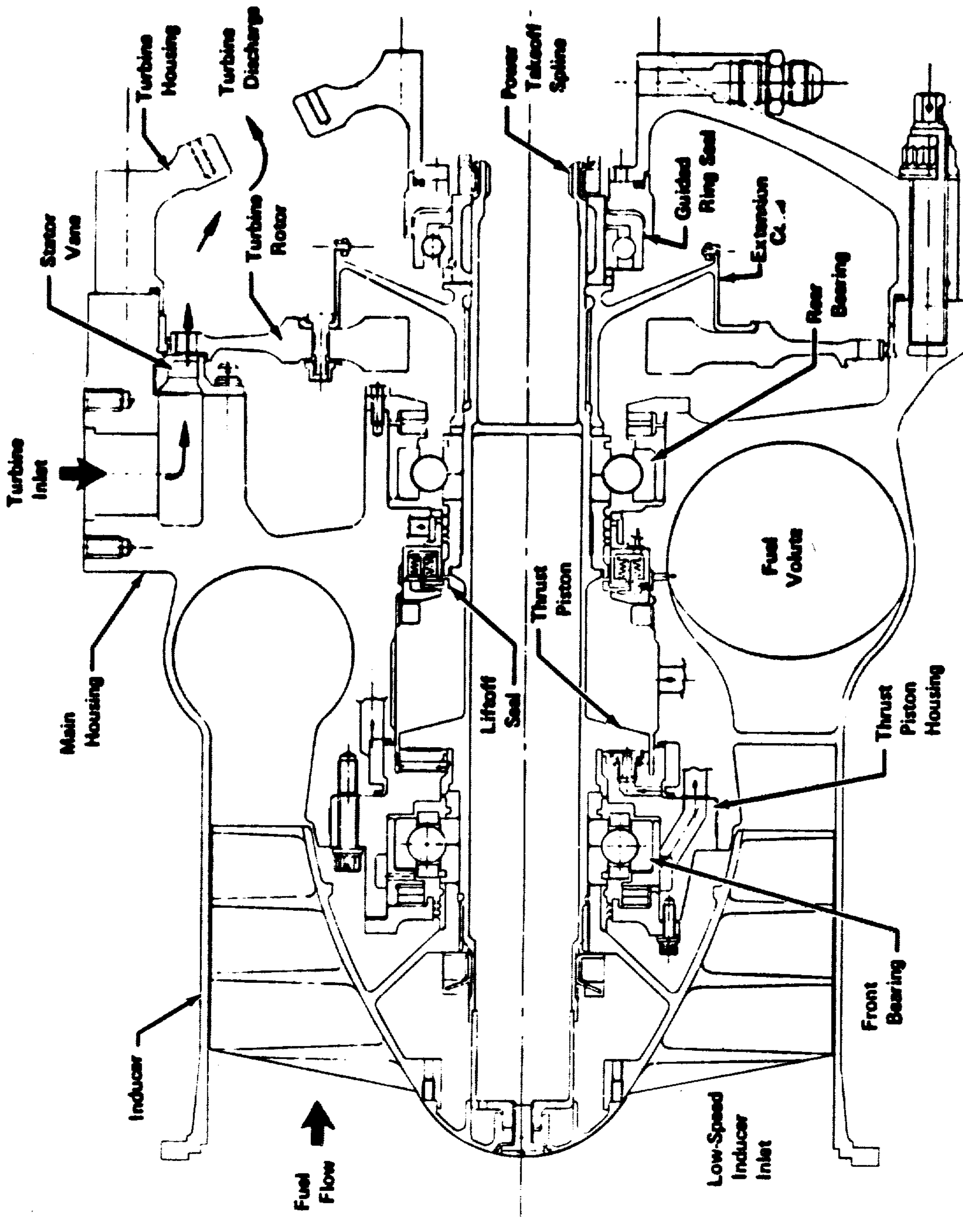
(C) The function of the fuel and oxidizer low-speed inducers is to supply propellants to the main turbopumps at a pressure (NPSH) sufficient to prevent cavitation. This permits the vehicle propellant tanks to be maintained at a lower pressure thus saving tank and vehicle weight. The overall goal of the fuel and oxidizer low-speed inducer subtasks is to demonstrate performance and operational capability for use in the demonstrator engine. The three-bladed fuel inducer is driven by a single-stage partial-admission turbine. The fuel inducer operates independently of the main turbopump and at a speed lower than the main turbopump. This permits the fuel inducer to operate with a low inlet NPSH without cavitation. The fuel inducer is capable of operating with an NPSH as low as 60 feet, ove. a hydrogen inlet temperature range, from 36°R to 45°R This inducer is designed for a suction specific speed of 48,400 rpm $GPM^{1/2}/ft^{3/4}$ and a maximum pressure rise of 90 psid. The fuel low-speed inducer consists of bearings, shaft and thrust piston, turbine, and housings and is shown in figure 9. The oxidizer low-speed inducer is a single shaft axial flow unit with high suction specific speed. It is driven by a variable-admission, single-stage, radial-inflow, hydraulic turbine. The turbine is driven by fluid supplied from the discharge of the main oxidizer turbopump. The oxidizer inducer is of helical design with three blades, and is attached to the drive shaft and turbine assembly as shown in figure 10. The shaft axial thrust imbalance is absorbed by a single acting thrust balance piston. The oxidizer inducer was designed to operate at a minimum NPSH of 16 feet over an oxygen inlet temperature range from 162°R to 180°R with a suction specific speed of 44,000 rpm $GPM^{1/2}/ft^{3/4}$ and a maximum pressure rise of 197 psid. Both inducers are designed for 100 reuses and a 10-hour life between overhaul.

I. FUEL TURBOPUMP

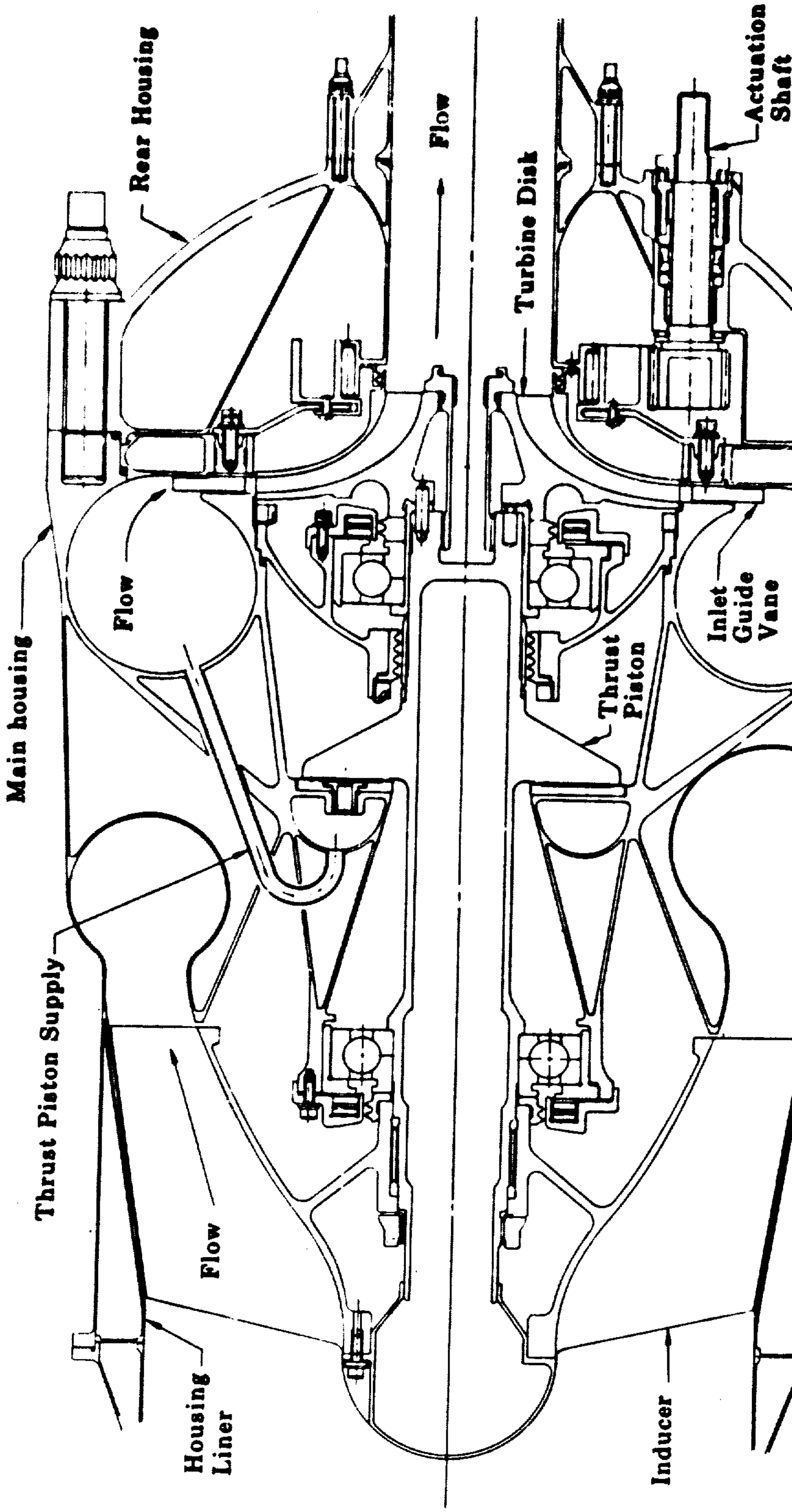
(C) The fuel turbopump supplies liquid hydrogen to the primary nozzle, the two-position nozzle and to the preburner injector at sufficient pressure and flowrates for engine operation from 20% to 100% maximum thrust and at mixture ratios from 5.0 to 7.0. The overall goal of the fuel turbopump subtask is to demonstrate an operational capability for use in the demonstrator engine program. The demonstrator engine requires the fuel turbopump to deliver liquid hydrogen at a flowrate of 91.3 lb/sec at a pressure of 5654 psia at its design point of 100% thrust and a mixture ratio of 5. The design of the fuel turbopump is shown in figure 11. The two-stage turbine delivers approximately 49,900 horsepower to the pump and operates at a maximum inlet temperature of 2325°R at 100% thrust and a mixture ratio of 7. The fuel pump must also demonstrate satisfactory starting capability and stable operation over the engine operating range of 20 to 100% thrust and mixture ratio range of 5 to 7. Pump life is based on 10 hours between overhaul and 100 reuses. Major components of the fuel turbopump are the pump, turbine, rotor assembly, and housings. The pump section includes the inducer, two bearings and mount systems, two impellers, and the thrust balance system. The high-speed inducer has three equally spaced helical blades. The roller bearings are 55 x 96.5 mm and are hydrogen cooled. The two pump impellers have 24 equally spaced, curved blades divided into increments of 6 long blades, 6 medium length splitter blades, and 12 short splitter blades. The thrust balance system, designed to compensate for any net axial imbalance during operation, provides a force of 50,000 pounds using a 2100 psia pressure difference. The thrust balance system consists of the thrust piston, the thrust piston housing, and the rear bearing housing. The two-stage turbine is cantilevered from the rear bearing assembly and consists of two turbine stages, inlet ducting, a support structure, and inlet and exit ducting. The pump housings consist of the inducer housing, the main housing, the thrust piston housing, and the rear bearing housing. Inconel 600 (AMS 5665) is used for the inducer housing and Inconel 718 (AMS 5663) is used for the other housings.

J. OXIDIZER TURBOPUMP

(C) The oxidizer turbopump supplies liquid oxygen to the preburner injector and main burner injector at sufficient pressure and flowrates for engine operation from 20 to 100% of maximum thrust and at a mixture ratio range from 5 to 7. The overall goal of the oxidizer turbopump subtask is to demonstrate performance and operational capability for use in the demonstrator engine program. The demonstrator engine requires the oxidizer turbopump to deliver liquid oxygen at a flowrate of 481 lb/sec at a pressure of 4800 psia at 100% thrust and mixture ratio of 7. The design of the oxidizer turbopump assembly is shown in figure 12. The oxidizer turbopump is a single-shaft unit with a single-stage shrouded centrifugal impeller driven by a two-stage, pressure-compounded turbine. The rotor shaft is supported by two antifriction 55 x 110 mm ball bearings. The forward bearing is located between the impeller and the thrust balance assembly, and the rear bearing is located in front of the turbine and separated from the turbine by a low leakage labyrinth seal. The rear bearing is cooled by liquid hydrogen and the front bearing is cooled by liquid oxygen. The bearings are separated by a low-leakage seal package that vents coolant leakage overboard. The seal package consists of one lift-off seal and five labyrinth seals. The two-stage turbine delivers a maximum of 18,000 horsepower to the pump and operates at a maximum inlet temperature of 2325°R at 100% and a mixture ratio of 7.0. The two-stage turbine consists of the turbine inlet duct, the two turbine stages, and the exit duct. Life is based on 10-hours time between overhaul and 100 reuses. The oxidizer turbopump consists of a pump, turbine, and housings. The pump section consists of the inducer impeller, bearings and mount system, thrust balance system, and the seals. The high speed inducer consists of three equally spaced helical blades having a constant tip diameter. The single stage impeller has 12 equally spaced, curved blades that are divided into increments of three full blades, three long splitter blades, and six short splitter blades. The impeller is completely shrouded and is fabricated from Inconel 718 (AMS 5563). The thrust balance system has a capability of 45,800 pounds at 100% thrust and a mixture ratio of 5. The housings consists of an inlet housing, center housing, and rear housing.



(U) Figure 9. Fuel Low-Speed Inducer



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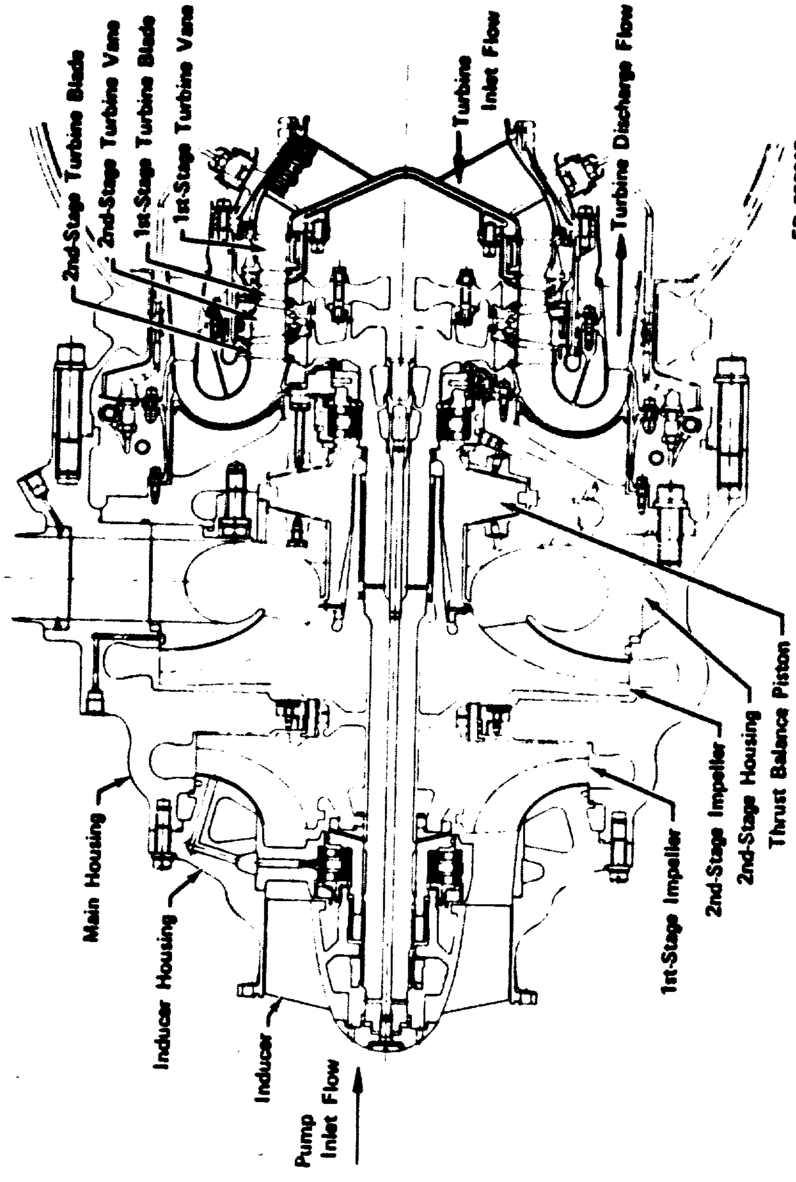
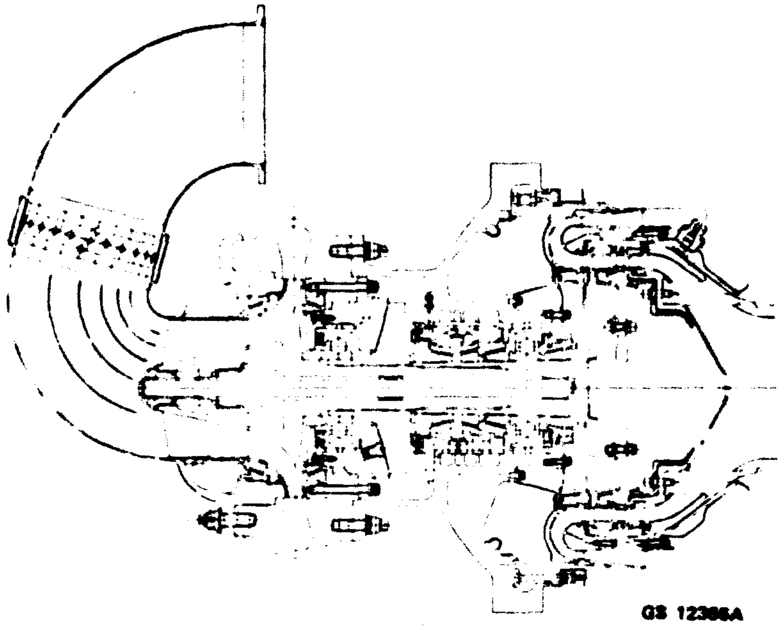
(U) Figure 10. Oxidizer Low-Speed Inducer



(U) Figure 11. Fuel Turbopump Assembly



(U) Figure 12. Oxidizer Turbopump Assembly

K. CONTROL SYSTEM

(U) A closed loop control system is required to ensure safe, precise, and responsive performance of the engine throughout its operating range. The planned system will accept vehicle or "man-in-the-loop" command signals at any rate or sequence, and provide rapid response within the functional and structural limits of the engine. The system will be stable at any setting and will respond smoothly to command.

(U) Four discrete electric current signals from the vehicle will accomplish engine starting, shutdown, and modulation of thrust and mixture ratio. The control signals may originate either in the vehicle guidance control or a pilot's command console in a manned vehicle. Response of the engine to these signals will be governed by an electronic Engine Command Unit (ECU). The demonstrator engine ECU will be a solid-state electronic component incorporating all flight engine control logic. The control valves, actuators, igniters, and plumbing will be lightweight, flight-type parts contained within the engine envelope. The closed loop control system will use flowmeters in both propellant lines to generate signals proportional to actual thrust and mixture ratio. These flowmeter signals will be compared to the vehicle input signals in the ECU and will automatically correct any difference between actual and desired values by modulating the engine propellant valves. An analysis of the XLR129-P-1 rocket engine cycle has established the following four control points are required for satisfactory steady-state operation: (1) preburner oxidizer valve, (2) preburner fuel valve (3) main-chamber oxidizer valve, and (4) oxidizer low-speed inducer, variable turbine, actuator. Several on-off sequenced valves are also used in the control system as follows: (1) nozzle-skirt coolant valve, (2) propellant vent valves, and

(3) helium system valves. The control system consists of a basic control computer that includes scheduled valve and oxidizer low-speed inducer turbine areas, with limited authority trim based on measured engine parameters. Various engine operating limits will be protected by override authority. Control capability protecting the engine is critical to a man-rated system. Within the demonstrator engine operating envelope, the propellant schedule in the control will prohibit operation beyond component limits. The principal control valves and subsystems have been designed, and are as follows: (1) preburner oxidizer valve, (2) preburner fuel valve, (3) main-chamber oxidizer valve, (4) two-position nozzle coolant supply system (5) propellant vent valves, and (6) helium supply system.

SECTION III
ENGINE SYSTEM DESCRIPTION AND PERFORMANCE

Page

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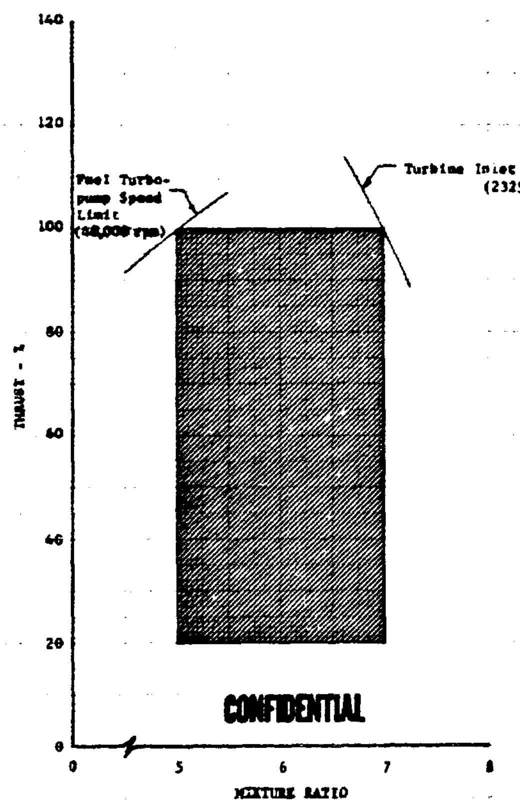
ENGINE SYSTEM DESCRIPTION AND PERFORMANCE

A. DESCRIPTION

(U) The staged-combustion, high-pressure demonstrator engine with a two-position bell-nozzle is a 250,000-lb thrust (class), throttleable, high-performance propulsion system. The operating envelope of thrust and mixture ratio is shown in figure 13 and engine characteristics are provided in table I. Nozzle interchangeability and the two-position nozzle concept permit operation of the same engine system with optimum nozzle area ratios for improving the performance of the lower or upper stages. This interchangeability is achieved by using the same turbomachinery power package and attaching the desired nozzle skirt for the various application requirements. A cutaway view of the engine is presented in figure 14. A propellant flow schematic illustrating the principal flowpaths is presented in figure 15.



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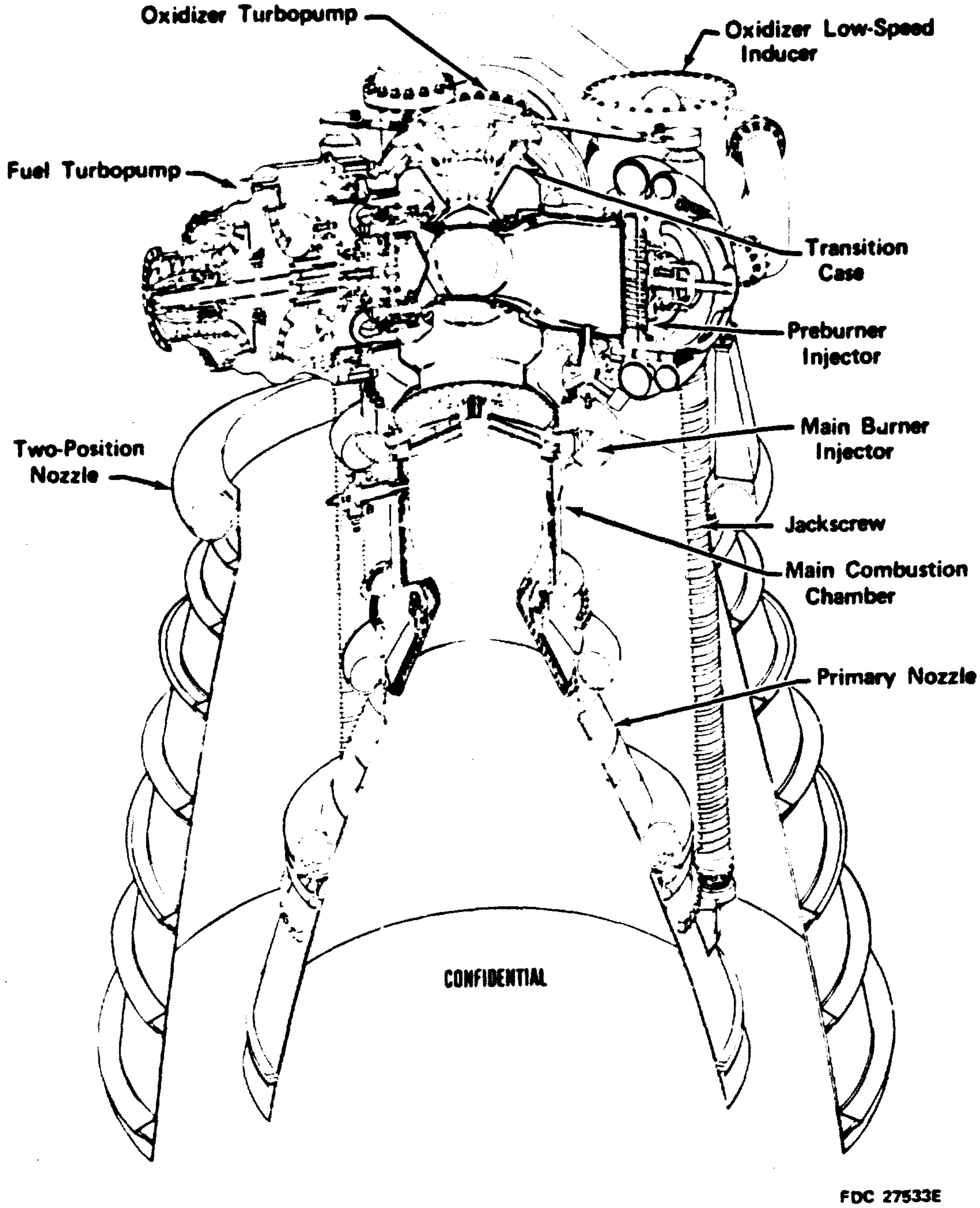
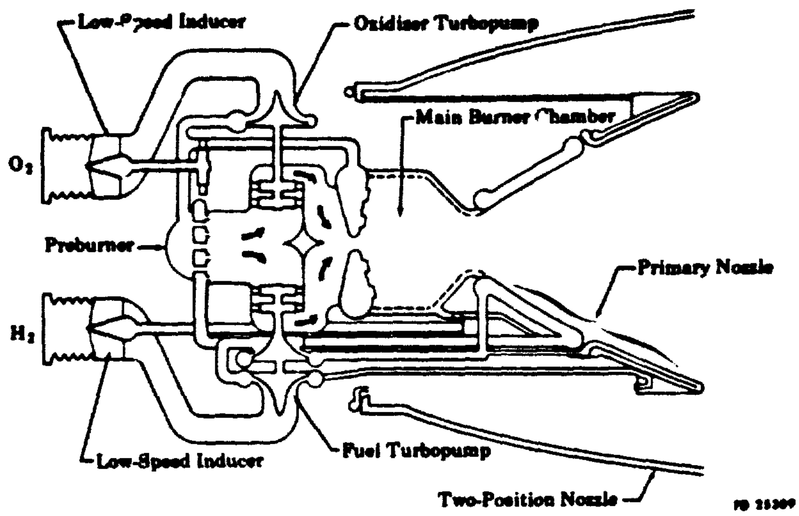
(U) Figure 13. Operating Range for Demonstrator Engine



(U) Figure 14. XLR129-P-1 Demonstrator Engine



(U) Figure 15. Demonstrator Engine Propellant Flow Schematic

(U) Hydrogen and oxygen enter at the engine-driven low-speed inducers. The low-speed inducers minimize vehicle tank pressure requirements while allowing high-speed main propellant pumps to obtain high turbopump efficiencies. The fuel low-speed inducer is a single shaft unit with a high specific speed, axial-flow inducer driven by a partial-admission, single-stage, hydrogen turbine. The oxidizer low-speed inducer is also a single shaft unit with a high specific speed, axial-flow inducer driven by a variable admission radial inflow single-stage liquid oxygen turbine.

(U) The main fuel turbopump is a single shaft unit with two back-to-back centrifugal pump stages driven by a two-stage, pressure-compounded turbine. A double-acting thrust balance piston is provided between the pump and turbine.

(U) The oxidizer turbopump is a single shaft unit with a single, centrifugal pump stage driven by a two-stage, pressure compounded turbine. A single-acting thrust balance piston is provided between the pump and turbine.

(U) The preburner injector consists of dual-orifice tangential-swirler oxidizer injection elements with concentric fixed-area fuel injection. A translating sleeve valve is incorporated at the rear of the injector assembly to vary the total oxidizer flow rate to adjust engine power level and to adjust the relative flow of the primary and secondary elements. The preburner combustion chamber is an integral part of the transition case, which contains the turbine drive gas ducts and a cooled outershell. The main turbopumps are mounted to the transition case with a plug-in arrangement of the turbines to provide maintainability.

(C) The main burner injector consists of a tangential-swirler oxidizer injection elements arranged in radial spraybars. The fuel side directs fuel-rich gas flow (preburner combustion products after expansion through the turbine) through slots in a porous faceplate. The combustion chamber wall is composed of a hydrogen-cooled liner extending from the injector face to an area ratio of 5.3. The liner is composed of porous plates providing transpiration cooling.

(U) The nozzle, which attaches at the end of the transpiration cooled section, is composed of two fixed regeneratively cooled sections and a retractable, low-pressure, dump-cooled section.

(U) The main fuel flow is pumped to system operating pressure levels by the main fuel pump and is ducted to cool the regeneratively cooled sections of the nozzle. The forward section is cooled with the majority of the fuel flow from the pump in a single pass heat exchanger. This flow exits from the nozzle and is ducted to the preburner. The regeneratively cooled rear section of the fixed nozzle is cooled with the remainder of the fuel flow in a two-pass heat exchanger. This flow is subsequently used as the working fluid to power the fuel low-speed inducer drive turbine and is then used to cool the porous main chamber walls.

(U) A small amount of fuel is ducted from the fuel pump interstage to cool the retractable nozzle skirt. This fuel is heated to high temperature in the skirt and expelled overboard through small nozzles at the ends of the coolant passages. A valve is provided to shut off the flow when the secondary nozzle is retracted.

(U) After being pumped to system operating pressure, the oxidizer is divided between the preburner and the main chamber. The smaller portion of the flow is supplied to the preburner and is burned with the fuel. The resulting combustion products provide the working fluid for the main turbines, which are arranged in parallel. The turbine exhaust gases are collected and directed to the main burner injector.

(U) The main burner oxidizer flow provides the oxidizer low-speed inducer turbine working fluid and uses the available pressure drop between the main oxidizer pump discharge pressure and the main chamber pressure for the turbine power. The oxidizer flow is then injected into the main burner chamber and is mixed and burned with the fuel-rich turbine exhaust gases. The resulting combustion gas is then expanded through the bell-nozzle.

(U) The primary engine control valves are located in the liquid oxygen supply lines to the preburner and the main chamber and in the liquid hydrogen supply line to the preburner.

B. OPERATING CHARACTERISTICS

1. Steady-State Operating Parameters

(C) The component and engine system steady-state operating parameters are given in table II, for mixture ratios of 5, 6, and 7 at 100%, 75, 50 and 20% thrust. (These operating parameters result from an iterative optimization process as described in paragraph G.)

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(U) Table II. Demonstrator Engine Operating Characteristics, Booster

Configuration	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	75% Thrust r = 5.0	75% Thrust r = 6.0	75% Thrust r = 7.0
Thrust, lb	244,000	244,000	244,000	183,000	183,000	183,000
Vacuum Specific Impulse, sec ϵ = 75	450	450	444	451	448	441
Sea Level Specific Impulse, sec ϵ = 35	387	386	380	370	367	364
Envelope:						
Diameter, in.	69.25	69.25	69.25	69.25	69.25	69.25
Length: Nozzle Extended/Retracted, in.	131.7/80.0	131.7/80.0	131.7/80	131.7/80.0	131.7/80.0	131.7/80.0
Nozzle Area Ratio: Extended/Retracted	75/35	75/35	75/35	75/35	75.35	75/35
Fuel Flow, lb/sec	90.3	77.5	68.8	67.7	58.4	51.8
Oxidizer Flow, lb/sec	451.5	465.2	481.4	338.4	350.1	362.8
Total Propellant Flow, lb/sec	541.9	542.8	550.1	406.1	408.5	414.6
Main Burner Chamber						
Throat Total Pressure, psia	2806	2740	2676	2101	2059	2002
Mixture Ratio (injector)	5.56	6.68	7.94	5.53	6.77	8.06
Specific Impulse Efficiency, %	96.7	97.0	96.9	96.9	96.8	96.7
Fuel Injector Pressure Loss, psi	164	134	125	97.7	89.7	86.3
Oxidizer Injector Pressure Loss, psi	851	910	979	496	532	572
Momentum Pressure Loss, psi	-1.6	0.8	-0.4	13.2	9.7	6.4
Transpiration Coolant Flow, lb/sec	6.42	5.36	5.49	4.24	4.34	4.50
Throat Diameter, in.	7.68	7.68	7.68	7.68	7.68	7.68
Preburner						
Total Pressure, psia	4778	4332	4152	3256	3100	3003
Mixture Ratio (preburner injector)	1.08	1.12	1.28	0.98	1.06	1.23
Temperature, °R	2026	2095	2345	1715	1984	2274
Fuel Injector Pressure Loss, psi	320.8	248.0	200.5	201.9	158	129
Oxidizer Injector and Control Valve Pressure Loss, psi	1141	944	599	1385	1065	966
Total Injector Propellant Flow, lb/sec	157.8	138.1	128.2	108.6	98.6	92.4
Combustion Efficiency, %	100	100	100	100	100	100
Primary Nozzle						
Transpiration Supply Section:						
Coolant Flow, lb/sec	7.75	6.44	6.59	5.06	5.17	5.36
Coolant Inlet Pressure, psia	5279	4723	4830	3595	3701	3836
Coolant Inlet Temperature, °R	142	133	139	114	120	129
Coolant Pressure Loss, psi	142	120	126	89	94	100
Coolant Temperature Rise, °R	266	338	341	336	350	349
Preburner Supply Section:						
Coolant Flow, lb/sec	76.5	65.5	56.7	57.7	48.4	41.7
Coolant Inlet Pressure, psia	5271	4712	4455	3561	3341	3199
Coolant Inlet Temperature, °R	142	133	142	114	121	131
Coolant Pressure Loss, psi	147	112	90.1	90	69	56.4
Coolant Temperature Rise, °R	35.7	44.2	48.5	40.7	48.8	53.4

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(U) Table II. Demonstrator Engine Operating Characteristics, Booster (Continued)

	50% Thrust r = 5.0	50% Thrust r = 6.0	50% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Configuration						
Thrust, lb	122,000	122,000	122,000	48,800	48,800	48,800
Vacuum Specific Impulse, sec $\epsilon = 75$	449	446	438	444	439	429
Sea Level Specific Impulse, sec $\epsilon = 35$	344	339	334	269	264	262
Envelope:						
Diameter, in.	69.25	69.25	69.25	69.25	69.25	69.25
Length: Nozzle Extended/Retracted, in.	131.7/80.0	131.7/80.0	131.7/80.0	131.7/80.0	131.7/80.0	131.7/80.0
Nozzle Area Ratio: Extended/Retracted	75/35	75/35	75/35	75/35	75/35	75/35
Fuel Flow, lb/sec	45.2	39.1	34.8	18.3	15.9	14.2
Oxidizer Flow, lb/sec	226.2	234.7	243.7	91.5	95.3	99.6
Total Propellant Flow, lb/sec	261.4	273.8	278.5	109.8	111.2	113.8
Main Burner Chamber						
Throat Total Pressure, psia	1396	1360	1329	552	537	526
Mixture Ratio (injector)	5.61	6.92	8.25	5.99	7.45	8.93
Specific Impulse Efficiency, %	96.8	96.6	96.4	96.3	96	95.9
Fuel Injector Pressure Loss, psi	543	53.5	52.8	17.4	18.0	18.3
Oxidizer Injector Pressure Loss, psi	230	246	265	13.1	42	45.6
Momentum Pressure Loss, psi	15.2	11.6	9.5	9.0	7.6	7.1
Transpiration Coolant Flow, lb/sec	2.95	3.23	3.36	1.45	1.57	1.62
Throat Diameter, in.	7.68	7.68	7.68	7.68	7.68	7.68
Preburner						
Total Injector Pressure, psia	2026	1970	1926	741	728	719
Mixture Ratio (preburner injector)	0.80	1.01	1.21	0.76	1.00	1.24
Temperature, °R	1548	1901	2223	1464	1883	2255
Fuel Injector Pressure Loss, psi	106	86	74	30.687	27.810	22.382
Oxidizer Injector and Control Valve Pressure Loss, psi	1093	1105	1102	583	652	677
Total Injector Propellant Flow, lb/sec	67.3	62.2	58.8	23.8	22.24	21.3
Combustion Efficiency, %	100	100	100	100	100	100
Primary Nozzle						
Transpiration Supply Section:						
Coolant Flow, lb/sec	3.43	3.81	3.95	1.66	1.80	1.86
Coolant Inlet Pressure, psia	2474	2668	2779	1156	1251	1302
Coolant Inlet Temperature, °R	93.8	105	116	75.6	84.8	92.6
Coolant Pressure Loss, psi	60.7	67.9	71.9	28.7	31.5	33.2
Coolant Temperature Rise, °R	375	365	365	392	387	392
Preburner Supply Section:						
Coolant Flow, lb/sec	37.7	31.3	26.9	13.7	11.2	9.6
Coolant Inlet Pressure, psia	2185	2100	2035	785	768	750
Coolant Inlet Temperature, °R	95.4	107	117	75.0	82.0	87.6
Coolant Pressure Loss, psi	44.8	35.8	30.0	10.9	9.6	7.8
Coolant Temperature Rise, °R	45.9	55.8	60.8	53.0	72.0	74.9

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(U) Table II. Demonstrator Engine Operating Characteristics, Booster (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	75% Thrust r = 5.0	75% Thrust r = 6.0	75% Thrust r = 7.0
Low-Speed Inducer						
Fuel Inducer:						
Flow Rate, lb/sec	90.31	77.5	68.8	67.67	58.4	51.8
Speed, rpm	19,823	18,146	17,699	16,150	15,742	15,618
Pressure Rise, psi	90.0	88.7	100.1	74.8	86.7	97.5
NPSH, ft	60.2	60.2	60.2	60.2	60.2	60.2
Efficiency, %	61.6	60.1	56.7	60.0	55.2	50.7
Oxidizer Inducer:						
Flow Rate, lb/sec	451.5	465.2	481	338.4	350.0	362.8
Speed, rpm	5417	4935	4904	4857	4659	4989
Pressure Rise, psi	258	197	186	254	224.2	257
NPSH, ft	16.0	16.0	16.0	16.0	16.0	16.0
Efficiency, %	57.2	60.7	61.4	52.9	55.8	54.1
Fuel Low-Speed Inducer Turbine						
Pressure Ratio	1.48	1.43	1.46	1.42	1.45	1.49
Flow Rate, lb/sec	5.55	4.62	4.73	3.64	3.72	3.87
Speed, rpm	19,823	18,146	17,699	16,150	15,742	15,618
Efficiency, %	63.1	60.3	58.8	57.4	55.1	53.7
Oxidizer Low-Speed Inducer Turbine						
Pressure Drop, psi	772	523.0	478	940	727.6	875
Flow Rate, lb/sec	368	391	408	286	298	310
Speed, rpm	5417	4935	4904	4857	4659	4989
Efficiency, %	68.5	72.7	73.8	52.9	63.7	62.5
Preburner Fuel Valve						
Flow, lb/sec	77	66	57	58.0	49.0	42.5
Pressure Drop, psi	270	200	550	162	475	745
Effective Area, in^2	3.48	3.48	1.83	3.45	1.69	1.20
Preburner Oxidizer Valve						
Flow, lb/sec	75	66	68	43	43	44
Pressure Drop, psi	670	580	230	1220	900	795
Effective Area, in^2	0.54	0.49	0.82	0.215	0.260	0.286
Main Chamber Oxidizer Valve						
Flow, lb/sec	370	390	410	283	300	310
Pressure Drop, psi	630	1120	1520	520	847	1100
Effective Area, in^2	1.74	2.13	2.96	1.58	1.86	2.49

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(U) Table II. Demonstrator Engine Operating Characteristics, Booster (Continued)

	50% Thrust r = 5.0	50% Thrust r = 6.0	50% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Low-Speed Inducer						
Fuel Inducer:						
Flow Rate, lb/sec	45.2	39.1	34.8	18.3	15.9	14.2
Speed, rpm	12,473	12,630	12,663	7748	7978	8043
Pressure Rise, psi	58.0	70.3	77.5	32.3	37.2	39.5
NPSH, ft	60.2	60.2	60.2	60.2	60.2	60.2
Efficiency, %	54.6	47.8	42.9	35	30.2	26.9
Oxidizer Inducer:						
Flow Rate, lb/sec	226.2	234.7	243.7	91.5	95.3	99.6
Speed, rpm	3745	3906	4115	1971	2077	2163
Pressure Rise, psi	176	192	213	54	60	65
NPSH, ft	16.0	16.0	16.0	16.0	16.0	16.0
Efficiency, %	48.9	48.8	48.2	36.0	36.1	36.4
Fuel Low-Speed Inducer Turbine						
Pressure Ratio	1.44	1.49	1.52	1.51	1.55	1.57
Flow Rate, lb/sec	2.51	2.75	2.86	1.20	1.30	1.35
Speed, rpm	12,473	12,630	12,663	7748	7978	8043
Efficiency, %	50.3	48.7	47.5	34.5	34.4	33.9
Oxidizer Low-Speed Inducer Turbine						
Pressure Drop, psi	835	923.2	1062	573	650.1	695.4
Flow Rate, lb/sec	195	202	210	80.6	83.5	87.1
Speed, rpm	3745	3906	4115	1971	2077	2163
Efficiency, %	48.6	48.2	47.3	28.8	28.3	28.5
Preburner Fuel Valve						
Flow, lb/sec	38.5	32.0	27.5	14	12	10
Pressure Drop, psi	348	628	808	380	500	570
Effective Area, in^2	1.55	0.98	0.76	0.56	0.43	0.35
Preburner Oxidizer Valve						
Flow, lb/sec	23	24	25	5	5.5	6
Pressure Drop, psi	1020	1040	1040	580	640	680
Effective Area, in^2	0.130	0.140	0.140	0.04	0.04	0.04
Main Chamber Oxidizer Valve						
Flow, lb/sec	194	202	210	80	85	90
Pressure Drop, psi	360	530	640	120	140	150
Effective Area, in^2	1.42	1.60	2.02	1.22	1.29	1.46

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(U) Table II. Demonstrator Engine Operating Characteristics, Booster (Continued)

	100% Thrust r = 5.0	100% Thrust r = 6.0	100% Thrust r = 7.0	75% Thrust r = 5.0	75% Thrust r = 6.0	75% Thrust r = 7.0
Two-Position Nozzle						
Coolant Flow, lb/sec	2.33	2.24	2.24	2.02	2.02	1.99
Thrust, lb	901	905	922	760	783	792
Fuel Turbopump						
Pump:						
Number of Pump Stages	2	2	2	2	2	2
Speed, rpm	48,043	44,548	44,490	38,783	38,724	39,404
Pressure Rise, psi	5493	4845	4915	3686	3723	3841
Overall Efficiency, %	65.9	65.3	63.7	650	63.1	60.6
Impeller Tip Velocity, 1st Stage, ft/sec	2226	2064	2061	1797	1794	1826
Impeller Tip Velocity, 2nd Stage, ft/sec	2641	2449	2446	2132	2129	2166
Temperature Rise, °R	91.3	83.1	88.6	66.1	71.9	80.5
Inlet Flow, lb/sec	91.3	78.5	69.7	67.7	58.4	51.8
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.59	1.49	1.47	1.46	1.43	1.42
Inlet Temperature, °R	2011	2079	2326	1702	1967	2252
Inlet Pressure, psia	4721	4283	4106	3220	3066	2971
Temperature Drop, °R	177	157.0	165	124	131.7	144
Mean Wheel Velocity, ft/sec	1488	1380	1378	1201	1200	1221
Efficiency, %	78.1	77.9	77.8	76.5	76.3	76.2
Inlet Flow, lb/sec	110.6	96.6	89.5	76.1	68.9	64.5
Oxidizer Turbopump						
Pump:						
Number of Stages	1	1	1	1	1	1
Speed, rpm	25,727	23,399	22,612	20,839	19,972	19,595
Pressure Rise, psi	5732	5139	4628	4397	3952	3726
Efficiency, %	55.6	65.5	66	61.8	63.0	63.8
Impeller Tip Velocity, ft/sec	952	866	837	771	739	725
Temperature Rise, °R	40.1	31.7	28.1	29.2	25.3	23.4
Inlet Flow, lb/sec	619.4	545.9	558.2	413.0	421.2	432.1
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.59	1.49	1.46	1.45	1.42	1.42
Inlet Flow, lb/sec	48.1	42.3	39.4	33.2	30.3	28.5
Inlet Temperature, °R	2011	2079	2326	1702	1967	2252
Inlet Pressure, psia	4730	4290	4113	3226	3071	2975
Temperature Drop, °R	156.1	137	142.0	109	114	122
Mean Wheel Velocity, ft/sec	1123	1021	987	912	871	855
Efficiency, %	69.4	68.5	67.6	67.5	66.3	65.2

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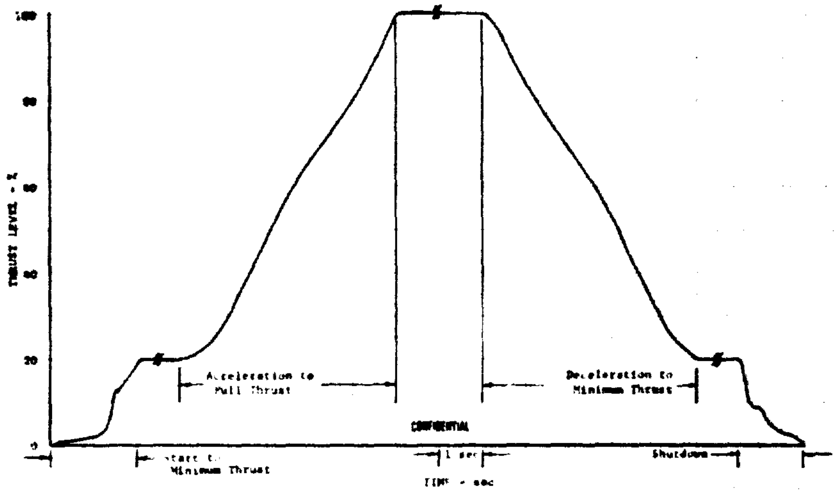
(U) Table II. Demonstrator Engine Operating Characteristics, Booster (Concluded)

	50% Thrust r = 5.0	50% Thrust r = 6.0	50% Thrust r = 7.0	20% Thrust r = 5.0	20% Thrust r = 6.0	20% Thrust r = 7.0
Two-Position Nozzle						
Coolant Flow, lb/sec	1.77	1.75	1.70	1.42	1.35	1.27
Thrust, lb	625	640	640	406	408	402
Fuel Turbopump						
Pump:						
Number of Pump Stages	2	2	2	2	2	2
Speed, rpm	31,195	32,474	33,591	20,843	22,262	23,357
Pressure Rise, psi	2450	2630.6	2736	1092	1183	1233
Overall Efficiency, %	61.8	58.2	55.3	49	44.9	42.0
Impeller Tip Velocity, 1st Stage, ft/sec	1445	1505	1556	966	1032	1082
Impeller Tip Velocity, 2nd Stage, ft/sec	1715	1785	1847	1146	1224	1284
Temperature Rise, °R	47.4	58.2	68.1	28.6	37.1	44.3
Inlet Flow, lb/sec	45.2	39.1	34.8	18.8	16.4	14.7
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.37	1.37	1.37	1.27	1.28	1.29
Inlet Temperature, °R	1534	1881	2197	1442	1850	2211
Inlet Pressure, psia	2004	1949	1905	734	721	712
Temperature Drop, °R	92.2	108.5	123	60.2	76.3	89.9
Mean Wheel Velocity, ft/sec	966	1006	1041	646	690	724
Efficiency, %	74.0	73.6	73.6	66.4	65.7	65.6
Inlet Flow, lb/sec	47.2	43.6	41.1	16.8	15.7	15.0
Oxidizer Turbopump						
Pump:						
Number of Stages	1	1	1	1	1	1
Speed, rpm	16,576	16,476	16,365	10,431	10,650	10,712
Pressure Rise, psi	2921	2862	2796	1229	1279	1291
Efficiency, %	58.5	59.4	60.3	44.9	45.5	46.5
Impeller Tip Velocity, ft/sec	613	610	605	385	394	396
Temperature Rise, °R	20.1	19.3	18.5	11.0	11.3	11.2
Inlet Flow, lb/sec	287.7	295.7	304.1	132.5	137.0	141.5
Turbine:						
Number of Stages	2	2	2	2	2	2
Pressure Ratio	1.37	1.36	1.37	1.27	1.3	1.28
Inlet Flow, lb/sec	20.7	19.2	18.2	7.4	6.9	6.66
Inlet Temperature, °R	1534	1881	2197	1442	1850	2211
Inlet Pressure, psia	2008	1952	1908	735	722	713
Temperature Drop, °R	79.8	91.8	102	49.4	61.1	70.0
Mean Wheel Velocity, ft/sec	723	719	714	455	465	467
Efficiency, %	64.6	62.8	61.6	55.3	53.5	52.1

2. Start, Shutdown, and Throttle Transients

(U) Estimated start, shutdown, and throttle transient data are presented in figure 16.



DFC 75786

(U) Figure 16. Demonstrator Engine Estimated Start, Shutdown, and Throttle Transient Data

C. LAYOUT AND SCHEMATIC

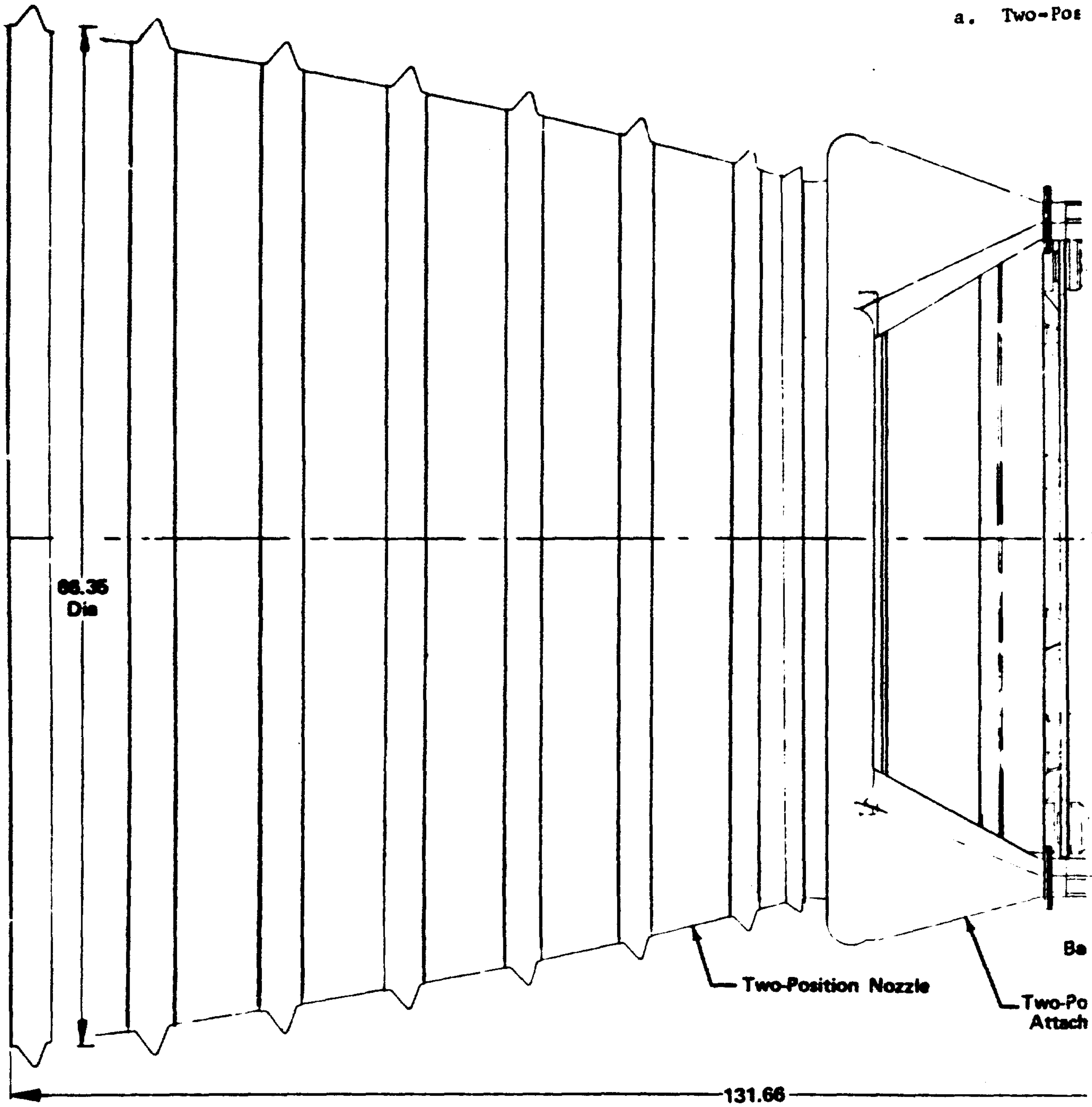
(U) The engine layout is illustrated in figure 17. The complete engine system schematic shown in figure 18 illustrates the helium supply system and the primary propellant flow paths and the interrelationship of all of the major components.

D. WEIGHT

(U) The estimated demonstrator engine weight based on lightweight rather than flightweight component designs is presented in table III. The targeted demonstrator engine weight is 3380 pounds. Component weights are discussed in more detail in the component sections of this report.

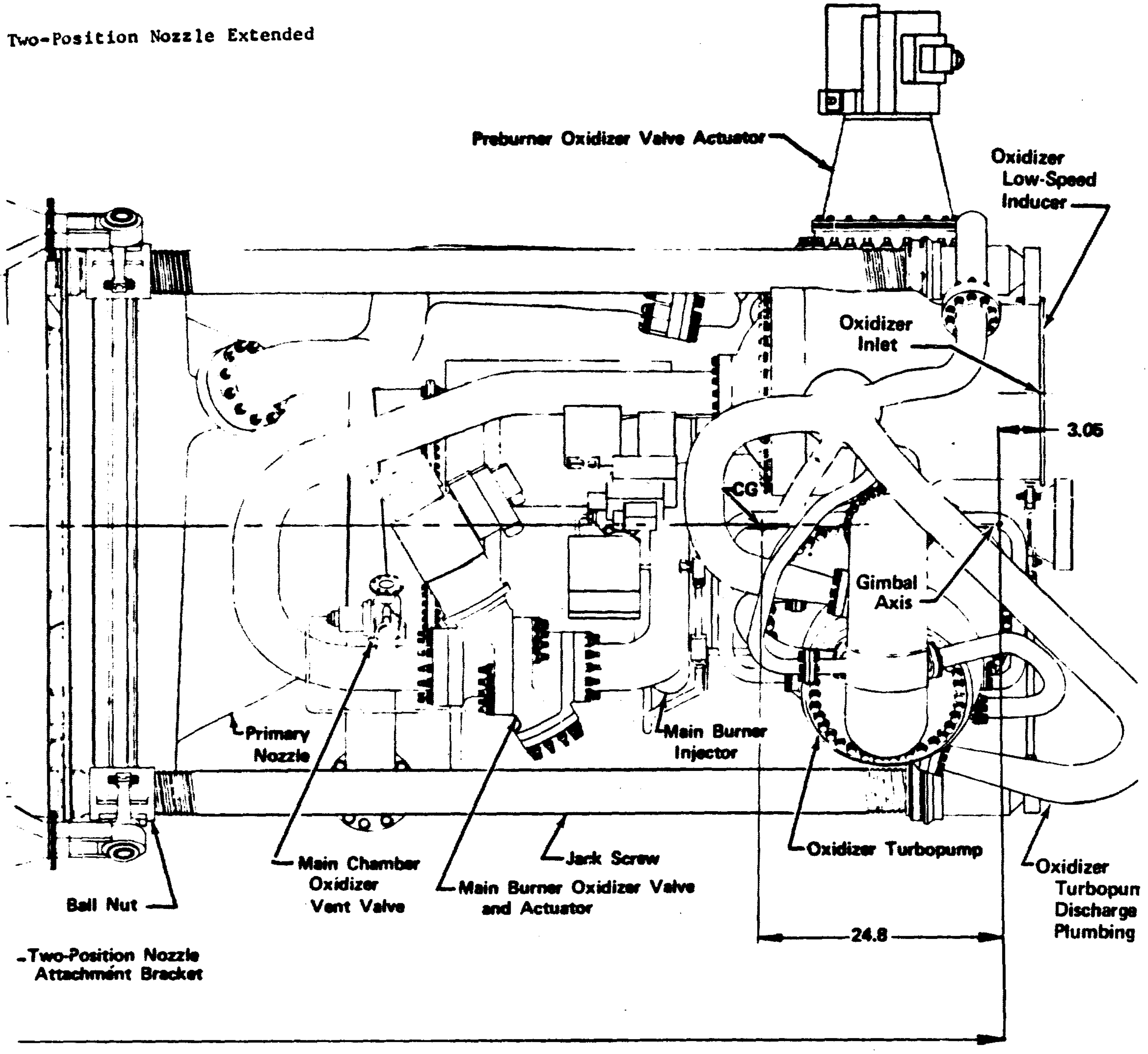
E. INTERFACES

(U) The ranges of temperature, pressure, and NPSH conditions required at the inlet to the fuel and oxidizer low-speed inducers are shown in figures 19 and 20.



(U) Figure 17. Demonstrator Engine Layout

Two-Position Nozzle Extended



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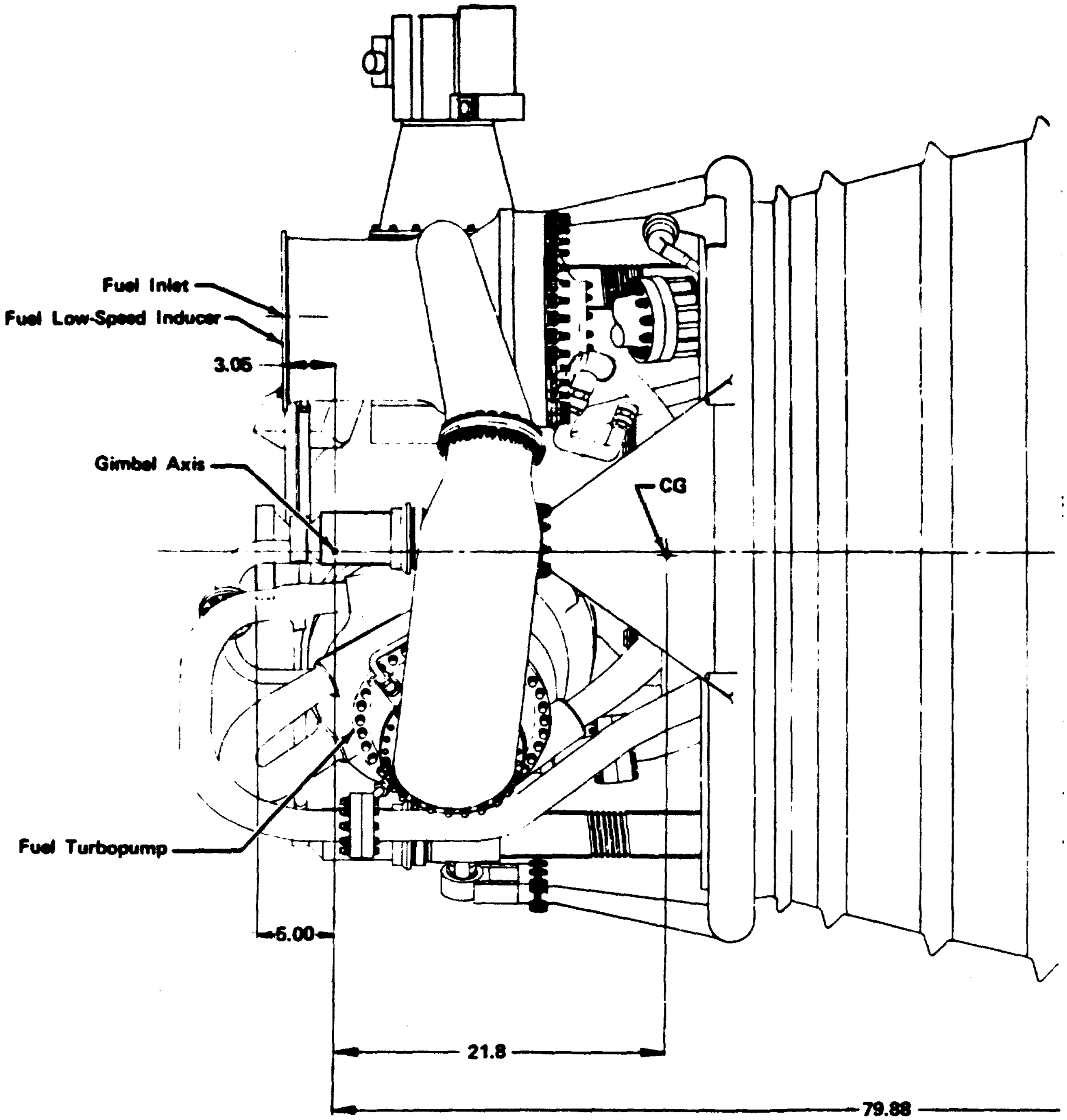
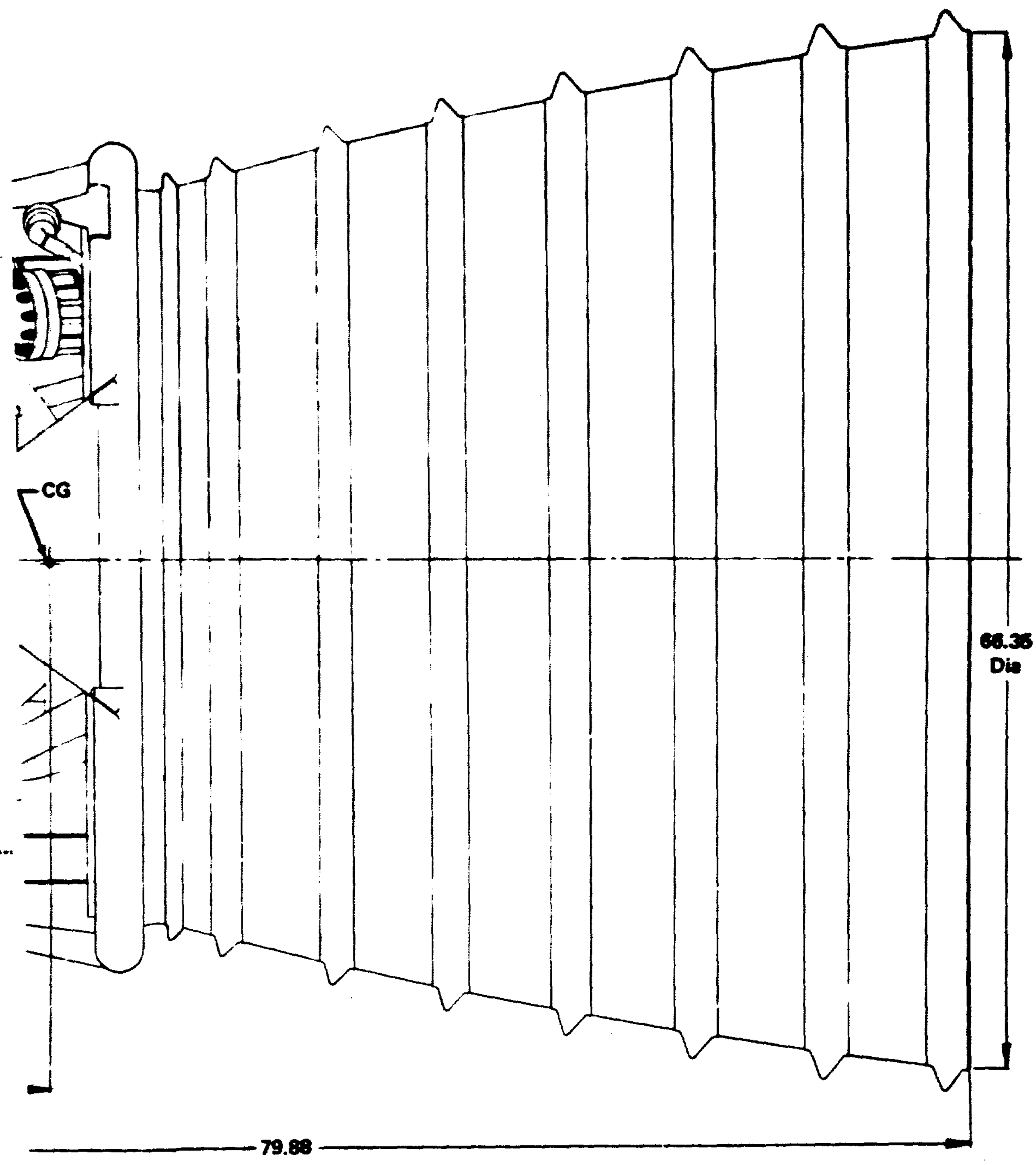
2

b. Two-Position Nozzle Stowed

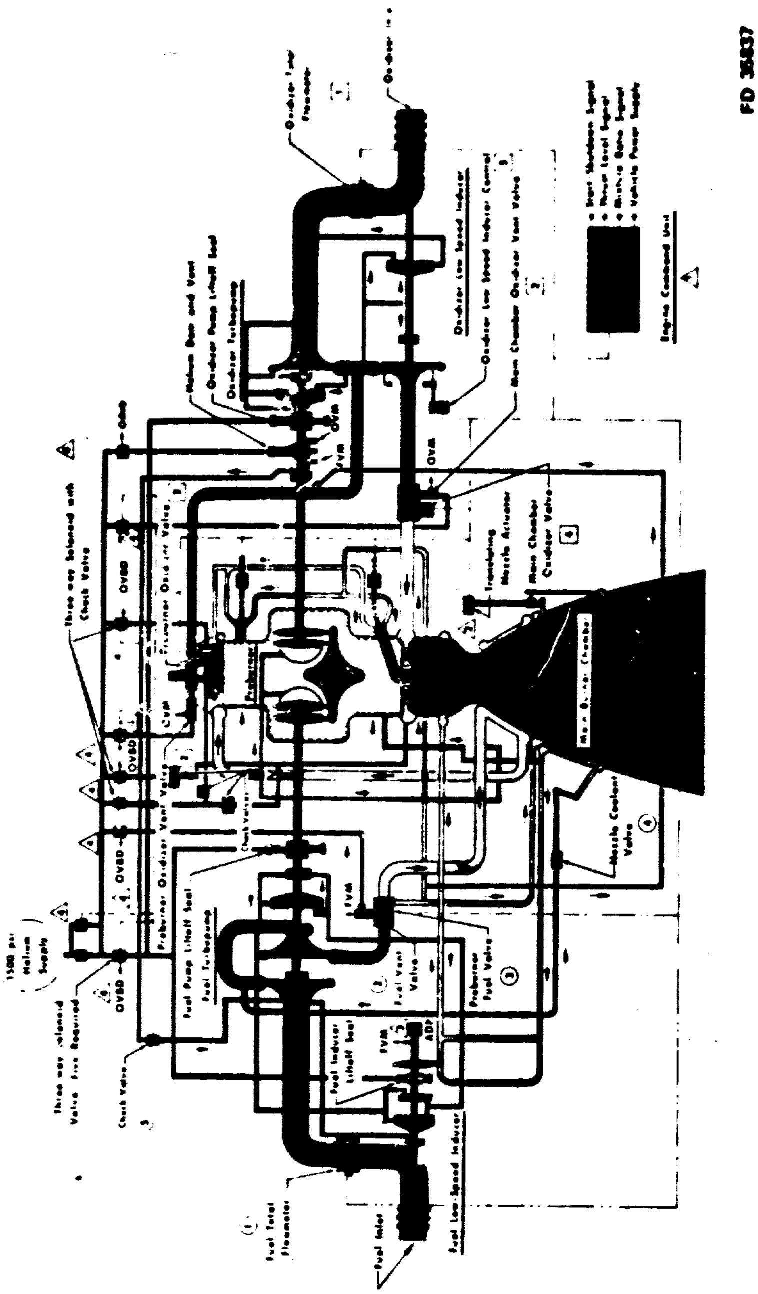


(U) Figure 17 --- Concluded

•. Two-Position Nozzle Stowed



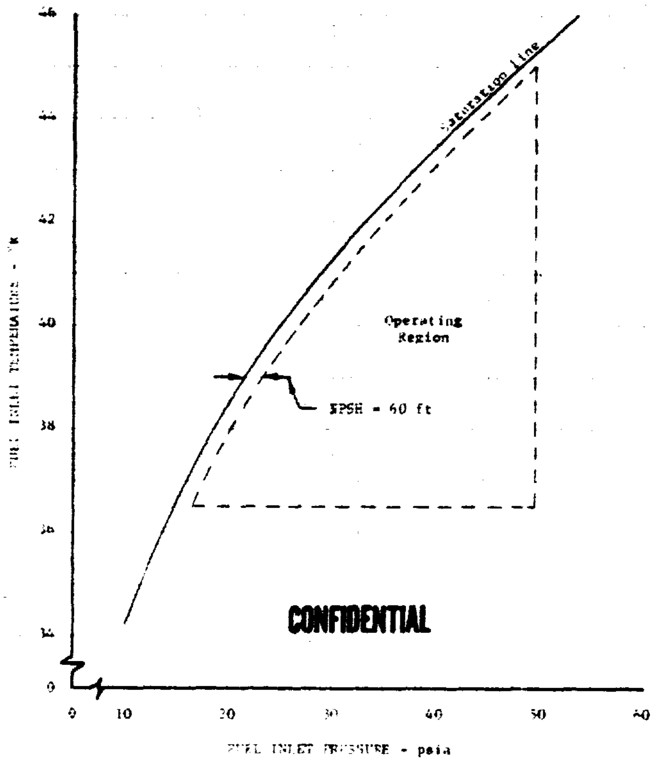
2



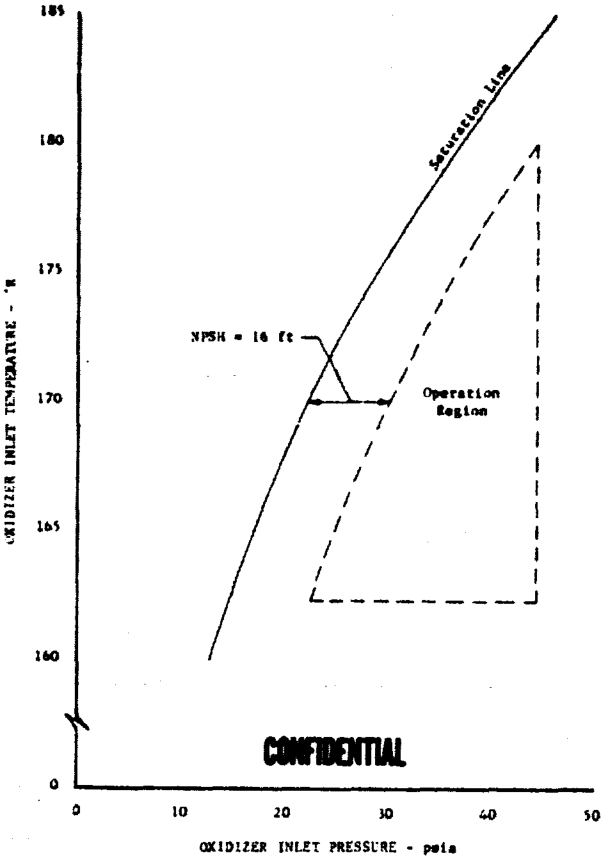
(U) Figure 18. Demonstrator Engine Complete Propellant Flow Schematic

(U) Table III. Demonstrator Engine Estimated Weight

Item	Estimated Weight, lb	Targeted Weight, lb
Preburner and Hardware	92	90
Transition Case and Gimbal	324	370
Main Burner Injector and Hardware	99	115
Main Burner Chamber	410	425
Nozzle and Actuation	652	640
Fuel Turbopump	554	480
Oxidizer Turbopump	383	335
Low-Speed Inducers	348	235
Controls	230	305
Plumbing	290	310
Miscellaneous	50	75
Total	3432	3380



(U) Figure 19. Fuel Inlet Operating Region



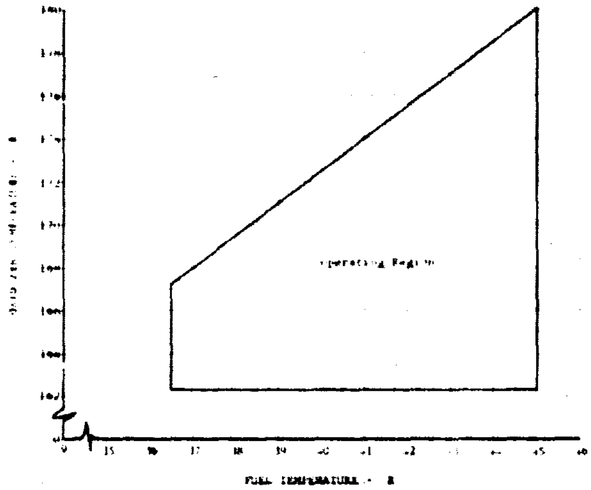
DFC 69933

(U) Figure 20. Oxidizer Inlet Operating Region

(U) A study selected the inlet propellant temperatures used for the design cycle analysis of the engine. Engine power requirements were found to vary significantly with engine inlet temperature. The required turbine inlet temperature varied approximately 96°R, and the fuel pump speed varied approximately 3000 rpm over the full range of inlet temperature specified for the demonstrator engine.

(U) The highest proposed inlet conditions were selected for component design to assure the engine power requirements can be met under the most severe operating conditions.

(C) The relationship required between fuel temperature and oxidizer temperature, so that the maximum turbine inlet temperature (2325°R) will not be exceeded at 100% thrust is shown on figure 21.



DFC 70473

(U) Figure 21. Propellant Temperature Limits for Fuel Trim Capability

(U) The engine/vehicle main structural interface is a ring flange on the thrust ball cone with a 6.0-in. diameter bolt circle. Eighteen equally spaced 0.2493-in. diameter holes are provided for bolt attachment of the engine to the vehicle. (Refer to Section IV, Paragraph B for a description of the gimbal thrust ball.)

(U) Gimbal acutators for control of pitch and yaw rate are attached by 0.5625-in. thread, UNF-3A, 12-point shoulder bolts to two gimbal actuator brackets on the main burner chamber pressure shell.

F. DESIGN CRITERIA

(U) Presented in Appendix I are the structural design criteria and limits for the major components of the XLR129-P-1 rocket engine.

G. SYSTEMS ANALYSIS

1. General

(U) System analyses of the demonstrator engine are being conducted throughout the program to define component design requirements, to estimate the capabilities of the integrated engine system, and to incorporate the results of component and engine tests.

(U) An initial analytical study was conducted to define those component design criteria that meet the design requirements of engine performance (as shown in table II) within the integrated engine system. These design data were derived by steady-state and transient analyses over the complete engine operating range using digital computer mathematical models.

The steady-state analysis consists of studies that establish a cycle balance between design limitations and component performance. The transient studies define the engine design requirements based on dynamic requirements and operating sequences for engine start, throttling, propellant utilization, and shutdown.

(U) As component and engine test data became available, the steady-state and transient analyses have been updated as required to provide the design data necessary to improve the component and module designs.

(U) The system design resulted from an iterative optimization process between mechanical and anayltical studies, and component and engine test data. Using a digital computer, a cycle balance program defines component design point data and determines engine performance for the design and off-design point operating conditions. These design data are used in the completion of design layouts. Specific component limitations defined in the process of designing the individual components are again input into the cycle balance and the iteration is continued until an optimum design is established.

2. Initial System Analysis

a. Analysis Method

(U) A balance was established between component thermodynamic performance, mechanical design requirements, and engine operating requirements. This balance was established by using an optimization procedure in which component geometry and performance are varied to maximize mechanical design margin while meeting the engine operational goals. Table IV summarizes the various inputs, engineering considerations, and results of this process.

b. Analysis Criteria

(U) The engine characteristics presented in table I were used for the systems analysis of the demonstrator engine. They represent the targets toward which the demonstrator engine program are being directed. Technology limits used for the system analysis were set at the state-of-the-art level per data obtained in subscale and full-scale combustion testing under Air Force contract AF04(611)-11401 and turbopump testing under NASA contracts NAS8-20540 and NAS8-11714. Because an accurate estimate of the anticipated component performance was known prior to the design analysis, the engine was designed with confidence that the structural margins and performance levels will be sufficient to allow maximum flexibility of engine operation.

c. Steady-State Cycle Optimization

(U) The cycle balance program used for steady-state cycle optimization is configured to afford flexibility in the integration of component requirements over the thrust and mixture ratio ranges. Features include: (1) a means of selecting the optimum component design points, (2) a procedure for optimizing the turbine area match for required engine operating range, and (3) a method of evaluating component off-design performance effects on overall cycle performance.

(U) Table IV. Cycle Definition Procedure

Component	Inputs or Specifications	Engineering Considerations	Results
Combustion Chamber and Nozzles	Nominal vacuum thrust	Component performance available	Structural Dimensions - chamber size, tube diameters, injector areas, etc.
	Chamber pressure (target value)	Mechanical design limits	
	Limiting engine dimensions	Weight tradeoffs	Nozzle expansion ratio
	Minimum specific impulse efficiency	Cooling requirements	Operating Performance - thrust, specific impulse, propellant flow rates, cooling flows, etc.
	Nominal mixture ratio	Component intergration	
	Durability requirements		Operating limits
	Environmental conditions		Weight
Turbopump Power Package	Pressure requirements	Component performance available	Structural Dimensions - pump and turbine diameters, injector areas, bearing sizes, etc.
	Propellant flow rates	Mechanical design limits	
	Thrust and mixture ratio range	Weight tradeoffs	Operating Performance - preburner temperature, pump pressures, speeds, efficiencies, coolant flow rates, NPSH, etc.
	Environmental and interface conditions	Design and off-design characteristics of engine cycle	
		Component integration	Operating limits
			Weight

(U) Table IV. Cycle Definition Procedure (Concluded)

Component	Inputs or Specifications	Engineering Considerations	Results
Control System	Engine operating modes	Design and off-design characteristics of engine cycle	Control point locations
	Operating Limits		Structural dimensions
	Engine thrust and mixture ratio accuracy	System pressure drops	Valve functions
		Mechanical design limits - area turndown, valve accuracy, response, etc.	Valve sequences
	Environmental and interface conditions		Valve area schedules
			Weight

(U) The initial step of the cycle balance program defines the chamber and nozzle geometry necessary to provide the required thrust at the nominal mixture ratio within the allowable design limitations. The main chamber combustion and nozzle efficiencies were maintained at a level consistent with the goals of the demonstrator engine program. The chamber pressure was fixed at the maximum level consistent with turbopump design limitations.

(C) The chamber geometry is established to provide 244,000-lb thrust for a booster stage vehicle application with an area ratio of 75:1.

(U) The nozzle coolant flow rates and passage sizes were varied until a balance was achieved between coolant pressure loss, nozzle weight, and coolant flow rate. The coolant pressure loss in the regenerative nozzles is important because it adds directly to fuel pump pressure. The nozzle skirt and transpiration coolant flow rates are important because these flows are not available for providing turbopump power. These flows also bypass the main injector, which increases the chamber mixture ratio and tends to decrease the overall specific impulse efficiency.

(U) The engine flow rates and pressures defined in the nozzle/chamber design calculations provide the basic data used to design the turbopump power package.

(U) The design approach taken in the cycle studies to obtain the required mixture ratio operating range was to use the extreme of the mixture ratio range as the power package design points.

(U) The fuel pressure requirement controls the power balance at the lowest mixture ratio, and the oxidizer pressure requirement controls the balance at the highest mixture ratio. In addition, the minimum available turbine power occurs at the highest mixture ratio where the fuel flow is at a minimum. At the extremes in mixture ratio, where one pump controls the power match, the other pump is in an "overspeed" condition. Overspeed means that the pressure provided exceeds the pressure required to satisfy the flow conditions.

(C) A reduction in efficiency results when a centrifugal pump is operated at flow rates and rotor speeds other than the pump design point. By selecting the design point of the main pumps at their respective maximum flow conditions, two advantages are obtained. First, the best efficiency point of the pump coincides with the engine operating point where the respective pump is controlling the power balance; second, the reduced efficiency at the low flow condition (i.e., the other extreme mixture ratio point) tends to minimize overspeed and to minimize the control system corrections required. Thus, the fuel pump is designed for a mixture ratio of 5.0 and rated thrust, and the oxidizer pump is designed for a mixture ratio of 7.0 and rated thrust. Use of this cycle optimization technique results in appreciably reduced pump pressure and speed requirements.

(U) The basic turbopump design variables other than efficiency, namely turbine areas, pump impeller diameters, speed, and turbine inlet temperature, are optimized through an iterative procedure. Turbine areas and pump diameters are varied to meet the cycle pressure requirements within rotor speed and turbine temperature limitations.

(C) The maximum turbine inlet temperature occurs at the maximum mixture ratio point, where the preburner fuel flow is at a minimum. The maximum allowable temperature is approximately 2325°R and is determined by the turbine stresses, which vary as functions of the turbine diameter, speed, and fluid bending forces.

(U) Variation in the total turbine area (fuel turbine area plus the oxidizer turbine area) affects the total power through pressure ratio, whereas the ratio of turbine areas (fuel turbine area/oxidizer turbine area) affects the division of turbine power. As these areas are changed, the pump head requirements vary, and the pump impeller diameters are then sized to provide the required pump pressures within allowable design limitations.

(U) At a particular value of total turbine area, the ratio of oxidizer turbine area to fuel turbine area is established to balance the turbopump power at the maximum allowable turbine inlet temperature (high mixture ratio). With this area ratio fixed, the cycle is rebalanced at the low mixture ratio extreme. Because of the turbopump power trends, the fuel and oxidizer turbopump speeds increase at the low mixture ratio. If the speeds are greater than allowable, the pump diameters are changed to hold the speed within the mechanical limitations defined by critical speed, turbine wheel speed, and bearing DN.

(U) Because a modification to either the fuel or oxidizer pump at low mixture ratio will affect its operating requirements at high mixture ratio, the turbine area ratio at high mixture ratio may need to be changed. This process (changing components at one mixture ratio and checking the effects at the other mixture ratio) was continued until an optimized cycle balance was obtained.

(U) In balancing the engine cycle, the components that maximized chamber pressures within the restraints of pump speed and pressure and turbine maximum temperature were selected.

(U) Combustion performance and injector characteristics are considered in conjunction with control pressure loss scheduling. In the preburner and main burner injector designs, the major performance considerations are the fluid velocities, the momentum exchange between the fuel and oxidizer, and the injector pressure potentials for combustion stability. In the control system designs, the major considerations are pressure drop for flow control potential and valve turndown ratio. Off-design performance characteristics are used to obtain basic injector and control design data to ensure that sufficient pressure drop to satisfy both the stability and control system pressure requirements is provided. At each balanced operating point the required injector and control pressure drop is maintained or exceeded.

(U) Design characteristics of the low-speed inducers were also determined within the cycle balancing effort. The inducer design discharge pressure must satisfy the main turbopump NPSH requirements and the estimated line losses between the low-speed inducers and pump.

(U) The low-speed inducer diameter and drive turbine speed were selected to provide the necessary inducer discharge pressure and turbine efficiency while remaining consistent with the available low-speed inducer inlet NPSH and inducer suction characteristics. The turbine areas were sized to provide sufficient energy to drive the inducers without increasing the maximum pressure of the main pumps.

d. Transient Analysis

(U) The transient characteristics of the XLR129-P-1 engine were investigated to identify any component design limits and to define the optimum control sequences to provide rapid, safe, and repeatable start throttling and shutdown transients. Variations in environmental conditions, such as inlet and ambient temperatures and pressures, were considered. Evaluation of the effects of engine component performance and control system on the system transients are an integral part of the design process.

(U) The basis for the control system used in the System Analysis was a detailed controls study performed under Contract NAS8-11427. This study evaluated several control systems and 17 control points for an advanced high pressure rocket engine system using a preburner cycle.

(1) Throttle Transient Analysis

(U) Transient analyses of the engine system within the extremes of main stage thrust and mixture ratio define the engine and control system dynamics, and define engine transient response and component protection required during rapid transients.

(C) Engine throttle transients were simulated using representative control systems that use engine parameters as input and/or feed back to the control areas. The preburner oxidizer valve was selected to provide closed-loop thrust control by using oxidizer and fuel flowmeter flows (summed) as a thrust indication. The ratio of the flowmeter flows was fed back to the main chamber oxidizer valve to provide closed-loop mixture ratio control. Accelerations and decelerations between idle and rated thrust in 2 seconds were simulated at engine mixture ratios of 5 and 7. Three second mixture ratio excursions between 5 and 7 were simulated at 100% thrust. Throttling transient analysis revealed no limitations that would require hardware or control mode changes from that established during steady-state analysis.

(2) Start and Shutdown Transient Analysis

(U) Similar mathematical models were used to simulate the start, throttling and shutdown modes of operation. Additional calculations in the start and shutdown simulations include: (1) the propellant filling processes, (2) fluid properties for phase transitions from gas to two-phase to liquid flow, (3) preburner and main chamber ignition, and (4) low-speed performance of the main turbopumps and low-speed inducers. Similar models have been used extensively during Phase I, Contract AF04(611)-11401, Module Design task and also in conjunction with the preburner and staged combustion test programs. A summary of the conclusions of these studies is presented below:

(a) Start Transient

(C) 1. The engine can be safely started to 20% thrust in approximately 2 seconds using a time-sequenced control method.

(U) 2. The orifice restriction in the primary flow path of the preburner oxidizer valve must be made smaller than that established during steady-state cycle analysis to reduce the preburner temperature spike when the primary cavity fills.

(U) 3. The use of helium purges in the secondary cavity of the preburner oxidizer injector and the main oxidizer injector is recommended to prevent back flow of the combustion product predicted to occur during the start transients.

(b) Shutdown Transient

(C) 1. A shutdown analysis showed that the engine can be shut down safely from 20% thrust at all mixture ratios by using a single time-based propellant valve sequence that schedules all shutoff valves to their fully closed position in a maximum of 1.5 sec.

(U) 2. The valve sequence can be modified to adjust the rate of preburner temperature decay during shutdown, if necessary for turbine stress and cycle life considerations.

(U) 3. The shutdown transient analysis revealed no limitations that would require hardware or control mode changes from that established during steady-state cycle analysis.

3. Special Design Cycle Studies

a. Design Point Trade Studies

(U) Trade studies were made to establish the sensitivity of engine characteristics to component performance levels to identify the critical component characteristics and minimize any undesirable effects. The trade factors presented in table V were determined by varying each parameter separately and rematching engine components to provide maximum chamber pressure within component limitations and cycle ground rules. The change in preburner temperature required to maintain a constant chamber pressure with variations in component performance was also calculated by using the trade factor for chamber pressure and turbine temperature defined in table V.

(U) The turbine area changes required to reoptimize (rematch) the cycle for full mixture ratio range at the indicated changes in chamber pressure were also established. For example, if main fuel turbine efficiency were increased 1 point, chamber pressure could be increased 12.6 psia over the full mixture ratio range provided the fuel turbine area was reduced by 0.25% and the main oxidizer turbine area was increased by 0.87%. The trade factors presented in table V may be assumed to be linear for small component variations.

b. Maximum Oxidizer Pump Discharge Pressure

(C) An analytical study was made to determine the optimum method of obtaining a maximum chamber pressure. Engine design cycles were established with components matched for maximum allowable oxidizer pump discharge pressures of 6500, 7000, and 7250 psia.

(C) The following major factors were noted:

1. The maximum excess thrust capability is available for an engine designed for a peak oxidizer pump discharge pressure of 7050 psia.

2. The maximum design chamber pressure increases with increasing maximum oxidizer pump discharge pressure.

3. As the peak oxidizer pump discharge pressure is decreased, the assumed fuel pump speed limit of 48,000 rpm is approached at 100%, r = 7.

4. With increasing oxidizer pump discharge pressure, the assumed preburner temperature limit of 2325°R is approached at 100%, r = 5.

5. Overall specific impulse at a mixture ratio of 5 decreases with increasing maximum oxidizer pump discharge pressure as a result of increased transpiration cooling flow.

(U) Table V. XLR129-P-1 Design Trades

Design Change	Magnitude of Change	Chamber Pressure psia	Preburner Temperature - °R P_c = Constant	Main Fuel Turbine Area, %	Main Oxidizer Turbine Area, %
1. Prebuner Temperature	+ 10°R at r = Max	+ 6.9	- - -	+ 0.48	+ 0.50
2. First Stage Fuel Pump Efficiency	+ 1 point	+ 18.0	- 26.0	- 0.37	+ 1.13
3. Second Stage Fuel Pump Efficiency	+ 1 point	+ 8.1	- 12.0	- 0.08	+ 0.76
4. Main Fuel Turbine Efficiency	+ 1 point	+ 12.6	- 18.0	- 0.24	+ 0.87
5. Fuel Side Pressure Loss (Pump discharge through preburner injector)	+ 10 psi at r = Min	- 3.3	+ 4.7	+ 0.36	+ 0.48
6. Fuel Turbine Upstream Housing Loss	+ 10 psi at r = Min	- 3.0	+ 4.3	+ 0.08	- 0.19
7. Fuel Turbine Downstream Housing Loss	+ 10 psi at r = Min	- 5.9	+ 8.5	+ 0.13	- 0.37
8. Main Oxidizer Pump Efficiency	+ 1 point	+ 8.3	- 12.0	+ 0.49	- 1.24
9. Main Oxidizer Turbine Efficiency	+ 1 point	+ 9.1	- 13.0	+ 0.57	- 1.00
10. Oxidizer Side Pressure Loss (Pump discharge through preburner injector)	+ 10 psi	- 3.0	+ 4.3	- 0.17	- 0.21

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(U) Table V. XLR129-P-1 Design Trades (Concluded)

Design	Magnitude of Change	Chamber Pressure, psia	Preburner Temperature - °R P_c = Constant	Main Fuel Turbine Area, %	Main Oxidizer Turbine Area, %
11. Maximum Oxidizer Pump Discharge Pressure	+ 100 psi at r = Min	+ 13.8	- 20.0	+ 0.76	+ 2.60
12. Maximum Fuel Pump Pressure Rise	+ 100 psi at r = Min	+ 21.0	- 30.0	- 3.0	- 5.3
13. Preburner Pressure Loss (Upstream of Turbine)	+ 10 psi	- 6.0	+ 8.6	Not Available	Not Available
14. Main Burner Hot Gas Duct Loss (Downstream of Turbine)	+ 10 psi	- 10.0	+ 14.0	Not Available	Not Available
15. Preburner/Turbine Bypass Flow	- 1.0 lb/sec	+ 22.0	- 32.0	+ 1.6	+ 1.6

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(C) Consideration of all the above factors indicates that an engine designed for a maximum oxidizer pump discharge pressure of 7050 psia would provide the optimum design. Although a slight loss in specific impulse would result at a mixture ratio of 5, no significant loss will be encountered at a mixture ratio of 6 and above, and the chamber pressure level attainable would be consistent with present design goals. An engine designed for higher oxidizer pump pressures could operate at slightly higher chamber pressure, but the specific impulse at mixture ratios of both 5 and 6 and excess thrust capability would be reduced.

c. Transistion Case Coolant Flow Source

(U) A design analysis of the transition case cooling passage indicated that structural problems may exist during the shutdown transient if the cooling is obtained from the preburner fuel valve discharge. This cycle analysis confirmed the acceptability of the alternative supply source located at the transpiration supply heat exchanger discharge.

(C) For the analysis, the transpiration supply heat exchanger configuration was maintained and its cooling flow rate was increased by the level of transition case cooling flow. The increased cooling flow at 100% thrust, mixture ratio of 7, did the following: (1) reduced the transpiration wafer inlet temperature 63°R, (2) reduced required wafer cooling flow 0.25 lb/sec, (3) increased specific impulse 0.3 second, and (4) decreased required preburn temperature 18.2°R.

(U) Rerouting of the flow reduced the maximum available fuel low-speed inducer turbine power 16% because of the increased turbine inlet line loss (higher flow) and decreased turbine temperature.

d. Two-Position Nozzle Flow Source

(U) This study investigated three engine locations for supplying the two-position nozzle coolant flow with and without a control valve. The locations investigated were: (1) the fuel pump interstage, (2) the fuel low-speed inducer discharge, and (3) the fuel preburner supply. For operation with the control valve in the system, the minimum coolant flow was scheduled into the nozzle. For operations without a control valve, an orifice was sized to provide the minimum coolant requirements at a critical engine operating point and allowed overcooling at all other operating conditions.

(U) The fuel pump interstage was chosen to supply cooling flow for the two-position nozzle because (1) acceptable nozzle cooling was provided without requiring a control valve, (2) the source was insensitive to variations in engine inlet conditions (the LSI tapoff was very sensitive to them), and (3) the slight penalty in chamber pressure and overall impulse efficiency caused by the overcooling characteristic inherent in the orifice configuration was acceptable.

(U) The engine characteristics with the three candidate locations are presented in figures 22 through 27.

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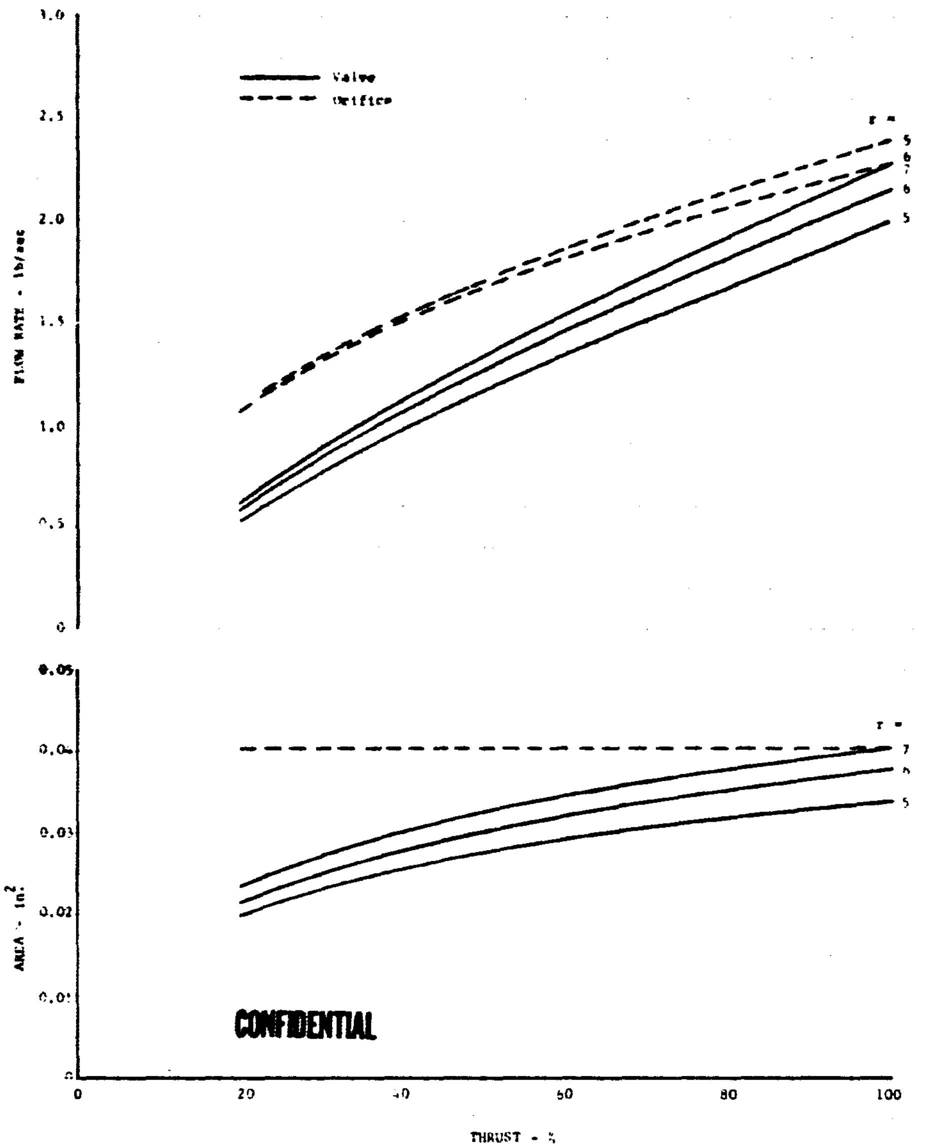

DFC 65466

(U) Figure 22. Effect of Pump Interstage Tap-Off Location on Lightweight Heat Exchanger Coolant Valve

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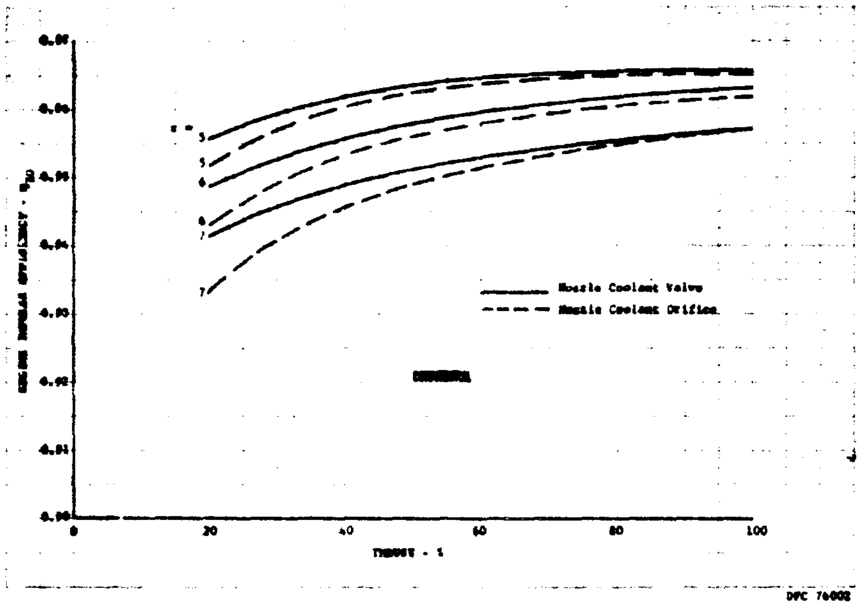
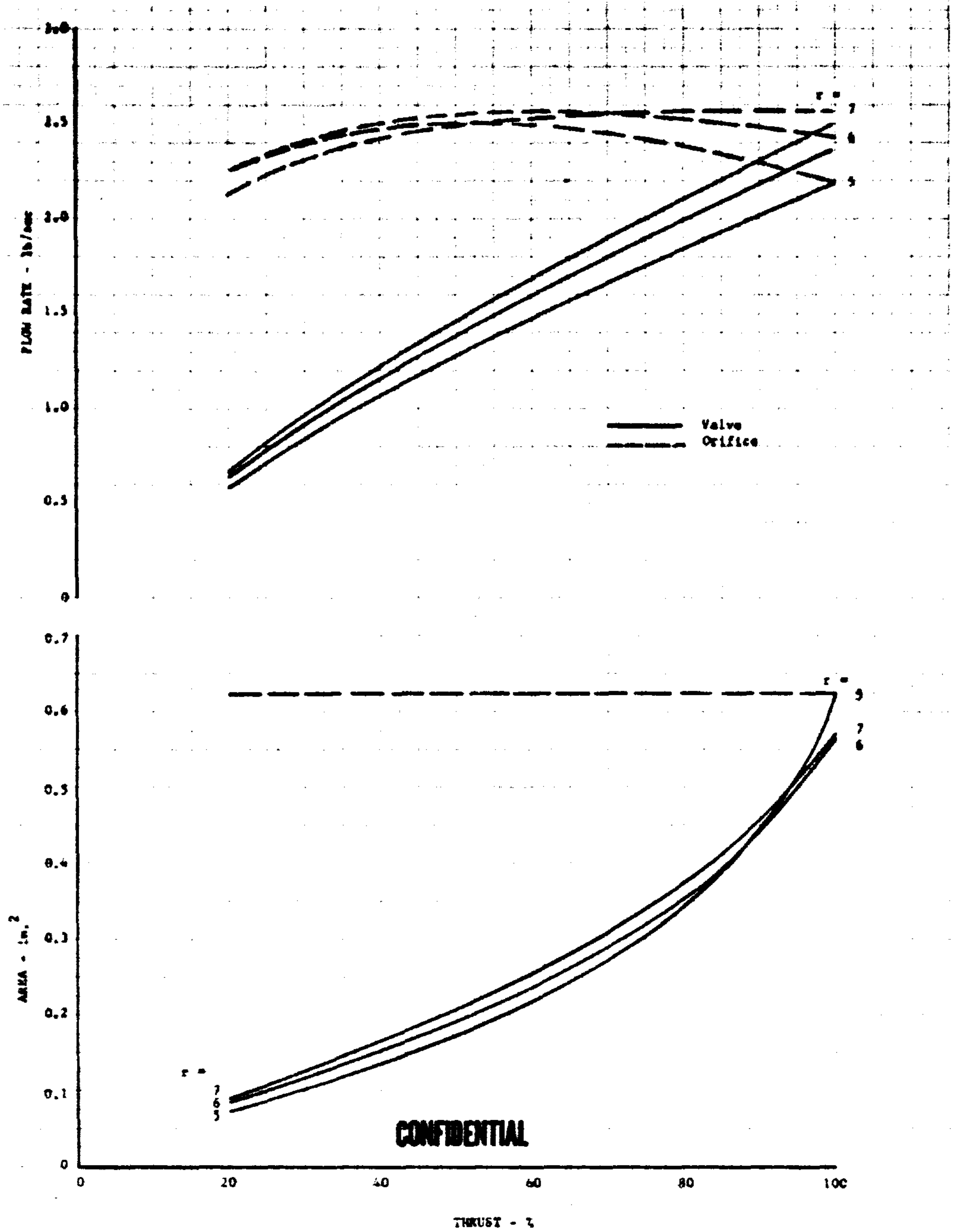


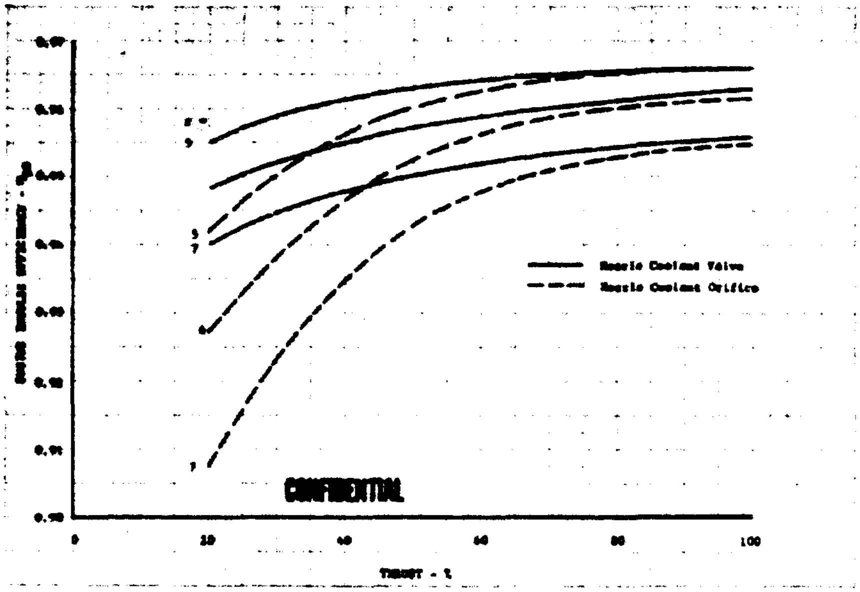
DFC 76002

(U) Figure 23. Effect of Pump Interstage Tap-Off Location on Engine Impulse Efficiency



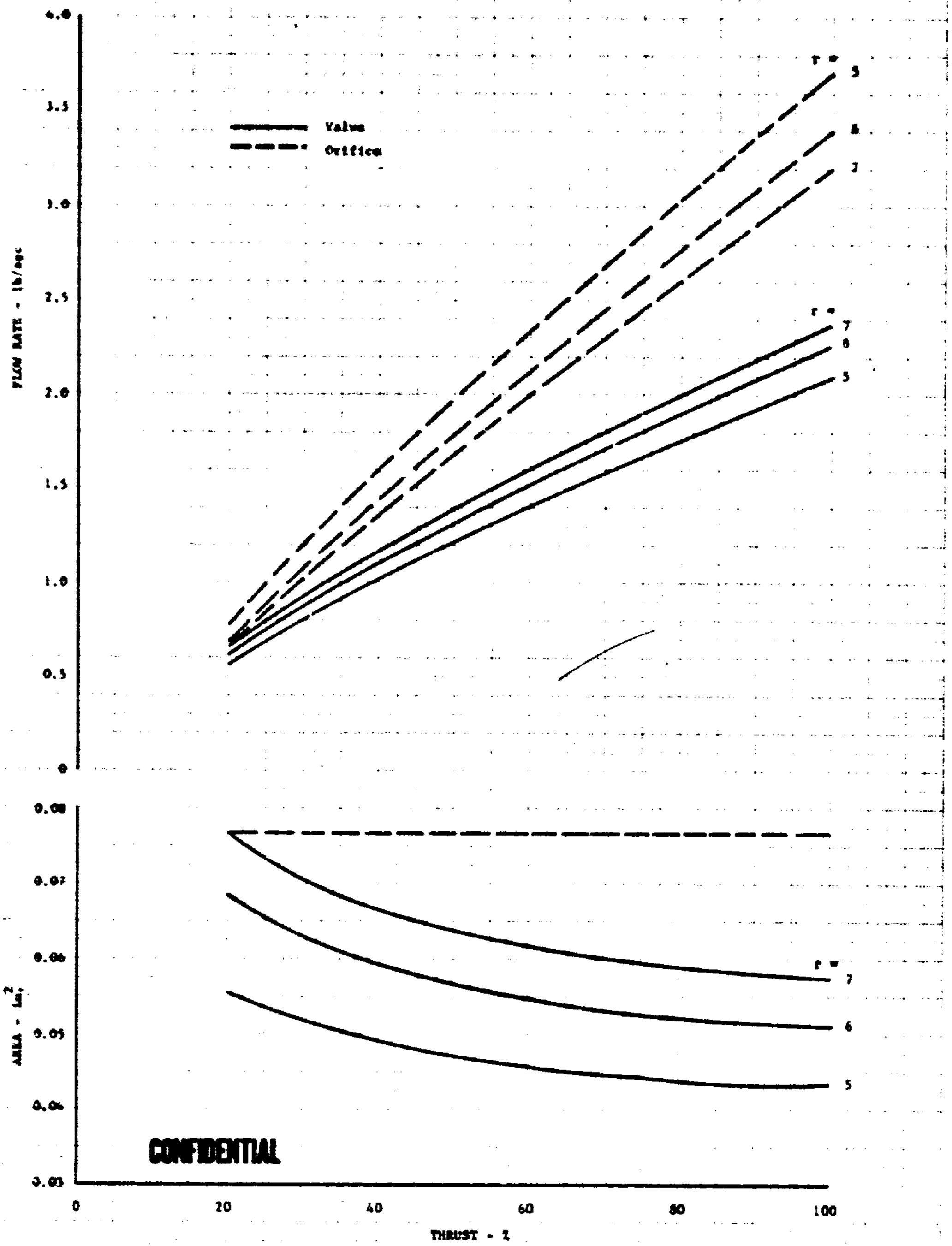
DF 76004

(U) Figure 24. Effect of Low-Speed Inducer Tap-Off Location on Lightweight Heat Exchanger Coolant Valve



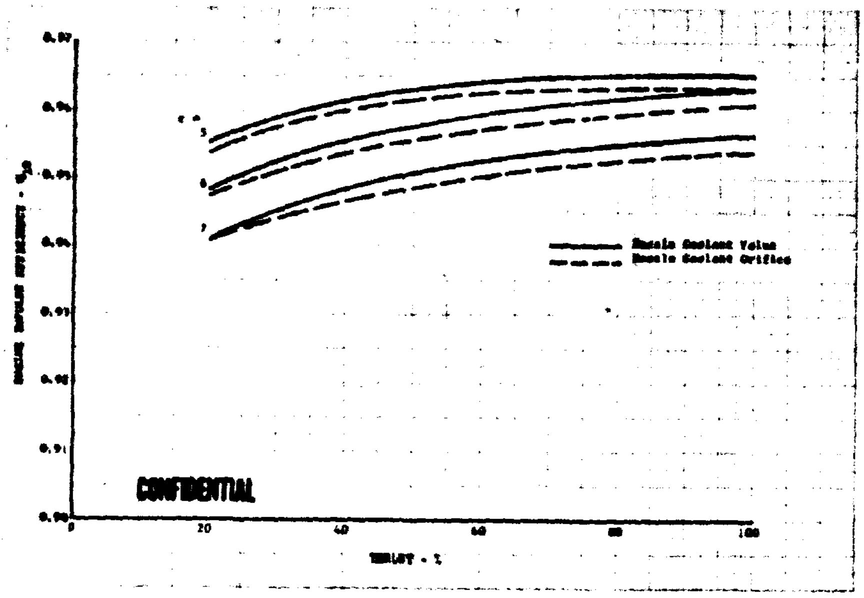
DFC 65467

(U) Figure 25. Effect of Low-Speed Inducer Tap-Off Location on Engine Impulse Efficiency



DFC 65468

(U) Figure 26. Effect of Fuel Preburner Supply Tap-Off Location on Lightweight Heat Exchanger Coolant Valve

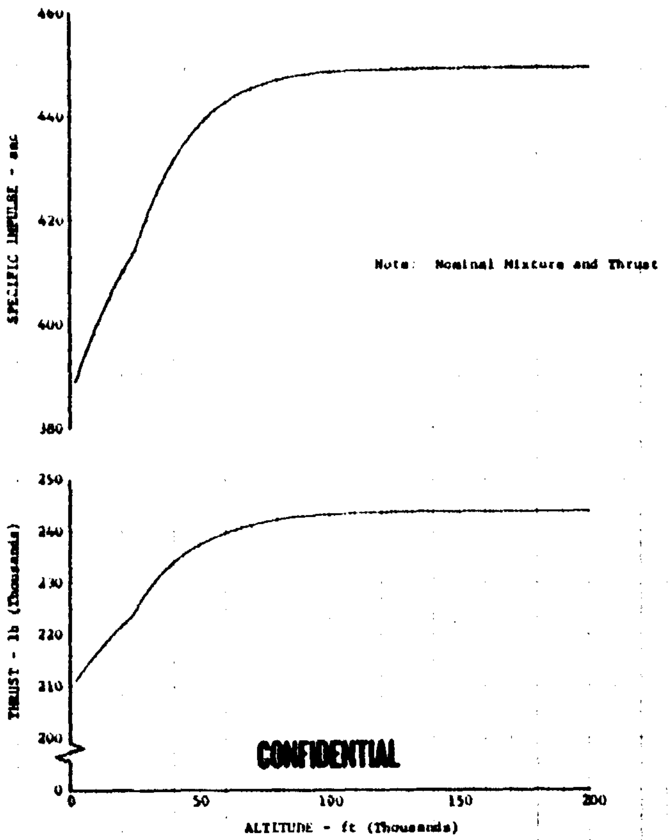


DFC 76003

(U) Figure 27. Effect of Fuel Preburner Supply Tap-Off Location on Engine Impulse Efficiency

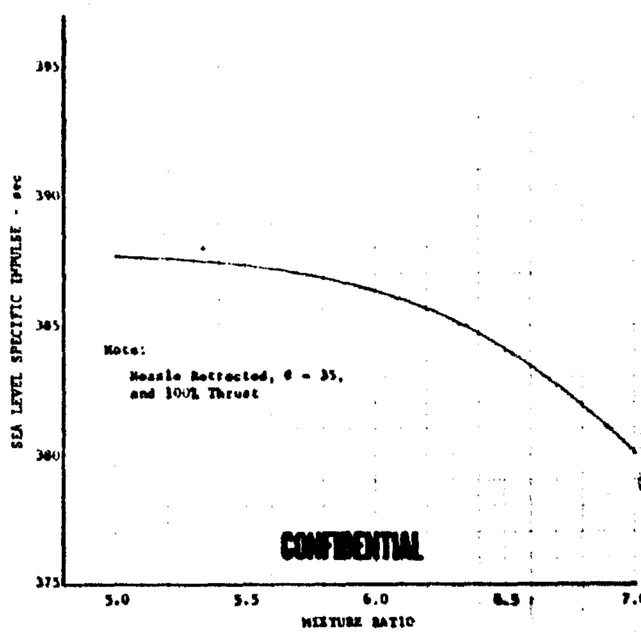
H. PERFORMANCE DATA

(C) The booster configuration operating at sea level, utilized the nozzle in the retracted position resulting in an expansion ratio of 35, which improves the thrust and specific impulse. At an altitude of 20,000 ft the two-position nozzle is translated to the extended position to provide an area ratio of 75 for improved altitude engine specific impulse. Use of the two-position nozzle provides nearly optimum performance for each operating regime. Altitude performance, i.e., thrust and specific impulse, for the booster configuration is presented in figure 28. The variation in sea level specific impulse with mixture ratio is shown in figure 29. The vacuum specific impulse variation with thrust and mixture ratio is given in figure 30.



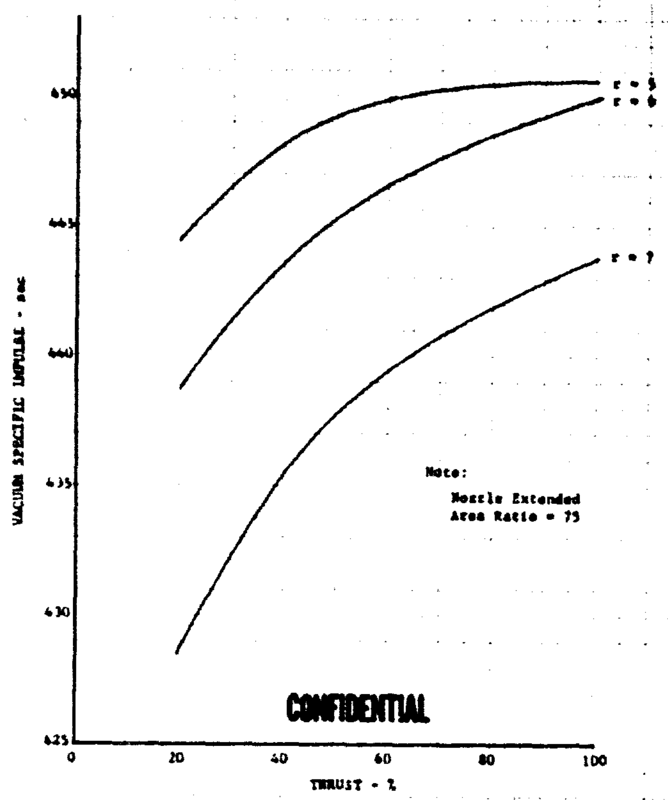
DFC 70090

(U) Figure 28. Altitude Performance for Demonstrator Engine (Booster Application)



DFC 70699

(U) Figure 29. Sea Level Specific Impulse vs Mixture Ratio (Booster Application)



DFC 70091

(U) Figure 30. Vacuum Specific Impulse vs Thrust (Booster Application)