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**Material in this publication relating to
DUAL ORIFICE INJECTION, LAMINATED
CHAMBER COOLING MEANS, and A VARIABLE
AREA INJECTOR CONCEPT**

reveals subject matter contained in U.S. Patent Application Serial Nos. 426,711, 319,047, and 390,521 entitled "Rocket Injector," "High Pressure Rocket and Cooling Means" and "Controllable Injector for Rockets," respectively, which have been placed under Secrecy Orders issued by the Commissioner of Patents. These Secrecy Orders have been modified by SECURITY REQUIREMENTS PERMITS and a PERMIT "A", respectively.

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AFRPL-TR-67-298-VOL III

(UNCLASSIFIED TITLE)

**ADVANCED
CRYOGENIC ROCKET ENGINE PROGRAM
STAGED-COMBUSTION CONCEPT
FINAL REPORT**

**R. R. ATHERTON
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DIVISION OF UNITED AIRCRAFT CORPORATION
FLORIDA RESEARCH AND DEVELOPMENT CENTER**

**TECHNICAL REPORT AFRPL-TR-67-298-VOL III
DECEMBER 1967**

DECLASSIFIED AFTER 15 YEARS, RDS BIR. 8888.18

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APPENDIX I ACCELERATION LOAD ANALYSIS

(U) A study of the 250X engine system with a fixed regeneratively cooled nozzle was made to determine the limiting values of acceleration loads for the basic engine components and assemblies. Particular attention was given to the rotating assemblies of the turbopumps and the major flanges between components.

(U) All engine components are capable of withstanding loads in excess of those required. (See figure 621.) The allowable accelerations (table LIII) were computed using the loads derived from the engine environment, including gimbal loads, vibration loads, pressure loads, and bolt preloads. This analysis did not include items such as control actuators, which have not been defined in mechanical detail.

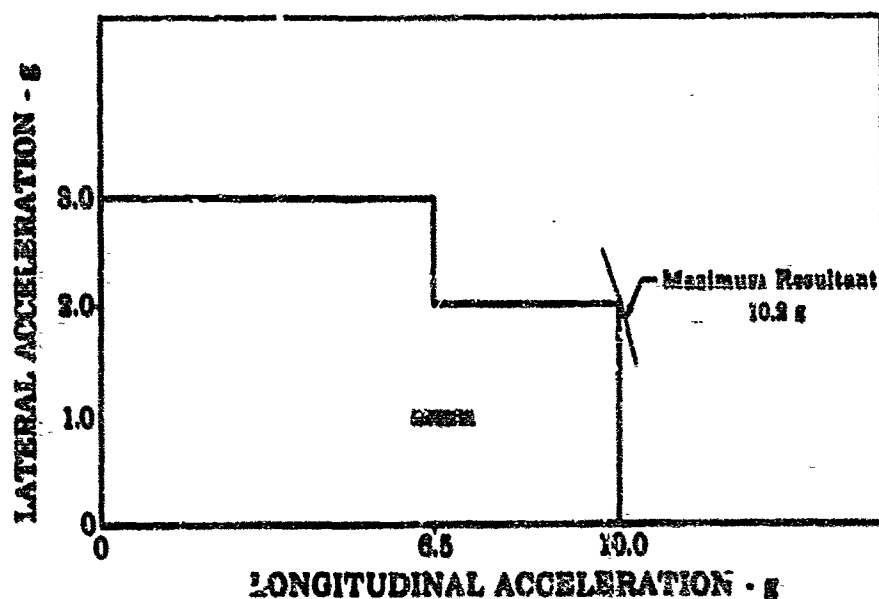


Figure 621. Operating or Nonoperating Engine
Acceleration Requirements

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(C) Table L121. Allowable g Load (10 hr)

Component	Longitudinal	Lateral	Resultant
Oxidizer Pump:			
Front Bearing			19
Rear Bearing			30
Housing			100+
Transition Case Flange			71
Transition Case Bolts			49
Shaft			*
Seals			15.8
			(Parallel to Centerline)
Fuel Pump:			
Front Bearing			37
Rear Bearing			30
Thrust Balance Piston			15+
			(Parallel to Centerline)
Housing			100+
Transition Case Flange			71
Transition Case Bolts			49
Shaft			*
Transition Case:			
Gimbal Actuator Loads	15+	3+	
Liner-Ducting	*	*	
Main Chamber Flange Bolts	15+	5	
Preburner Flow Divider Valve:			
Shaft Bearing			100+
Housing			187
Main Combustion Chamber:			
Nozzle Flange Bolts at $\phi = 4.75$	15+	5	
Nozzle:			
Transpiration Heat Exchanger:			
Outer Shell	15+	21	
Tube Buckling	15+	38	

*The allowable acceleration for these items is very large compared to the area of interest.

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(C) Table LIII. Allowable g Load (10 hr) (Continued)

Component	Longitudinal	Lateral	Resultant
Controls:			
Mixture Ratio Control:			
Shaft and Bearings	*		132
Housing			
Thrust Control:			
Butterfly Attachment			30+
Chamber Coolant Valve:			
Housing			20+
Low-Speed Inducers:			
Bearing Thrust Load	15+	2	

*The allowable acceleration for these items is very large compared to the area of interest.

(U) The main turbopump bearings are life and load limited. The oxidizer turbopump bearings used for this analysis were single ball bearings in both the front and rear locations. The fuel pump bearings were roller bearings in both locations.

(C) From this analysis the following conclusions can be made:

1. The engine system is capable of meeting the proposed maximum acceleration and environmental loads.
2. Accelerations to 15g longitudinal and 5g lateral can be accepted with no major change in the engine configuration. Minor changes in the transition case design to increase the structural rigidity may be required to accept the increased lateral loading.

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APPENDIX II
A SIMULATION FOR CONTROL STUDY OF THE
ROCKET ENGINE

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APPENDIX II
SIMULATION FOR CONTROL STUDY OF THE
ADVANCED CRYOGENIC ROCKET ENGINE

A. INTRODUCTION

(U) An analog mathematical model of the High Pressure Rocket Engine was developed in conjunction with a dynamically compatible digital model to study engine characteristics critical to the establishment of suitable control system. The decision to provide both an analog and digital engine simulation was based on the relatively high operational cost of the more accurate digital transient program and the fixed resolution problem associated with throttling the analog program over the required operational range. It is, therefore, anticipated that the digital program will be used to evaluate transient characteristics of the engine-control system, while the analog program will be used to determine steady-state stability and response.

(U) This report describes (1) the engine cycle and engine operational limits or component physical constraints that cannot be exceeded throughout the required engine operating range, (2) the complete mathematical description of the analog program, including equations, constants, functions, and block diagrams, and (3) predicted engine performance obtained from the analog model. The assumptions and bases used in deriving the equations for the analog mathematical model and the P&WA-FRCC computer wiring diagram are included.

(U) Paragraph E describes in detail the equivalent digital mathematical model.

B. ENGINE CYCLE DESCRIPTION

(U) The High Pressure Rocket Engine uses a staged-combustion cycle in which the fuel is burned with a portion of the oxygen in a preburner. These gaseous products are used to drive the main turbopumps before final combustion with the remainder of the oxygen in the main chamber. A propellant flow schematic illustrating the principle flow paths and functional component arrangement of this engine is shown in figure 622.

(U) Hydrogen enters at the engine-driven fuel low-speed inducer where sufficient pressure rise must be provided to satisfy the main fuel pump NPSH requirements. The low-speed inducers are used to minimize vehicle NPSH (i.e., tank pressure) requirements and allow high-speed main propellant pump operation for high turbopump efficiencies. Hydrogen is pumped to the system operating pressure by the main fuel pump. It is then ducted to cool the regenerative sections of the nozzle. The principal hydrogen flow from the pump is used in the rear regeneratively cooled nozzle section, and then ducted to the preburner. The remainder of the hydrogen is used in the forward regeneratively cooled nozzle section. This heated hydrogen then passes through the low-speed fuel inducer drive turbine prior to being passed into the main chamber as a transpiration coolant. A small amount of hydrogen is bled off at the fuel low-speed inducer discharge to provide dump coolant for the translatable secondary

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nozzle. This cooling flow is ducted from the discharge of the fuel low-speed inducer through a flow regulating valve and a quick-disconnect fitting, which is provided to stop the flow when the secondary nozzle is in the retracted position.

(U) Oxygen enters at the oxidizer low-speed inducer where sufficient pressure rise must be produced to satisfy the main oxidizer pump NPSH requirements. The oxygen is then pumped to system operating pressure levels by the main oxidizer pump. Pump discharge flow is then divided between the preburner and the main chamber. The main chamber oxygen flow is the principal oxidizer flow and is used as the oxygen low-speed inducer turbine working fluid. The smaller portion of the oxygen is ducted to the preburner where it is burned with the hydrogen. The resulting combustion products are ducted to the two main turbines, which are arranged in parallel, where the work required to drive the main pumps is extracted. The turbine exhaust gases rejoin in the transition case and pass through the main injector where they mix and burn with the main chamber oxidizer flow. The resulting combustion gas is then expanded through the bell nozzles.

(U) The low-speed fuel turbopump is a single shaft unit with an axial-flow inducer driven by a single-stage, gaseous hydrogen turbine. The oxygen low-speed turbopump is also a single shaft unit with an axial-flow inducer driven by a variable-admission, 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.

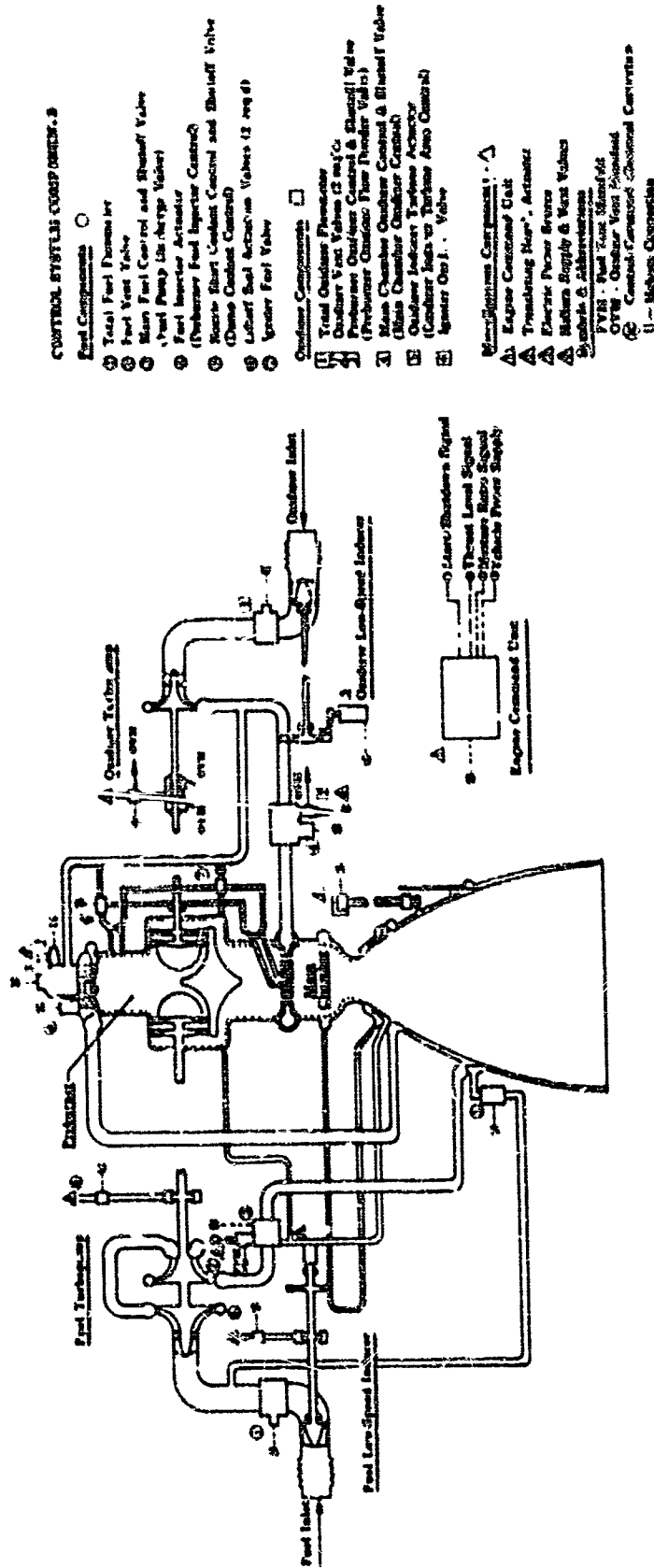
(U) The oxidizer turbopump is a single shaft unit with a single, shrouded centrifugal pump stage driven by a two-stage, pressure-compounded turbine. A single-acting thrust balance piston is provided between the pump and turbine.

(U) The preburner injector consists of dual-orifice oxidizer injector elements and variable-area fuel injector elements. A flow divider valve is incorporated at the inlet to the injector assembly to vary the total preburner oxidizer flow rate, and to regulate the flow split to the dual orifice oxidizer elements. The preburner combustion chamber is an integral part of the transition case.

(U) The main chamber propellants are supplied through fixed area injection elements. The fuel side (preburner products after expansion through the turbine) guides the fuel-rich gas around the oxidizer elements. The main combustor chamber wall consists of a hydrogen transpirationally cooled liner extending from the injector face to a point downstream of the chamber throat.

(U) The exhaust nozzle attaches immediately downstream of the transpiration cooled section and is composed of two regenerative cooled sections followed by the translating secondary nozzle.

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Figure 622. Demonstration Engine Control System

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(U) Preliminary analysis of the 250K rocket engine cycle, AF-1111A, indicates the following six control points are required for steady-state operation. These six control points as shown in figure 622 are:

1. Fuel pump discharge valve
2. Preburner fuel injector control
3. Dump coolant control
4. Preburner oxidizer flow divider valve
5. Main chamber oxidizer control
6. Oxidizer inducer turbine area control.

1. Fuel Pump Discharge Valve

(U) The main fuel control is located downstream of the fuel pump in the hydrogen coolant supply to the lower regenerative nozzle. The function of this control is to distribute the hydrogen flow between the preburner and the chamber transpiration supply section. This control regulates most of the hydrogen flow. Therefore, it has a strong influence on chamber mixture ratio and a minimal influence on thrust, because fuel flow is a small fraction of total propellant flow. By regulating the fuel flow entering the preburner, it also influences available turbine power. Because the main fuel valve controls the magnitude of transpiration flow, it will influence the low-speed inducer available power (i.e., main fuel pump NPSH) as well as the adequacy of the transpiration cooling. Further, the discharge valve is positioned to produce sufficient pressure loss to ensure fuel system stability at all operating conditions.

2. Preburner Fuel Injector Control

(U) The preburner fuel control is an integral part of the fuel preburner injector. Modulation of this control varies the area of the fuel injector.

(U) The velocity of the fuel entering the preburner must be controlled at all flow rates to ensure stable, efficient preburner combustion. This fuel injector area is in series with the main fuel control and affects the fuel system in a similar manner.

3. Dump Coolant Control

(U) The dump coolant control is located in the dump cooled two-position nozzle hydrogen supply. This control is used to regulate the fuel supply to the secondary nozzle. The percentage of total fuel flow regulated by this area is small, and its influence on the overall system is small.

4. Preburner Oxidizer Flow Divider Valve

(U) The preburner oxidizer flow divider valve regulates the total oxidizer flow to the preburner as well as the flow split to the primary and secondary elements of the injector. Oxidizer preburner flow determines available turbine power by controlling turbine flow rate and inlet temperature. The oxidizer flow split determines the velocity of the oxidizer entering the preburner and must be controlled to ensure stable efficient preburner combustion.

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5. Main Chamber Oxidizer Control

(U) The main chamber oxidizer control is located in the oxidizer supply to the main chamber. Because this control regulates most of the oxidizer flow, and in turn the majority of the total propellant flow, it has a strong influence on both chamber mixture ratio and thrust.

6. Oxidizer Inducer Turbine Area Control

(U) This control schedules the area of the low-speed inducer turbine that is in series with the main oxidizer control, and therefore affects the system in a similar manner. Modulation of this control affects available main oxidizer pump NPSH because turbine area determines turbine nozzle velocity, which influences turbine power and low-speed inducer performance.

7. Engine Operating Limits

(C) The operational limits and engine/control accuracy requirements for this engine-control system are as follows:

Thrust

1. Range - 100% to 20%
2. Engine/Control Accuracy
Engine/Control Accuracy at 100% thrust - $\pm 3\%$ full scale
Engine/Control Accuracy at 20% thrust - $\pm 3\%$ full scale

Mixture Ratio

1. Range - 5 to 7
2. Engine/Control Accuracy⁷
Engine/Control Accuracy at 100% thrust - 3% of mixture ratio
Engine/Control Accuracy at 20% thrust - 3% of mixture ratio

a. Thrust and Mixture Ratio Transient Response

(C) Changes between any combination of thrust and mixture ratio must be accomplished in less than 5 seconds within the limits as specified in the following paragraph.

b. Operational Limits

(C) The accompanying operational envelope (figure 623) reflects the steady-state component limitations for this engine cycle within the required operation ranges. These limits are as indicated below:

1. Main fuel turbopump maximum speed (NFP) - 48,000 rpm
2. Main oxidizer turbopump maximum speed (NLP) - 25,800 rpm
3. Main turbine inlet temperature - 2325°R.

⁷ Engine/Control Accuracy is defined as total inaccuracy and includes deviations in engine trim, control valve areas, computer nonrepeatability and control sense precision.

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(C) In addition to the steady-state limitations determined by the operational envelope:

1. A 4.5% minimum pressure drop exists for the main oxidizer injector, main fuel injector and preburner oxidizer and fuel injectors. This limitation will ensure combustion stability throughout the operating range.
2. Steady-state maximum transpiration cooling flow will not exceed the schedule shown in figure 624.

(C) Other physical limitations that apply to transient as well as steady-state operation are:

1. Minimum mixture ratio is set at 2.0 for the main chamber and 0.25 for the preburner.
2. Maximum main chamber mixture ratio is set at 8.0.
3. Minimum required net positive suction head for the main fuel pump and main oxidizer pump is a function of pump operation condition, as shown in figures 625 and 626.
4. Minimum transpiration cooling flow is a function of main chamber operating conditions and will be defined at a later date.

C. ANALOG SIMULATION

1. General

(U) In addition to the equations, constants, and functions that define this analog simulation, results from operation of this simulation on the P&WA-FRDC computer, and details of this operation, are presented in the following paragraphs.

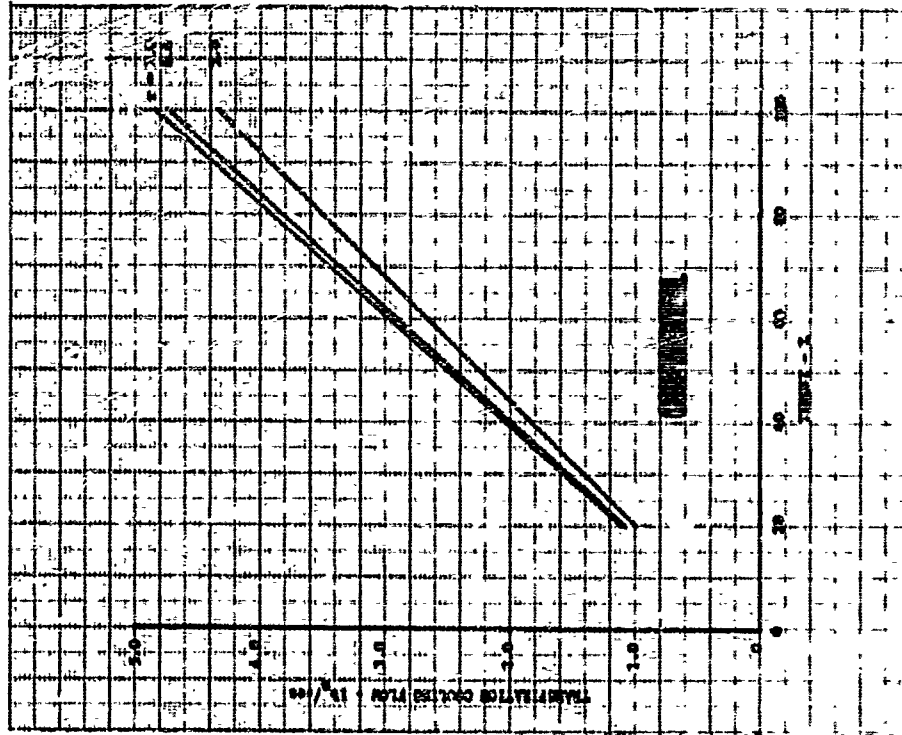
2. Equations, Constants, and Functions

(C) The analog simulation of the High Performance Rocket Engine is defined in this section. The simulation consists of nine point-programs that allow limited operation about combinations of thrust at 100%, 50%, or 20% with mixture ratios of 5, 6, or 7.

(U) The building block method of simulation has been used. Individual component functions are separately developed and then interrelated into the engine system. The complete engine analog definition is accomplished within 12 groups of related engine components. These groups, called Engine Analog Sections, are illustrated and identified by letter and name in the accompanying Engine Section Diagram, figure 627. The parameter symbols used in this diagram show the location of parameters within the engine system. These same parameter symbols are used in the analog equations appearing as part of the analog definition within each of these Engine Analog Sections.

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Figure 624. Transpiration Cooling Flow vs Thrust and Mixture Ratio

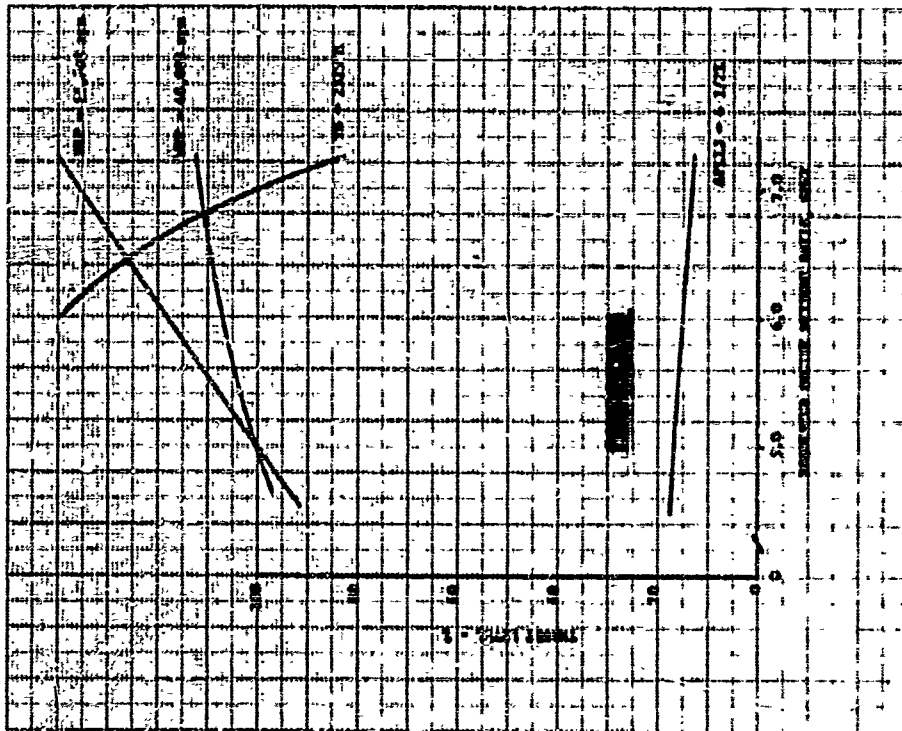


Figure 623. Operational Envelope

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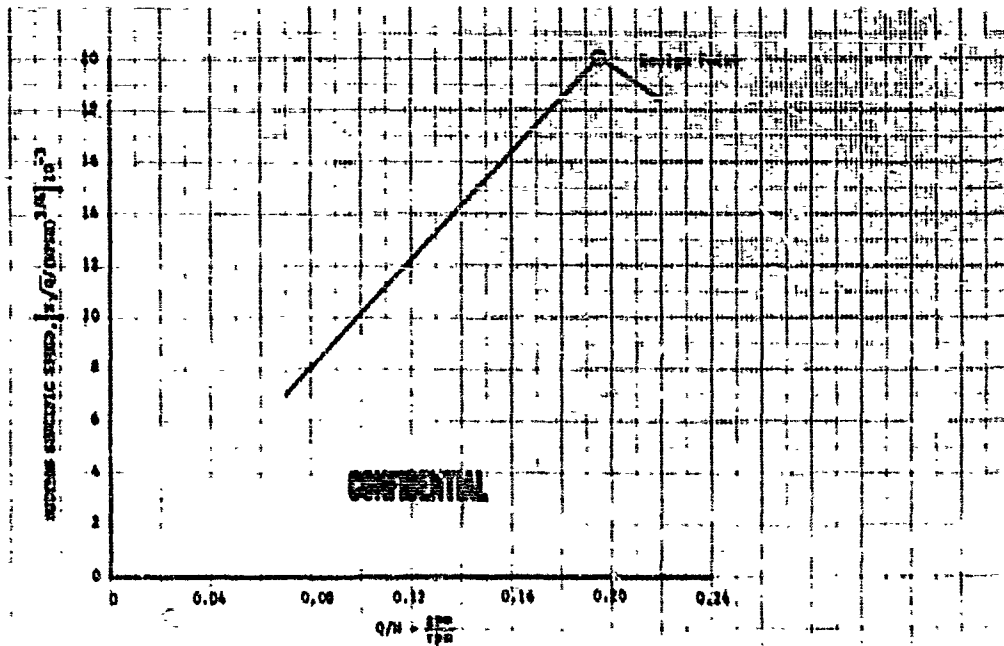


Figure 625. Main Fuel Pump Design
Suction Capability

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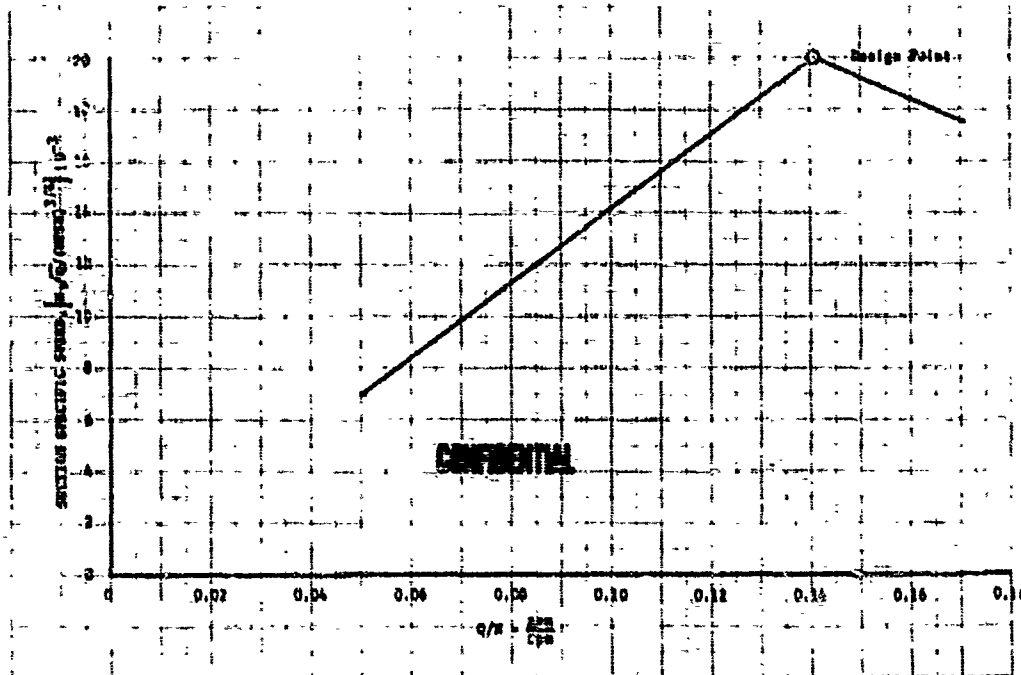


Figure 626. Main Oxidizer Pump Design
Suction Capability

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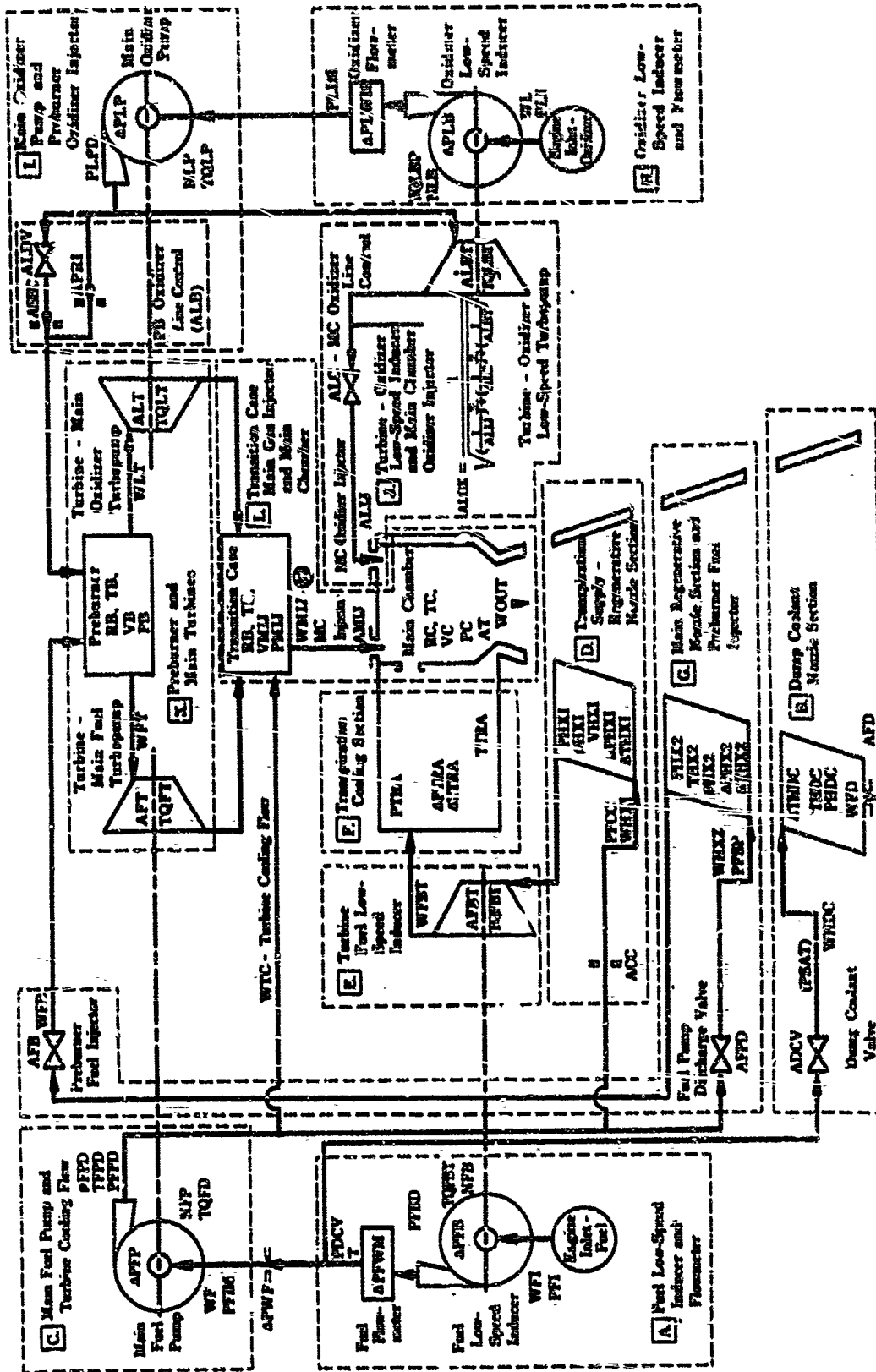


Figure 6A7. Engine Section Diagram

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(U) The complete analog definition for each lettered Engine Analog Section appears within the following, correspondingly lettered, paragraphs. In addition to a set of analog equations expressed in terms of constants, functions, and parameter symbols, the definition includes numerical values for the constants, curves for the functions, and word definitions for the parameter symbols. Also included is a representative block diagram for the fuel low-speed inducer and flowmeter section.

(U) For analog operation at F&WA-FRDC the univariate functions were matched as straight lines between the points listed in the tables shown for each univariate curve. Records of the F&WA-FRDC match achieved for each of the bivariate functions (the eight pump curves) are included as broad (wavy) lines on each bivariate curve.

(U) **FUEL LOW-SPEED INDUCER AND FLOWMETER**

Refer to figure 628 for block diagram.

EQUATIONS

$$\begin{aligned} WFI &= WF + WHDC \\ \dot{NFB} &= C_5 \cdot (TQFBT - TQFBF) \\ TQFBF &= f_{1-x} \\ \Delta PFB &= f_{2-x} \\ PFED &= PFI + \Delta PFB \\ \Delta PFWM &= C_6 \cdot (WFI)^2 \\ PDCV &= PFED - \Delta PFWM \end{aligned}$$

CONSTANTS AND FUNCTIONS

$$\begin{aligned} C_5 &= 9.55/JFBF = 0.267 \times 10^3 \\ C_6 &= (1.496)^2 / [\rho_F \cdot (AWFM)^2] = 0.1776 \times 10^{-2} \\ f_{1-x} &= f(NFB, WFI): \text{See figure 629 for } f_{1-100} \\ f_{2-x} &= f(NFB, WFI): \text{See figure 630 for } f_{2-100} \\ PFI &= 0.3180 \times 10^2 \end{aligned}$$

PARAMETER DEFINITIONS

WFI = Flow - Fuel, Inlet, (Engine Total)	: lb _m /sec
WF = Flow - Fuel, Main Pump	: lb _m /sec
WHDC = Flow - (Fuel), Heat Exchanger, Dump Coolant	: lb _m /sec
NFB = Speed - Fuel Low-Speed Inducer	: rpm
$\dot{NFB}$ = Speed Change Rate - Fuel Low-Speed Inducer	: d(rpm)/d(t)

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PARAMETER DEFINITIONS (Continued)

TQFST = Torque - Fuel Low-Speed Inducer Turbine (Delivered)	: lb _f · ft
TQFSP = Torque - Fuel Low-Speed Inducer (Required)	: lb _f · ft
ΔPFS = Pressure Rise - Fuel Low-Speed Inducer (Pump)	: lb _f /in. ²
PFBD = Pressure - Fuel Low-Speed Inducer Discharge	: lb _f /in. ²
PFI = Pressure - Fuel, (Engine) Inlet	: lb _f /in. ²
ΔPFTM = Pressure Drop - Fuel Flowmeter	: lb _f /in. ²
PDCV = Pressure - Dump Coolant Valve (Upstream)	: lb _f /in. ²
JFBI = Rotor Polar Moment of Inertia, Fuel Low-Speed Inducer	: ft · lb _f · sec ²
ρ _f = Fuel Density, (Low-Speed Inducer Inlet)	: lb _m /ft ³
AWFM = Area (Effective), Fuel Flowmeter	: in. ²

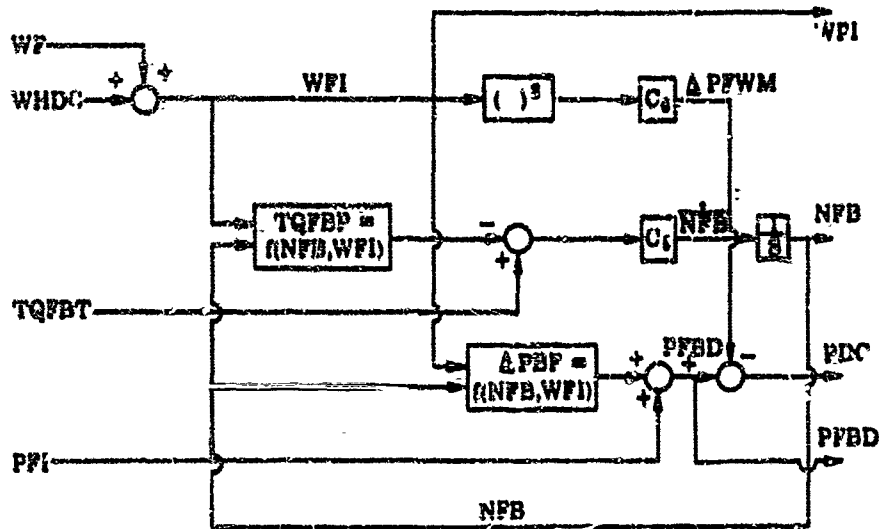


Figure 623. Fuel Low-Speed Inducer and Flowmeter Analog Section

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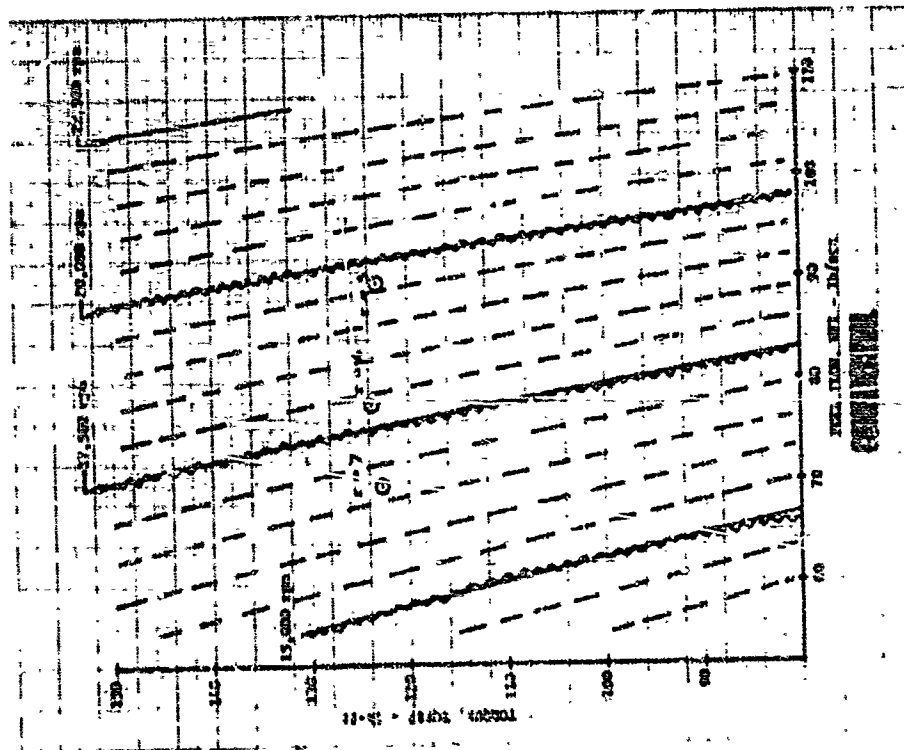


Figure 629. Fuel Low-Speed Inducer Characteristics Curve (Sheet 1)

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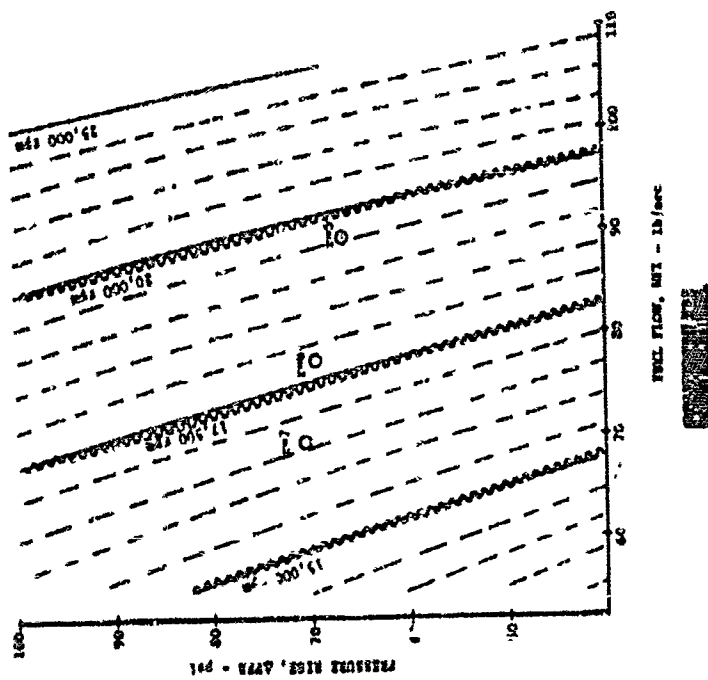


Figure 630. Fuel Low-Speed Inducer Characteristics Curve (Sheet 2)

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(U) **B.** DUMP COOLANT NOZZLE SECTION (HEAT EXCHANGER, DUMP COOLANT)

EQUATIONS

$$\begin{aligned} WHDC &= C_7 \cdot ADCV \cdot \sqrt{(PDCV - C_{93})} \\ \Delta THDC &= C_8 \cdot (PFG)^{0.87} \cdot f_{14} / WHDC \\ (PFG)^{0.97} &= (C_{81-x} \cdot PFG) + C_{82-x} \quad (\text{See Section D, Heat Exchanger No. 1}) \\ THDC &= (C_{89} + \Delta THDC) \cdot 1 / (1 + C_9 \cdot S) \\ PHDC &= C_{10} \cdot THDC \cdot (WHDC - WFD) \\ WFD &= C_{11} \cdot f_6 \cdot PHDC / THDC \end{aligned}$$

CONSTANTS AND FUNCTIONS

$$\begin{aligned} C_7 &= \sqrt{PFED} / 1.496 = 0.1375 \times 10 \\ C_8 &= 0.2704 \times 10^4 \\ C_{9-x} &= (\text{Temp. Time Constant}): C_{9-100} = 0.2500 \\ C_{10} &= RH / VHDC = 0.7460 \times 10 \\ C_{11} &= AFD = 0.9714 \times 10 \\ C_{89} &= TFBD = 0.4220 \times 10^2 \\ ADCV_{-x/y} &: ADCV_{-100/5} = 0.1683, ADCV_{-100/6} = 0.1527, ADCV_{-100/7} = 0.1344 \end{aligned}$$

$$\begin{aligned} f_{14} &: KOFHT = f(OFC), \text{ Refer to table LIV} \\ f_6 &: W \cdot \sqrt{T/A \cdot P} = f(PHDC/PAMB), \text{ Refer to table LV} \\ C_{93} &= PSAT = 3.300 \end{aligned}$$

(U) Table LIV f_{14} (KOFHT) as a Function of (OFC)

OFC	f_{14}	OFC	f_{14}
2.0	0.61	6.5	0.96
2.5	0.75	7.0	0.912
3.0	0.865	7.5	0.86
3.5	0.955	8.0	0.81
4.0	1.008	8.5	0.76
4.5	1.03	9.0	0.71
5.0	1.04	9.5	0.655
5.5	1.028	10.0	0.605
6.0	1.00		

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(U) Table LV. $f_6 \left[\frac{W\sqrt{T}}{A\sqrt{P}} \right]$ as a Function of (PHDC/PAMB)

$\gamma = 1.39$			
$\frac{PHDC}{PAMB}$	f_6	$\frac{PHDC}{PAMB}$	f_6
1.03	0.045	1.45	0.1315
1.05	0.06	1.6	0.1370
1.10	0.0829	1.75	0.139
1.15	0.097	1.89	0.1399
1.25	0.115	2.0	0.1399
1.35	0.1252		

PARAMETER DEFINITIONS

WHDC = Flow - (Fuel), Heat Exchanger, Pump Coolant (Inlet)	: lb_m/sec
ADCV = Area - Dump Coolant Valve	: in^2
PDCV = Pressure - Dump Coolant Valve (Upstream), See Section A	: lb_f/in^2
$\Delta THDC$ = Temperature Rise - Heat Exchanger, Dump Coolant	: $^{\circ}R$
THDC = Pressure - Heat Exchanger, Dump Coolant (Exit)	: $^{\circ}R$
PHDC = Pressure - Heat Exchanger, Dump Coolant (Exit)	: lb_f/in^2
WFD = Flow - Fuel, Dump (Heat Exchanger, Dump Coolant, Exit)	: lb_m/sec
ρ_{FBD} = Density - Fuel (Main) Pump Discharge (See Section C)	: lb_m/ft^3
RH = Gas Constant for Hydrogen (See Section D)	
VHDC = Volume - (Fuel Passage), Heat Exchanger, Dump Coolant	: in^3
AFD = Area (Effective) - Heat Exchanger, Dump Coolant	: in^2
TFRD = Temperature - Fuel Low-Speed Inducer Discharge	: $^{\circ}R$
PAMB = Pressure, Ambient - 14.7	: lb_f/in^2

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(U) C. MAIN FUEL PUMP AND TURBINE

EQUATIONS

$$\Delta PWF = C_{12} \cdot (WF)^2$$

$$PFIM = PDCV - \Delta PWF$$

$$WF = WHX2 + WHX1 + WTC$$

$$\frac{1}{NFP} = C_{14} \cdot (TQFT - TQFP)$$

$$TQFP = f_{8-x}$$

$$\Delta PFP = f_{9-x}$$

$$PFPD = PFIM + \Delta PFP$$

$$WTC = f_{49}$$

$$TFPD = (C_{75} \cdot PFPD) + C_{76}$$

$$\sqrt{PFPD} = (C_{77-x} \cdot PFPD) + C_{78-x}$$

CONSTANTS AND FUNCTIONS

$$C_{12} = (1.496)^2 / \left[\rho_F \cdot (AFPI)^2 \right] = 0.8474 \times 10^{-3}$$

$$C_{14} = 9.55/JFP = 0.1230 \times 10^3$$

$$C_{75} = 0.1400 \times 10^{-1}$$

$$C_{76} = 0.5110 \times 10^0$$

$$C_{77-x} : C_{77-100} = 0.2800 \times 10^{-4}$$

$$C_{78-x} : C_{78-100} = 1.917$$

$$f_{8-x} = f(NFP, WF) : \text{See figure 631 for } f_{8-100}$$

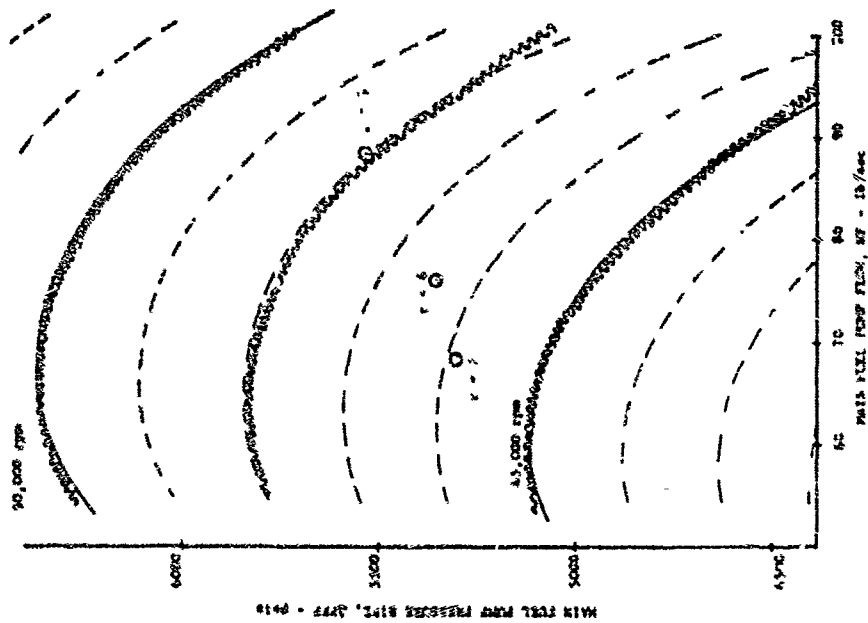
$$f_{9-x} = f(NFP, WF) : \text{See figure 632 for } f_{9-100}$$

$$f_{49} = WTC = f(PFPD) : \text{Refer to table LVZ.}$$

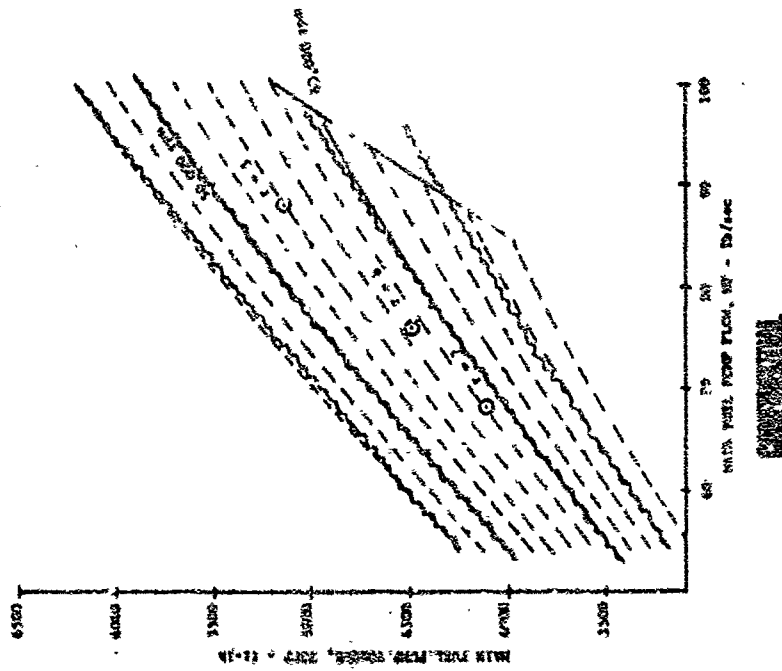
PARAMETER DEFINITIONS

ΔPWF = Pressure Drop - Fuel Flow (Main Pump Inlet Line)	: lb_f/in^2
WF = Flow - Fuel, (Main Pump)	: lb_m/sec
$PFIM$ = Pressure - Fuel Inlet, Main (Pump)	: lb_f/in^2
$PDCV$ = Pressure - Dump Coolant Valve (Upstream)	: lb_f/in^2
$WHX2$ = Flow - (Fuel), Heat Exchanger No. 2 (Main Regenerative Nozzle)	: lb_m/sec

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Figure 631. Main Fuel Pump and Turbine Characteristics Curve (Sheet 1) DF 59709 Figure 632. Main Fuel Pump and Turbine Characteristics Curve (Sheet 2) DF 59710

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PARAMETER DEFINITIONS (Continued)

WHX1 = Flow - (Fuel) Heat Exchanger No. 1 (Transpiration Supply Regenerative Nozzle)	: lb_m/sec
WTC = Flow - (Fuel), Turbine Cooling (Etc.)	: lb_m/sec
NFP = Speed - (Main) Fuel (Turbo) Pump	: rpm
$\dot{NFP}$ = Speed Change Rate - (Main) Fuel (Turbo) Pump	: $d(rpm)/d(t)$
TQFT = Torque - (Main) Fuel Turbine (Delivered), (See Section K)	: $lb_f \cdot ft$
TQFP = Torque - (Main) Fuel Pump (Required)	: $lb_f/in.^2$
ΔPFP = Pressure Rise - Fuel Pump (Main)	: $lb_f/in.^2$
PFPD = Pressure - Fuel (Main) Pump Discharge	: $lb_f/in.^2$
TFPD = Temperature - (Main) Fuel Pump Discharge	: $^{\circ}R$
ρ_{FPD} = Density - Fuel, (Main) Pump Discharge	: lb_m/ft^3
ρ_f = Fuel Density, (Main Fuel Pump Inlet)	: lb_m/ft^3
AFPI = Area (Effective), Main Fuel Pump Inlet Line	: $in.^2$
JFP = Rotor Polar Moment of Inertia, Main Fuel Turbopump	: $ft \cdot lb_f \cdot sec^2$

(U) Table LVI. $f_{49}(WTC)$ as a Function
of (PFPD)

WTC	PFPD
1.85	900
2.07	1130
2.75	1925
3.32	2655
4.40	4000
5.42	5443
6.54	7000
7.25	8000

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(U) **TRANSPIRATION SUPPLY REGENERATIVE NOZZLE SECTION**
(HEX. NO. 1 - COOLANT SIDE)

EQUATIONS

$$\begin{aligned} WHX1 &= C_{15} \cdot \sqrt{\rho FPD} \cdot \sqrt{(PTPD - PFCC)} \\ PFCC &= PHX1 + \Delta PHX1 \\ \overline{PHX1} &= C_{18} \cdot THX1 \cdot (WHX1 - WFBT) \\ \Delta PHX1 &= C_{74} \cdot (PFG)^{0.05} \cdot (WHX1)^2 / \rho HX1 \\ \rho HX1 &= C_{100} \cdot PHX1 / THX1 \\ \Delta THX1 &= C_{16} \cdot (PFG)^{0.87} \cdot f_{14} \cdot (1 / WHX1) \\ THX1 &= (TFPD + \Delta THX1) \cdot \left[1 / (1 + C_{17} \cdot s) \right] \\ (PFG)^{0.05} &= (C_{79-x} \cdot PFG) + C_{80-x} \\ (PFG)^{0.87} &= (C_{81-x} \cdot PFG) + C_{82-x} \\ PFG &= PC / [C_{84-x} - (C_{83-x} \cdot OFER)] \end{aligned}$$

CONSTANTS AND FUNCTIONS

$$\begin{aligned} C_{15} &= ACC / 1.496 = 0.7192 \times 10^{-1} \\ C_{16} &= 0.1671 \times 10^4 \\ C_{17-x} &= (\text{Temperature Time Constant}) : C_{17-100} = 0.2500 \\ C_{18} &= RH / VHXL = 9200 / 300 = 0.3067 \times 10^2 \\ C_{74} &= 0.7252 \times 10 \\ C_{79-x} : C_{79-100} &= 0.4930 \times 10^{-4} \\ C_{80-x} : C_{80-100} &= 0.9507 \\ C_{81-x} : C_{81-100} &= 0.8667 \\ C_{82-x} : C_{82-100} &= 0.1333 \\ C_{83-x} : C_{83-100} &= 0.6130 \times 10^2 \\ C_{84-x} : C_{84-100} &= 0.3094 \times 10^4 \\ C_{100} &= 0.1878 \\ f_{14} : KOFHT &= f(OF), \text{ Refer to table LIV.} \end{aligned}$$

PARAMETER DEFINITIONS

PFCC = Pressure - Fuel, Transpiration Orifice, (Orifice-Downstream) : $lb_f/in.^2$
PHX1 = Pressure - (Fuel), Heat Exchanger No. 1, (Exit) : $lb_f/in.^2$
THX1 = Temperature - (Fuel) Heat Exchanger No. 1 (Exit) : $^{\circ}R$

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PARAMETER DEFINITIONS (Continued)

WFBT = Flow - Fuel, Low-Speed Inducer Turbine - See Section E	: lb _m /sec
WHX1 = Flow - Fuel, Heat Exchanger No. 1 (Transpiration Supply)	: lb _m /sec
ρFPD = Density - Fuel Pump (Main Discharge), (See Section C)	: lb _m /ft ³
PFPD = Pressure - Fuel Pump (Main) Discharge, (See Section C)	: lb _f /in. ²
ΔPHX1 = Pressure Loss - (Fuel), Heat Exchanger No. 1	: lb _f /in. ²
PFG = PC/(PC @ 100% Thrust Level)	: D'less
ρHX1 = Density - (Fuel), Heat Exchanger No. 1	: lb _m /ft ³
ΔTHX1 = Temperature Rise - Heat Exchanger No. 1 (Trans. Supply Fuel)	: °R
TFPD = Temperature - Fuel (Main) Pump Discharge	: °R
OFER = Mixture Ratio - Engine (Overall), Requested	: D'less
ACC = Area (Effective), Transpiration Orifice	: in. ²
RH = Gas Constant for Hydrogen	: in. · lb _f /lb _m · °R
VHX1 = Volume - Heat Exchanger No. 1 (Fuel Passage)	: in. ³
KOFHT = Correction Factor - Mixture Ratio, (Heat Exchanger Fuel Temperature)	: D'less
OFC = Mixture Ratio - (Main) Chamber, (Chamber Overall), See Sec. L	: D'less
PC = Pressure - (Main) Chamber, See Section L	: lb _f /in. ²
S = Laplace Operator	

(U) E. TURBINE - FUEL LOW-SPEED INDUCER

EQUATIONS

$$\begin{aligned}
 WFBT &= C_{19} \cdot f_{15} \cdot PHX1 \cdot (1/\sqrt{THX1}) \\
 \sqrt{\Delta HFBT} &= f_{17} \cdot \sqrt{THX1} \cdot \sqrt{CPBT} \\
 \sqrt{CPBT} &= (C_{20-x} \cdot THX1) + C_{21-x} \\
 TQFBT &= WFBT \cdot \left[(C_{22-x} \cdot \sqrt{\Delta HFBT}) - (C_{23-x} \cdot NFB) \right]
 \end{aligned}$$

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CONSTANTS AND FUNCTIONS

Where:

$$\begin{aligned}
 C_{19} &= AFET = 0.2006 \\
 C_{20-x} : C_{20-100} &= 0.7550 \times 10^{-3} \\
 C_{21-x} : C_{21-100} &= 0.2330 \times 10 \\
 C_{22-x} : C_{22-100} &= 0.3389 \times 10 \\
 C_{23-x} : C_{23-100} &= 0.6546 \times 10^{-3} \\
 f_{15} : W \cdot \sqrt{T} / (A \cdot P) &= f(PHX1/PTRA), \text{ Refer to table LVII} \\
 f_{17} : \sqrt{1 - (PTRA/PHX1)^{\gamma-1/\gamma}} &= f(PTRA/PHX1), \text{ Refer to table LVIII.}
 \end{aligned}$$

(U) Table LVII. $f_{15} \left[W \sqrt{T} / (AP) \right]$ as a Function of $(PHX1/PTRA)$

$\gamma = 1.39$			
$\frac{PHX1}{PTRA}$	f_{15}	$\frac{PHX1}{PTRA}$	f_{15}
1.03	0.045	1.45	0.1315
1.05	0.06	1.6	0.1370
1.10	0.0829	1.75	0.139
1.15	0.097	1.89	0.1399
1.25	0.115	2.00	0.1399
1.35	0.1252		

(U) Table LVIII. $f_{17} \left(\sqrt{1 - (PTRA/PHX1)^{\gamma-1/\gamma}} \right)$ as a Function of $(PTRA/PHX1)$

$\frac{PTRA}{PHX1}$	f_{17}	$\frac{PTRA}{PHX1}$	f_{17}
0.0	1.0	0.3	0.5354
0.015	0.8500	0.75	0.2784
0.03	0.8000	0.9	0.1706
0.05	0.7540	0.95	0.1195
0.1	0.6898	1.00	0.0
0.2	0.6028		

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PARAMETER DEFINITIONS

WFBT = Flow - Fuel, Low-Speed Inducer Turbine	: lb _m /sec
ΔHFBT = Enthalpy Drop - Fuel, Low-Speed Inducer Turbine	: Btu/lb _m
CPBT = Specific Heat at Constant Pressure - (Fuel), Low-Speed Inducer Turbine	: Btu/lb _m · °R
TQFBT = Torque - Fuel Low-Speed Inducer Turbine (Delivered)	: lb _f · ft
THX1 = Temperature - (Fuel) Heat Exchanger No. 1 (Exit) (See Section D)	: °R
NFB = Speed - Fuel Low-Speed Inducer, (See Section A)	: rpm
γ = Ratio of Specific Heats	: D'less
AFBT = Area (Effective) - Fuel Low-Speed Inducer	: in. ²
PHX1 = Pressure - (Fuel) Heat Exchanger No. 1 (Exit) (See Section D)	: lb _f /in. ²
PTRA = Pressure - (Fuel) Transpiration Cooled Section Inlet (See Section D)	: lb _f /in. ²

(U) F TRANSPIRATION COOLING SECTION

EQUATIONS

$$\begin{aligned}\Delta T_{TRA} &= C_{67} \cdot (PFG)^{0.87} \cdot f_{43} \cdot (1/WFBT) \\ T_{TRA} &= THX1 + \Delta T_{TRA} \\ \Delta P_{TRA} &= C_{25} \cdot (PFG)^{0.2} \cdot f_{50} \cdot (WFBT)^2 \cdot T_{TRA}/PC \\ P_{TRA} &= PC + \Delta P_{TRA} \\ (PFG)^{0.2} &= (C_{85-x} \cdot PFG) + C_{86-x}\end{aligned}$$

CONSTANTS AND FUNCTIONS

$$\begin{aligned}(PFG)^{0.87} &= (C_{81-x} \cdot PFG) + C_{82-x}, \text{ (See Section D)} \\ C_{25} &= 0.3976 \times 10^2 \\ C_{67} &= 0.6730 \times 10^4 \\ C_{81-x} &: \text{ (See Section D)} \\ C_{82-x} &: \text{ (See Section D)} \\ C_{85-x} &: C_{85-100} = 0.1950 \\ C_{86-x} &: C_{86-100} = 0.8050 \\ f_{43} &: KOFTT = f(OPC), \text{ Refer to table LIX.}\end{aligned}$$

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(U) Table LX. f_{50} (KOFTT) as a Function of (OFC)

OFC	f_{43}	OFC	f_{43}
2.0	0.28	6.5	1.025
3.0	0.52	7.0	1.044
3.5	0.63	7.5	1.06
4.0	0.73	8.0	1.08
4.5	0.82	8.5	1.097
5.0	0.90	9.0	1.11
5.5	0.96	9.5	1.13
6.0	1.0	10.0	1.145

PARAMETER DEFINITIONS

$\Delta TTRA$	= Temperature Rise - Transpiration Cooled Section (Fuel)	: °R
$TTRA$	= Temperature (Fuel) - Transpiration Cooled Section (Exit)	: °R
ΔPTR	= Pressure Drop (Fuel) - Transpiration Cooled Section	: lb_f/in^2
$PTRA$	= Pressure (Fuel) - Transpiration Cooled Section (Inlet)	: lb_f/in^2
$THX1$	= Temperature - (Fuel), Heat Exchanger No. 1 (Exit) (See Section D)	: °R
PFG	= Thrust Ratio (See Section D)	: D'less
f_{50}	= KOFTP = $f(OFC)$, Refer to table LX.	
KOFTP	= Correction Factor - Mixture Ratio, Transpiration (Cooled Section) Pressure	: D'less
PC	= Pressure - (Main) Chamber, (See Section L)	: D'less
OFER	= Mixture Ratio - Engine (Overall) - Requested	: D'less
OFC	= Mixture Ratio - Main Chamber	: D'less
KOFTT	= Correction Factor - Mixture Ratio, Transpiration (Cooled Section) Temperature	: D'less
WFET	= Flow - Fuel, Low-Speed Inducer Turbine (Calculated in Section E)	: lb_m/sec

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(U) Table LX. $f_{20}(\text{KOFTP})$ as a Function of (OFC)

OFC	KOFTP	OFC	KOFTP
2.0	0.927	6.6	0.960
2.6	0.971	7.0	0.9214
3.6	1.008	7.6	0.840
4.0	1.016	8.0	0.778
4.6	1.020	8.6	0.673
5.0	1.0198	9.0	0.600
5.6	1.012	9.6	0.490
6.0	1.0	10.0	0.413

(U) **2.** IN REGENERATIVE NOZZLE SECTION (HEX NO. 2); AND
PREBURNER INJECTOR (FUEL SIDE)

EQUATIONS

$$\begin{aligned} \text{WHX2} &= C_{27} \cdot \text{AFPD}_{x/y} \sqrt{\rho \text{FPD}} \cdot \sqrt{\text{PFDP} - \text{PFSP}} \\ \text{PFSP} &= \text{PHX2} + \Delta \text{PHX2} \\ \Delta \text{PHX2} &= C_{28} \cdot (\text{PFG})^{0.05} \cdot (\text{WHX2})^2 / \rho \text{HX2} \\ \rho \text{HX2} &= C_{100} \cdot \text{PHX2} / \text{THX2} \\ \Delta \text{THX2} &= C_{29} \cdot (\text{PFG})^{0.87} \cdot f_{14} / (\text{WHX2}) \\ \text{THX2} &= (\text{TFPD} + \Delta \text{THX2}) \cdot \left[1 / (1 + C_{30} \cdot S) \right] \\ \text{PHX2} &= C_{31} \cdot \text{THX2} \cdot (\text{WHX2} - \text{WFB}) \\ \text{WFB} &= \text{AFB}_{x/y} \cdot f_{24} \cdot \text{PHX2} / \sqrt{\text{THX2}} \end{aligned}$$

CONSTANTS AND FUNCTIONS

Where:

$$\begin{aligned} C_{27} &= 1/1.496 = 0.6684 \\ C_{28} &= 0.6242 \times 10^{-1} \\ C_{29} &= 0.3972 \times 10^4 \\ C_{30-x} &: \text{Main Regenerative Heat Exchanger} \\ &\quad \text{Temperature Time Constants, } C_{30-100} = 0.2500 \\ C_{31} &= \text{RH/VHX2} = 9200/1683 = 0.5466 \times 10 \\ C_{100} &= 0.1878 \\ f_{14} &: \text{KOFHT} = f(\text{OFC}), \text{ Refer to table LIV.} \end{aligned}$$

PARAMETER DEFINITIONS

$$\begin{aligned} \text{WHX2} &= \text{Flow - (Fuel), Heat Exchanger No. 2} \\ &\quad \text{(Coolant In)} \quad : \text{ lb}_m/\text{sec} \\ \rho \text{FPD} &= \text{Density - Fuel (Main) Pump Discharge} \\ &\quad \text{(See Main Fuel Pump) } \boxed{C} \quad : \text{ lb}_m/\text{ft}^3 \end{aligned}$$

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PARAMETER DEFINITIONS (Continued)

PFPD = Pressure - Fuel Pump Discharge (See Main Fuel Pump) C	: lb_f/in^2
PFSP = Pressure - Fuel (Heat Exchanger No. 2 Inlet)	: lb_f/in^2
PHX2 = Pressure - (Fuel), Heat Exchanger No. 2 (Exit)	: lb_f/in^2
Δ PHX2 = Pressure Drop - (Fuel), Heat Exchanger No. 2	: lb_f/in^2
ρ HX2 = Density - (Fuel), Heat Exchanger No. 2 (Exit)	: lb_m/ft^3
Δ THX2 = Temperature Rise - (Fuel), Heat Exchanger No. 2	: $^\circ\text{R}$
THX2 = Temperature - (Fuel), Heat Exchanger No. 2, (Exit)	: $^\circ\text{R}$
AFPD = Area - Fuel Pump Discharge (Valve), (Effective)	: in^2
S = Laplace Operator	
WFB = Flow - Fuel, Preburner (Injector)	: lb_m/sec
AFB = Area - Fuel (Side), Preburner (Injector) (Effective)	: in^2
AFB _{-x/y} : AFB _{-100/5} = 2.661, AFB _{-100/6} = 1.814 AFB _{-100/7} = 1.403	
f_{24} : $W \cdot \sqrt{T}/A \cdot P = f(\text{PHX2/PB})$, Refer to table LXI.	
AFPD = Area (Effective) - Fuel Pump Discharge (Valve)	: in^2
AFPD _{-x/y} : AFPD _{-100/5} = 3.237, AFPD _{-100/6} = 3.237, AFPD _{-100/7} = 3.216	
(PFG) ^{0.05} = (C _{79-x} · PFG) + C _{80-x} , See Section D	: D'less
TFPD = Temperature - (Main) Fuel Pump Discharge, Section C	: $^\circ\text{R}$
RH = Gas Constant, Hydrogen	: $\text{in} \cdot \text{lb}_f/\text{lb}_m \cdot ^\circ\text{R}$
VHX2 = Volume - (Main) Regenerative Nozzle Section - Coolant Side) Heat Exchanger No. 2	: in^3
PFG = PC/(PC at 100% Thrust Level), See Section D	: D'less

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(U) Table LXI. $f_{24} \left[\left(\frac{W}{A} \sqrt{T/AP} \right) \right]$ as a Function of $(FHX2/PB)$

$\frac{FHX2}{PB}$	f_{24}	$\frac{FHX2}{PB}$	f_{24}
1.05	0.06	1.45	0.1315
1.078	0.0748	1.6	0.1370
1.10	0.0829	1.75	0.139
1.15	0.097	1.89	0.1399
1.25	0.115	2.0	0.1399
1.35	0.1252		

$$\gamma = 1.39$$

(U) **H.** OXIDIZER LOW-SPEED INDUCER AND FLOWMETER

EQUATIONS

$$WL = WL_B + WL_C$$

$$NLB = C_{43} \cdot (TQIET - TQLBP)$$

$$TQLBP = f_{27-x}$$

$$\Delta PLB = f_{28-x}$$

$$PLBD = PLI + \Delta PLB$$

$$\Delta PLWM = C_{44} \cdot (WL)^2$$

$$PLIM = PLBD - \Delta PLWM$$

CONSTANTS AND FUNCTIONS

$$PLI = 0.3760 \times 10^2$$

$$C_{43} = 9.55/JLBP = 0.2870 \times 10^3$$

$$C_{44} = (1.496)^2 / \left[\rho_L \cdot (AWLM)^2 \right] = 0.2912 \times 10^{-3}$$

$$f_{27-x} = f(NLB, WL) \quad \text{See figure 633 for } f_{27-100}$$

$$f_{28-x} = f(NLB, WL) = \text{See figure 634 for } f_{28-100}$$

PARAMETER DEFINITIONS

JLBP = Rotor Polar Moment of Inertia,
Oxidizer Low-Speed Inducer

$$: \text{ft} \cdot \text{lb}_f \cdot \text{sec}^2$$

AWLM = Area (Effective), Oxidizer Flowmeter

$$: \text{in}^2$$

ρ_L = Oxidizer Density, (Low-Speed Inducer Inlet)

$$: \text{lb}_m/\text{ft}^3$$

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PARAMETER DEFINITIONS (Continued)

WL = Flow - Oxidizer, (Engine - Total)	: lb _m /sec
WLB = Flow - Oxidizer, Preburner (Injector)	: lb _m /sec
WLC = Flow - Oxidizer, Main Chamber (Injector)	: lb _m /sec
NLP = Speed - Oxidizer Low-Speed Inducer	: rpm
TQLBP = Torque - Oxidizer Low-Speed Inducer (Required)	: lb _f -ft
TQLBT = Torque - Oxidizer Low-Speed Inducer Turbine (Delivered) Section J	: lb _f -ft
ΔPLB = Pressure Rise - Oxidizer Low-Speed Inducer	: lb _f /in. ²
PLBD = Pressure - Oxidizer Low-Speed Inducer Discharge	: lb _f /in. ²
ΔPLWM = Pressure Drop - Oxidizer Flowmeter	: lb _f /in. ²
PLIM = Pressure - Oxidizer Inlet, Main (Pump)	: lb _f /in. ²
PLI = Pressure - Oxidizer, (Engine) Inlet	: lb _f /in. ²

(U) I MAIN OXIDIZER PUMP AND PREBURNER INJECTOR (OXIDIZER SIDE)

EQUATIONS

$$\dot{NLP} = C_{45} \cdot (TQLT - TQLP)$$

$$TQLP = f_{29-x}$$

$$\Delta PLP = f_{30-x}$$

$$PLPD = PLIM + \Delta PLP$$

$$WLB = C_{46} \cdot ALB \cdot \sqrt{(PLPD - PB)}$$

$$ALB = C_{47} + \left| 1 / \sqrt{C_{90} + [1 / (ALDV)^2]} \right|$$

CONSTANTS AND FUNCTIONS

$$C_{45} = 9.55/JLP = 9/40 \times 10^2$$

$$C_{46-y} : C_{46-5} = 5.470, C_{46-6} = 5.524, C_{46-7} = 5.539$$

$$C_{47} \Delta PRI = 0.4150 \times 10^{-1}$$

$$C_{90} = 1 / (ASEC)^2 = 0.8044$$

$$f_{29-x} = f(NLP, WL) = \text{See figure 635 for } f_{29-100}$$

$$f_{30-x} = f(NLP, WLB) = \text{See figure 636 for } f_{30-100}$$

$$ALB_{-x/y} : ALB_{-100/5} = 0.3237, ALB_{-100/6} = 0.3939, ALB_{-100/7} = 0.5284$$

$$ALDV_{-x/y} : ALDV_{-100/5} = 0.2917, ALDV_{-100/6} = 0.3715, ALDV_{-100/7} = 0.3412$$

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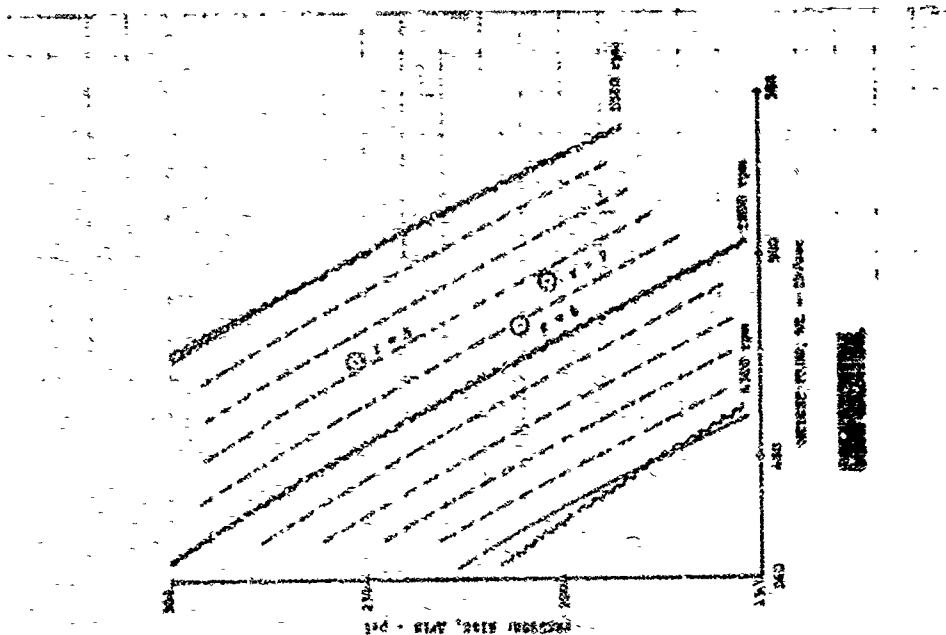


Figure 634. Oxidizer Low-Speed Inducer Characteristics Curve (Sheet 2)

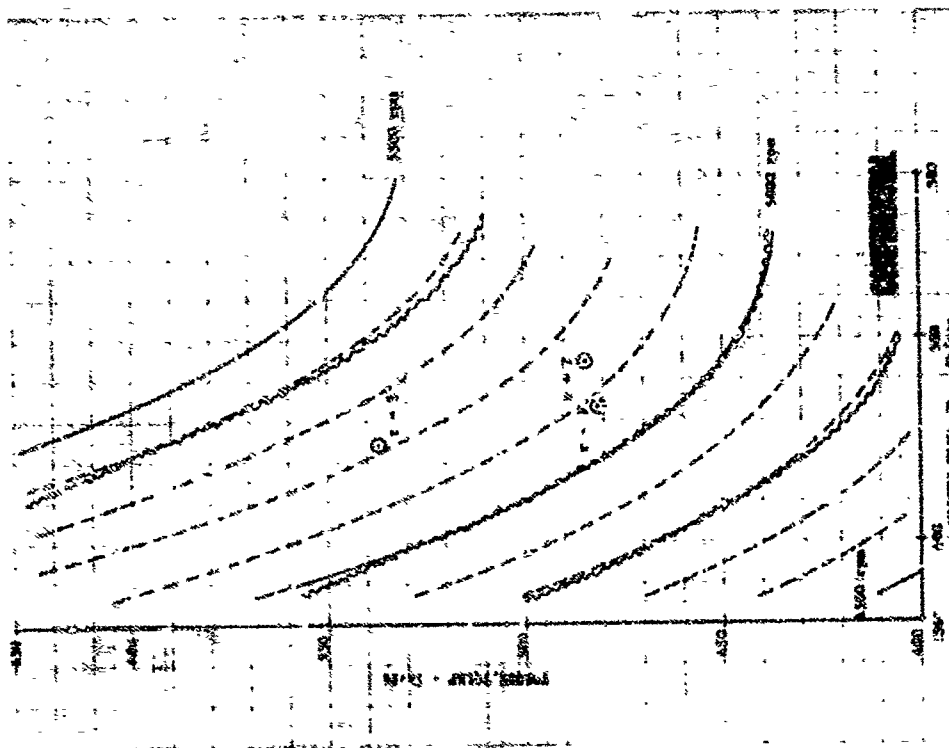


Figure 633. Oxidizer Low-Speed Inducer Characteristics Curve (Sheet 1)

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PARAMETER DEFINITIONS

NLP = Speed - (Main) Oxidizer (Turbo) Pump	: rpm
TQLP = Torque - (Main) Oxidizer Pump, (Required)	: ft-lb _f
TOLT = Torque - (Main) Oxidizer Turbine, (Delivered) (See Section K)	: ft-lb _f
ΔFLP = Pressure Rise - Oxidizer Pump (Main)	: lb _f /in. ²
PLPD = Pressure - Oxidizer, (Main) Pump Discharge	: lb _f /in. ²
PLIM = Pressure - Oxidizer Inlet, Main (Pump)	: lb _f /in. ²
WLB = Flow - Oxidizer, Preburner (Injector)	: lb _f /sec
ALB = Area - Oxidizer, Preburner (Line and Control - Effective)	: in. ²
APMI = Area - Primary (Preburner Injector) (Effective)	: in. ²
ASEC = Area - Secondary (Preburner Injector) (Effective)	: in. ²
ALDV = Area - Oxidizer, (Flow) Divider Valve, (Effective)	: in. ²
PB = Pressure - Preburner (See Section K)	: lb _f /in. ²
PLPD = Pressure - Oxidizer, Main Pump Discharge	: lb _f /in. ²
JLP = Rotor Polar Moment of Inertia, Main Oxidizer Pump	: ft lb _f sec ²

(U) **J.** TURBINE - OXIDIZER LOW-SPEED INDUCER AND MAIN CHAMBER OXIDIZER INJECTOR

EQUATIONS

$$\begin{aligned}\sqrt{\Delta HLB} &= C_{48} \cdot (WLC/ALBT) \\ TQLBT &= WLC \cdot \left[(C_{49-x} \cdot \sqrt{\Delta HLB}) - (C_{50-x} \cdot NLB) \right] \\ WLC &= C_{51} \cdot ALOX \cdot \sqrt{(PLPD - PC)} \\ ALOX &= 1 / \sqrt{C_{52} + (1/ALC^2) + (1/ALBT^2)}\end{aligned}$$

CONSTANTS AND FUNCTIONS

where:

$$\begin{aligned}C_{48-x/y} &: C_{48-100/5} = 0.9675 \times 10^{-2}, C_{48-100/6} = 0.9489 \times 10^{-2} \\ &C_{48-100/7} = 0.9435 \times 10^{-2} \\ C_{49-x} &: C_{49-100} = 2.919\end{aligned}$$

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CONSTANTS AND FUNCTIONS (Continued)

$$C_{50-x} : C_{50-100} = 0.2677 \times 10^{-3}$$

$$C_{51-x/y} : C_{51-100/5} = 5.470, C_{51-100/6} = 5.524, \\ C_{51-100/7} = 5.539$$

$$C_{52-x/y} = (1/ALIJ^2) : C_{52-100/5} = 0.1778, C_{52-100/6} = 0.1742, \\ C_{52-100/7} = 0.1714$$

$$ALOX-x/y : ALOX-100/5 = 1.034, ALOX-100/6 = 1.276, \\ ALOX-100/7 = 1.550$$

$$ALBT-x/y : ALBT-100/5 = 2.514, ALBT-100/6 = 2.862, \\ ALBT-100/7 = 3.058$$

$$ALC-x/y : ALC-100/5 = 1.292, ALC-100/6 = 1.774, \\ ALC-100/7 = 2.694$$

PARAMETER DEFINITIONS

ΔH_{LBT} = Enthalpy Drop - Oxidizer Low-Speed Inducer Turbine : Btu/lb_m

$ALBT$ = Area - Oxidizer Low-Speed Inducer Turbine (Variable) : $in.^2$

$TQLBT$ = Torque - Oxidizer Low-Speed Inducer Turbine (Delivered) : $ft-lb_f$

WLC = Flow - Oxidizer, (Main) Chamber (Injector) : lb_m/sec

$ALOX$ = Area - Oxidizer (Effective Total Area - Main Chamber Injector and Controls) : $in.^2$

ALC = Area - Oxidizer (Main Chamber Line) Control : $in.^2$

$ALIJ$ = Area - Oxidizer (Main Chamber) Injector : $in.^2$

NLB = Speed - Oxidizer Low-Speed Inducer : rpm

$PLPD$ = Pressure - Oxidizer Low-Speed Inducer Discharge : $lb_f/in.^2$

PC = Pressure - Main Chamber : $lb_f/in.^2$

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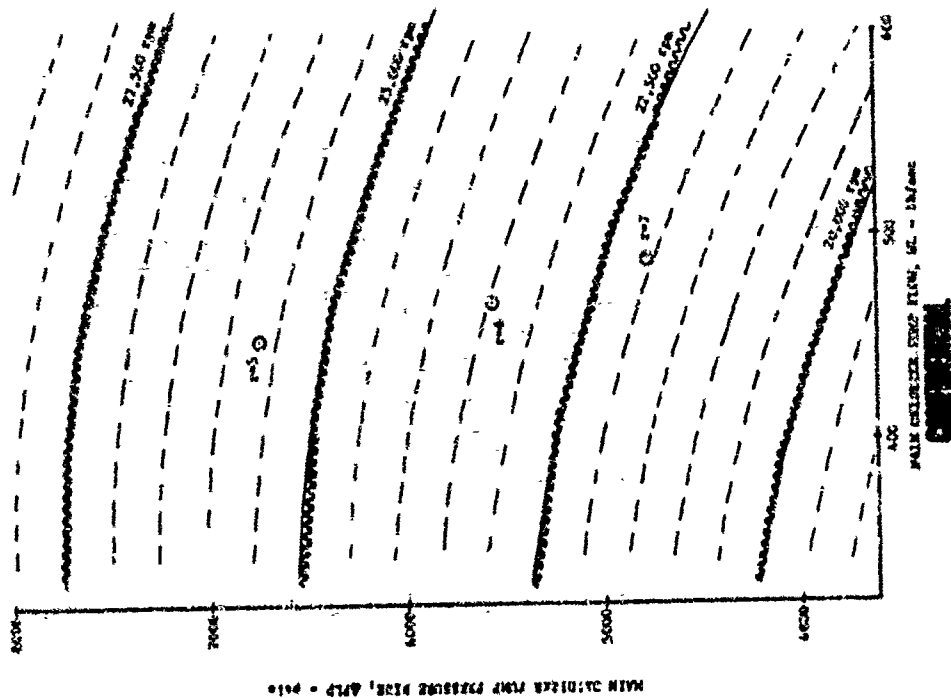


Figure 636. Main Oxidizer Pump and Preburner Injector Characteristics Curve (Sheet 2)

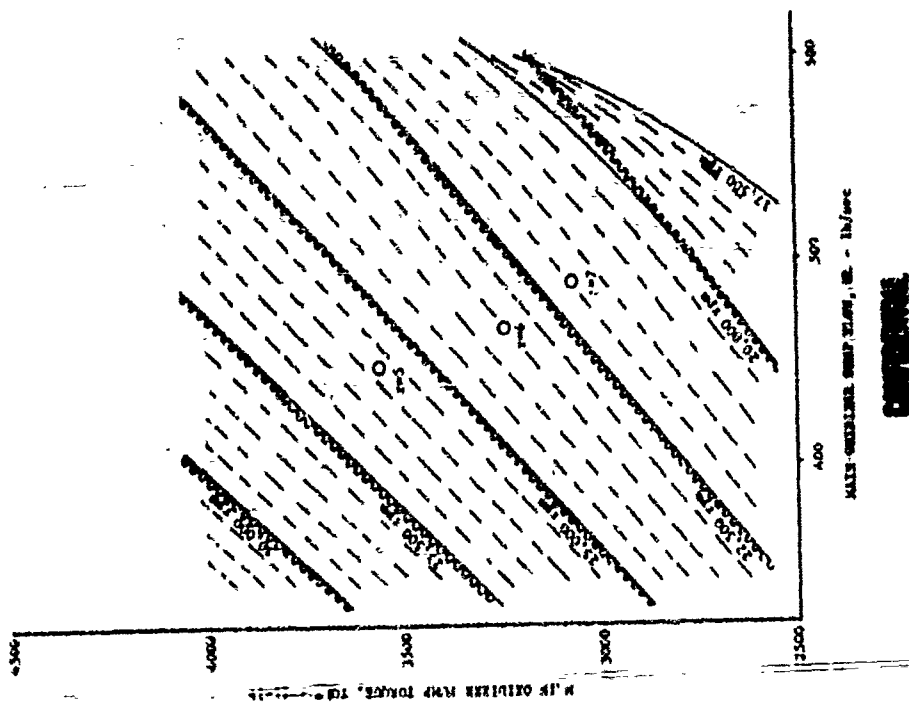


Figure 635. Main Oxidizer Pump and Preburner Injector Characteristics Curve (Sheet 1)

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(U) **K.** PREBURNER AND MAIN TURBINES

EQUATIONS

$$\begin{aligned}\dot{P}_B &= C_{56} \cdot f_{31} \cdot (WFB + WLB - WFT - WLT) \\ FLO &= f_{32} \cdot f_{33} \cdot PB \\ \sqrt{\Delta HP} &= f_{34} \cdot f_{35} \\ OFB &= WLB/WFB \\ WFT &= C_{57-x} \cdot FLO \\ TQFT &= WFT \cdot \left[(C_{58-x} \cdot \sqrt{\Delta HP}) - (C_{59-x} \cdot NFP) \right] \\ WLT &= C_{60-x} \cdot FLO \\ TQLT &= WLT \cdot \left[(C_{61-x} \cdot \sqrt{\Delta HP}) - (C_{62-x} \cdot NLP) \right]\end{aligned}$$

CONSTANTS AND FUNCTIONS

where:

$$\begin{aligned}C_{56} &= 1/VB = 0.7143 \times 10^{-3} \\ C_{57-x} &= AFT: C_{57-100} = 1.581 \\ C_{58-x/y} &= C_{58-100/5} = 4.715, C_{58-100/6} = 4.678, \\ &C_{58-100/7} = 4.618 \\ C_{59-x} &= C_{59-100} = 0.9517 \times 10^{-2} \\ C_{60-x} &= ALT: C_{60-100} = 0.7643 \\ C_{61-x/y} &: C_{61-100/5} = 6.324, C_{61-100/6} = 6.255, \\ &C_{61-100/7} = 6.165 \\ C_{62-x} &: C_{62-100} = 0.2205 \times 10^{-2} \\ f_{31} &: RB \cdot TB = f(OFB) \text{ Refer to table LXII} \\ f_{32} &: 1/\sqrt{RB \cdot TB} = f(OFB) \text{ Refer to table LXIII} \\ f_{33} &: W \cdot \sqrt{R \cdot T/(A \cdot P)} = f(PB/FMJ) \text{ Refer to table LXIV} \\ f_{34} &: \sqrt{CPB \cdot TE} = f(OFB) \text{ Refer to table LXV} \\ f_{35} &: \sqrt{1 - (FMJ/PB)^{\gamma-1/\gamma}} = f(FMJ/PB) \text{ Refer to table LXVI.}\end{aligned}$$

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(U) Table LXII. $f_{31}(RB \cdot TB)$ as a Function of (OFB)

OFB	f_{31}	OFB	f_{31}
0.4	4.85	0.9	7.90
0.5	5.70	1.0	8.27
0.6	6.40	1.2	8.90
0.7	7.00	1.35	9.26
0.8	7.50	1.45	9.55

(U) Table LXIII. $f_{32}(1/\sqrt{RB \cdot TB})$ as a Function of (OFB)

OFB	f_{32}	OFB	f_{32}
0.4	1.435	0.8	1.155
0.45	1.365	0.9	1.124
0.5	1.320	1.1	1.078
0.6	1.250	1.3	1.043
0.7	1.195	1.5	1.022

(U) Table LXIV. $f_{33}(W\sqrt{RT}/(AP))$ as a Function of $(PB/PMIJ)$

$\gamma = 1.37$			
$\frac{PB}{PMIJ}$	f_{33}	$\frac{PB}{PMIJ}$	f_{33}
1.0	1.00	1.40	12.3376
1.03	5.00	1.50	12.8195
1.10	7.9574	1.70	13.2747
1.15	9.3	1.8868	13.3645
1.20	10.2685	2.000	13.3645
1.30	11.5572		

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(U) Table LXV. $f_{34}(\sqrt{CPB \cdot TB})$ as a Function of (OFB)

OFB	f_{34}	OFB	f_{34}
0.4	46.0	0.8	57.0
0.5	49.4	0.9	58.8
0.6	52.4	1.0	60.3
0.7	54.9	1.4	65.5

(U) Table LXVI. $f_{35}(\sqrt{1 - (PMLJ/PB)^{\gamma-1/\gamma}})$ as a Function of (PMLJ/PB)

$\gamma = 1.37$			
$\frac{PMLJ}{PB}$	f_{35}	$\frac{PMLJ}{PB}$	f_{35}
0.0	1.000	0.7	0.3030
0.02	0.82	0.8	0.2418
0.05	0.7447	0.9	0.1674
0.1	0.6804	0.95	0.1172
0.2	0.5937	1.00	0.0000
0.3	0.5268		

PARAMETER DEFINITIONS

TB	Temperature - Preburner Combustion Chamber	: °R
PB	Pressure - Preburner	: lb_f/in^2
VB	Volume - Preburner Combustion Chamber	: in^3
FLO	Intermediate Calculation for Turbine Gas Flow	: ---
RB	Preburner Combustion Products - Gas Constant	: $in.-lb_f/lb_m \cdot ^\circ R$
ΔH_P	Enthalpy Drop - Main Turbines	: Btu/lb_m
PMLJ	Pressure - Main Fuel Injector (Transition) (Section [L])	: lb_f/in^2
OFB	Mixture Ratio - Preburner	: D'less
WFB	Flow - (Fuel) - Preburner Injector (Section [G])	: lb_m/sec
WFT	Flow (Gas) - (Main) Fuel (Turbopump) Turbine	: lb_m/sec
NFP	Speed - Main Fuel Turbopump (Section [C])	: rpm
TQFT	Torque - (Main) Fuel (Turbopump) Turbine	: lb_f-ft

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PARAMETER DEFINITIONS (Continued)

WLB = Flow - (Oxidizer) - Preburner Injector (Section I)	: lb _m /sec
WLT = Flow (Gas) - (Main) Oxidizer (Turbopump) Turbine	: lb _m /sec
NLP = Speed - Main Oxidizer Turbopump (Section I)	: rpm
TQLT = Torque - (Main) Oxidizer (Turbopump) Turbine	: lb _f -ft
ALT = Effective Area - Main Oxidizer Turbine	: in ²
AFT = Effective Area - Main Fuel Turbine	: in ²
CPB = Specific Heat (Constant Pressure) Preburner	: Btu/lb _m -°R
γ = Ratio of Specific Heats	: D'leas

(U) I. TRANSITION CASE, MAIN FUEL INJECTOR AND MAIN CHAMBER

EQUATIONS

$$\begin{aligned}
 \dot{M}_{IJ} &= C_{63} \cdot f_{31} \cdot (WFT + WLT + WTC - WMIJ) \\
 WMIJ &= C_{64-x} \cdot f_{32} \cdot f_{38} \cdot IMIJ \\
 WFIJ &= WMIJ \cdot (WFB + WTC) / (WFB + WTC + WLB) \\
 WLIJ &= WMIJ \cdot (WLB) / (WFB + WTC + WLB) \\
 OFC &= (WLIJ + WLC) / (WFIJ + WFBT) \\
 \dot{P}_C &= C_{65} \cdot f_{39} \cdot (WMIJ + WFBT + WLC - WOUT) \\
 WOUT &= C_{66-x} \cdot f_{40} \cdot PC \\
 OFE &= WL/WFI
 \end{aligned}$$

CONSTANTS AND FUNCTIONS

$$\begin{aligned}
 C_{63} &= 1/VMIJ = 0.5000 \times 10^{-3} \\
 C_{64-x/y} &= AMIJ : C_{64-100/5} = 5.068, C_{64-100/6} = 4.96, \\
 &C_{64-100/7} = 4.935 \\
 C_{65} &= 1/VC = 0.8333 \times 10^{-3} \\
 C_{66-x/y} &= AT \cdot g : C_{66-100/5} = 0.1496 \times 10^4, \\
 &C_{66-100/6} = 0.1502 \times 10^4, \\
 &C_{66-100/7} = 0.1514 \times 10^4
 \end{aligned}$$

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CONSTANTS AND FUNCTIONS (Continued)

$$f_{31} = RB \cdot TB = f(OFB), \text{ See Preburner, Section K}$$

$$f_{32} = 1/\sqrt{RB \cdot TB} = f(OFB), \text{ See Preburner, Section K}$$

$$f_{39} = RC \cdot TC = f(OFC), \text{ Refer to table LXVII}$$

$$f_{40} = 1/(\eta_c^* \cdot C^*) = f(OFC), \text{ Refer to table LXVIII}$$

$$f_{38} = W \cdot \sqrt{R \cdot T} / (A \cdot P) = f(PMIJ/PC) \text{ Refer to table LXIX.}$$

(U) Table LXVII. $f_{39}(RC \cdot TC)$ as a Function of (OFC)

OFC	f_{39}	OFC	f_{39}
2.0	10.23	4.25	10.05
3.0	10.20	4.50	9.95
4.0	10.12	10.00	7.08

(U) Table LXVIII. $f_{40}[1/(\eta_c^* \cdot C^*)]$ as a Function of (OFC)

OFC	f_{40}	OFC	f_{40}
2.0	0.0001295	4.75	0.0001271
2.35	0.0001275	5.25	0.0001283
2.75	0.0001263	5.75	0.0001300
3.25	0.0001259	7.50	0.0001390
3.75	0.0001259	10.00	0.0001538
4.75	0.0001262		

(U) Table LXIX. $f_{38}[W \sqrt{RT}/(AP)]$ as a Function of (PMIJ/PC)

$\frac{PMIJ}{PC}$	f_{38}	$\frac{PMIJ}{PC}$	f_{38}
1.0	0.00	1.40	12.3376
1.03	5.00	1.50	12.8195
1.10	7.9574	1.70	13.2747
1.15	9.3	1.8869	13.3645
1.20	10.2685	2.000	13.3645
1.30	11.5572		

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PARAMETER DEFINITIONS

PMIJ = Pressure - Main (Gas) Injector (Transition Case)	: lb_f/in^2
$\dot{P}MIJ$ = Pressure Change Rate - Main (Gas) Injector	: $d(PMIJ)/dt$
WMIJ = Flow (Gas) - Main (Gas) Injector	: lb_m/sec
WFB = Flow - Fuel, Preburner (Injector), See Section G	: lb_m/sec : lb_m/sec
WFIJ = Flow - Fuel, (Main Gas) Injector	: lb_m/sec
WTC = Flow (Fuel), Turbine Cooling (Etc.), See Section C	: lb_m/sec
OFC = Mixture Ratio (Main) Chamber (Chamber-Overall)	: D'less
η_c^* = Characteristic Velocity Efficiency	: D'less
PC = Pressure - (Main) Chamber	: lb_f/in^2
C^* = Characteristic Velocity	: ft/sec
WOUT = Flow - (Gas), (Main Chamber) Out	: lb_m/sec
WFT = Flow (Gas) - (Main) Fuel (Turbopump) Turbine Section K	: lb_m/sec
VMIJ = Volume - Main (Gas) Injector	: in^3
WLT = Flow (Gas) - (Main) Oxidizer (Turbopump) Turbine Section K	: lb_m/sec
VC = Volume - (Main) Chamber	: in^3
AMIJ = Area - (Effective) - Main Chamber Gas Injector	: in^2
OFE = Mixture Ratio - Engine (Overall)	: D'less
AT = Area - (Effective) - Throat, Main Chamber	: in^2
g = Gravitational Constant	: ft/sec^2
RC = Gas Constant, Main Chamber Gas	: $\text{in-lb}_f/\text{lb}_m\text{-}^\circ\text{R}$
TC = Temperature (Gas) - (Main) Chamber	: $^\circ\text{R}$
WLB = Flow - Oxidizer, Preburner (Injector), (See Section C)	: lb_m/sec
WLIJ = Flow - Oxidizer, (Main Gas) Injector	: lb_m/sec

1. EQUATION ASSUMPTIONS AND BASES

(1) A description of the assumptions and the bases for the equations used in the analog simulation of the Engine Cycle AF 1111A is included here along with a description of the technique employed in activating this simulation on the P&WA-FRDC analog computer.

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(U) Certain assumptions have been made in arriving at the equations used in this analog simulation. A summary of the equations used is shown in figure 637, and the more significant assumptions made, along with the bases of the equations involved are discussed in the following paragraphs.

1. Fluid Flow - General

(U) When simulating the fluid flows through the engine, the oxidizer (oxygen) is considered to be liquid from the tank through the oxidizer injectors, and the fuel (hydrogen) is considered to be liquid from the tank up to the inlet of the nozzle sections, (heat exchangers). Beyond these points the fluids are considered to be gases.

(U) Where Bernoulli's equation is used in determining flow rates or pressure drops, the flow is assumed to be steady, irrotational, one-dimensional, frictionless and adiabatic; and differences in component elevation as well as any influence of external forces such as vehicle accelerations are disregarded.

(U) Certain line pressure drops have been consolidated with adjacent component pressure drops in this Engine Analog Simulation to reduce the amount of analog computer equipment involved. The following areas of consolidation should be noted:

- a. Upstream line loss is included with the transpiration cooling flow orifice (ACC)
- b. Downstream line loss is included with the main fuel pump discharge valve (AFPD)
- c. Upstream line loss is included with the fuel preburner injector (AFB)
- d. Downstream line loss is included with the transpiration heat exchanger
- e. Downstream line loss is included with the fuel inducer turbine
- f. Upstream line loss is included with the oxidizer preburner injector (ALB)
- g. Upstream line loss is included with the oxidizer inducer turbine
- h. Upstream line loss is included with the main oxidizer injector (ALIJ)
- i. Upstream loss is included with the main turbines
- j. Upstream loss is included with the main gas injector (AMIJ).

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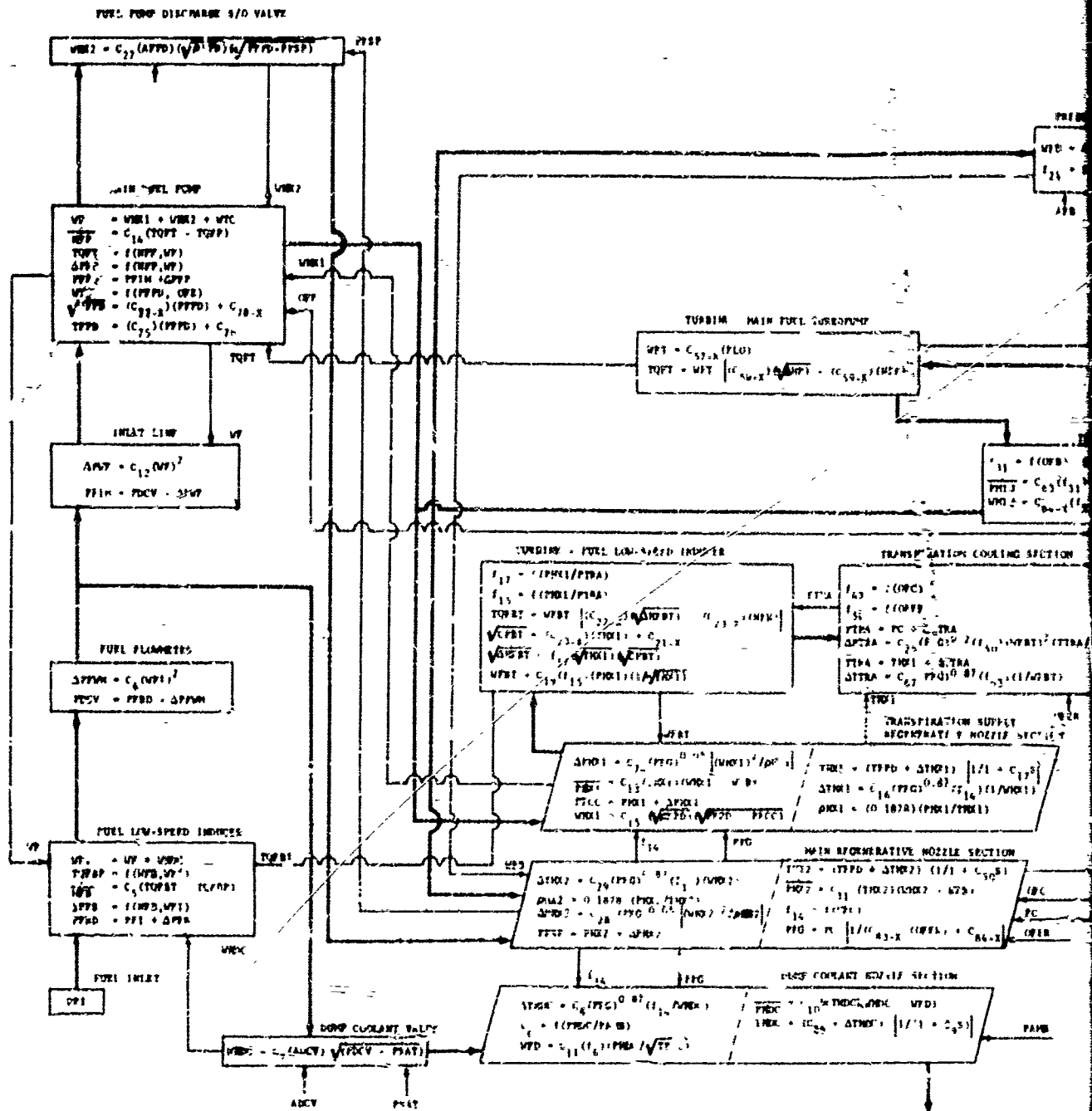


Figure 6-17. Logic Diagram Used for Analog Simulation



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(C) Because flow through the dump coolant valve (ADCV) is subject to phase change during operation, particularly at the lower (20%) thrust levels, this valve is treated as a cavitating venturi type of valve with saturation pressure assumed to exist at the throat. The effective throat area (ADCV) was established on this basis.

(U) All flow areas used in the analog simulation were calculated using cycle flows and, where applicable, the consolidated pressure drops noted above.

2. Liquid Flow

(U) All liquid flow in this program is considered incompressible. The liquid flow rates up to and through the fuel valves and the oxidizer injectors are simulated in the following form by the combination of Bernoulli's equation and the Continuity equation.

$$\text{Bernoulli's equation: } V = \sqrt{\frac{2g_c \Delta P}{\rho}}$$

$$\text{Continuity equation: } \dot{w} = \rho A_{\text{ed}} V$$

therefore:

$$\dot{w} = \frac{A_{\text{ed}}}{1.496} \sqrt{\rho \Delta P}$$

where:

$\dot{w}$ = Flow rate, lb_m/sec

A_{ed} = Effective area, in.²

ρ = Density, lb_m/ft.³

ΔP = Pressure drop, lb_f/in.²

g_c = Gravitational constant, ft-lb_m/lb_f-sec²

3. Gas Flow

(U) Bernoulli's equation as formulated for an isentropic, perfect gas flow process is used in determining the gas flow rates through the preburner - fuel and main-gas injectors as well as through the main turbines and the fuel boost turbine. Bernoulli's equation in this gas flow form is:

$$\frac{\dot{w} \cdot \sqrt{R \cdot T_1}}{A_{\text{ed}} \cdot P_1} = \frac{\sqrt{2g_c \gamma / (\gamma - 1)}}{(P_1/P_2)^{\gamma+1/2\gamma}} \left[(P_1/P_2)^{\gamma-1/\gamma} - 1 \right] = \text{Gas Flow Parameter}$$

where:

$\dot{w}$ = Flow rate, lb/sec

T_1 = Upstream temperature, °R

R = Gas constant, ft-lb_f/lb_m-°R

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A_{cd} = Effective area, in.^2

P_1 = Upstream pressure, $\text{lb}_f/\text{in.}^2$ (abs)

γ = Specific heat ratio (C_p/C_v), d'less

g_c = Gravitational constant, $\text{ft-lb}_m/\text{lb}_f\text{-sec}^2$

(U) For hydrogen gas flow, the hydrogen gas constant, R , is constant and is included when calculating the hydrogen gas flow parameter as follows:

$$\frac{\dot{w} \cdot \sqrt{T_1}}{A_{cd} \cdot P_1} = \frac{\sqrt{\left[2g_c \gamma / R(\gamma-1)\right] \left[(P_1/P_2)^{\gamma-1/\gamma} - 1\right]}}{(P_1/P_2)^{\gamma+1/2\gamma}} = \text{Hydrogen Gas Flow Parameter}$$

These flow parameters are calculated for the pertinent specific heat ratio (γ) and as functions of the total to static pressure ratio = P_1/P_2 . It should be noted that in the analog simulation the pressure ratios used are total-to-total, which are not consistent with the flow parameters definition given above. This incompatibility has been found to be negligible in most cases, and when it was necessary it was compensated for in the program constants (i.e., turbine torque constants). This is the basis for these functions as used in the analog simulation.

(U) It was assumed that the preburner temperature can be used when calculating the main-gas injector flow (AMIJ) by slightly (3.5%) changing the main-gas injector area (AMIJ). The small sacrifice in dynamic accuracy resulting from this assumption is compensated for by the elimination of inaccuracies in the additional operations otherwise required to determine and incorporate the turbine downstream temperature into this gas flow calculation.

4. Gas Pressures-

(U) The instantaneous gas pressures within such engine volumes as the preburner transition case, main chamber and the nozzle section fuel passages are obtained by continuously integrating the following expression.

$$\dot{P} = (R/V)T(\dot{w}_1 - \dot{w}_2)$$

where:

$\dot{P}$ = Rate of change of pressure = $\text{lb}_f/\text{in.}^2\text{-sec}$

V = Volume = in.^3

T = Temperature = $^{\circ}\text{R}$

$\dot{w}_1$ = Flow rate, into volume = lb_m/sec

$\dot{w}_2$ = Flow rate, out of volume = lb_m/sec

R = Gas constant = $\text{in.-lb}_f/\text{lb}_m\text{-}^{\circ}\text{R}$

(U) IC (initial condition) values are assigned to these pressures before the computer is switched to "compute." Thereafter these pressure values are controlled by computer integration.

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(U) The above differential expression is derived from the perfect gas law as follows:

$$P = RTm/V$$

where:

P = Pressure - $\text{lb}_f/\text{in}^2\text{-sec}$

R = Gas constant - $\text{in.-lb}_f/\text{lb}_m\text{-}^\circ\text{R}$

T = Temperature - $^\circ\text{R}$

m = Mass density - lb_m/in^3

V = Volume - in^3

Differentiating with respect to time

$$d(P)/dt = (RT/V)d(m)/dt + (mR/V)d(T)/dt$$

Because the temperature of an ideal gas is a function of the gas energy only, no change in stagnation temperature results from consideration of reversible adiabatic flow in and out of the volume. Therefore, dT/dt is zero and consequently

$$\dot{P} = d(P)/dt = (RT/V)dm/dt = (RT/V)(\dot{w}_1 - \dot{w}_2).$$

5. Pumps

(U) The pump characteristics are depicted by the Pressure Rise vs Flow and Torque vs Flow curves that are bivariate with speed. These curves are provided for each of the four pumps.

(U) Angular momentum is used in determining the pump speeds as follows:

$$J \cdot \dot{N} = T$$

where:

T = Torque

$\dot{N}$ = Time rate of change of angular velocity (angular acceleration)

J = Polar moment of inertia for turbopump rotor assembly

therefore:

$$\dot{N} = T/J = (\text{turbine torque} - \text{pump torque})/J.$$

(U) The analog computer continuously integrates this expression to develop the instantaneous pump speeds.

6. Regenerative Nozzle Sections (Heat Exchangers)

(U) The expressions for pressure drop and temperature rise of the coolant (fuel) passing through the different nozzle sections and the main chamber transpiration cooling section are empirically derived. This includes the time constant which adds a dynamic factor to these expressions.

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(U) Another dynamic factor is introduced in the simulation by considering the equivalent volume of the nozzle sections and their manifolds as existing between two flow-regulating areas. The instantaneous gas pressures within these volumes are developed from the conservation of mass relationship as discussed in paragraph 4, Gas Pressures. The relatively small volume of the main chamber transpiration cooling section has been added to the volume of the transpiration supply nozzle section to provide for its dynamic effect on the transpiration flow.

7. Preburner and Main Turbines

(U) Pressure in the preburner is derived from the conservation of mass relationship discussed in paragraph 4, Gas Pressures.

(U) The FLO expression is an intermediate step in the calculation of gas flow through the main turbine areas as noted in paragraph 3, Gas Flow.

(U) The adiabatic flow of gases through a turbine was expressed as:

$$\Delta h' = C_p (T_1 - T_2)$$

where $\Delta h'$ is the ideal enthalpy drop across the turbine, C_p is the specific heat of the gas at constant pressure, and T_1 , T_2 are the turbine inlet and outlet total temperatures. For isentropic flow of a perfect gas, the total pressure ratio corresponding to the temperature ratio is given by the equation

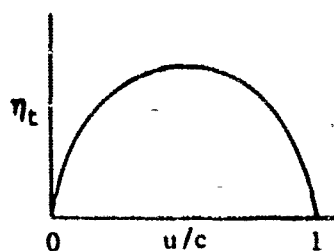
$$(P_1/P_2) = (T_1/T_2)^{\gamma/(\gamma-1)}$$

therefore, combination of these equations gives

$$\Delta h' = C_p T_1 \left[1 - \left(\frac{P_2}{P_1} \right)^{(\gamma-1)/\gamma} \right]$$

This is the expression used in simulating the ideal enthalpy drop across the turbines.

(U) The efficiency of an impulse turbine is defined in terms of the ratio of mean wheel speed to the ideal nozzle exit velocity (u/c) as shown by the following parabolic curve.



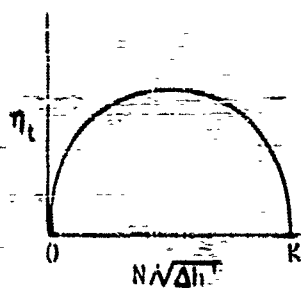
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The parameter u/c is proportional to $(N/\sqrt{\Delta h'})$ where N is the turbine speed in rpm and $\Delta h'$ is the ideal enthalpy drop across the turbine.

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(U) Therefore, the efficiency can be expressed in terms of the parameter $(N/\sqrt{\Delta h^*})$ by a similar parabolic curve as follows:



The equation defining this curve will have a paraboloid characteristic originating at the point (0,0) as follows:

$$\eta_T = K_2 \frac{N}{\sqrt{\Delta h^*}} - K_1 \frac{N^2}{\Delta h^*} \quad (9)$$

Turbine torque is defined by the relationship:

$$TQ = \frac{\Delta h^* \times \eta_T \times W_T}{0.00013466 \times N} \quad (10)$$

where W_T = turbine mass flow, lb_m/sec and the constant 0.00013466 is the units conversion required to convert rpm to rad/sec and $ft \cdot lb_f$ to $ft \cdot lb_m$.

Combination of equations (9) and (10) yields:

$$TQ = \frac{\Delta h^* \times W_T}{0.00013466 \times N} K_2 \frac{N}{\sqrt{\Delta h^*}} - K_1 \frac{N^2}{\Delta h^*} \quad (11)$$

which can be reduced to the form

$$TQ = W_T \left[\left(K_2 \cdot \sqrt{\Delta h^*} \right) - \left(K_1 \cdot N \right) \right]$$

This is the form in which the torque equation is used in the engine analog simulation.

8. Main Combustion Chamber

(U) The characteristic velocity, c^* , is defined by the relation:

$$c^* = A_c P_c / \dot{w}$$

where:

A_c = Throat area, in^2

P_c = Chamber pressure, lb_f/in^2

g = Gravitational constant, $32.17 \text{ ft} \cdot lb_m / lb_f \text{ sec}^2$

$\dot{w}$ = Mass flow rate, lb_m/sec

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And from the choking equation for a perfect gas

$$c^* = A_c P_c g/\dot{w} = \sqrt{\frac{RT_g}{M}} f(\gamma)$$

where:

R = Universal gas constant, 1544 ft. lb./lb-mole °R

T_g = Total gas temperature = °R

M = Molecular weight of gas.

The effect of γ is slight, so c^* depends primarily upon the propellant combination, which determines the stagnation temperature and molecular weight of the product gases. The last expression gives a theoretical value of c^* as calculated from the properties of the hot combustion gases and which is a function of mixture ratio.

(U) The characteristic exhaust velocity efficiency, η_c^* , is the ratio of the actual c^* (realized) to the theoretical c^* . The theoretical c^* values used in this simulation are based on calculations assuming complete chemical reaction within the chamber and a homogeneous gas mixture with equilibrium expansion to the throat. The η_c^* values are based on empirical results based on P&WA-FRDC test experience. Therefore, in terms of theoretical c^* and η_c^* , the above expression becomes

$$c^* = \eta_c^* A_c P_c g/\dot{w}$$

or

$$\frac{(\dot{w}/F)}{A_c P_c} = \frac{1}{c^* \eta_c^*} = \frac{1}{66} f_{40} \text{ PC}$$

which is the analog equation used to calculate flow out of the main chamber. The function, $1/c^* \eta_c^*$, is based on the particular propellant combination used and on predicted efficiencies and can be plotted as a function of the oxidizer to fuel mixture ratio.

(U) Because the gas flow from the preburner (volume) through the turbines (area), transition case (volume), and main gas injector (area) into the main chamber is subject to the dynamic of flow through two volumes and areas in series, and since these dynamics have been shown to have a significant effect on the main chamber mixture ratio (OFC), they were considered in the equation for OFC.

E. DIGITAL TRANSIENT PROGRAM

(U) The following sections describe the digital computer dynamic simulation of the demonstrator engine cycle. The engineering/mathematical representations of the engine components are essentially the same as those incorporated in the previously described analog program. Basic differences in the two programs are the manner in which the time solutions are obtained and the operational procedures involved in execution of simulations. The analog solutions are based on continuous time, while the digital program solutions are based on numerical integration techniques. Results from both simulations show excellent agreement of data over comparable operations.

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(H) This program was written for the IBM System 360 Model 65 and requires the following equipment:

1. Fortran IV, Level II
2. 100,000 Bytes or 25,000 words of storage
3. An additional 16,000 Bytes or 4000 words for systems subroutines, e.g., I/O control, exponential, zero, log, max - min
4. Extended binary coded decimal language
5. Multiple entry and return for subroutines
6. Sense light subroutine for testing
7. 120 characters per line of output
8. Initialization of data tables in assembly language
9. Assembly language subroutine to count arguments in a calling list.

1. Program Input Parameters

(U) The digital program input parameters are used to establish (1) the duration of the engine transient calculation (STIM-To), (2) time increments to be taken during the transient (DT), and (3) optional data print time (PTIM).

1. To Engine Transient Start Time
2. DT Time increment
3. PTIM Increment of T Between Printout
4. STIM Engine Transient Stop Time

(U) The following engine control areas will be input through a separate subroutine and constitute the transient forcing parameters around which the control system will be developed.

5. ALDV	Control Area	Preburner oxidizer flow divider valve	in ²
6. ALS	Control Area	Main oxidizer line control valve	in ²
7. ALBT	Control Area	Oxidizer low-speed inducer turbine	in ²
8. AFPD	Control Area	Fuel valve at pump discharge	in ²
9. AFB	Control Area	Preburner fuel injector	in ²
10. ADCV	Control Area	Dump coolant control valve	in ²

(C) Engine initial conditions are required to establish the starting point of the transients and are input for each case. The following engine initial conditions for 20, 50 and 100 percent are contained in table LXX.

11. ORV	Mixture Ratio	Into the low-speed inducers	lb/lb _{ss}
12. PBS	Pressure	Preburner injector face (static)	lb _f /in ²
13. PB	Pressure	Preburner total	lb _f /in ²

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14.	PC	Pressure	Main chamber throat total	lb _f /in ²
15.	PCP	Pressure	Main chamber throat total (steady-state)	lb _f /in ²
16.	PCS	Pressure	Main chamber injector face (static)	lb _f /in ²
17.	PHDC	Pressure	Dump coolant heat exchanger discharge	lb _f /in ²
18.	PHX2	Pressure	Downstream of the main nozzle regenerative heat exchanger	lb _f /in ²
19.	PMIJ	Pressure	Upstream of main chamber injector	lb _f /in ²
20.	PTCV	Pressure	Turbine cooling flow volume	lb _f /in ²
21.	PTRA	Pressure	Transpiration section supply	lb _f /in ²
22.	RHX1	Density	Small nozzle heat exchanger discharge	lb _m /ft ³
23.	RHX2	Density	Main nozzle heat exchanger discharge	lb _m /ft ³
24.	ROFB	Density	Fuel low-speed inducer discharge	lb _m /ft ³
25.	ROFI	Density	Fuel Pump Interstage	lb _m /ft ³
26.	RFPD	Density	Main fuel pump discharge (2nd stage)	lb _m /ft ³
27.	ROL	Density	Main oxidizer pump discharge	lb _m /ft ³
28.	ROLE	Density	Oxidizer low-speed inducer discharge	lb _m /ft ³
29.	NFP	Speed	Main fuel turbopump	rpm
30.	NFB	Speed	Fuel low-speed inducer	rpm
31.	OLP	Speed	Main oxidizer turbopump	rpm
32.	NLB	Speed	Oxidizer low-speed inducer	rpm
33.	THDC	Temperature	Dump cooling nozzle exit	°R
34.	TFBTD	Temperature	Fuel low-speed inducer turbine discharge	°R
35.	THX1	Temperature	Small nozzle heat exchanger discharge	°R
36.	THX2	Temperature	Main nozzle heat exchanger discharge	°R
37.	TMJ	Temperature	Upstream of main chamber injector (dome)	°R
38.	WDC	Flow	Out of the transpiration section volume	lb _m /sec
39.	WFI	Flow	Fuel low-speed inducer	lb _m /sec
40.	WL	Flow	Oxidizer low-speed inducer	lb _m /sec
41.	FFI	Pressure	Fuel low-speed inducer inlet	lb _f /in ²
42.	WHDC	Flow	Into the dump cooling heat exchanger	lb _m /sec
42.	HFI	Enthalpy	Fuel inlet	Btu/lb _m
44.	OFC	Mixture Ratio	Main chamber, including all transpiration flow	D'ies
45.	PLIM	Pressure	Main oxidizer pump inlet	lb _f /in ²
46.	TLI	Temperature	Oxidizer inlet	°R
47.	PLI	Pressure	Oxidizer low-speed inducer inlet	lb _f /in ²
48.	WMIJ	Flow	Exit from the dome volume	lb _m /sec
49.	TACC	Time Constant	Main chamber	sec

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(C) Table LXX. Engine Initial Conditions at Engine Mixture Ratio of 6 and Thrust Levels of 20, 50, and 100 Percent

	THRUST LEVEL		
	20%	50%	100%
OFV	$\left(\frac{93.84}{15.64}\right) = 6.000$	$\left(\frac{231.52}{38.92}\right) = 5.900$	$\left(\frac{445.77}{77.63}\right) = 5.000$
	721.8	1,976.8	4,516.5
Pe	$\left(\frac{721.8}{1.00836}\right) = 715.8$	$\left(\frac{1976.8}{1.00836}\right) = 1960.4$	$\left(\frac{4516.5}{1.00836}\right) = 4479.1$
Pc	538.1	1,355.3	2,726.4
PCP	538.1	1,355.3	2,725.4
PCS	558.8	1,407.2	2,830.9
PHDC	10.9	25.5	48.6
PHX2	924.8	2,385.6	4,374.3
PHXJ	577.2	1,467.6	3,003.6
PTCY	700.	1,700.	3,580.
PTRA	767.	1,737.3	3,350.5
RHX1	0.47	6.90	1.42
RHX2	1.17	2.41	3.33
ROFB	4.20	4.21	4.23
ROF1	3.95	4.11	4.24
RFPD	3.80	4.04	4.28
ROL	68.38	68.77	68.29
ROL3	69.00	69.09	69.20
NFP	20,984.	32,125.	46,471.
NFB	6,099.	11,002.	17,861.
NLP	9,931.	15,817.	23,509.
NLS	1,697.	3,004.	5,087.
THDC	1,901.	1,799.3	1,726.
TFBTD	368.3	416.5	458.5
THX1	375.2	428.0	474.9
THX2	142.6	158.8	187.
TMJ	1,279.2	1,494.5	1,847.6
WCC	1.33	2.68	4.76
WFL	15.64	38.92	77.63
WL	93.84	233.2	465.77

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(C) Table LXX. Engine Initial Conditions at Engine Mixture Ratio of 6 and Thrust Levels of 20, 50, and 100 Percent (Continued)

	THRUST LEVEL		
	20%	50%	100%
PFI	31.8	31.8	31.8
WHDC	0.34	0.81	1.59
HVI	-97.08	-97.08	-97.08
OFC	6.125	6.125	6.125
PLIM	76.3	118.0	185.7
TLI	175.6	175.6	175.6
PLI	37.6	37.6	37.6
WHIJ	23.29	64.56	147.95
TC	0.01	0.01	0.01

2. Program Output Parameters

(U) The output parameter list is arranged according to the output format shown below. The block of output data is printed at time intervals specified by PTIM and is labeled with the actual time in the engine simulation.

PROGRAM OUTPUT FORMAT

1. PFI	PHXI	PBS	PLI	WFI	WL	ROFB	TFBD	TQFT	QXF	NFP
2. PFBD	PFBD	PS	P.LD	WHDC	WLLK	RFPD	TFPD	TQFT	QXL	NLP
3. PFIM	PFIM	PFTIN	PLIM	WFD	WLC	RHX2	THX2	TQFST	QXFB	NFB
4. XFN 2nd	FTCV	PFTD	XLNPSF	WF	WLB	RHX1	THX1	TQLBT	QXLB	NLB
5. PFFD		PLTIK	PLPD	WHX2	WFT	ROLB	TFBD	DPFB	DPFDV	ALDV
6. PFSL	OFL	PLTD	PLSP	WFB	WLT	ROL	TLBD	DPLP	DPLCV	ALC
7. PFSP	OFC	PHIJ	PLETU	WICS	WFTD	WHJI	TLD	DPFB	DPLBT	ALBT
8. PHX2	OFB	PCS	PLETD	WTC	WLTG	WHIJ	TE	DPLE	DPFV	APFD
9. PFB1	OFV	PC	PLCV	WFBT	WFJI	WFIJ	TMJ	HPFP	DPFBI	AFB
10. PDCV	PFC	PCP	PLIJ	WCC	WCCX	WLCH	THDC	HPLP	DPDCV	ADCV

PROGRAM OUTPUT PARAMETERS

PFI	Pressure	Fuel low-speed inducer inlet	lb _f /in ²
PFBD	Pressure	Fuel low-speed inducer discharge	lb _f /in ²
PFIM	Pressure	Main fuel pump inlet (1st Stage)	lb _f /in ²
XFNPSF	Pressure	Excess NPSF at main fuel pump inlet	lb _f /in ²
PFFD	Pressure	Main fuel pump discharge (2nd stage)	lb _f /in ²
PFSL	Pressure	Split point of the cooling flows and P/B fuel flow	lb _f /in ²
PFSP	Pressure	Downstream of the fuel control valve	lb _f /in ²
PFB1	Pressure	Downstream of the main nozzle regenerative heat exchanger	lb _f /in ²

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PROGRAM OUTPUT PARAMETERS (Continued)

PFB1	Pressure	Inlet to the preburner fuel injector	lb_f/in^2
PDCVU	Pressure	Inlet to the dump coolant control	lb_f/in^2
PHX1	Pressure	Fuel low-speed inducer turbine inlet	lb_f/in^2
PFHTD	Pressure	Fuel low-speed inducer turbine discharge	lb_f/in^2
PTRA	Pressure	Transpiration section supply	lb_f/in^2
PTCV	Pressure	Turbine cooling flow volume	lb_f/in^2
OFI	Mixture Ratio	Main chamber injector	D'less
OFC	Mixture Ratio	Main chamber, including all transpiration flow	D'less
OFB	Mixture Ratio	Preburner	D'less
OFV	Mixture Ratio	Into the low speed inducers	D'less
PPG	Ratio	Chamber pressure/design chamber pressure	D'less
PBS	Pressure	Preburner injector face (static)	lb_f/in^2
PB	Pressure	Preburner total	lb_f/in^2
PF1IN	Pressure	Main fuel turbine inlet	lb_f/in^2
PF1D	Pressure	Main fuel turbine discharge	lb_f/in^2
PL1IN	Pressure	Main oxidizer turbine inlet	lb_f/in^2
PL1D	Pressure	Main oxidizer turbine discharge	lb_f/in^2
PM1J	Pressure	Upstream of main chamber injector	lb_f/in^2
PCS	Pressure	Main chamber injector face (static)	lb_f/in^2
PC	Pressure	Main chamber throat total	lb_f/in^2
PCP	Pressure	Main chamber throat total (steady state)	lb_f/in^2
PL1	Pressure	Oxidizer low-speed inducer inlet	lb_f/in^2
PL1D	Pressure	Oxidizer low-speed inducer discharge	lb_f/in^2
FL1M	Pressure	Main oxidizer pump inlet	lb_f/in^2
XLNPSP	Pressure	Excess NPSP at main oxidizer pump inlet	lb_f/in^2
PLPD	Pressure	Main oxidizer pump discharge	lb_f/in^2
PLSP	Pressure	Split point of the oxidizer P/B and main chamber flows	lb_f/in^2
PLBTU	Pressure	Oxidizer low-speed inducer turbine inlet	lb_f/in^2
PLBTD	Pressure	Oxidizer low-speed inducer turbine discharge	lb_f/in^2
PLCV	Pressure	Downstream of the main oxidizer control	lb_f/in^2
PL1J	Pressure	Inlet to the main oxidizer injector	lb_f/in^2
WFI	Flow	Fuel low-speed inducer	lb_m/sec
WHDC	Flow	Into the dump cooling heat exchanger	lb_m/sec
WFD	Flow	Discharge of the dump cooling nozzle	lb_m/sec
WF	Flow	Main fuel pump	lb_m/sec
WHX2	Flow	Fuel pump valve	lb_m/sec
WFB	Flow	Preburner fuel flow	lb_m/sec
WTCB	Flow	Into the turbine cooling volume	lb_m/sec
WTC	Flow	Out of the turbine cooling volume	lb_m/sec
WFTB	Flow	Into the transpiration section volume	lb_m/sec
WCC	Flow	Out of the transpiration section volume	lb_m/sec

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PROG 2A. OUTPUT PARAMETERS (Continued)

WLO	Flow	Oxidizer low speed inducer	lb/sec
WLEK	Flow	Oxidizer at which dumps overboard	lb ^m /sec
WLC	Flow	Oxidizer main chamber injector	lb ^m /sec
WLB	Flow	Oxidizer into the preburner	lb ^m /sec
WFT	Flow	Preburner products through the main fuel turbine	lb ^m /sec
WLT	Flow	Preburner products through the main oxidizer turbine	lb ^m /sec
WFTD	Flow	Main fuel turbine discharge including cooling	lb ^m /sec
WLTG	Flow	Main oxidizer turbine discharge including cooling	lb ^m /sec
WFLI	Flow	Fuel flow in the transition case	lb ^m /sec
WCCX	Flow	Excess transpiration cooling flow	lb ^m /sec
ROFB	Density	Fuel low-speed inducer discharge	lb ^m /ft ³
RFPD	Density	Main fuel pump discharge (2nd stage)	lb ^m /ft ³
RHX2	Density	Main nozzle heat exchanger discharge	lb ^m /ft ³
RHX1	Density	Small nozzle heat exchanger discharge	lb ^m /ft ³
ROLB	Density	Oxidizer low-speed inducer discharge	lb ^m /ft ³
ROL	Density	Main oxidizer pump discharge	lb ^m /ft ³
WMJI	Flow	Total turbine discharge into the dome volume	lb ^m /sec
WMIJ	Flow	Exit from the dome volume	lb ^m /sec
WFIJ	Flow	Fuel flow out of the main chamber injector	lb ^m /sec
WLCH	Flow	Total oxidizer flow in the main chamber	lb ^m /sec
TFBD	Temperature	Fuel low-speed inducer discharge	°R
TFPD	Temperature	Main fuel pump discharge	°R
THX2	Temperature	Main nozzle heat exchanger discharge	°R
THX1	Temperature	Small nozzle heat exchanger discharge	°R
TFTD	Temperature	Fuel low-speed inducer turbine discharge	°R
TLBD	Temperature	Oxidizer low-speed inducer turbine discharge	°R
TLD	Temperature	Main oxidizer pump discharge	°R
TB	Temperature	Preburner	°R
TLJ	Temperature	Upstream of main chamber injector (dome)	°R
THDC	Temperature	Dump cooling nozzle exit	°R
TQFT	Torque	Main fuel turbine	lb-ft
TQLT	Torque	Main oxidizer turbine	lb ^m -ft
TQFBT	Torque	Fuel low-speed inducer turbine	lb ^m -ft
TQLBT	Torque	Oxidizer low-speed inducer turbine	lb ^m -ft
DRFP	Pressure Rise	Main fuel pump	lb ₂ /in ₂
DPLP	Pressure Rise	Main oxidizer pump	lb ₂ /in ₂
DRFB	Pressure Rise	Fuel low-speed inducer	lb ₂ /in ₂
DPLB	Pressure Rise	Oxidizer low-speed inducer	lb ₂ /in ₂
HPFP	Horsepower	Main fuel pump	hp
HPLP	Horsepower	Main oxidizer pump	hp
CKF	Excess Torque	Main fuel (turbine torque - pump torque)	lb ^m -ft

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PROGRAM OUTPUT PARAMETERS (Continued)

QXL	Excess Torque	Main oxidizer (turbine torque - pump torque)	lb _m -ft
QXPB	Excess Torque	Fuel boost low-speed inducer - (turbine torque - pump torque)	lb _m -ft
QXL2	Excess Torque	Oxidizer low-speed inducer - (turbine torque - pump torque)	lb _m -ft
DFFDV	Pressure Drop	Preburner oxidizer flow divider valve	lb _f /in ²
DPLCV	Pressure Drop	Main oxidizer line control valve	lb _f /in ²
DPLBT	Pressure Drop	Oxidizer low-speed inducer turbine	lb _f /in ²
DFFV	Pressure Drop	Fuel valve at pump discharge	lb _f /in ²
DFFBI	Pressure Drop	Preburner fuel injector	lb _f /in ²
DFDCV	Pressure Drop	Pump coolant control	lb _f /in
NFP	Speed	Main fuel turbopump	rpm
NLP	Speed	Main oxidizer turbopump	rpm
NFB	Speed	Fuel low-speed inducer	rpm
NLB	Speed	Oxidizer low-speed inducer	rpm
ALDV	Control Area	Preburner oxidizer flow divider valve	in ²
ALC	Control Area	Main oxidizer line control valve	in ²
ALBT	Control Area	Oxidizer low-speed inducer turbine	in ²
AFPD	Control Area	Fuel valve at pump discharge	in ²
AFB	Control Area	Preburner fuel injector	in ²
ADCV	Control Area	Pump coolant control valve	in

3. Program Formulation

(U) The following engineering formulation includes the series of calculation of which the program execution cycles through each time increment (DT). The numerical integration technique has proven quite accurate, with DT = 0.001 second. However, some calculations have required mathematical iteration loops to assure proper balances in each program cycle pass. This is particularly true in the liquid flow > pressure balances where extremely small DT's would be required to eliminate mathematical oscillations. These iteration loops are designated in the formulation.

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(U) 1. Fuel side

$$PFG = PC/2716.4$$

Assume WFI

$$DPFB = f_1(MFB, WFI, ROPB)$$

$$PFBD = PFI + DPFB$$

$$PFIM = PFBD - 0.01139605 \frac{(WFI)^2}{ROPB}$$

$$PDCVU = PFBD - (0.005698 \cdot WFI^2 + 4.28 \cdot WHDC^2)/ROPB$$

$$WHDC = 1.3645 \cdot ADCV \cdot \sqrt{PDCVU - 33}$$

$$WF = WFI - WHDC$$

$$DPFP = f_2(MFP, WF, ROPF, RSTB)$$

$$PFPD = PFIM + DPFP$$

$$PFSL = PFPD - 0.01 \cdot (WF)^2/ROPF$$

$$ATCS = f(PFG)$$

Assume FICV

$$WTCS = ATCS \sqrt{(PFSL - FICV) \cdot RFPD/1.486}$$

$$WTC = 0.1228 \sqrt{(FICV - RCL1) \cdot RTCV}$$

$$\text{where: } RTCV = f(FICV)$$

$$FICVX = FICVP + RTCV \cdot (WTCS - WTC) \cdot 8.64 \cdot (DT)/RFPD$$

$$\text{where: } FICVP = FICV \text{ of time } = T - DT$$

$$RTCV = f_4(FICV)$$

Iterate on FICV until FICVX = FICV ± tolerance.

Assume WCC

$$PTRA = 0.5(PC + \sqrt{PC^2 + 787.378 \cdot SOT \cdot PC \cdot TFBD})$$

$$\text{where: } SOT = 32.119(WCC/PTRA/PC) = 0.182$$

$$WFET = WCC + \frac{0.022814 \cdot (PTRA - PTRAP)}{PTRA(DT)}$$

$$\text{where: } PTRAP = PTRA \text{ of time } = T - DT$$

$$PFBDT = \sqrt{PTRA^2 + 22.20267 \cdot (WFET)^2 \cdot TFBD}$$

$$PHX1 = 476.799 \cdot WFET \cdot \sqrt{PHX1/f_3(PFBDT)}$$

$$\text{where: } PFBDT = 0.3(PFBDT +$$

$$\sqrt{PFBDT^2 + 43.71 \cdot TFBDT \cdot (WFET)^2})$$

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$$PFSTX = \sqrt{PFST^2 - 10.2674 \cdot TAX1 \cdot (WFBT)^2}$$

$$\text{where: } PFST = PFSL \cdot$$

$$\left(\frac{151.385}{RFPD} + \frac{0.4}{RHX1} + \frac{21.3}{(RFPD - RHX1)} \right) WFBT^2$$

Iterate on WOC until PFSTX = RHX1 ± tolerance

$$WFBT = WF + WTCB + WFBX$$

$$PFSP = PFSL - \frac{2.218}{RFPD} \cdot \left(\frac{WFBT}{RFPD} \right)^2$$

$$DPFV = PFSL - PFSP$$

$$PHDX = PFSP - WHX2^2 \left(\frac{5.02775}{RFPD} + \frac{5.04(PFG)^{0.05}}{RHX2} \right)$$

$$WFB = \sqrt{\frac{(PHDX - PBS) RHX2}{0.089 + \left(\frac{1.495}{APB} \right)^2}}$$

$$PFBI = PBS + \frac{2.158}{RHX2} \cdot \left(\frac{WFB}{APB} \right)^2$$

$$DPFBI = PFBI - PBS$$

$$PHX2 = PHDX + 1.02674 \cdot BPHX \cdot (WHX2 - WFB) \cdot (DT)/RHX2$$

$$\text{where: } BPHX = f_6(PHDX)$$

$$PHDXP = PHX2 \text{ of time } = T - DT$$

Iterate on WFI until PHX1 = PHDX ± tolerance.

$$TFBD = f_7(WFB, WFB)$$

$$\text{where: } WFB, WFB = f_1(WFI, WFB)$$

$$WFB = WFI + \frac{WFB}{776(WFB)}$$

$$WFBT = f_2(WFB, WFB)$$

$$WFB = f_3(WFB, WFB)$$

$$\text{where: } WFB, WFB = f_4(WFB, WFB)$$

$$WFB = WFB + \frac{WFB}{776(WFB)}$$

$$WFBT = f_5(WFB, WFB)$$

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$$THX1 = THXDP - (THXDP - THX2) \exp \left(-\frac{DT}{TAUX} \right)$$

$$\text{where: } HMXD = HF2 + 12840 (PFG)^{0.87} \text{ SKOF/WHX2}$$

$$SKOF = f_9(DFC)$$

$$THXDP = f_7(PHX2, HMXD)$$

$$TAUM = f_{10}(WFB)$$

$$PHX2 = f(PHX2, THX2)$$

$$THX1 = THXDP - (THXDP - THX1) \exp \left(-\frac{DT}{TAUX} \right)$$

$$\text{where: } HPHXD = HF2 + 6500 (PFG)^{0.87} \text{ SKOF/WFBT}$$

$$SKOF = f_9(DFC)$$

$$THXDP = f_7(PHXD, HPHXD)$$

$$TAUX = f_{10}(WFBT)$$

$$PHX1 = f_8(PHXD, THX1)$$

$$TQFBT = 7426.11 \cdot DHFBT \cdot ETFBT \cdot WFBT/NFB$$

$$\text{where: } DHFBT = 3.516 \cdot THX2 \left(1 - (PFBT/PHX1)^{0.2642} \right)$$

$$ETFBT = f_{11}(NFB, DHFBT)$$

$$QXFB = TQFBT - TQFB$$

$$\text{where: } RDFB, ETFB = f_1(WFI, NFB)$$

$$TQFB = \frac{1.34861 \cdot WFI \cdot RDFB}{NFB \cdot ETFB}$$

$$NFB = 266.74 \cdot QXFB \cdot DT + NFB$$

$$THX1 = THX1 - DHFBT \cdot ETFBT/3.516$$

$$WFD = f_2(THDC, PHDC)$$

$$PHDC = PHDC + 3.8325 \cdot (THFD + THDC) \cdot (THDC - WFD) \cdot DT$$

$$THDC = 1726. / (PFG)^{0.06}$$

$$DPDCV = PDCVU - PHDC - 28. (PFG)^{0.98} \cdot SKLP - 4.28 (WHDC)^2 / ROFB$$

$$PFBT = PFBT - 30. - \frac{ROFB}{144.} \left(4.391 \cdot 10^{-7} \cdot NFB^2 \cdot \sqrt{\frac{ROFB}{WFB}} \right)^{4/3}$$

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$$XLMFSP = FLIM + 30. = \frac{ROF}{14} + (3.18 \times 10^{-7} \frac{WL^2}{ROL})^{1/3}$$

$$WCCX = WCC = \frac{24130}{6600} \cdot \frac{0.87}{ALBD + BWBT}$$

$$\text{where: } BWBT = f_{22}(OPG)$$

(U) 2. Oxidizer Side

Assume WL

$$DPLB = f_{12}(WL, WLEK, ROLB)$$

$$PLBD = PLB + D/LP$$

$$FLIM = PLBD + 0.004542 \cdot (WL)^2 / ROLB$$

$$EPLP = f_{11}(FLIM, PLBD, ROLB)$$

$$PLPD = PLBD + EPLP$$

$$PLSP = PLPD + 0.011009 \cdot (WL)^2 / ROL$$

$$WLB = \sqrt{\frac{(PLSP - PBS) \cdot ROL}{0.3875 + \left(\frac{1.496}{ALBT}\right)^2}}$$

$$\text{where: } ALBT = 0.029345 \cdot \frac{1.496}{\sqrt{2.3993 + \left(\frac{1.496}{ALDV}\right)^2}}$$

$$ALC = \sqrt{\frac{(PLSP - PBS) \cdot ROL}{0.322637 + \left(\frac{1.496}{ALBT}\right)^2 + \left(\frac{1.496}{ALC}\right)^2}}$$

$$WLEK = 0.9375 \cdot (OPG) + 0.0625$$

$$WLPX = WLB + WLC + WLEK$$

Iterate on WL until WLPX = WL ± Tolerance

$$TLBD = TEL + (1.6/CTAB - 1.0) \cdot 0.0047697 \cdot (HDLB)$$

$$\text{where: } HDLB, ETAB = f_{12}(WLB, WL)$$

$$ROLB = f_{11}(PLBD, TLBD)$$

$$RLD = \frac{316.646 \cdot PLBD \cdot ROL + RL1 + 1.74 \cdot DPLP \cdot (1. - RL1)}{RL1(316.646 \cdot ROL + 0.33634 \cdot DPLP)}$$

$$\text{where: } RL1 = f_{13}(P, WL, ROL)$$

$$ROL = f_{14}(PLPD, RLD)$$

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$$PLBTU = PLSP - 0.0027184(WLC)^2/ROL$$

$$DPLBT = \frac{2.234}{ROL} \left(\frac{WLC}{ALBT} \right)^2$$

$$PLBTD = PLBTU - DPLBT$$

$$TQLBT = 7426.1 (DHLBT \cdot ETLBT \cdot WLC)/NLB$$

$$\text{where: } DHLBT = 0.185 DPLBT/ROL$$

$$ETLBT = f_{15}(NLB, \frac{1}{\sqrt{DHLBT}}, ALBT)$$

$$QXLB = TQLBT - TQLB$$

$$\text{where: } ROLB, ETLB = f_{12}(NLB, WL)$$

$$TQLB = \frac{1.54861 \cdot WL \cdot HDLB}{NLB \cdot ETLB}$$

$$NLB = 286.482 \cdot QXLB \cdot DT + NLB$$

$$PLIJ = PCS + 0.31694(WLC)^2/ROL$$

$$PLCV = PLIJ + 0.0036245(WLC)^2/ROL$$

$$DPLCV = PLBTD - PLCV$$

$$DPFDV = \left(\frac{1.496}{ALDV} \right)^2 \cdot \left(\left(1 - \frac{0.029345}{ALBI} \right) \cdot WLB \right)^2$$

(U) 3. Preburner and Main Turbines

a. Preburner

$$OPB = WLB/WFB$$

$$TB = f(OPB, THX2)$$

$$\text{Also: } GMB, CPB, RB = f_{16}(OPB, THX2)$$

$$(WFT + WLT) = \frac{7.452 \cdot CA}{1.3 \cdot RB} \cdot f_{17} \left(\frac{PB}{PLIJ}, GMB \right)$$

$$PE = 0.0007145 \cdot RB \cdot TB \cdot \left[TB + WLB - (WFT + WLT) \right] \cdot DT + PB$$

$$PBS = 1.00836 \cdot PB$$

b. Fuel Turbine

$$WFT = 0.67408 \cdot (WFT + WLT)$$

$$WFTD = WFT + 0.359 WTC$$

$$FFTD = \sqrt{FHLJ^2 + 8.0452 \times 10^{-6} \cdot (WFTD)^2 \cdot RB \cdot THJ}$$

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$$PFTIN = \sqrt{PB^2 - 2.0828 \times 10^{-6} \cdot (WFT)^2 \cdot RB \cdot TB}$$

$$TQFT = 7426.11 \cdot DHFT \cdot ETAFT \cdot WFT/NFP$$

where:

$$PFTS = 0.5 \left(PFTD^2 + \sqrt{PFTD^2 - 3.6444 \times 10^{-5} \cdot RB \cdot TMJ \cdot (WFTD)^2} \right)$$

$$DHFT = CPB \cdot TB \left[1. - \left(\frac{PFTS}{PFTIN} \right)^{\frac{GMB-1}{GMB}} \right]$$

$$ETAFT = f_{18} \left(NFP, \frac{1}{\sqrt{DHFT}} \right)$$

$$QXF = TQFT - TQRF$$

$$\text{where: } DHFP, BFP = f_2(WF, NFP)$$

$$TQRF = \frac{9.54861 \cdot WF \cdot DHFP}{NFP \cdot BFP}$$

$$NFP = 123.22 \cdot QXF \cdot DT + NFP$$

$$HPFP = TQRF \cdot NFP/5250$$

c. Oxidizer Turbine

$$WLT = (WFT + WLT) - WFT$$

$$WLTD = WLT + 0.359 WTC$$

$$PLTD = \sqrt{PMLJ^2 + 4.04672 \times 10^{-5} \cdot (WLTD)^2 \cdot RJ \cdot TMJ}$$

$$PLTIN = \sqrt{PB^2 - 2.12476 \times 10^{-5} \cdot (WLT)^2 \cdot RB \cdot TB}$$

$$TQLT = 7426.11 \cdot DHLT \cdot ETALT \cdot WLT/NLP$$

where:

$$PLTS = 0.5 \left(PLTD^2 + \sqrt{PLTD^2 - 1.29235 \times 10^{-5} \cdot RB \cdot TMJ \cdot (WLTD)^2} \right)$$

$$DHLT = CPB \cdot TB \left[1. - \left(\frac{PLTS}{PLTIN} \right)^{\frac{GMB-1}{GMB}} \right]$$

$$ETALT = f_{19} (NLP, \sqrt{DHLT})$$

$$QXL = TQLT - TQRL$$

$$\text{where: } DNLP, BLP = f_{13}(WL, NLP)$$

$$TQRL = \frac{9.54861 \cdot WL \cdot DNLP}{NLP \cdot BLP}$$

$$NLP = 97.4418 \cdot QXL \cdot DT + NLP$$

$$HPLP = TQRL \cdot NLP/5250$$

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(U) 4. Main Injector and Chamber Dome

$$WMIJ = WTC + WFT + WLT$$

$$TMJ = \frac{WFTD \cdot TFT + WLTD \cdot TLT}{\frac{3.816}{CPB} \cdot WTC + WFT + WLT}$$

$$\text{where: } TFT = TB - \frac{DHFT \cdot ETFT}{CPB}$$

$$TLT = TB - \frac{DHLT \cdot ETALT}{CPB}$$

Assume: WMIJ

$$PMIJ = PMIJP + 0.0005(WMIJ - WMIJ) \cdot RMIX \cdot TMJ \cdot DT$$

$$\text{where: } RMIX = \frac{9198 \cdot WTC + RB(WFT + WLT)}{WMIJ}$$

$$PMIJP = PMIJ \text{ of time } = T - DT$$

$$WMIJOP = \frac{21.837 \cdot PMIJ}{\sqrt{TMJ \cdot RMIX}} f_{17}(PMJ/PCS, GAMIX)$$

$$\text{where: } CPMIX = \frac{3.816 \cdot WTC + CPB(WFT + WLT)}{WMIJ}$$

$$GAMIX = \frac{CPMIX \cdot 9336}{CPMIX \cdot 9336 + RMIX}$$

Iterate on WMIJ until WMIJOP = WMIJ : Tolerance

$$WFIJ = \frac{WFT + WLT}{(OPB + 1)} + WTC$$

$$WFIJ = \frac{WMIJ}{\left(\frac{WMIJ - WFIJ}{WFIJ} \right) + 1}$$

$$WLCH = WLC + WMIJ - WFIJ$$

$$OFI = WLCH/WFIJ$$

$$OFC = \frac{WLCH}{WFIJ \cdot WCC}$$

$$PCP = (WLCH + WFIJ + WCC) \cdot CSTR \cdot ETAC/1497.3$$

$$\text{where: } CSTR = f_{20}(OFC, PC)$$

$$ETAC = f_{21}(OFI)$$

$$PC = PCP - (PCP - PC) \exp\left(-\frac{DT}{TC}\right)$$

$$PCS = PC \cdot 1.03633$$

$$OFV = WL/WFI$$

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F. PROGRAM OPERATING INSTRUCTIONS

(U) The 250K digital transient deck consists of the following program blocks:

1. A "START" subroutine will be provided as a source deck and will allow the users prime authority over the selection of available control areas and sense parameters. For example, the dump coolant control area (ADCV) can be programmed as a function of chamber pressure, and preburner fuel injector area (AFB) can be set to a constant. The named common block "CINPUT" is contained in the subroutine "START" and must be retained in its present form.
2. The engine mathematical simulation "ENGINE" will be provided as an object deck and will not require program changes unless cycle modifications are encountered.

(U) The input items in "CINPUT" are initialized by the first 40 cards of input. These cards are followed by a blank card to indicate the end of the data for the first case. The number in column 1-2 determines the location in the "CINPUT" common block where the value is to be stored. The number should be right adjusted in the field. The name in columns 25-30 is for references only. The value for each input can be in either a decimal form or decimal exponent form. This value must be punched between columns 5 and 20 and must contain a decimal point. The input listing always prints the input values in the decimal exponent form.

(U) The data are read in until the blank card is processed. The program then enters subroutine "START" to obtain the engine control areas. Subroutine "ENGINE" is called to perform the engine simulation calculations for the input start time. Upon completion of these calculations, the time increment (DT) is added to the start time and subroutine "START" is recalled to change the control areas, if necessary. Subroutine "ENGINE" is again entered and the computation for the new time is completed. At this point the sum of the time increments is checked against the print time (PTIM). If they are equal, the results of the simulation at that time are printed. The process of adding the time increment and completing the simulation is repeated until the stop time (STIM) is reached. At any time during any of the steps, if a situation occurs that would terminate the run, "EPRINT" is called to print the results at that point and a statement is printed indicating that Error Print was called.

(U) The next card is read for a second case. If multiple cases are desired they must be separated by a blank card and only the input items that are to be changed from the previous case need be included for the next case. If there is no additional case to be run, a card with -1 punched in columns 1-2 is placed following the blank card of the first case. The data package must have a blank card and a -1 card at the end.

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(U) The time increment, DT, (input card No. 2 of the data package) should remain 0.001 because the calculations and convergence loops in the program are based on this increment. The start time, T, (card No. 1) can be changed providing the initial conditions established by the other input cards are adjusted to give reasonable engine conditions at the start time input. The print time, PTIM, (card No. 3) may be changed to reduce the printed output, if desired. The print time must be some multiple of the time increment (DT).

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**APPENDIX III
PARAMETRIC DATA**

A. Performance Data	715
1. General	715
2. Performance	716
B. Weight and Envelope	717

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**APPENDIX III
PARAMETRIC DATA**

A. PERFORMANCE DATA

1. General

(C) These data are for high-pressure, two-position bell-nozzle engines, using the staged-combustion (preburner) cycle and are based on technology that could be realized during a full engine development program. The use of dump cooling downstream of an expansion ratio of 35 was assumed in computing these parametric engine data. For high expansion ratio nozzles, radiation cooling is used aft of the lowest expansion ratio allowed by heat flux consideration. (This expansion ratio varies over the range, but is approximately 300.) Performance, weight, and dimensional data are based on the following:

1. High pressure, staged-combustion, two-position bell-nozzle engines
2. Main turbopumps and preburner, mounted in a transition case
3. Low speed inducers with:
 - a. Minimum hydrogen net positive suction head (NPSH) = 60 ft
 - b. Minimum oxygen net positive suction head (NPSH) = 60 ft
4. Throttling capability - continuous between 100 and 20% of rated thrust
5. Mixture ratio range of 5 to 7 at all thrust levels
6. Thrust vector control provided by gimbaling
7. Durability of 10 hours time between overhaul (TBO), 100 reuses, 300 starts, 300 thermal cycles, 10,000 valve cycles
8. The lightweight extendable secondary nozzle translates to provide high sea level performance and altitude compensation capability.

(C) The ranges of engine parameters included in the parametric data are:

1. Vacuum thrust - 100,000 to 350,000 lb
2. Chamber pressure - 2000 to 3500 psia
3. Overall engine mixture ratio - 5 to 7
4. Overall expansion ratio - 50 to 400

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5. Nozzle contour - maximum performance (MC_g), base, and minimum surface area (MSA)

6. Primary expansion ratio - 35.

(U) Values of specific impulse are given for an altitude range from sea level to vacuum conditions.

(C) Data are presented for engines where the secondary nozzle skirt is translated from primary expansion ratios of 35 for a range of overall nozzle area ratios of 30 to 150. In addition, data are presented for primary area ratios that result in the minimum stowed engine length (i.e., the secondary nozzle skirt fully retracted). The primary expansion ratio for minimum stowed length varies with thrust level, chamber pressure, overall expansion ratio, and nozzle contour. For these data, the overall nozzle area ratio range is 80 to 400. The secondary nozzle can be translated over the turbopump for those engines where the primary area ratio is greater than 80. For the nozzles translating at primary area ratios between 35 and 80, the secondary nozzle is retracted to the point where it is limited by the turbomachinery.

(U) Parametric data over the complete thrust and chamber pressures range and for minimum surface area, base, and maximum performance nozzle contours are presented in the following paragraphs.

2. Performance

(U) Delivered vacuum specific impulse as a function of thrust, mixture ratio, chamber pressure, and nozzle area ratio is shown in figure 638 through 658 for minimum surface area, base, and maximum performance nozzle contours.

(C) Performance from sea level to 200,000 ft is shown as a ratio of delivered specific impulse to vacuum specific impulse plotted as a function of the ratio of chamber pressure to ambient pressure for primary area ratios of 35 (the data are independent of chamber pressure when primary area ratio is constant). These data are presented in figures 659 through 667. Altitude to vacuum performance for minimum stowed length engines (i.e., primary area ratio is a variable) is shown in figures 668 through 703 as a function of altitude.

(C) The delivered specific impulse at other than vacuum conditions can be calculated using figures 659 through 703, which show the ratio of altitude to vacuum performance as a function of altitude or pressure ratio and nozzle expansion ratio for mixture ratios of 5, 6, and 7. The delivered specific impulse at any altitude up to 200,000 ft (200,000 ft is the reference altitude for vacuum conditions) is calculated by:

$$I_{s_{alt}} = \left[\frac{I_{s_{alt}}}{I_{vac}} \right] I_{vac}$$

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where:

$I_{s_{alt}}$ = delivered specific impulse at the altitude of interest

$I_{s_{alt}} / I_{vac}$ = ratio of altitude to vacuum performance for the altitude and mixture ratio of interest (figures 659 through 703)

also

$$F_{alt} = \frac{I_{s_{alt}}}{I_{vac}} F_{vac}$$

(U) In cases where altitude performance is plotted versus pressure ratio, the pressure ratio can be calculated by:

$$PR = \frac{P_c}{P_a}$$

where:

P_c = chamber pressure, psia

P_a = ambient pressure at the altitude of interest, psia

(U) The curves present altitude performance for the engines with the secondary nozzle extended at high altitudes and with the secondary nozzle retracted at low altitudes. Low altitude primary nozzle data have not been included for cases where the primary nozzle exhaust would be expected to reattach to the secondary nozzle (acting as a shroud).

B. WEIGHT AND ENVELOPE

(C) This paragraph contains parametric data showing the effect of nozzle contour, area ratio, and mixture ratio changes on the weight and envelope of oxygen-hydrogen, high-pressure (3000-psia chamber pressure), pump-fed, staged combustion cycle rocket engines with transpiration cooled main combustion chambers and dump-cooled two-position nozzles. Data are presented for vacuum thrust levels of 100,000 lb to 350,000 lb, with nozzle contour, chamber pressure, and area ratio as independent variables. The nozzle contours covered are the following truncations of perfect bell nozzles: (1) minimum surface area (MSA); (2) base nozzle; and (3) maximum performance (MC_g).

(U) Engine weights are presented in figures 704 through 721.

(U) Figure 722 illustrates the engine configuration with a two-position nozzle. Engine length with a two-position nozzle is presented as stored length, minimum stored length (upper stage), and overall length (nozzle fully extended). These data are presented in figures 723 through 749.

(U) Exit envelope diameters are presented in figures 750 through 756.

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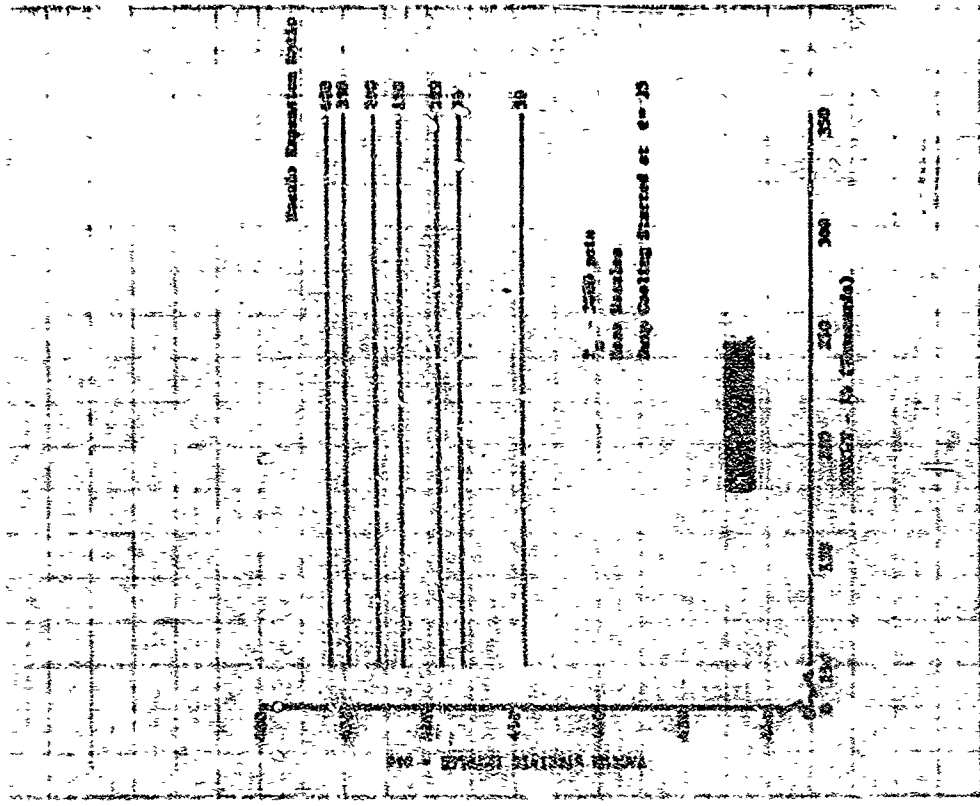


Figure 638. Vacuum Specific Impulse vs. Thrust With Base Contour Two-Position Nozzle ($\epsilon = 5.0$)

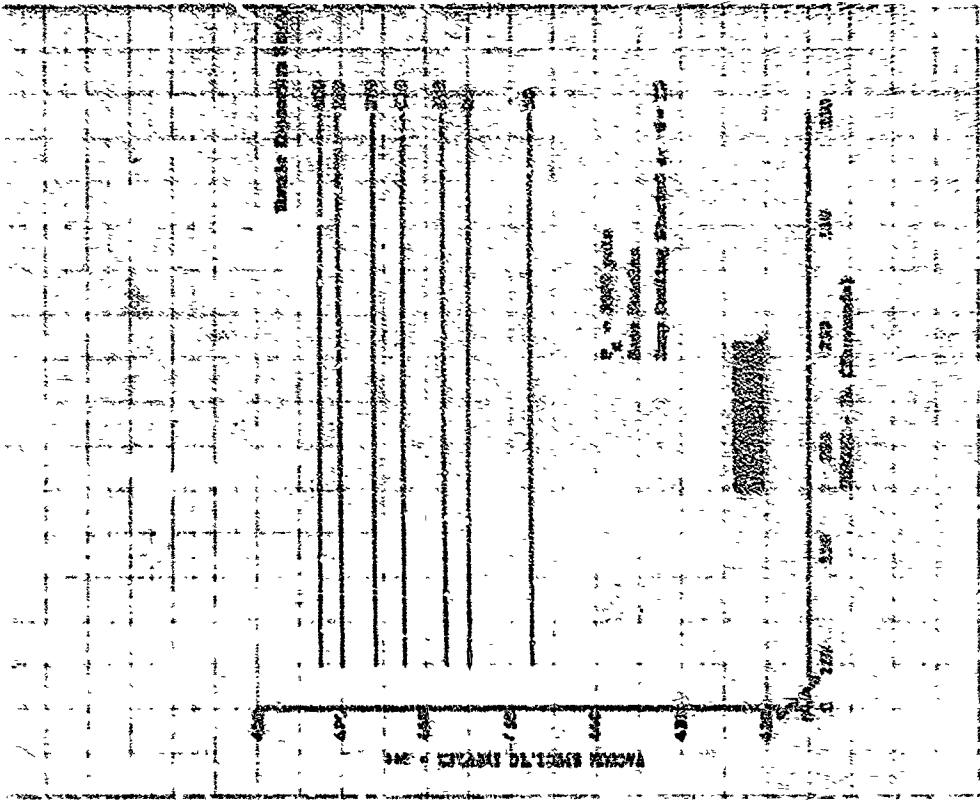


Figure 639. Vacuum Specific Impulse vs. Thrust With Base Contour Two-Position Nozzle ($\epsilon = 5.5$)

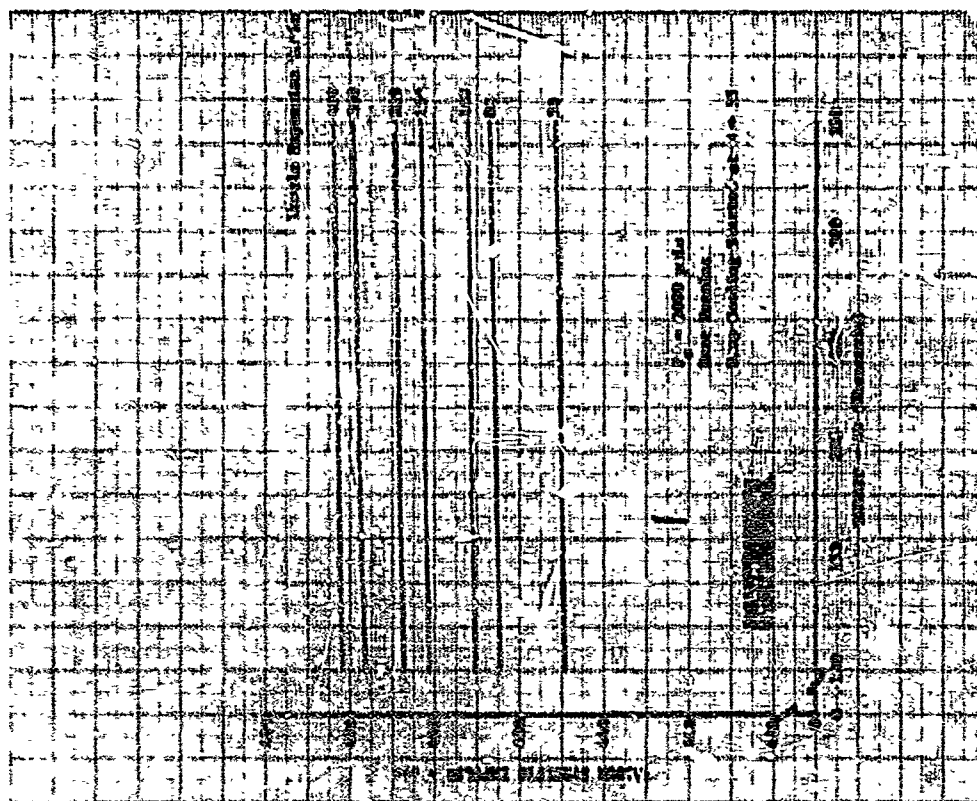


Figure 640. Vacuum Specific Impulse vs Thrust With Base Contour Two-Position Nozzle ($r = 6.0$)

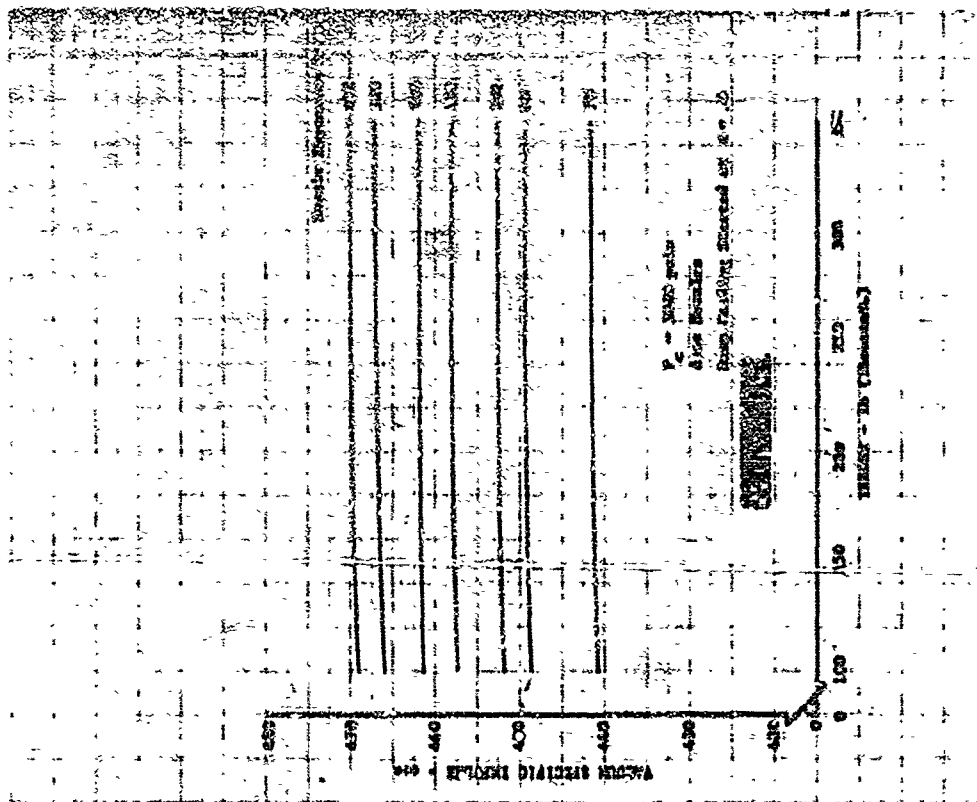


Figure 641. Vacuum Specific Impulse vs Thrust With Base Contour Two-Position Nozzle ($r = 6.5$)

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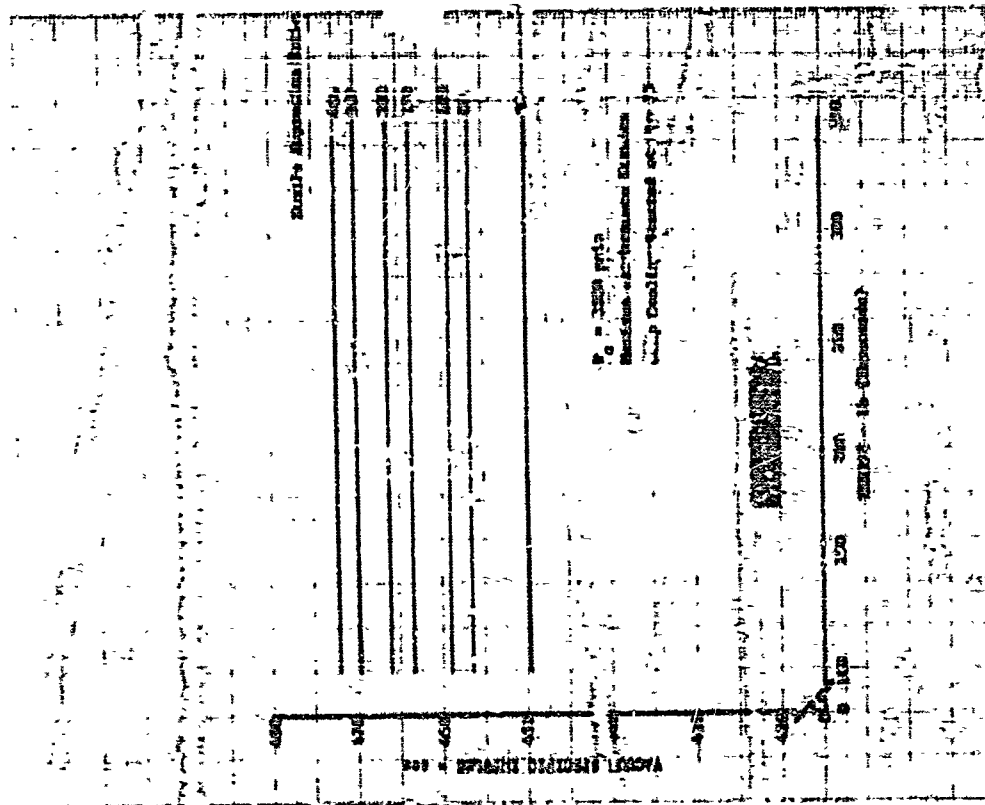


Figure 643. Vacuum Specific Impulse vs Thrust With Maximum Performance Contour Nozzle (r = 5.0) DF 56294

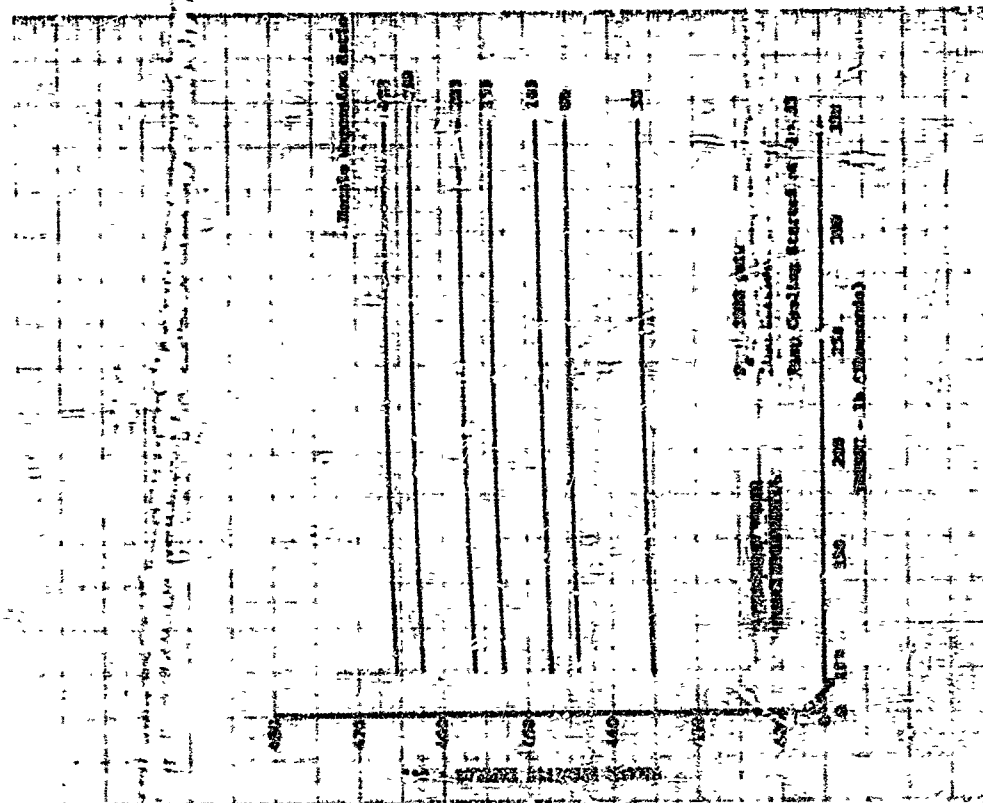


Figure 642. Vacuum Specific Impulse vs Thrust With Base Contour Nozzle (r = 7.0) DF 56288

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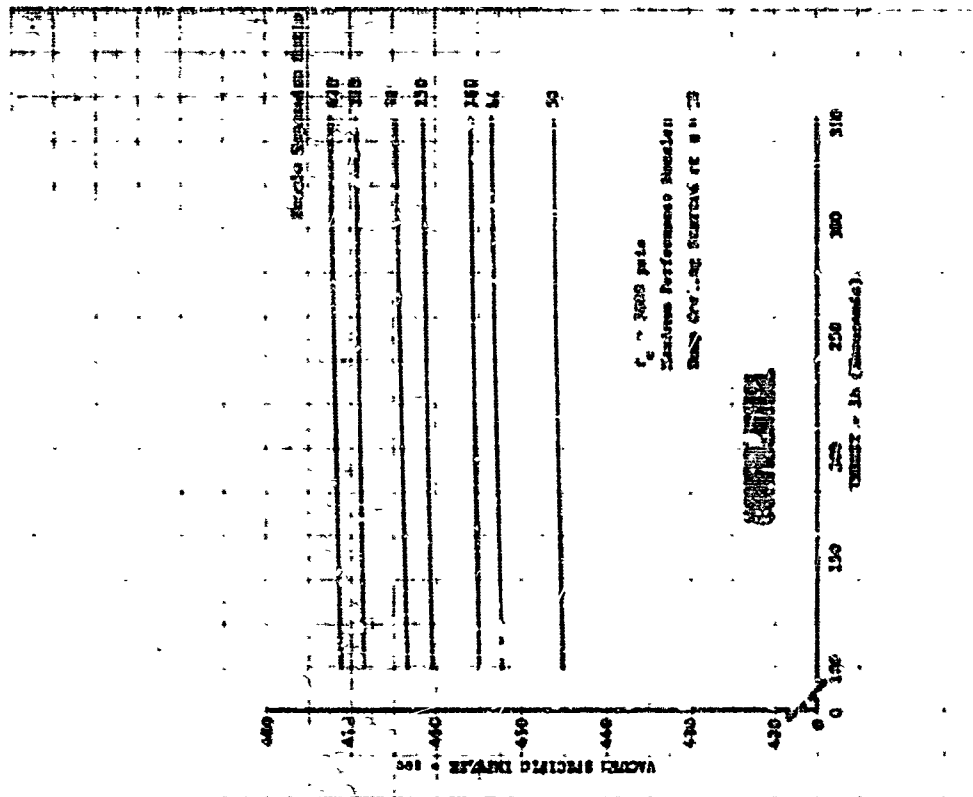


Figure 645. Vacuum Specific Impulse vs Thrust With Maximum Performance Contour Two-Position Nozzle ($r = 6.0$)

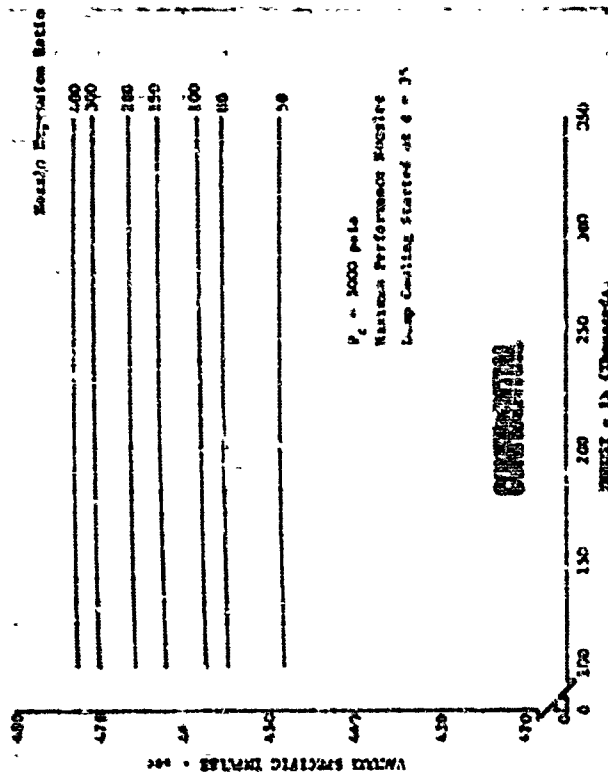


Figure 644. Vacuum Specific Impulse vs Thrust With Maximum Performance Contour Two-Position Nozzle ($r = 5.5$)

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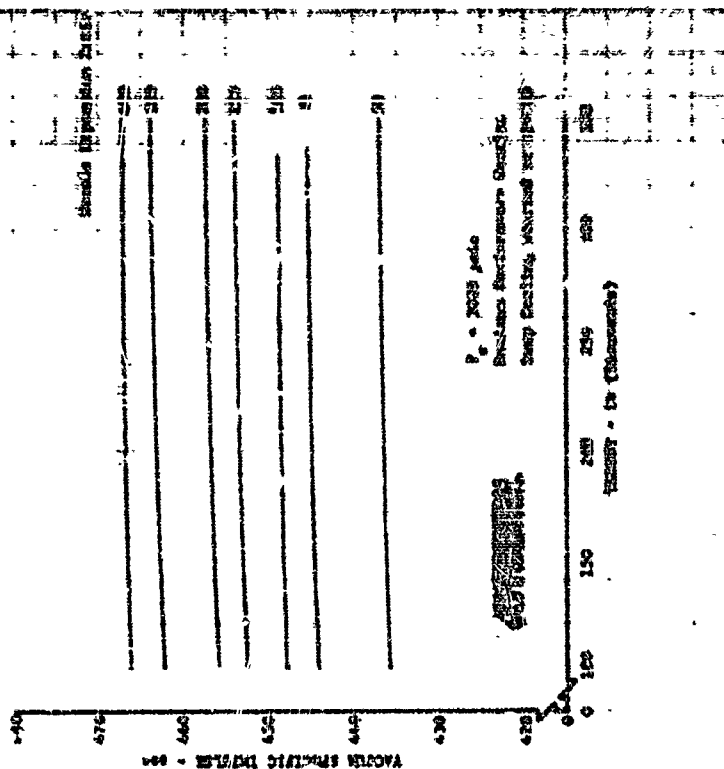


Figure 647. Vacuum Specific Impulse vs Thrust with Maximum Performance Contour Two-Position Nozzle ($r = 7.0$)

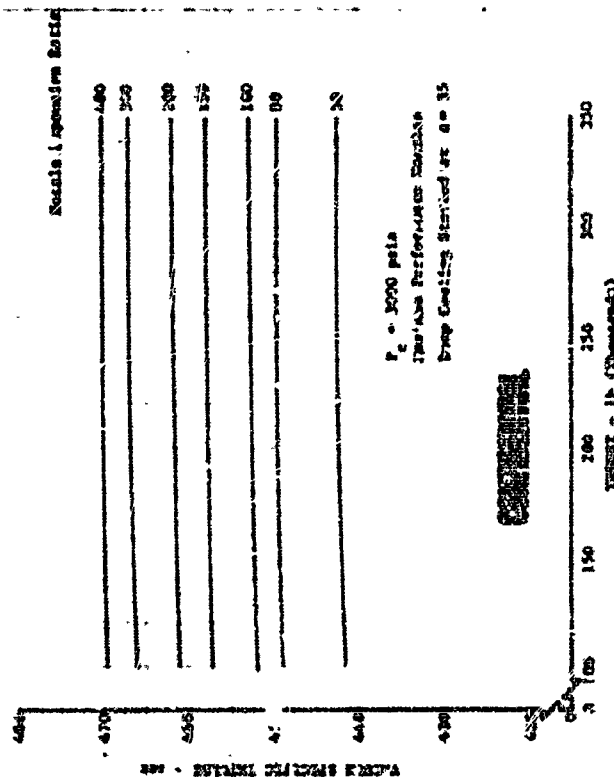


Figure 646. Vacuum Specific Impulse vs Thrust with Maximum Performance Contour Two-Position Nozzle ($r = 6.5$)

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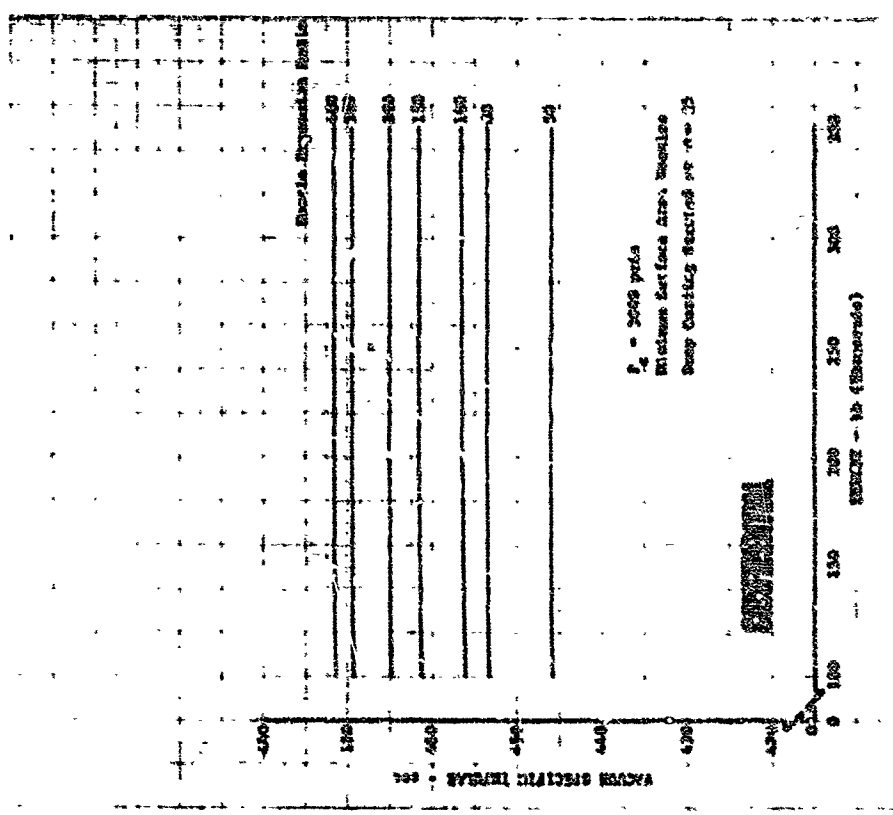


Figure 649. Vacuum Specific Impulse vs Thrust With Minimum Surface Area Contour Two-Position Nozzle (r = 5.5)

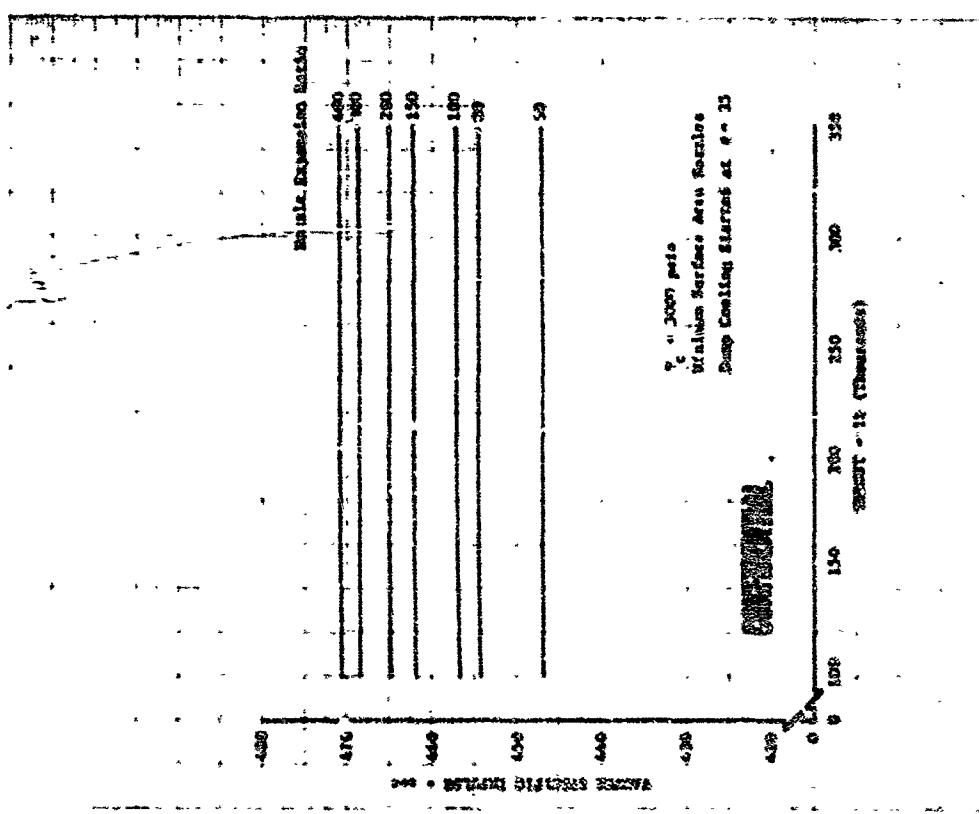


Figure 648. Vacuum Specific Impulse vs Thrust With Minimum Surface Area Contour Two-Position Nozzle (r = 5.0)

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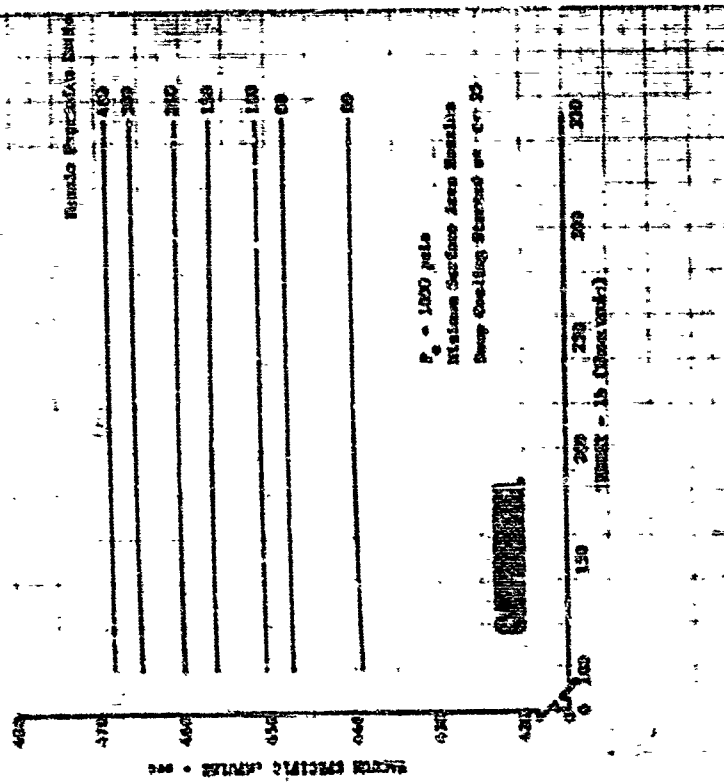


Figure 651. Vacuum Specific Impulse vs Thrust With Minimum Surface Area Contour Two-Position Nozzle (r = 6.5)

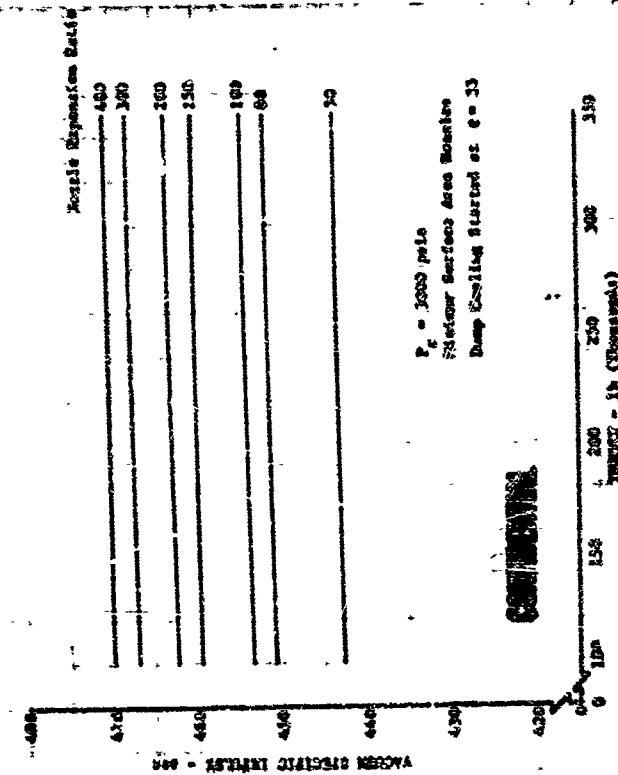
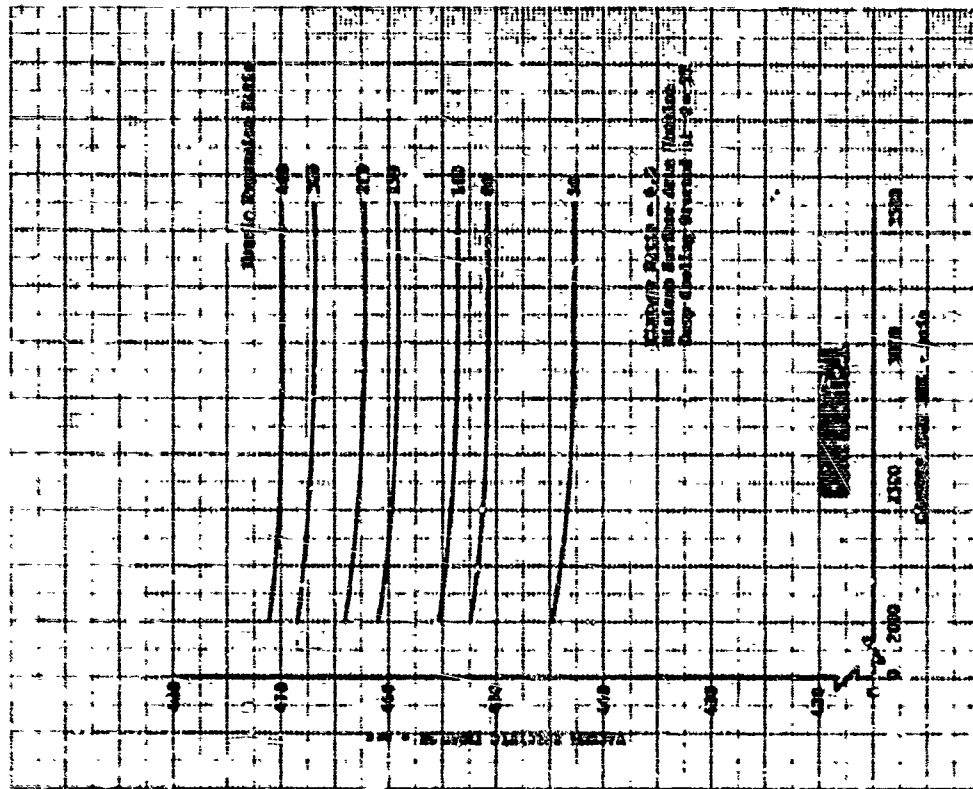


Figure 650. Vacuum Specific Impulse vs Thrust With Minimum Surface Area Contour Two-Position Nozzle (r = 6.0)

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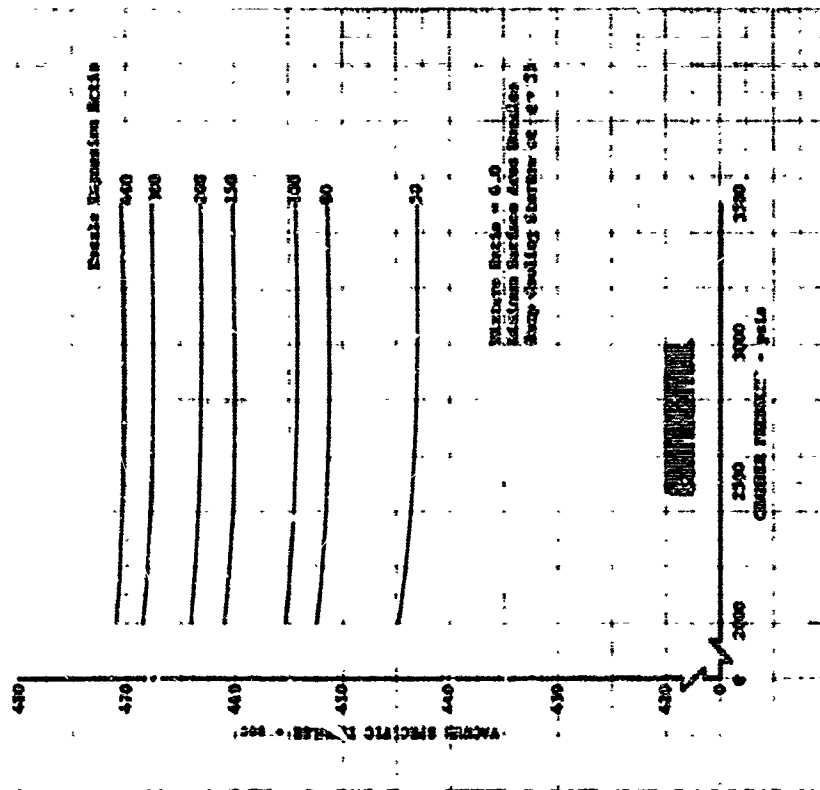


Figure 655. Vacuum Specific Impulse vs DP 56212
Chamber Pressure (200,000-lb Thrust)

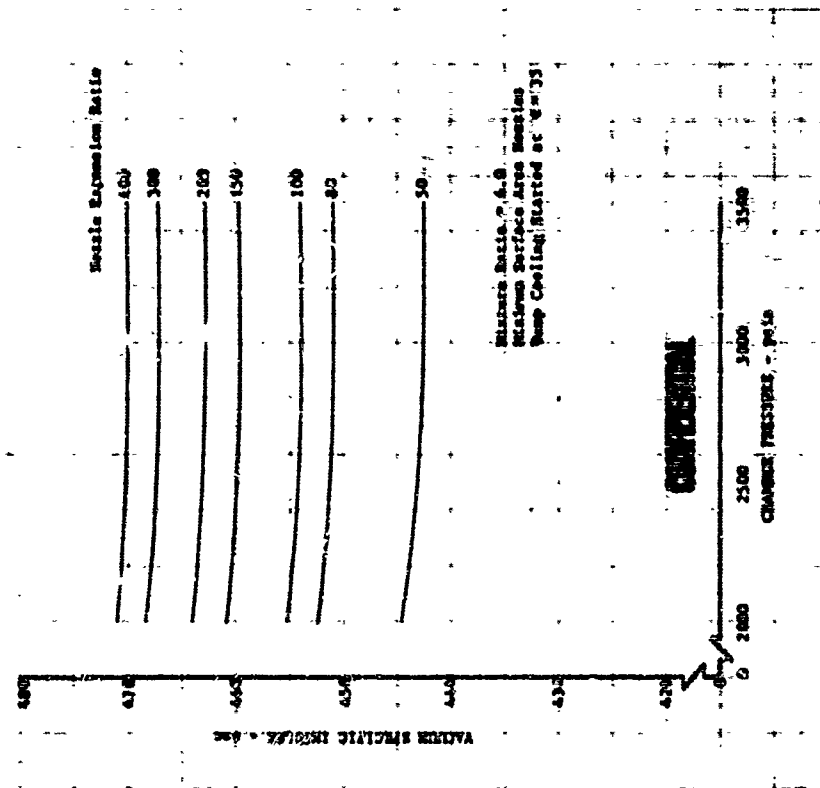


Figure 654. Vacuum Specific Impulse vs DP 56223
Chamber Pressure (150,000-lb Thrust)

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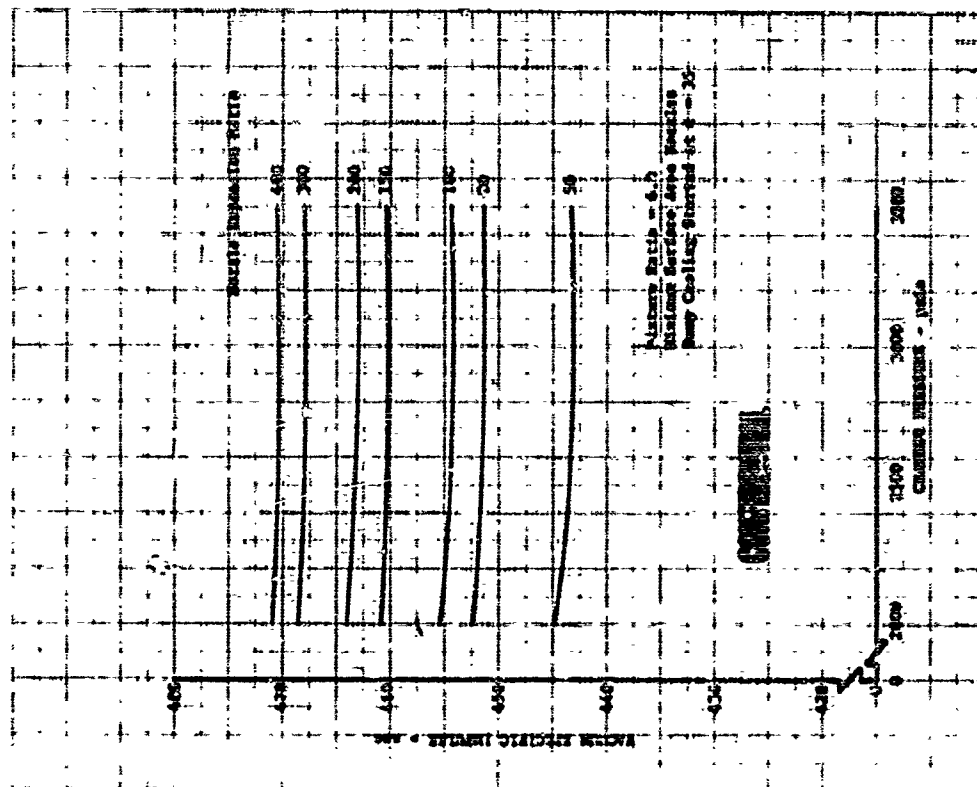


Figure 657. Vacuum Specific Impulse vs Chamber Pressure (300,000-lb Thrust) DF 56220

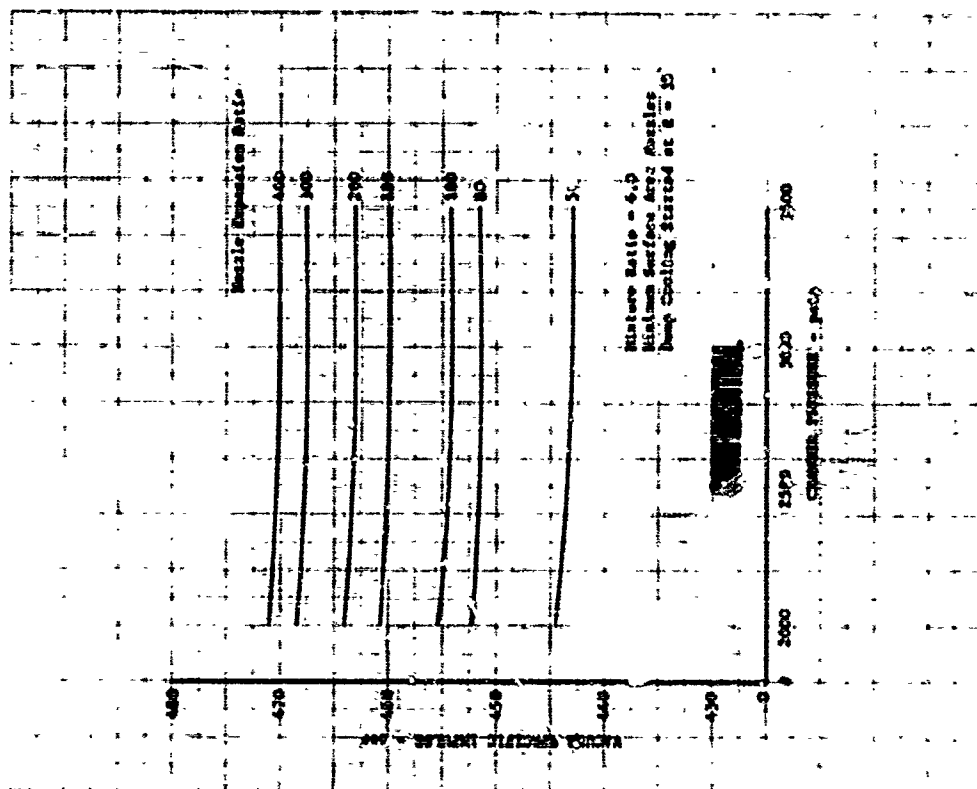


Figure 656. Vacuum Specific Impulse vs Chamber Pressure (250,000-lb Thrust) DF 56221

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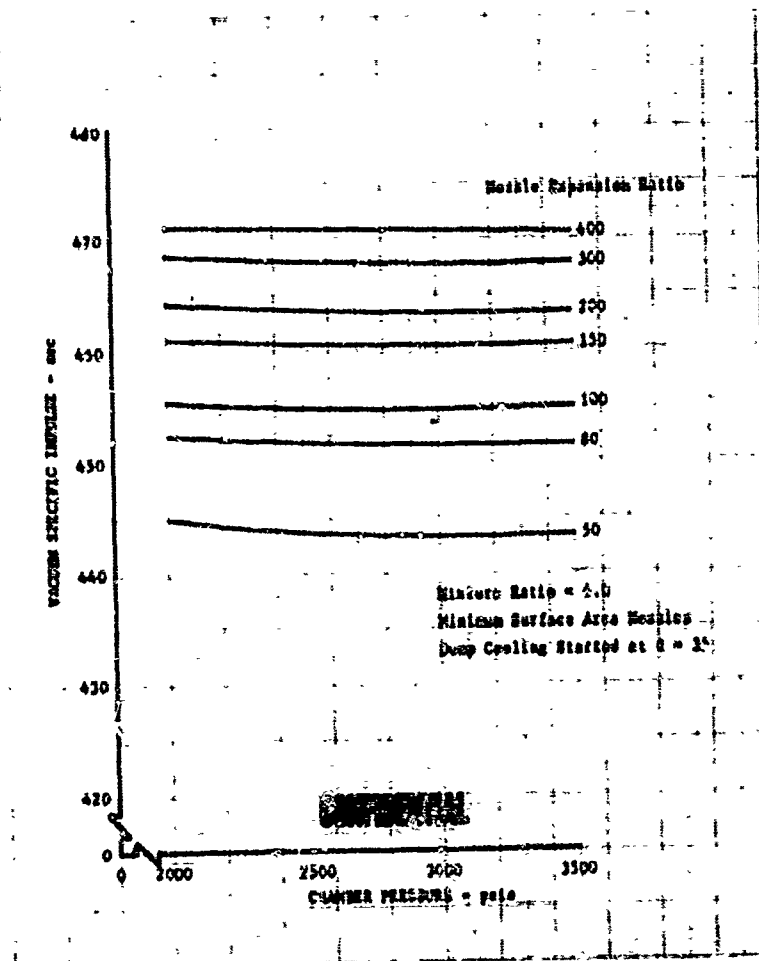
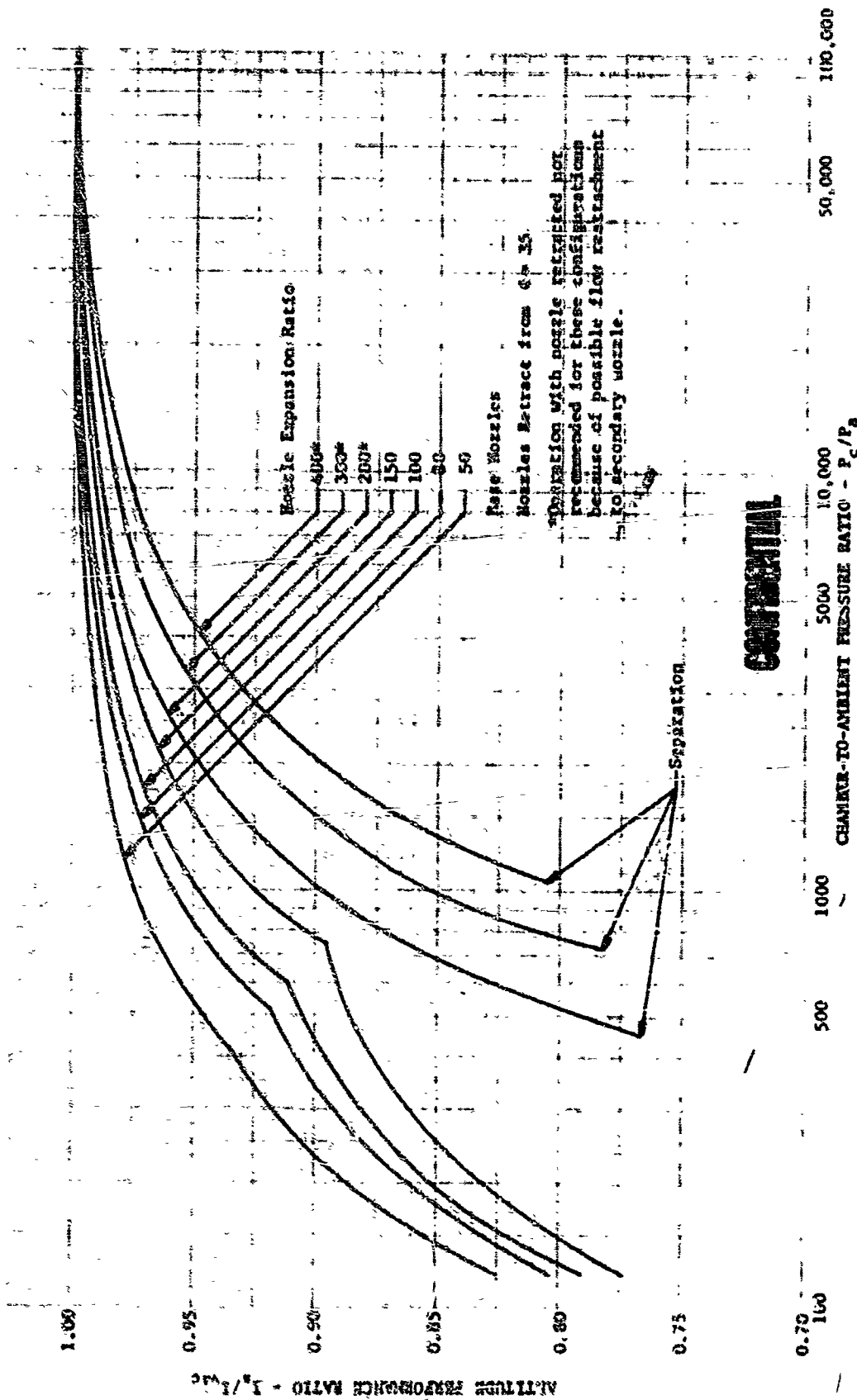


Figure 658. Vacuum Specific Impulse vs Chamber Pressure (350,000-lb thrust)

DF 56279

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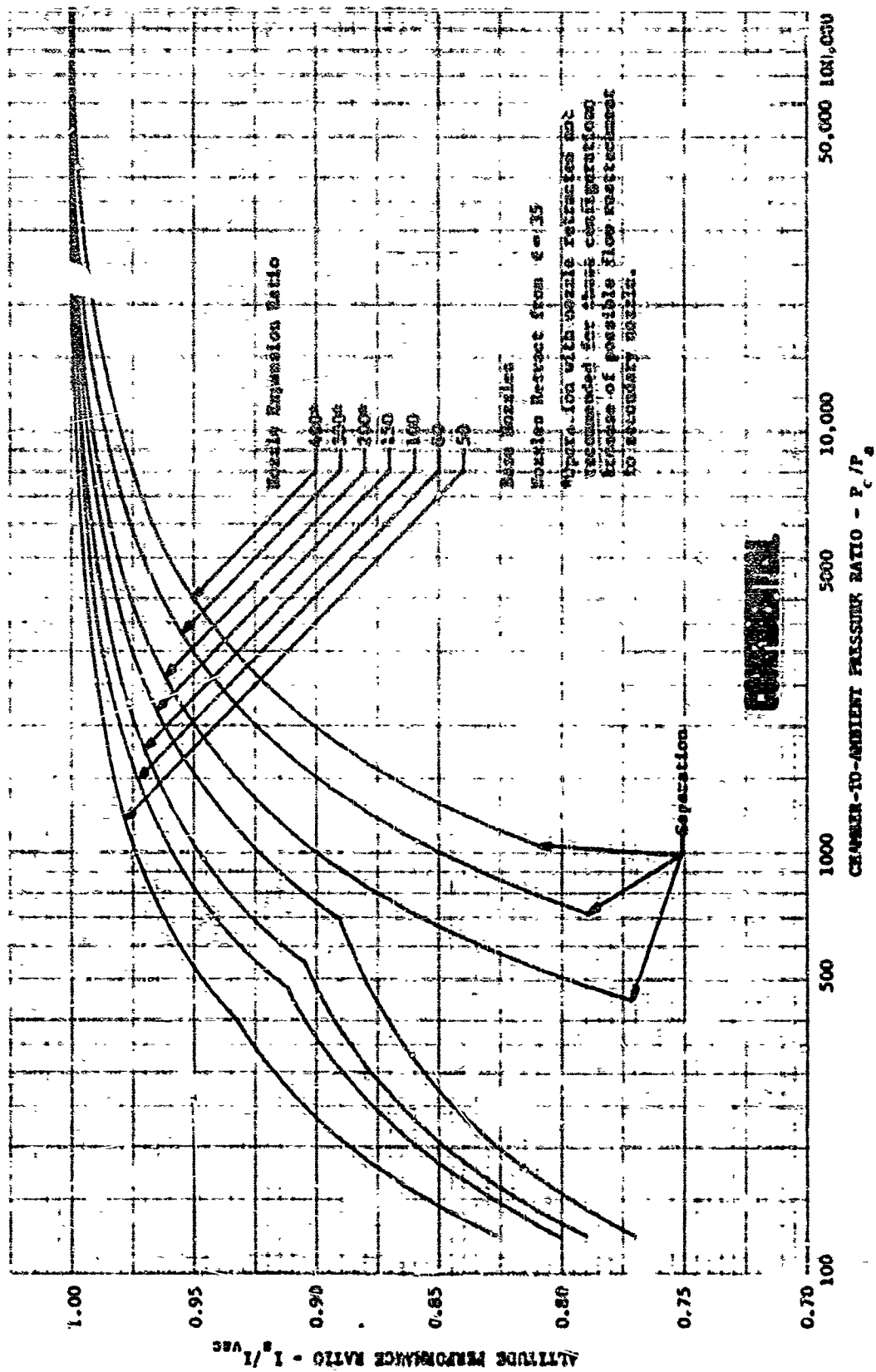
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Figure 659. Altitude Performance With Base Contour Two-Position Nozzle ($r = 5.0$)

DF 56031

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Figure 660. Altitude Performance With Base Contour Two-Position Nozzle ($r = 6.0$)

DF 56032

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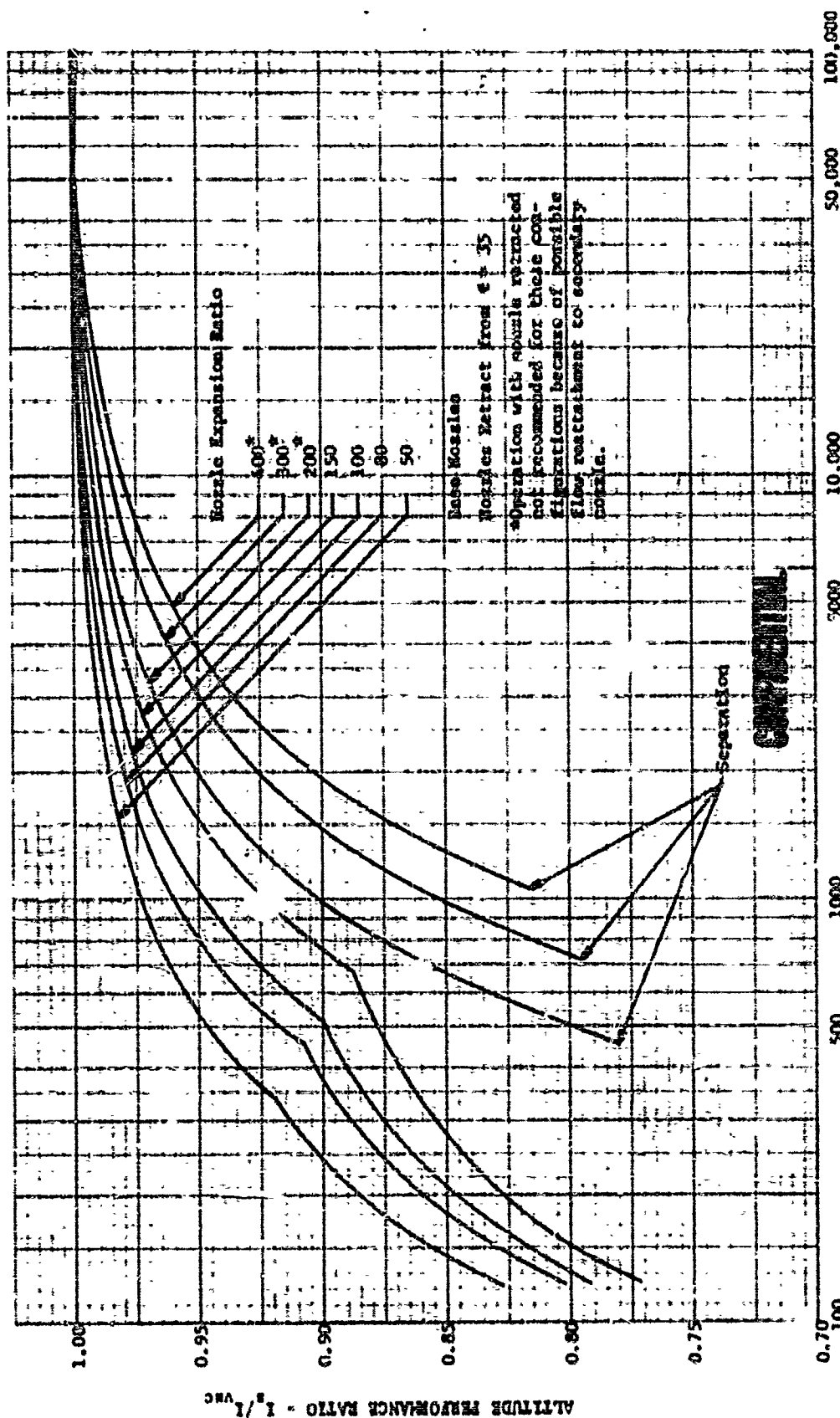


Figure 661. Altitude Performance With Base Contour Two-Position Nozzle ($x = 7.0$)

DF 56033

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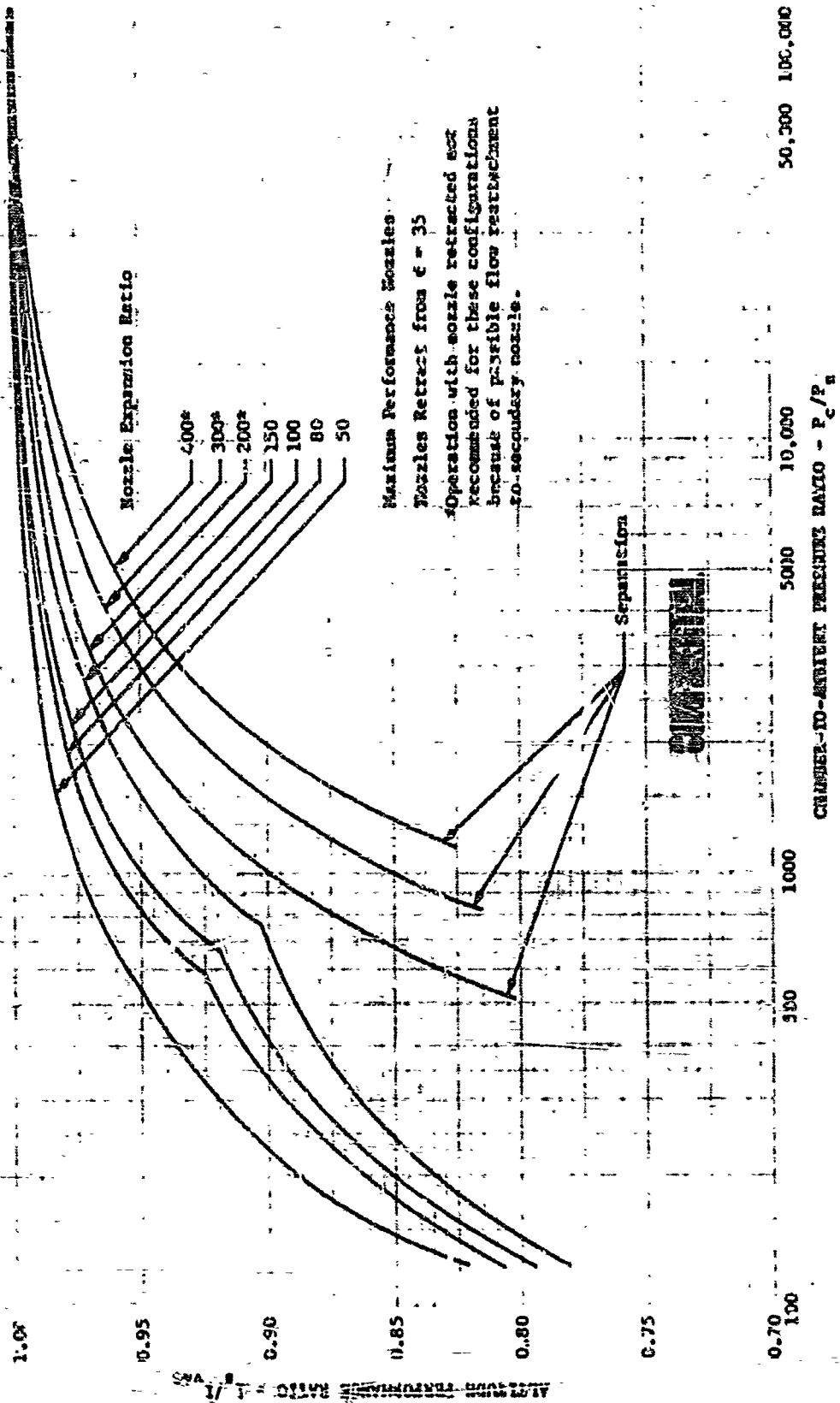
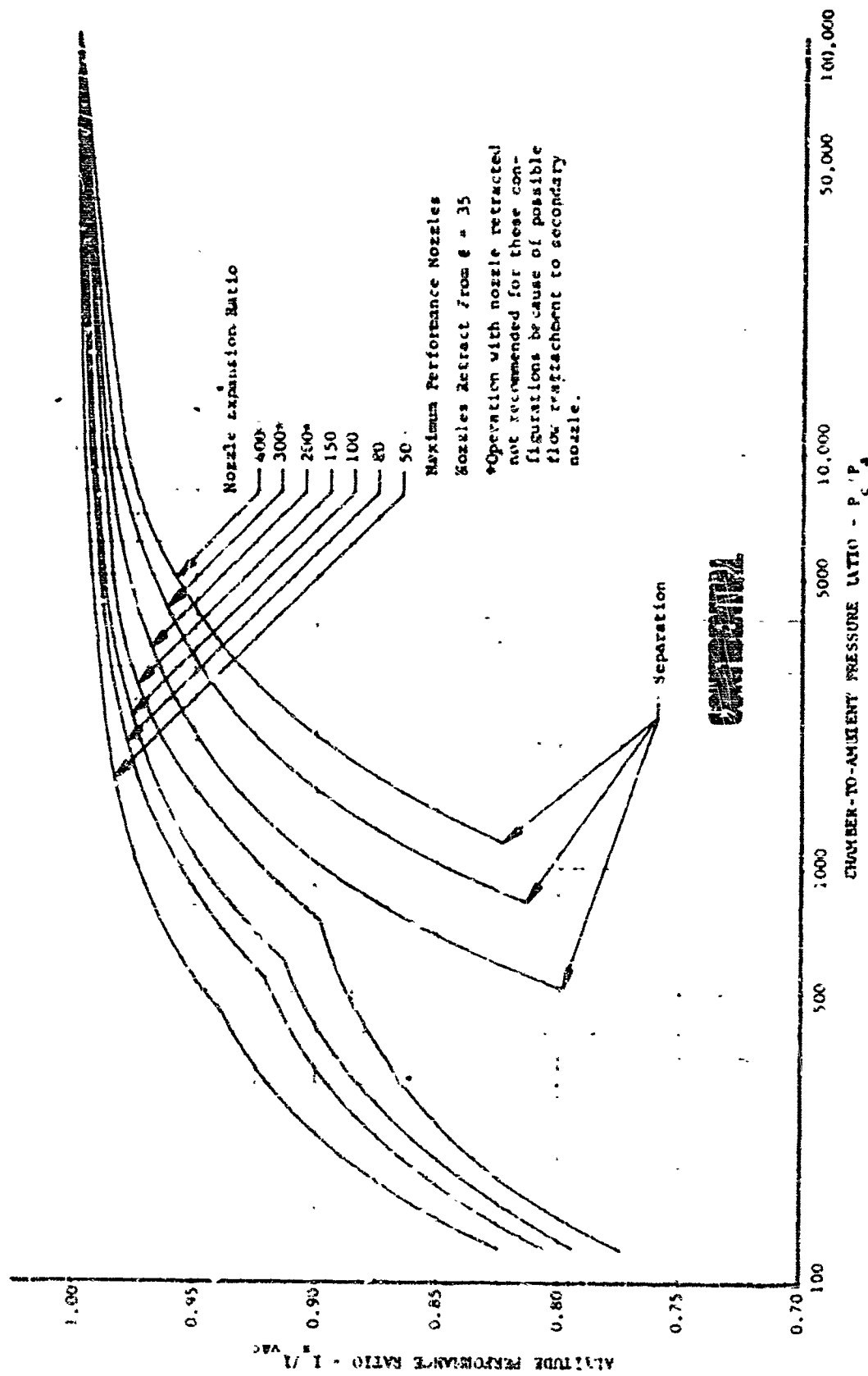


Figure 662. Altitude Performance With Maximum Performance Contour Two-Position Nozzle ($\epsilon = 5.0$) DR 56028

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Figure 663. Altitude Performance With Maximum Performance Contour Two-Position Nozzle ($\epsilon = 6.0$) DF 56029

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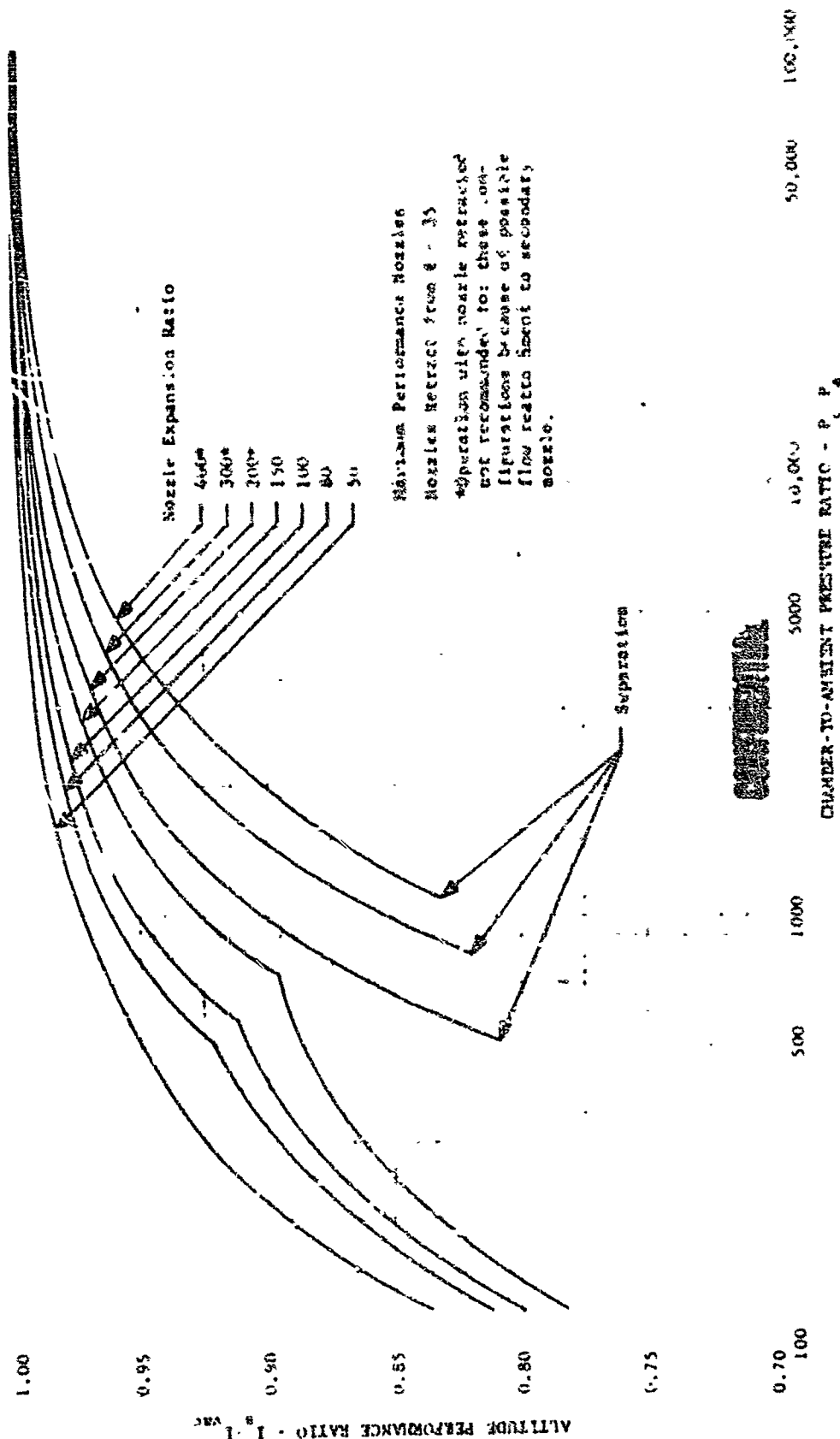


Figure 664. Altitude Performance With Maximum Performance Contour Two-Position Nozzle ($r = 7.0$) DP 560310

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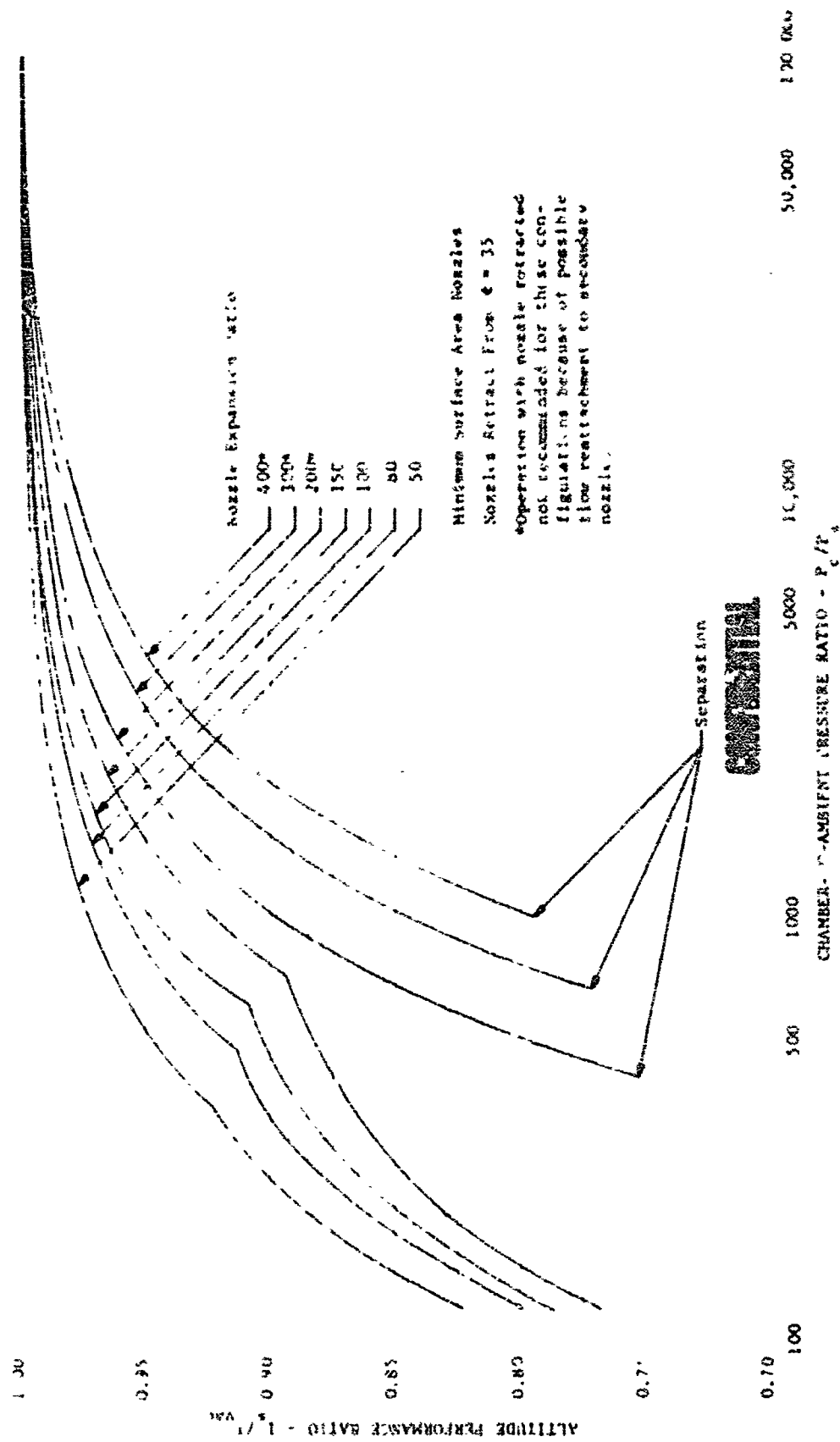
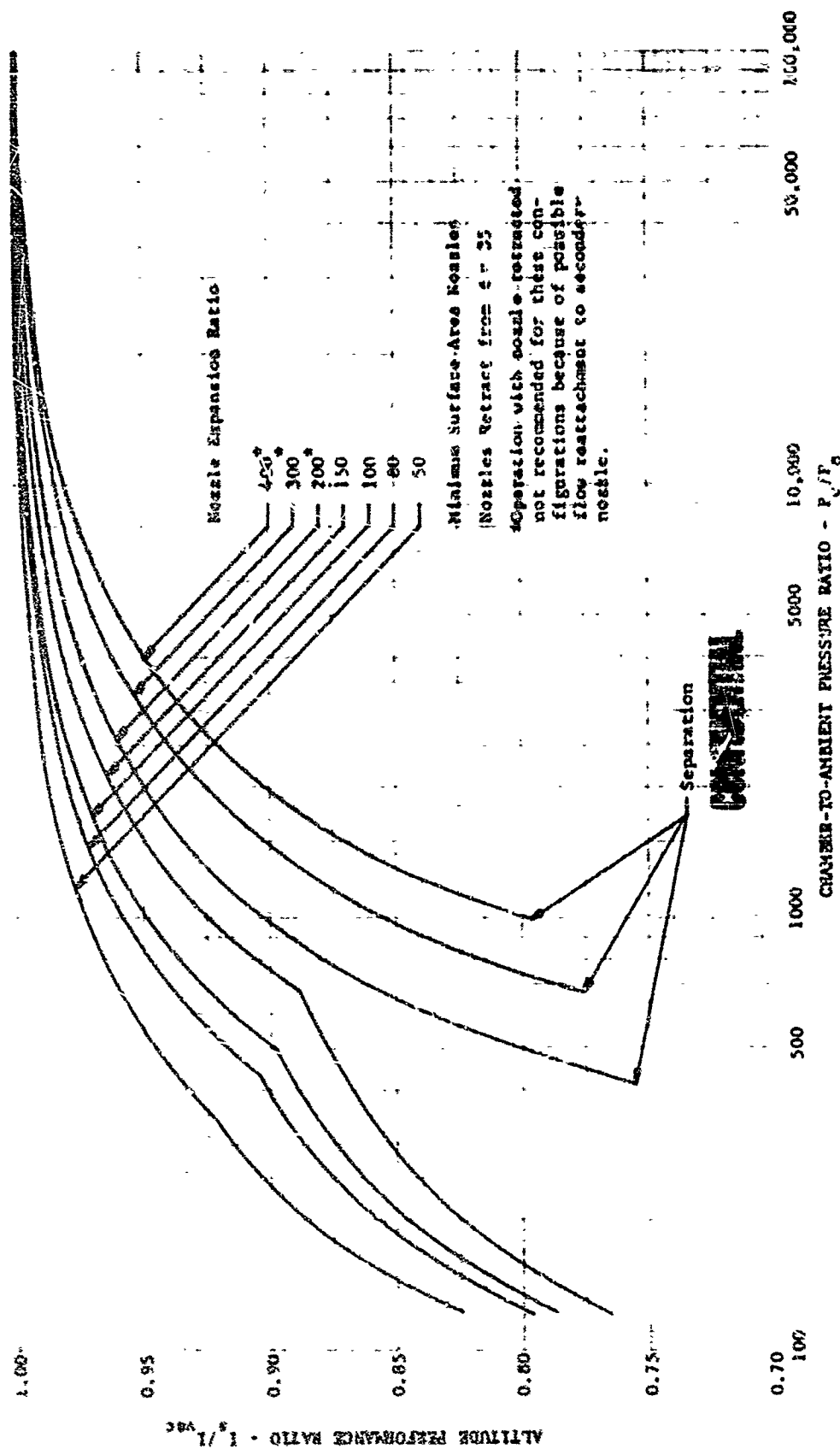


Figure 665. Altitude Performance With Minimum Surface Area Contour Two-Position Nozzle ($\epsilon = 5.0$) DF 54034

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Figure 666. Altitude Performance With Minimum Surface Area Conour Two-Position Nozzle ($r = 6.0$) DF 156013

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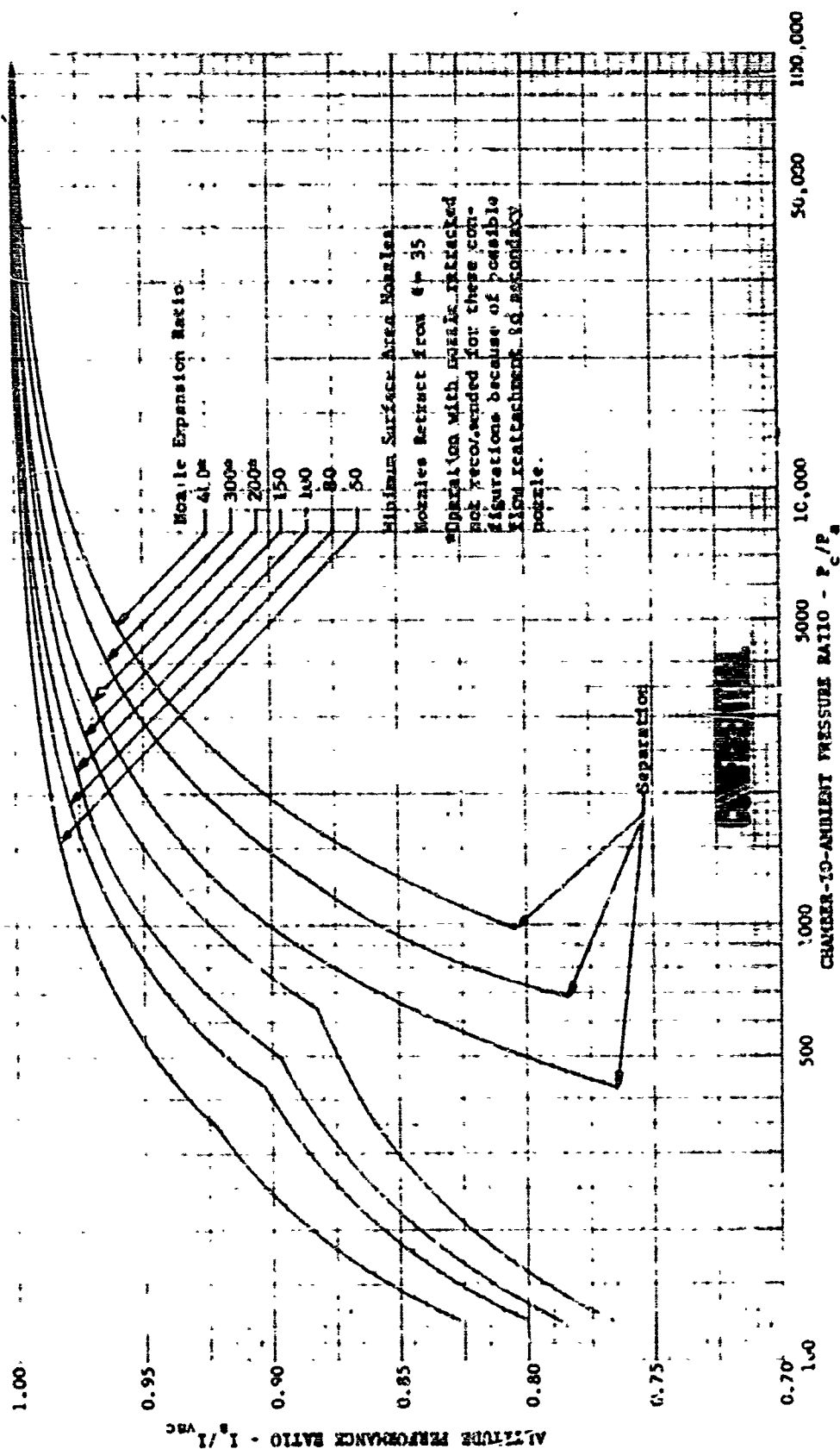


Figure 667. Altitude Performance With Minimum Surface Area Contour Two-Position Nozzle ($r = 7.0$) DF 56036

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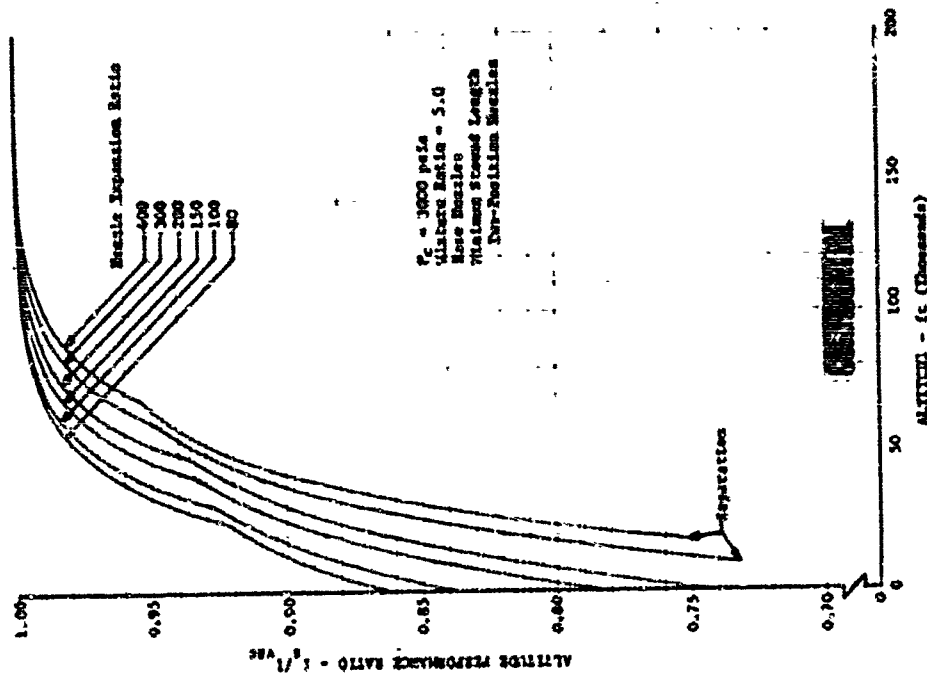


Figure 669. Altitude Performance With
Base Contour Nozzle,
250,000-lb Thrust, $r = 5.0$

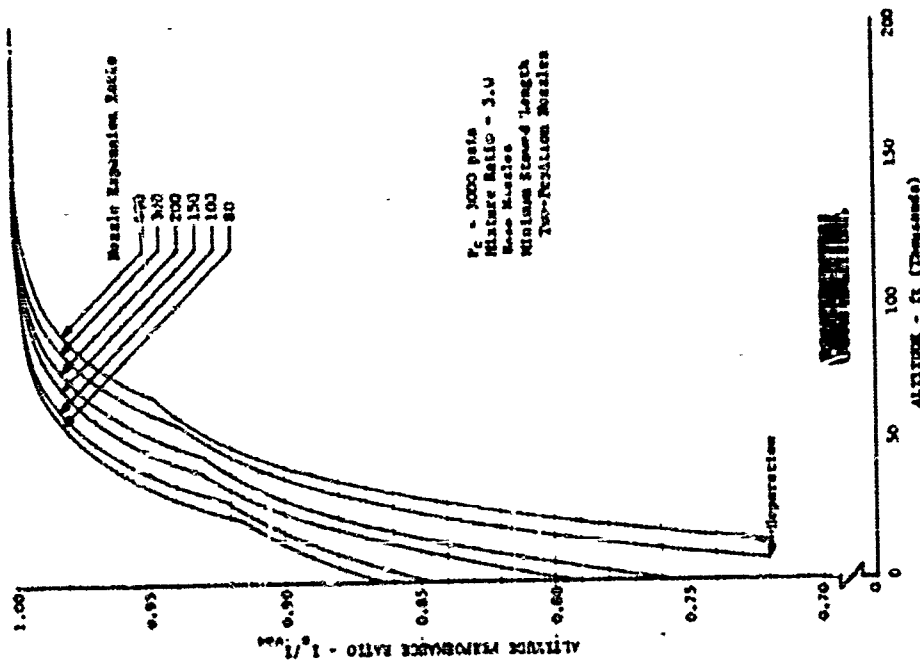
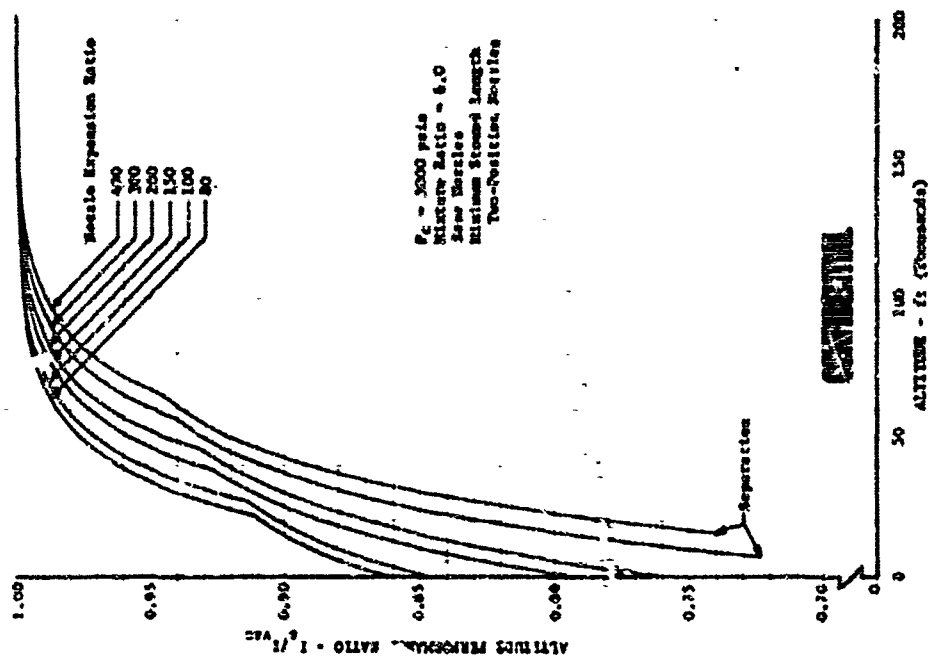


Figure 668. Altitude Performance With
Base Contour Nozzle,
100,000-lb Thrust, $r = 5.0$

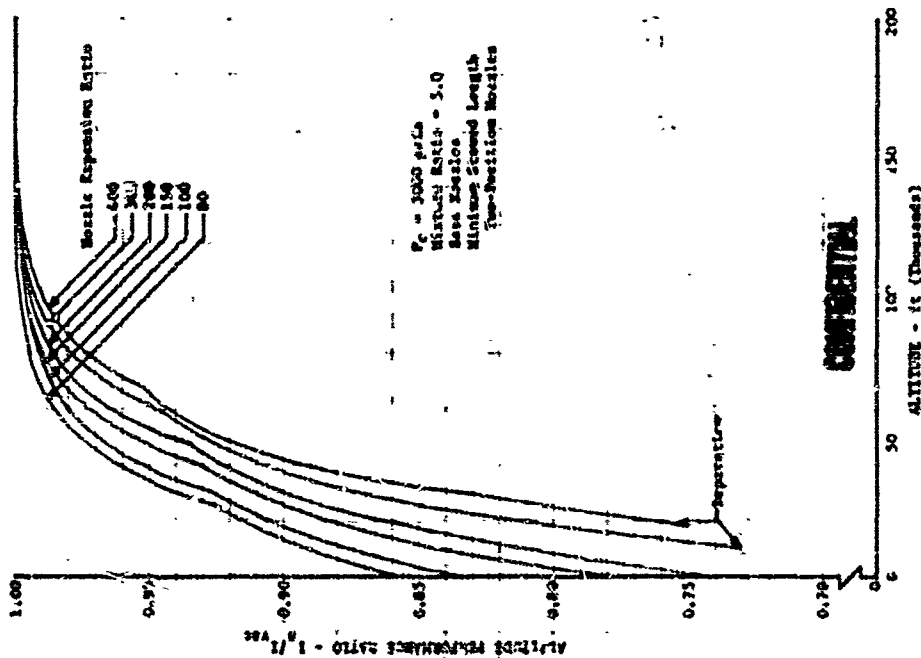
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Figure 671. Altitude Performance With DF 56169
Base Contour Nozzle,
100,000-lb Thrust, $r = 6.0$



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Figure 670. Altitude Performance With DF 56170
Base Contour Nozzle,
250,000-lb Thrust, $r = 5.0$

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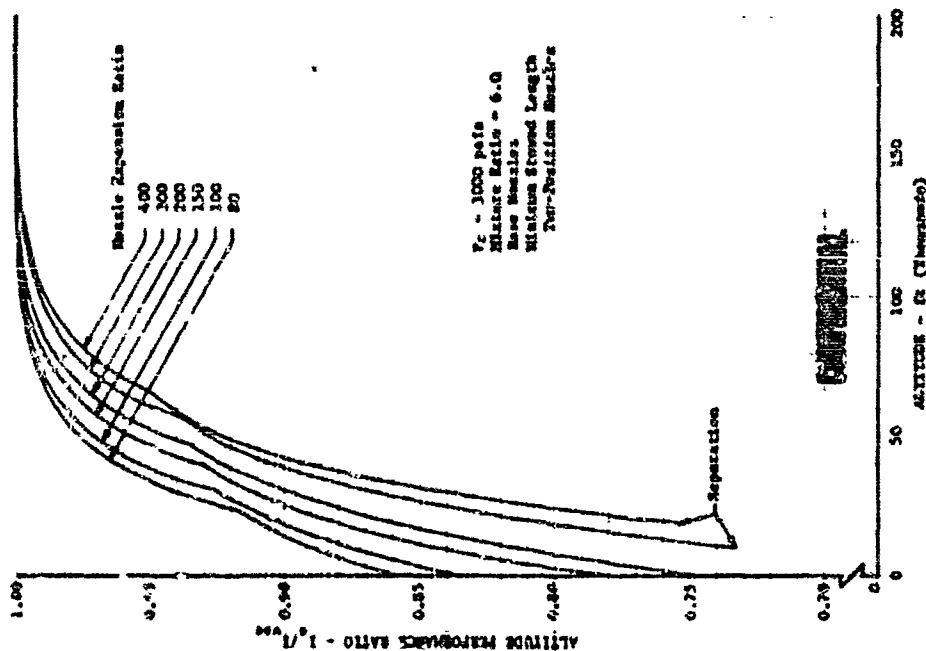


Figure 673. Altitude Performance With DP 56167
Base Contour Nozzle,
350,000-lb Thrust, $r = 6.0$

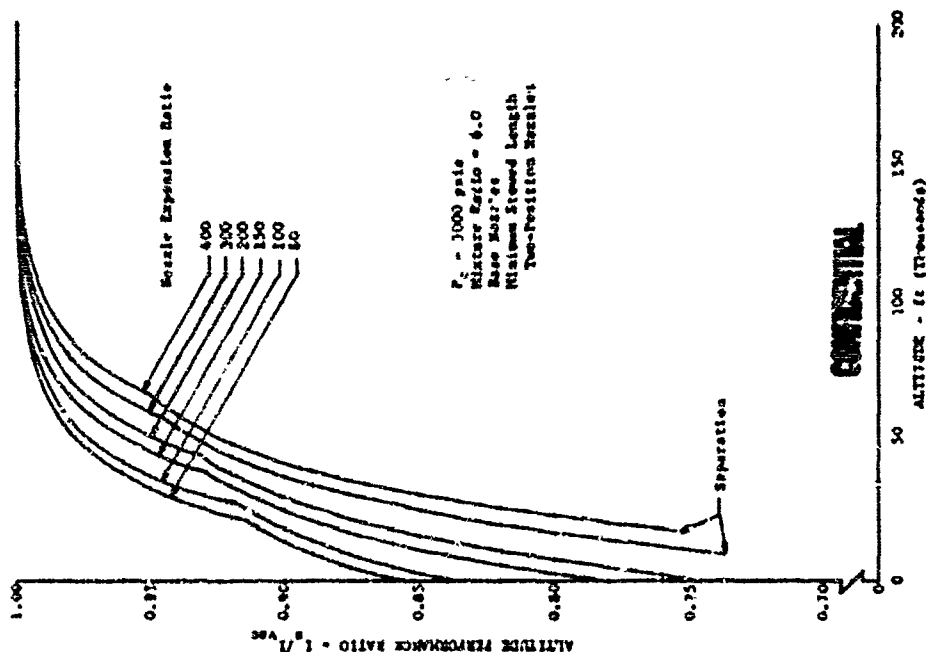


Figure 672. Altitude Performance With DF 56168
Base Contour Nozzle,
250,000-lb Thrust, $r = 6.0$

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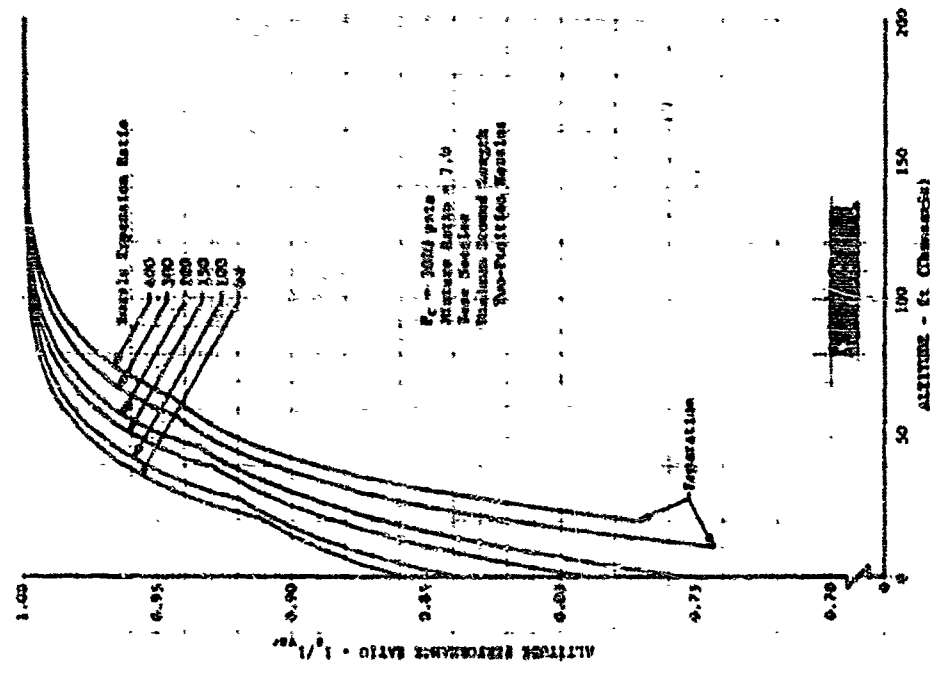


Figure 675. Altitude Performance With DP 56183
Base Contour Nozzle,
250,000-lb Thrust, $r = 7.0$

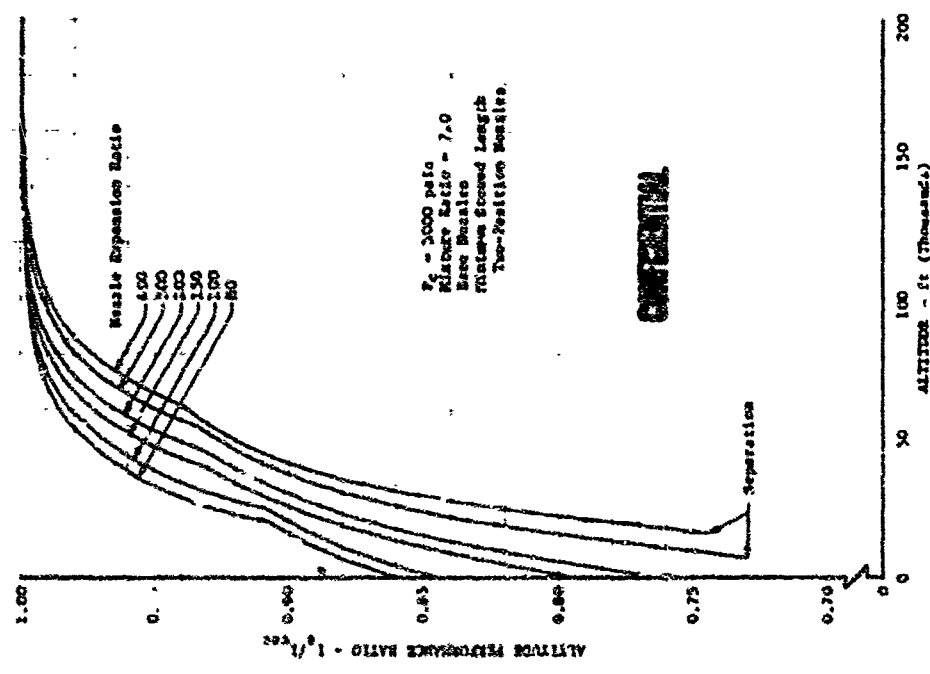


Figure 674. Altitude Performance With DP 56184
Base Contour Nozzle,
100,000-lb Thrust, $r = 7.0$

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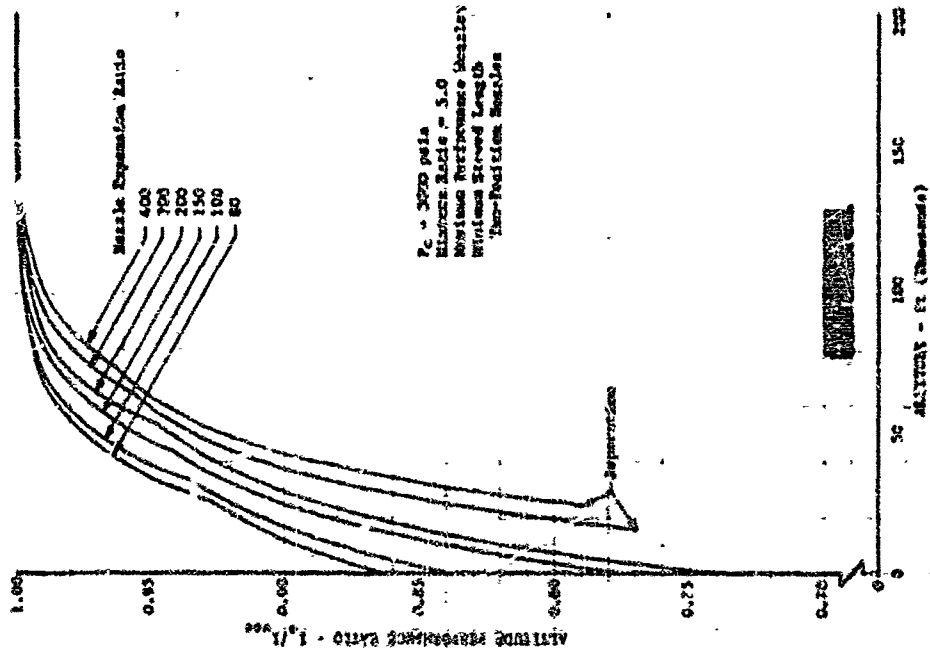


Figure 677. Altitude Performance With DF 56181.
 Maximum Performance Contour
 Nozzle, 100,000-lb Thrust,
 $r = 5.0$

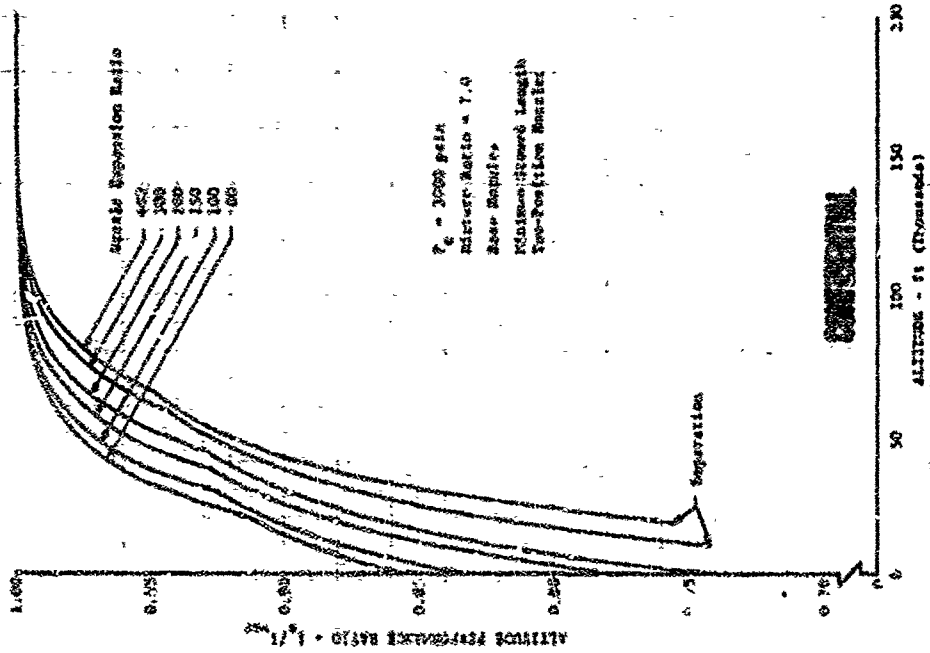


Figure 676. Altitude Performance With DF 56182
 Base Contour Nozzle,
 350,000-lb Thrust, $r = 7.0$

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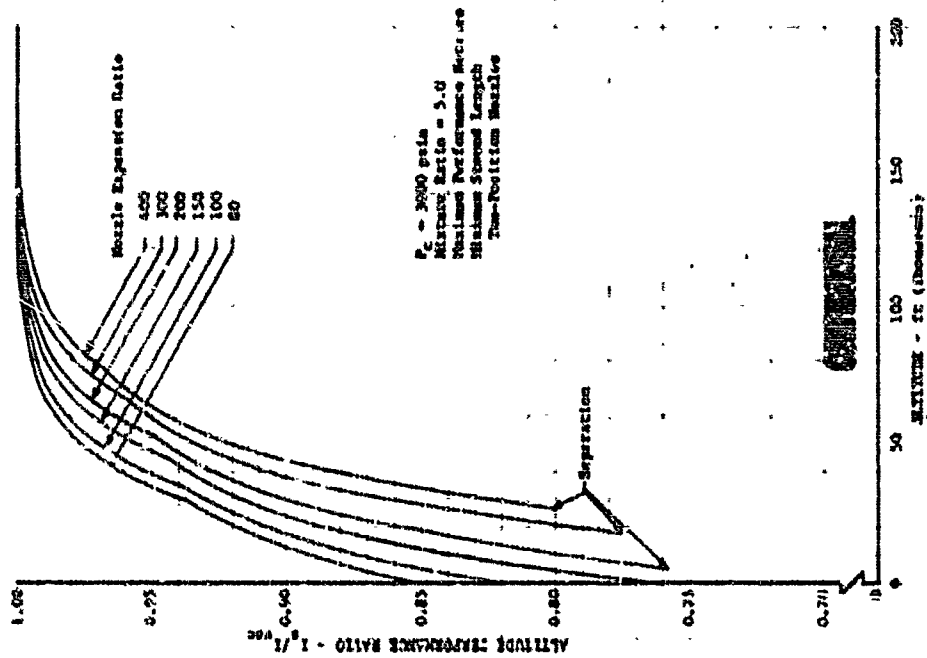


Figure 679. Altitude Performance With
Maximum Performance Contour
Nozzle, 350,000-lb Thrust,
 $r = 5.0$

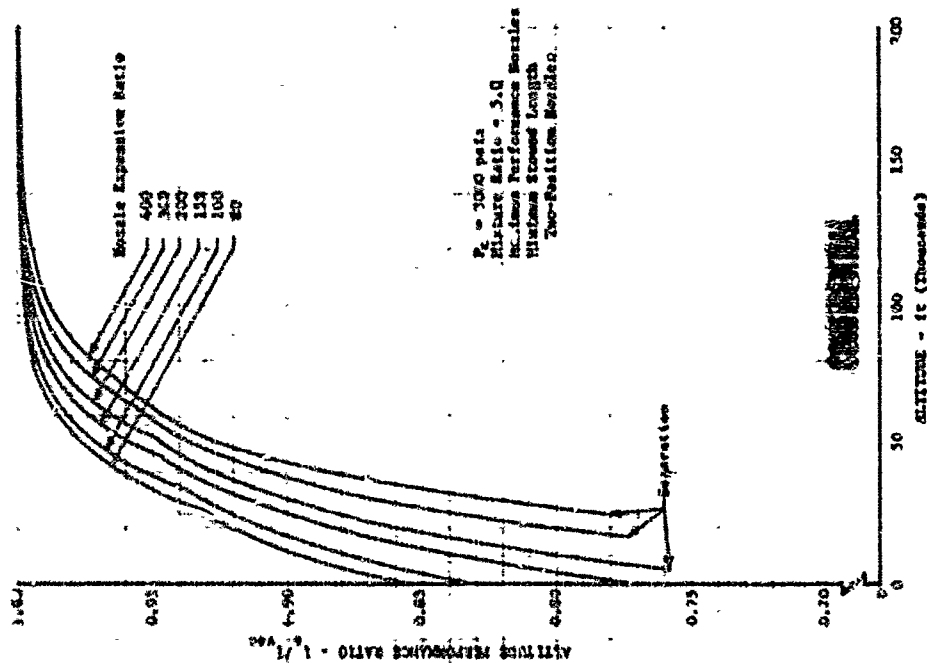


Figure 678. Altitude Performance With
Maximum Performance Contour
Nozzle, 250,000-lb Thrust,
 $r = 5.0$

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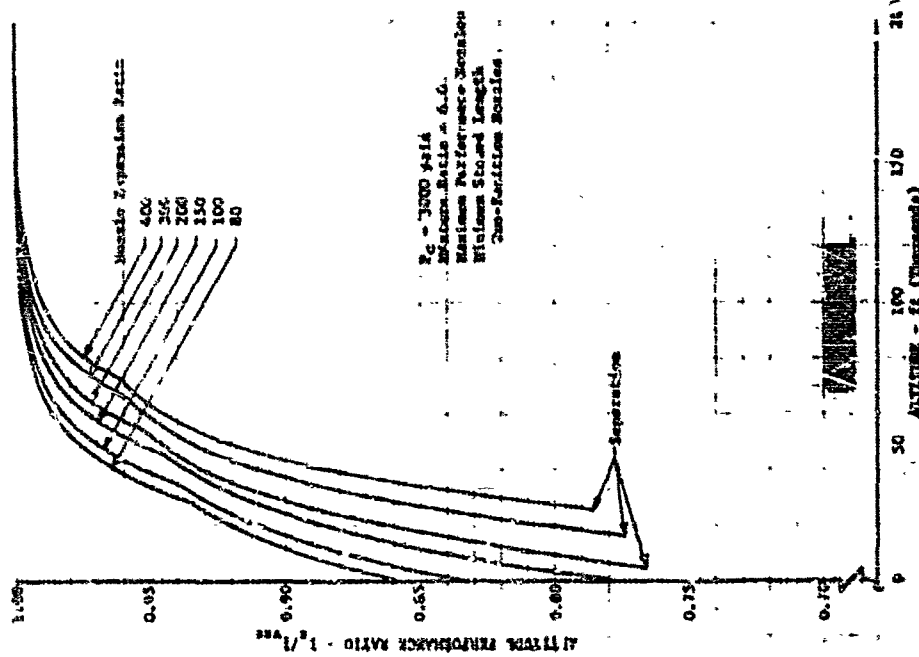


Figure 681. Altitude Performance With
Maximum Performance Contour
Nozzle, 250,000-lb Thrust,
 $r = 6.0$

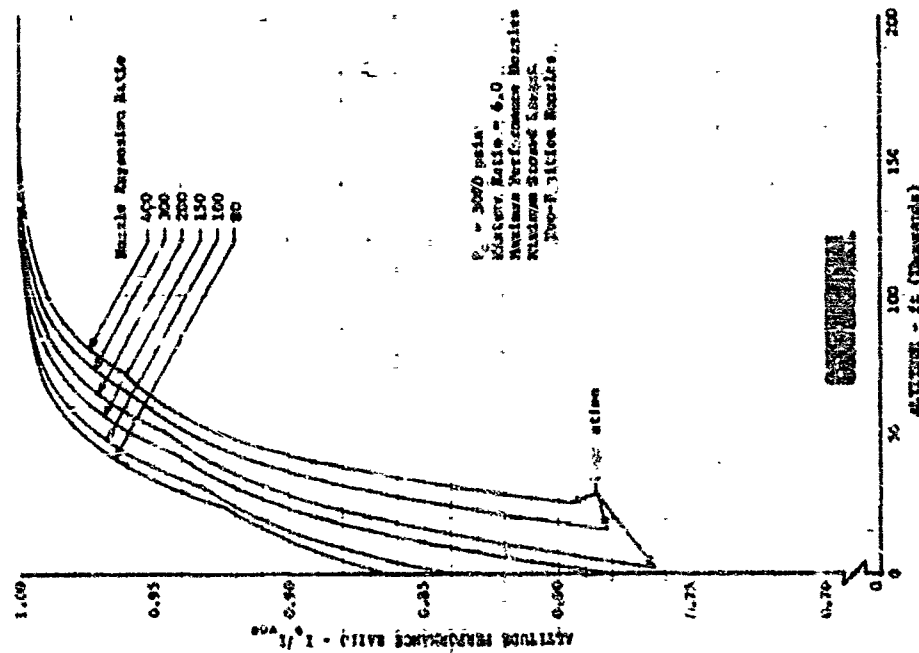


Figure 680. Altitude Performance With
Maximum Performance Contour
Nozzle, 100,000-lb Thrust,
 $r = 6.0$

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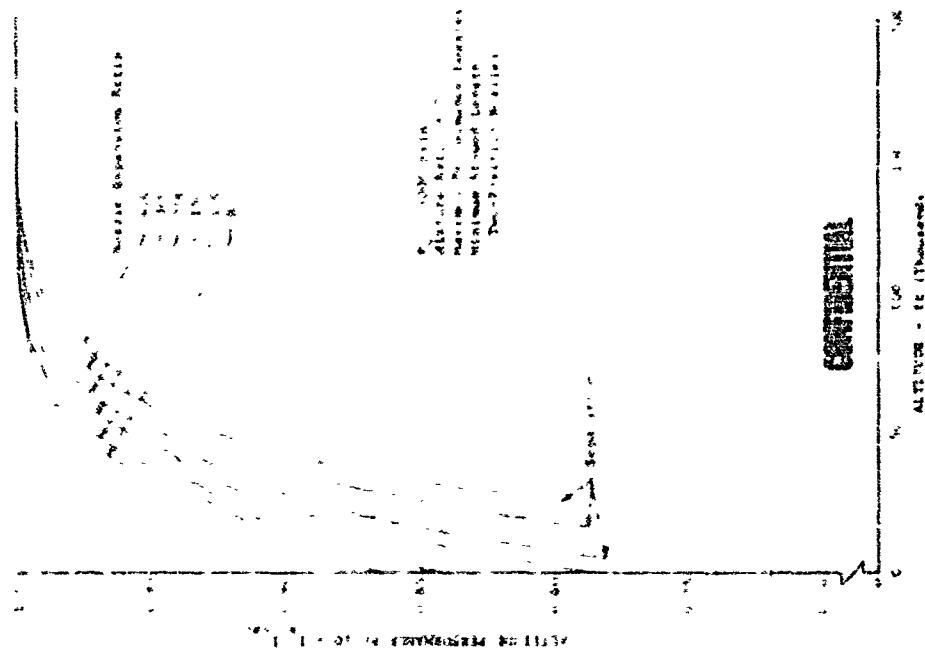


Figure 683. Altitude Performance With
Maximum Performance Contour
Nozzle, 100,000-lb Thrust,
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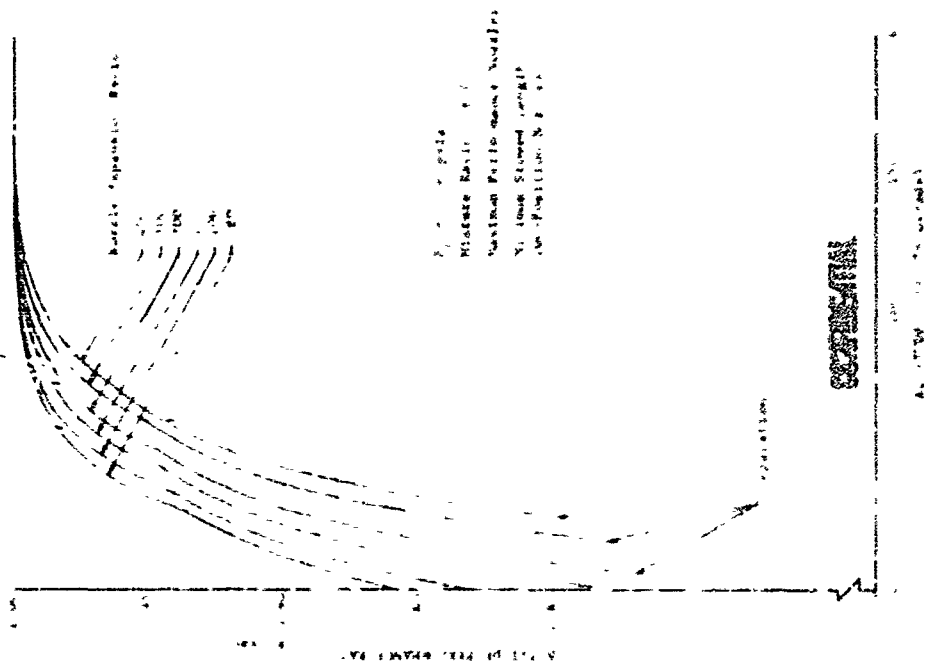


Figure 682. Altitude Performance With
Maximum Performance Contour
Nozzle, 350,000-lb Thrust,
 $r = 6.0$

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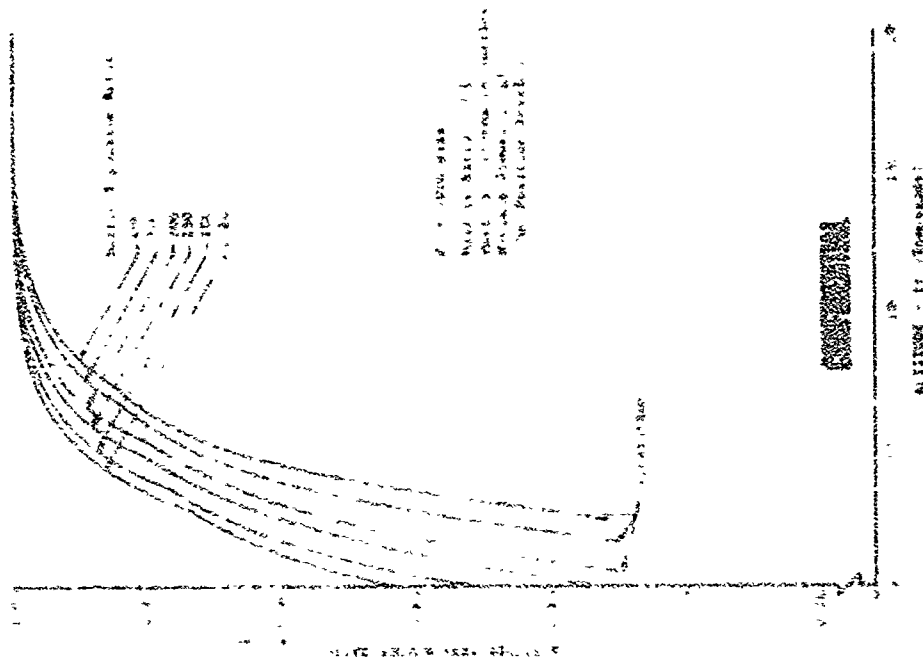


Figure 68. Altitude Performance With Maximum Performance Contour
Nozzle, 300,000-lb Thrust,
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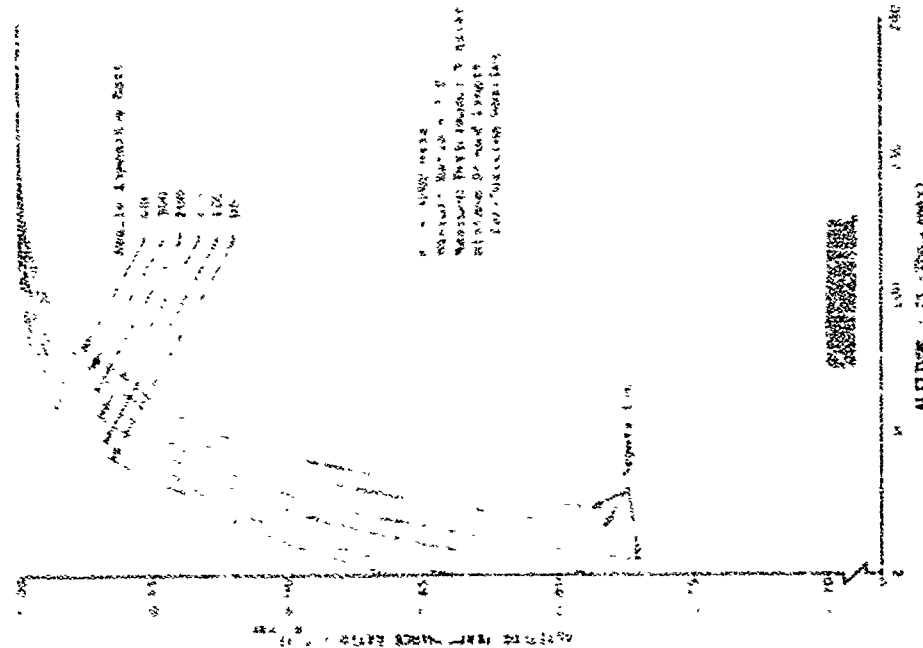
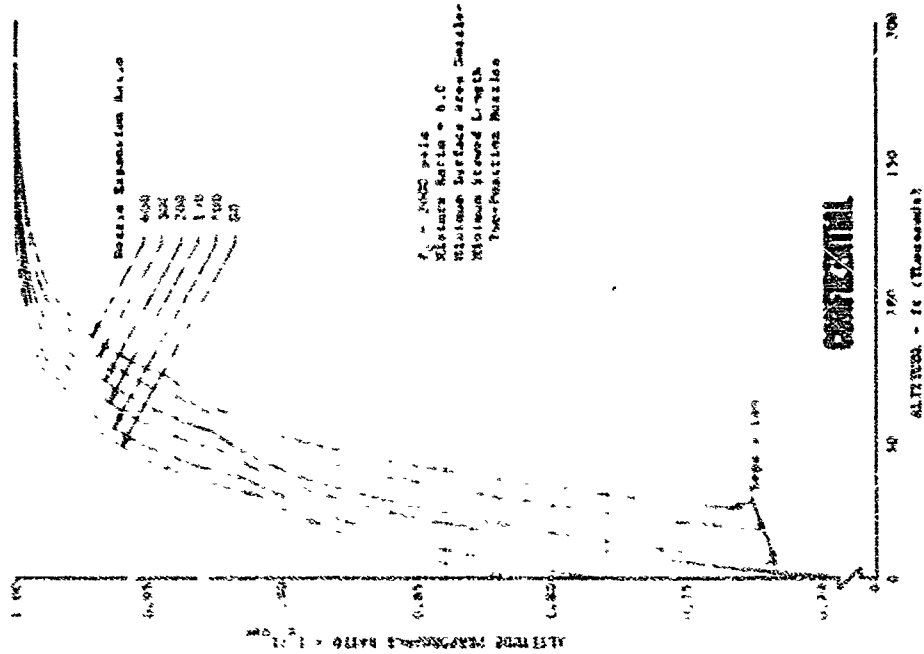


Figure 69. Altitude Performance With Maximum Performance Contour
Nozzle, 300,000-lb Thrust,
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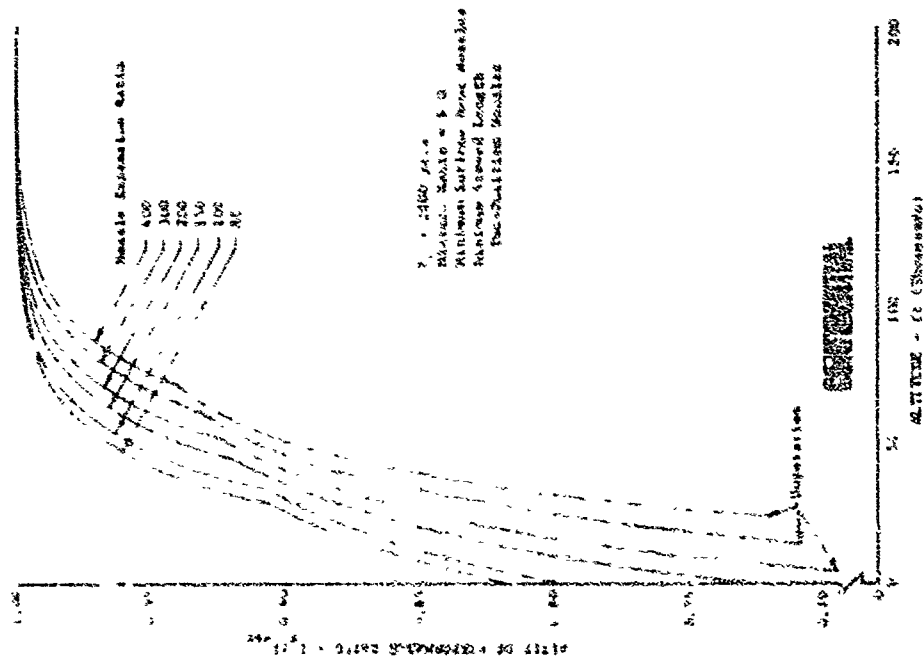
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Figure 687. Altitude Performance With DF 56180
Minimum Surface Area Contour
Nozzle, 250,000-lb Thrust,
 $r = 6.0$, $P_c = 2000$ psia



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Figure 688. Altitude Performance With DF 56189
Minimum Surface Area Contour
Nozzle, 100,000-lb Thrust,
 $r = 6.0$, $P_c = 2000$ psia

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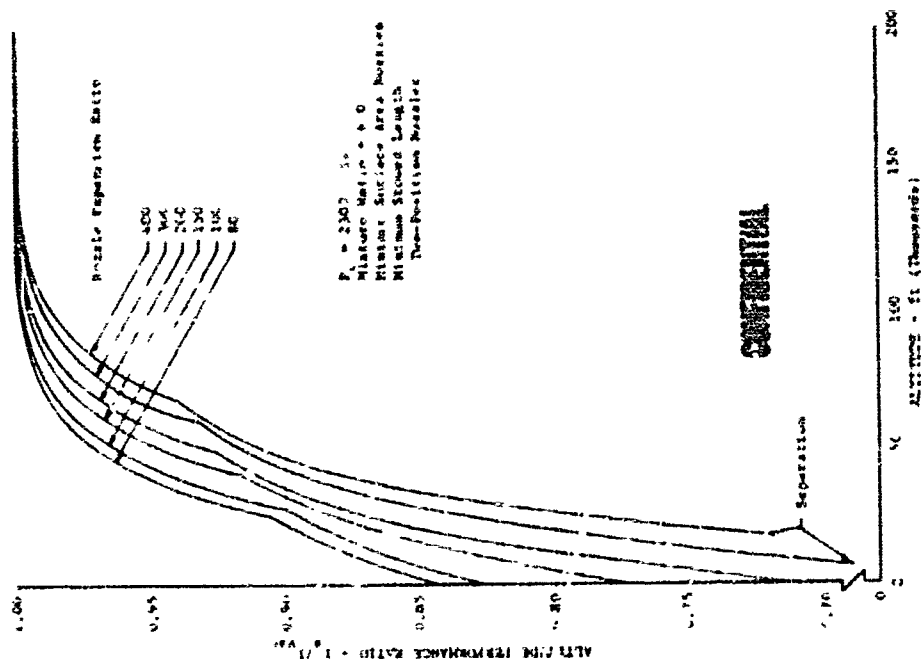


Figure 688. Altitude Performance With Minimum Surface Area Contour
Nozzle, 350,000-lb Thrust,
 $r = 5.0$, $P_c = 2000 \text{ psia}$

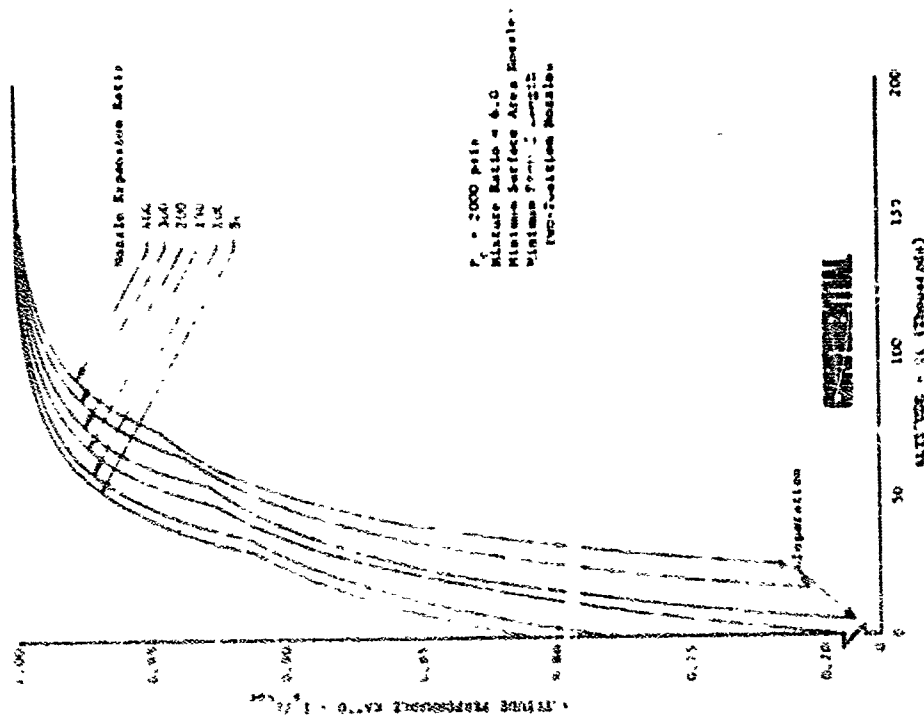


Figure 689. Altitude Performance With Minimum Surface Area Contour
Nozzle, 100,000-lb Thrust,
 $r = 6.0$, $P_c = 2500 \text{ psia}$

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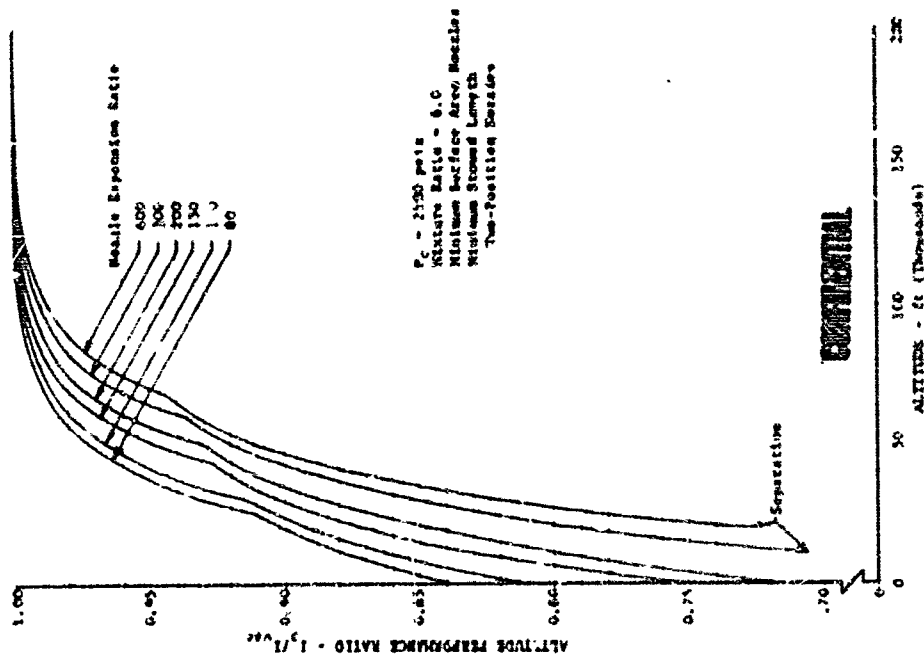


Figure 690. Altitude Performance With Minimum Surface Area Contour
 Nozzle, 250,000-lb Thrust,
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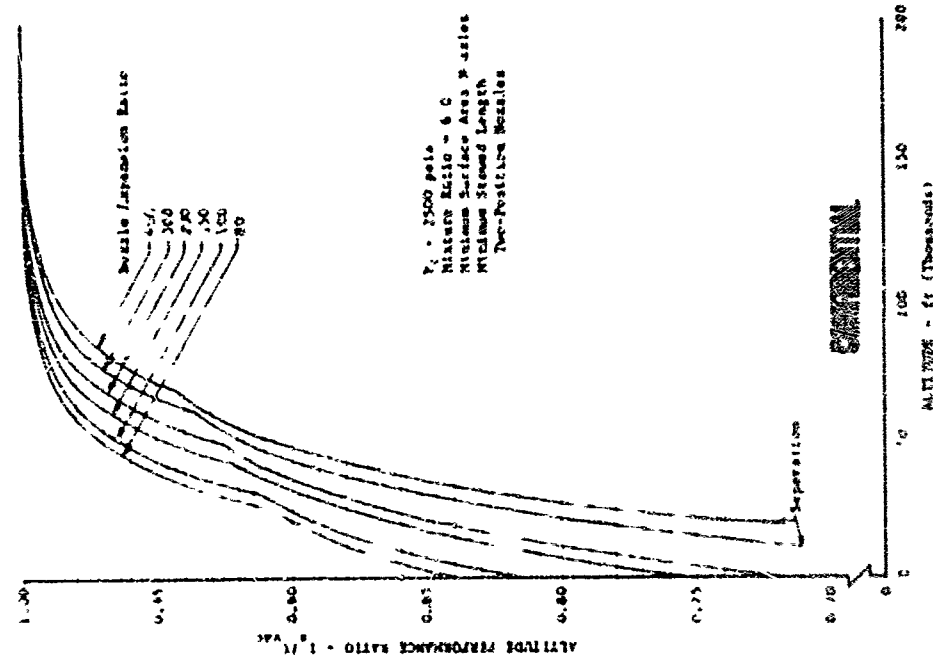


Figure 691. Altitude Performance With Minimum Surface Area Contour
 Nozzle, 350,000-lb Thrust,
 $r = 6.0$, $P_c = 2500$ psia

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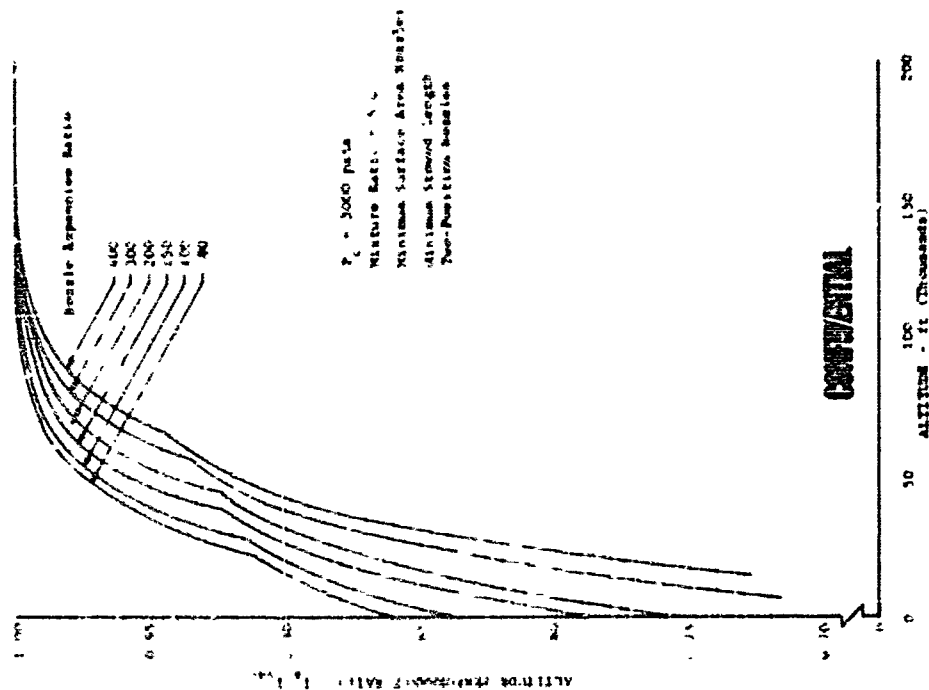


Figure 693. Altitude Performance With
Minimum Surface Area Contour
Nozzle, 250,000-lb Thrust,
 $r = 5.0$, $P_c = 3000$ psia

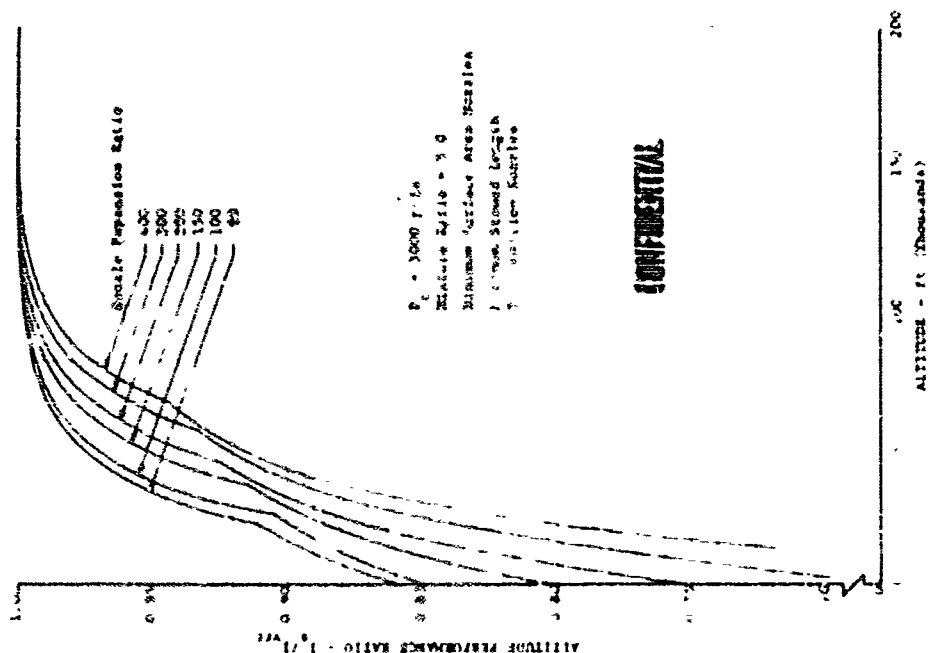


Figure 692. Altitude Performance With
Minimum Surface Area Contour
Nozzle, 100,000-lb Thrust,
 $r = 5.0$, $P_c = 3000$ psia

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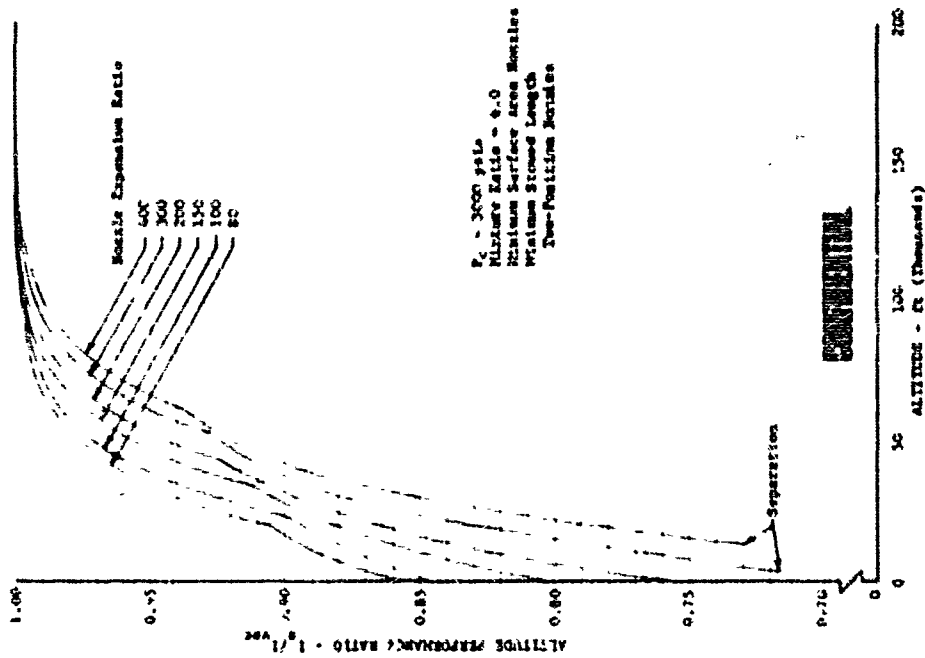


Figure 695. Altitude Performance With
Minimum Surface Area Contour
Nozzle, 10',000-lb Thrust,
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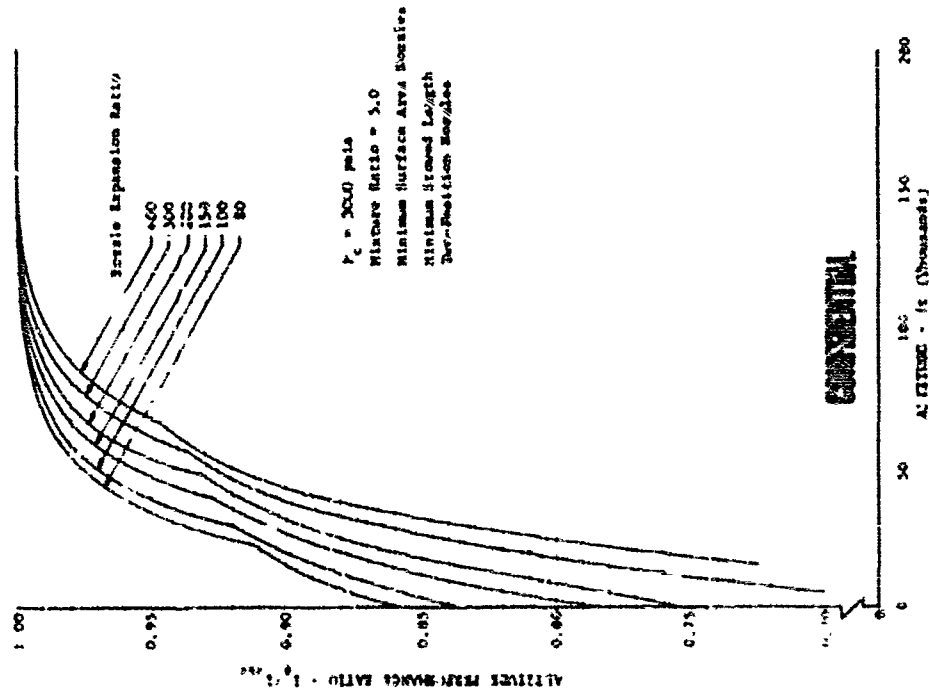


Figure 694. Altitude Performance With
Minimum Surface Area Contour
Nozzle, 350,000-lb Thrust,
 $r = 6.0$, $P_c = 3000$ psia

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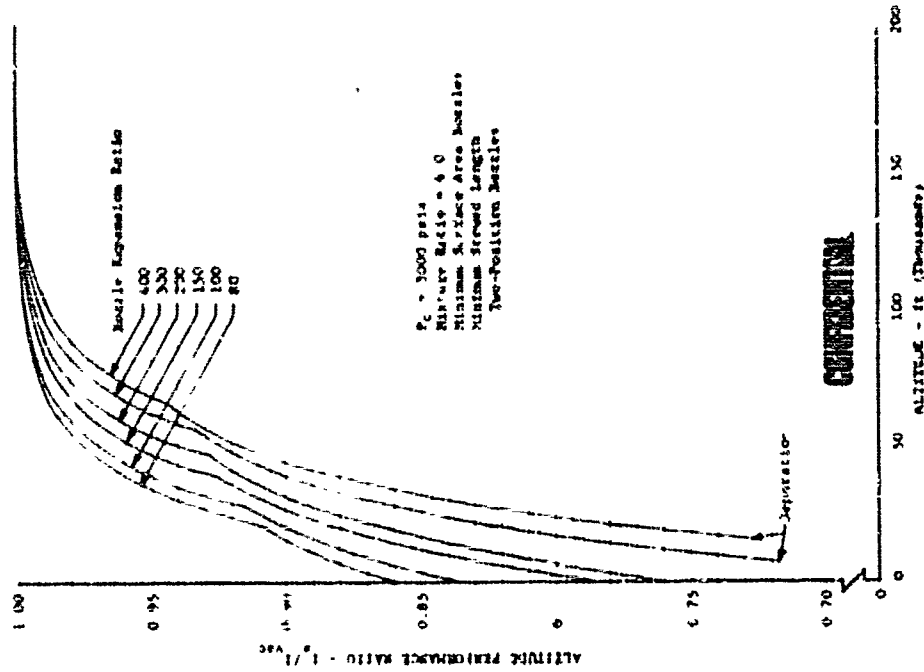


Figure 697. Altitude Performance With
 Minimum Surface Area Contour
 Nozzle, 350,000-lb Thrust,
 $r = 6.0$, $P_c = 3000 \text{ psia}$

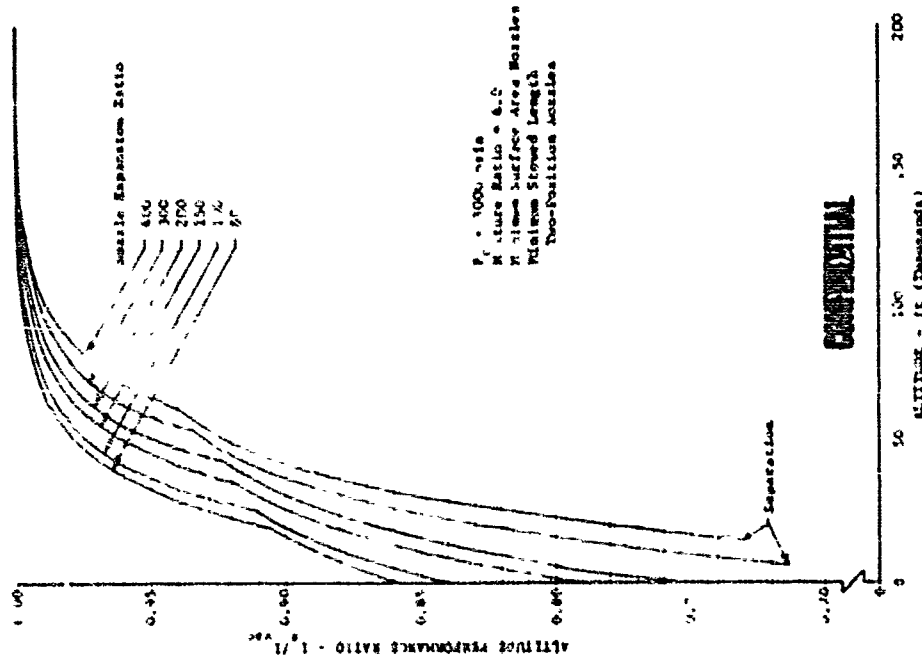


Figure 696. Altitude Performance With
 Minimum Surface Area Contour
 Nozzle, 250,000-lb Thrust,
 $r = 6.0$, $P_c = 3000 \text{ psia}$

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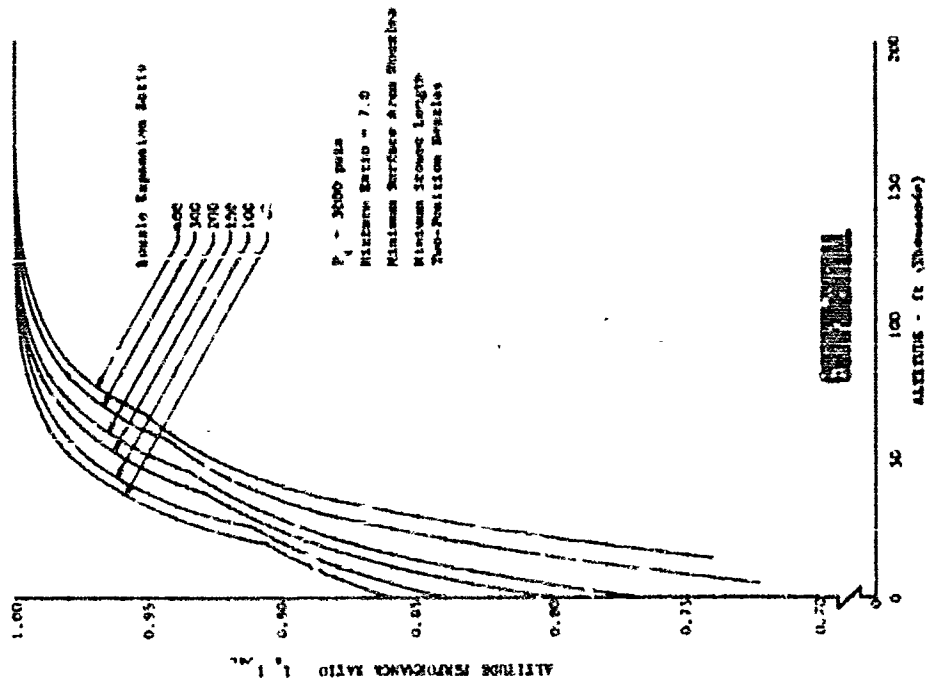


Figure 699. Altitude Performance With Minimum Surface Area Contour:
Nozzle, 250,000-lb Thrust,
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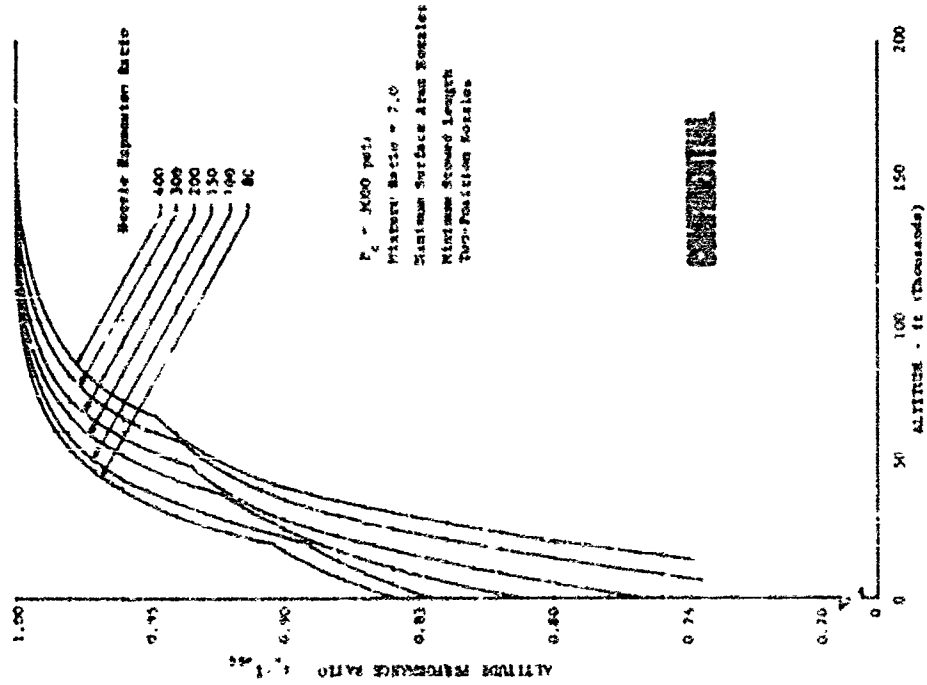


Figure 698. Altitude Performance With Minimum Surface Area Contour:
Nozzle, 100,000-lb Thrust,
 $r = 7.0$, $P_c = 3000$ psia

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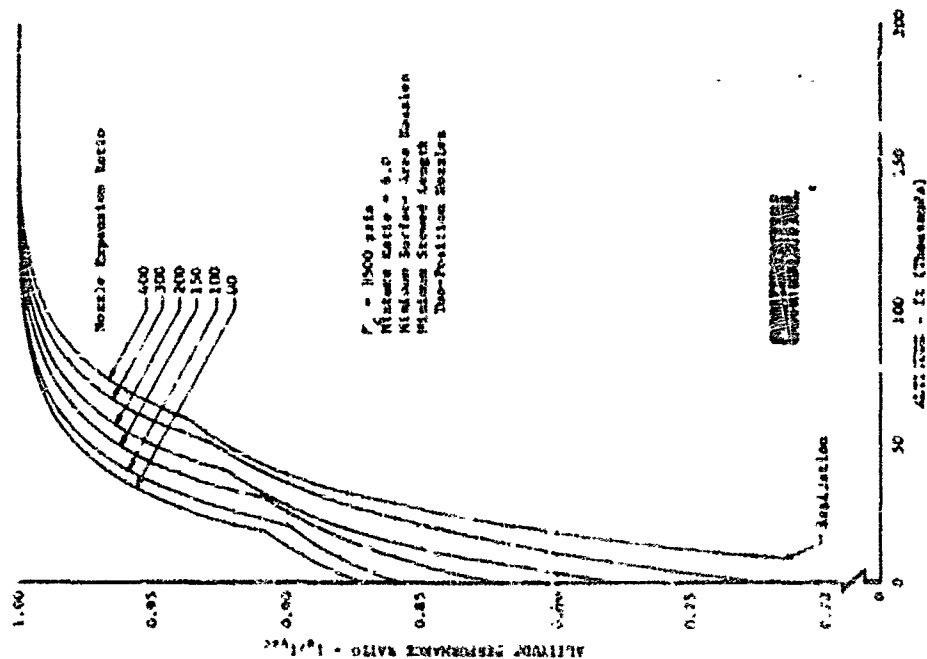


Figure 701. Altitude Performance With DF 56187
Minimum Surface Area Contour
Nozzle, 100,000-lb Thrust,
 $r = 6.0$, $P_c = 3500$ psia

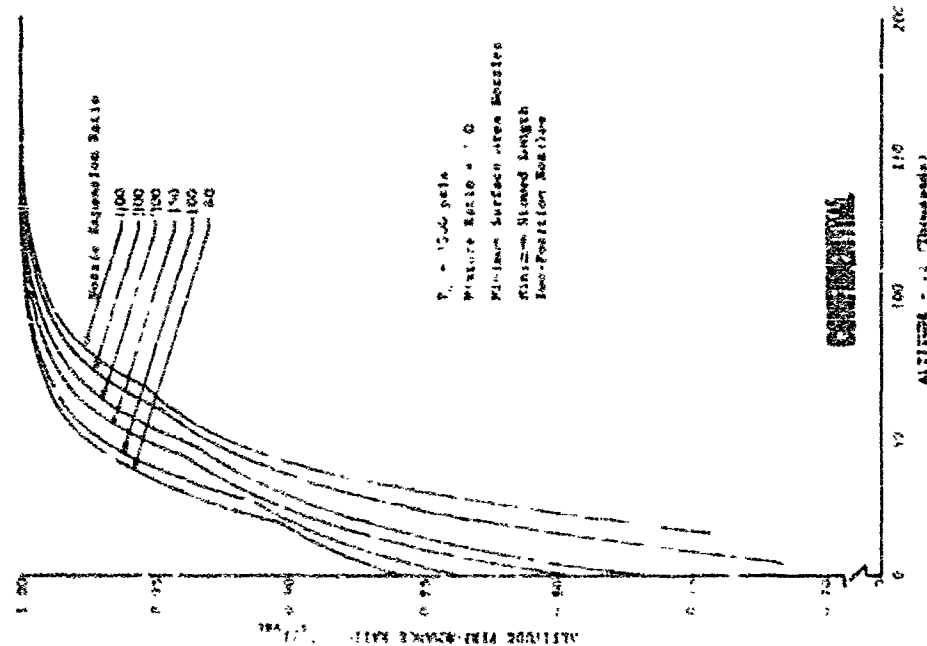


Figure 700. Altitude Performance With DF 56406
Minimum Surface Area Contour
Nozzle, 350,000-lb Thrust,
 $r = 7.0$, $P_c = 3000$ psia

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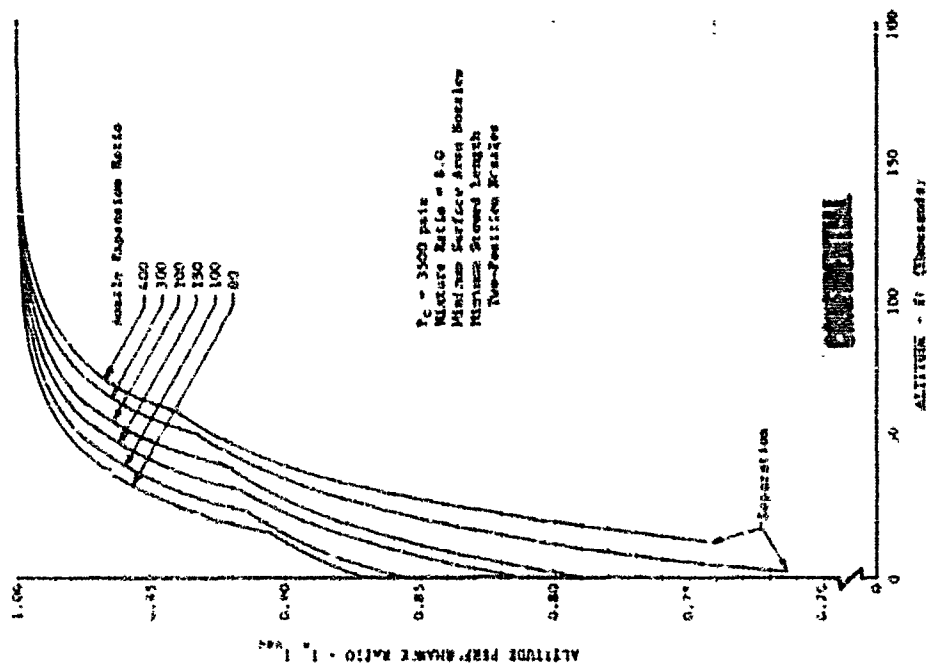


Figure 702 Altitude Performance With Minimum Surface Area Contour
Nozzle, 250,000-lb Thrust,
 $r = 6.0$, $P_c = 3500$ psia

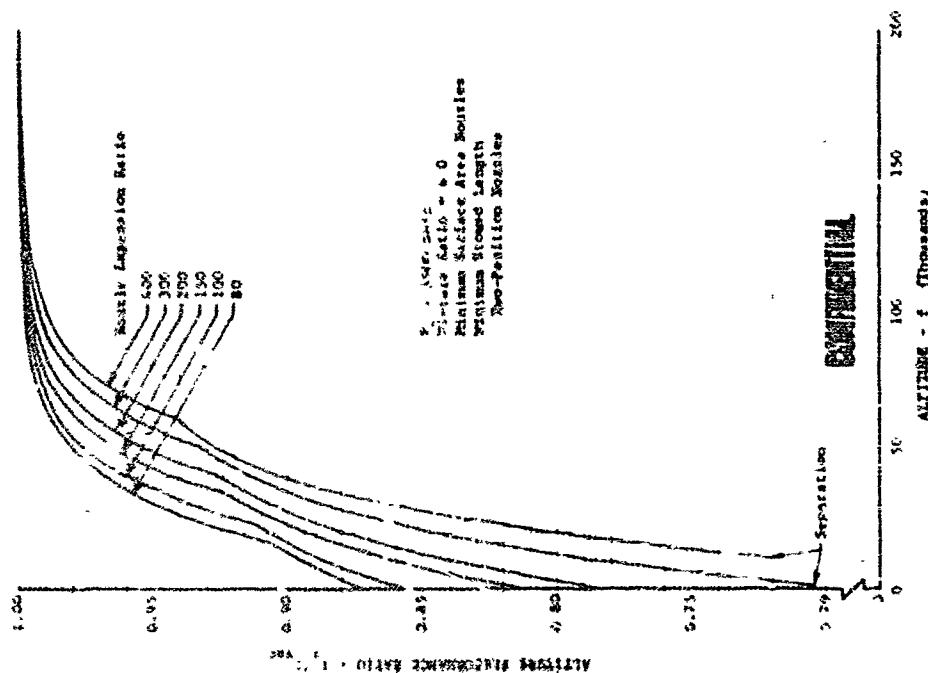


Figure 703 Altitude Performance With Minimum Surface Area Contour
Nozzle, 350,000-lb Thrust,
 $r = 6.0$, $P_c = 3500$ psia

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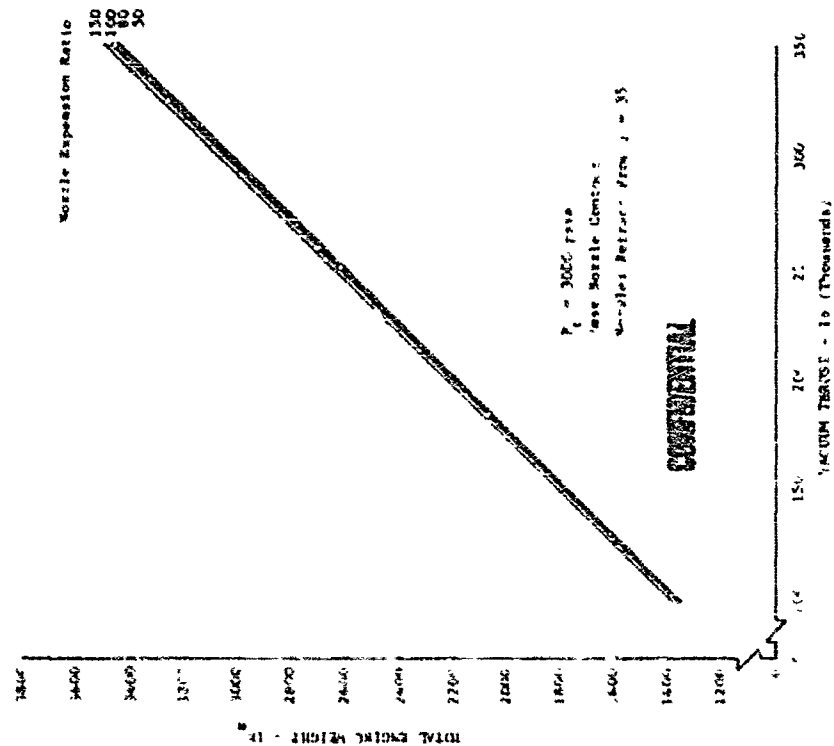
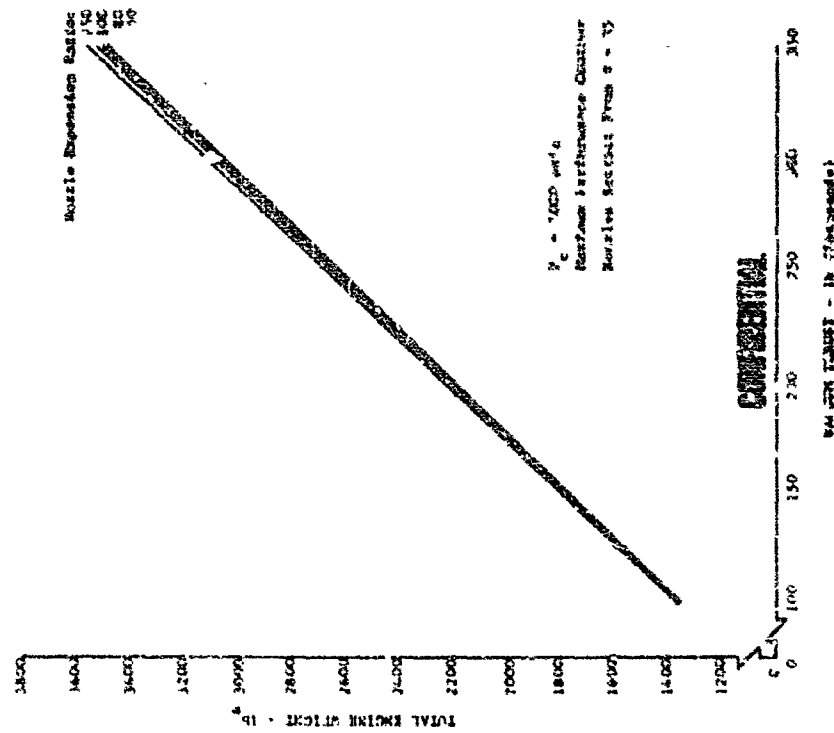


Figure 704. Total Engine Weight vs Vacuum Thrust for Engine With Base Contour Two-Position Nozzle ($\epsilon_p = 35$) DF 56357

Figure 705. Total Engine Weight vs Vacuum Thrust for Engine With Maximum Performance Contour Two-Position Nozzle ($\epsilon_p = 35$) DF 56356

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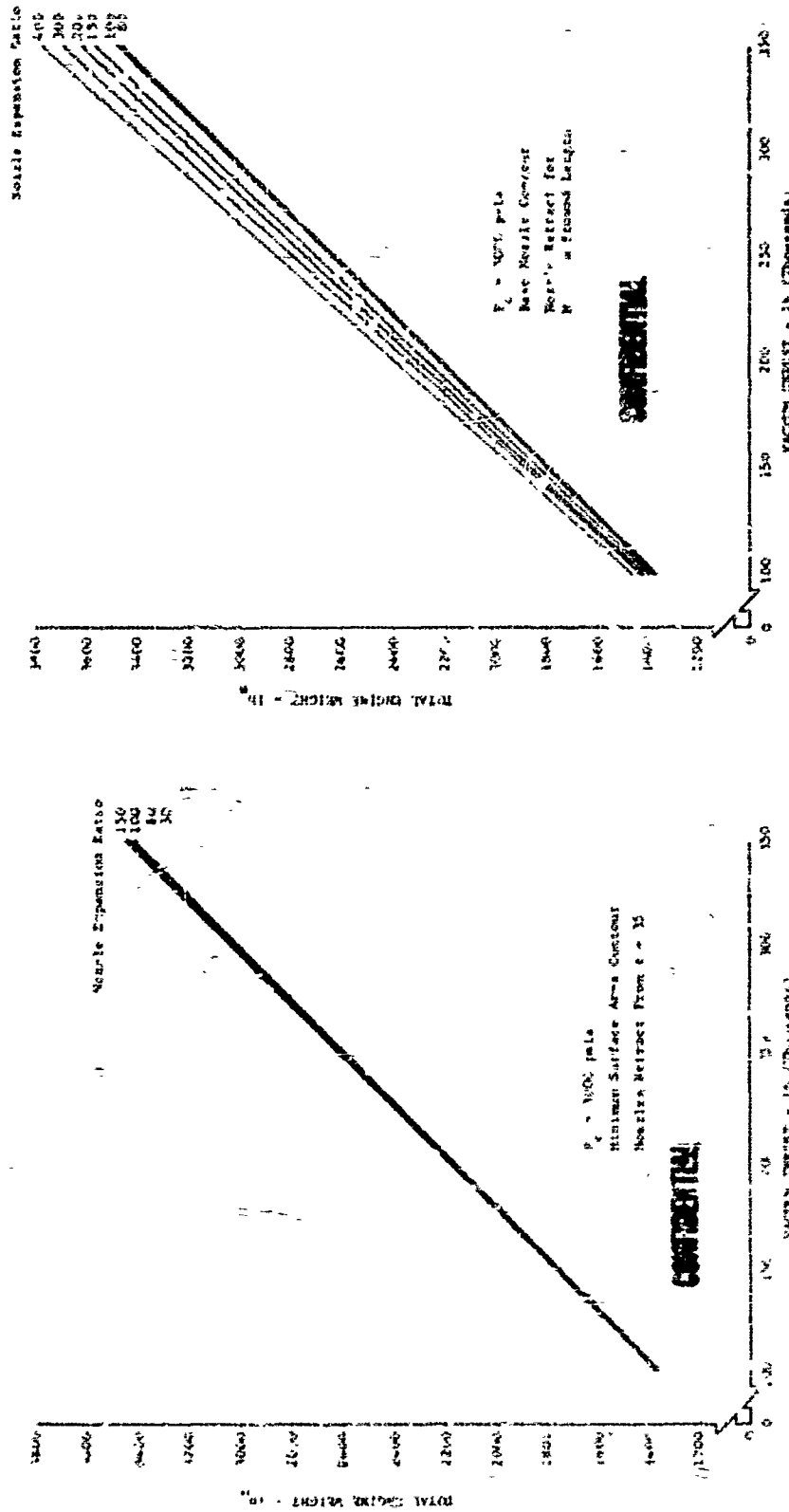


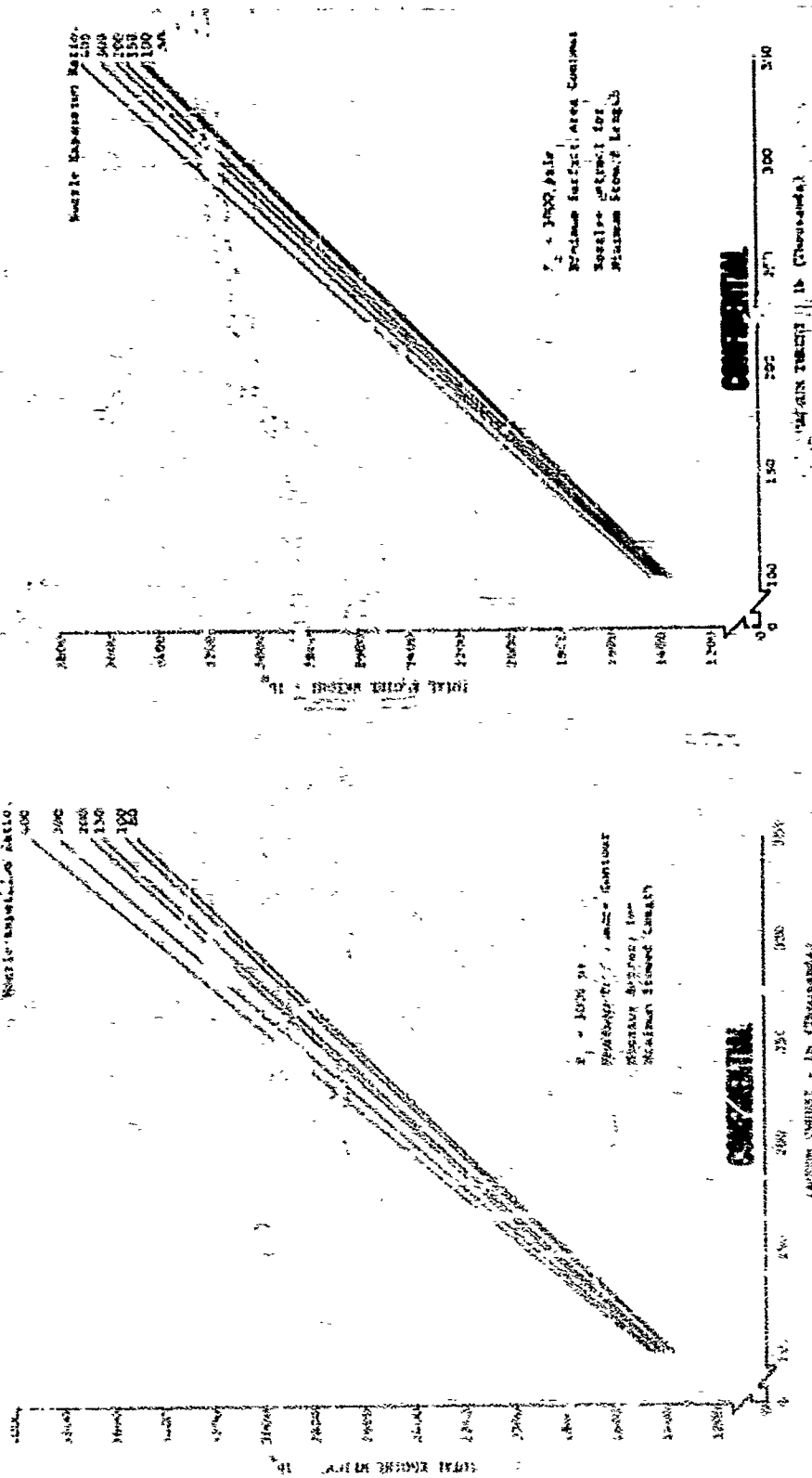
Figure 700. Total Engine Weight vs Vacuum Thrust for Engine With Minimum Surface Area Contour Two-Position Nozzle ($e_f = 35$)

Figure 701. Total Engine Weight vs Vacuum Thrust for Engine With Base Contour Two-Position Nozzle ($e_p = \text{Minimum}$)

DF 56358

DF 56354

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Figure 708. Total Engine Weight vs Vacuum Thrust for Engine With Maximum Performance Contour Two-Position Nozzle ($\epsilon_p = \text{Minimum}$) DF 56353

Figure 709. Total Engine Weight vs Vacuum Thrust for Engine With Minimum Surface Area Contour Two-Position Nozzle ($\epsilon_p = \text{Minimum}$) DF 56355

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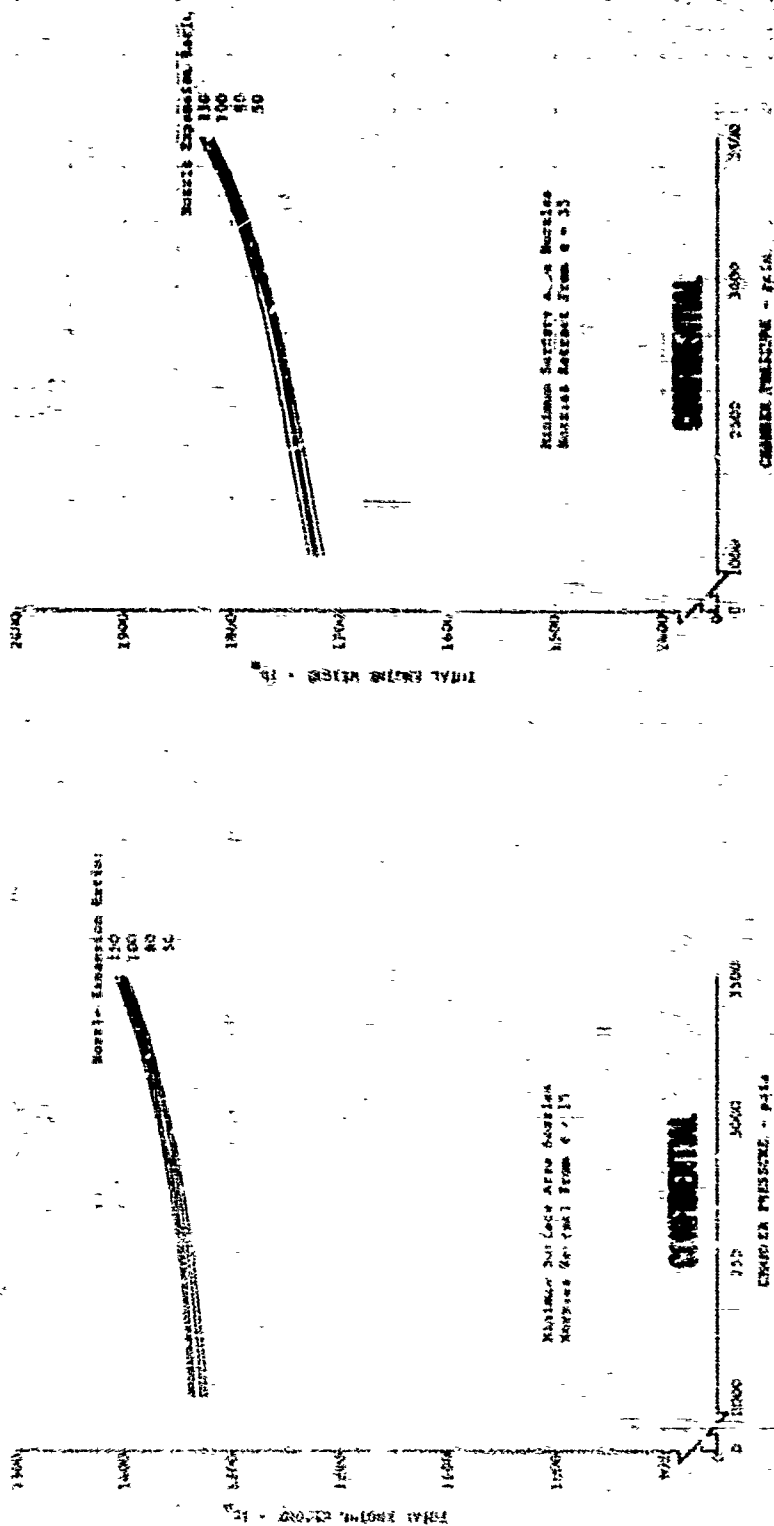
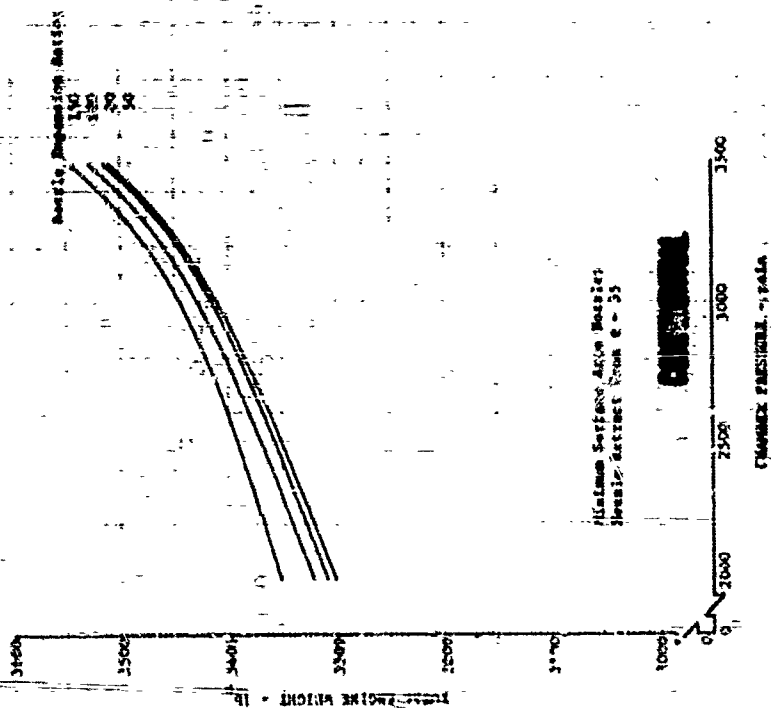


Figure 710. Total Engine Weight vs Chamber Pressure for 100,000-lb Thrust Engine ($\epsilon_p = 35$)

Figure 711. Total Engine Weight vs Chamber Pressure for 150,000-lb Thrust Engine ($\epsilon_p = 35$)

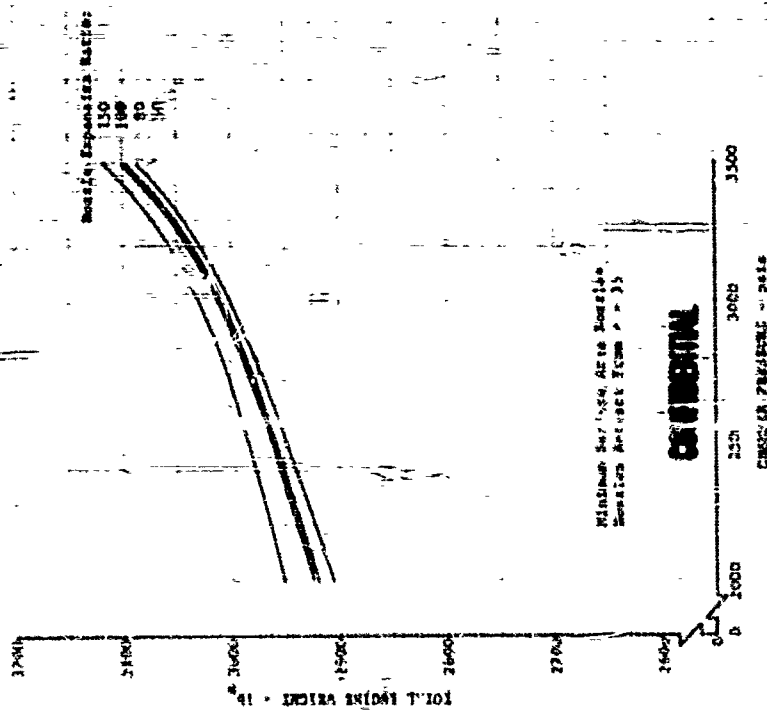
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DF 56352

Figure 715. Total Engine Weight vs Chamber Pressure for 350,000-lb Thrust Engine ($\epsilon_p = 35$)



DF 56351

Figure 714. Total Engine Weight vs Chamber Pressure for 300,000-lb Thrust Engine ($\epsilon_p = 35$)

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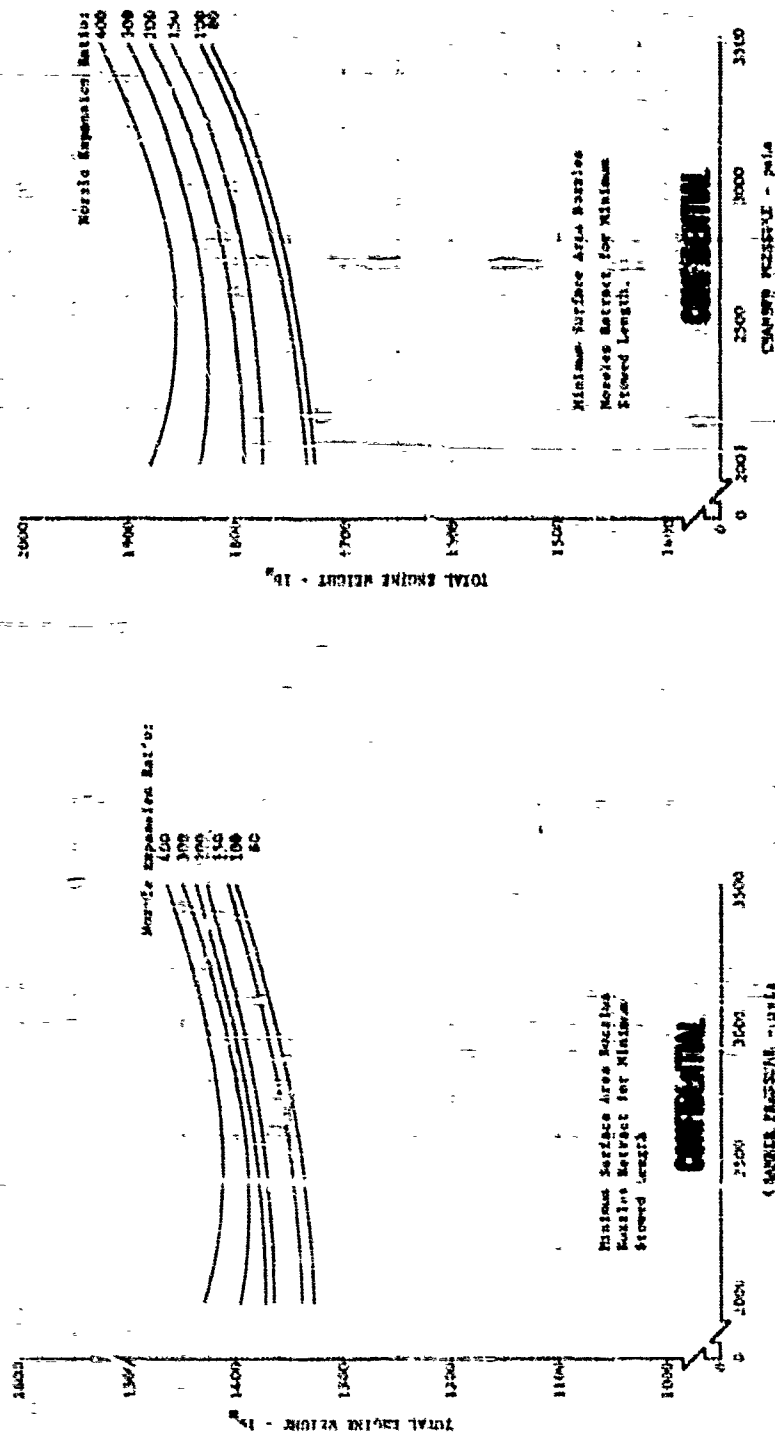


Figure 716. Total Engine Weight vs Chamber Pressure for 100,000-lb Thrust Engine (ϵ_p = Minimum)

DF 56342

DF 56359

Figure 717. Total Engine Weight vs Chamber Pressure for 150,000-lb Thrust Engine (ϵ_p = Minimum)

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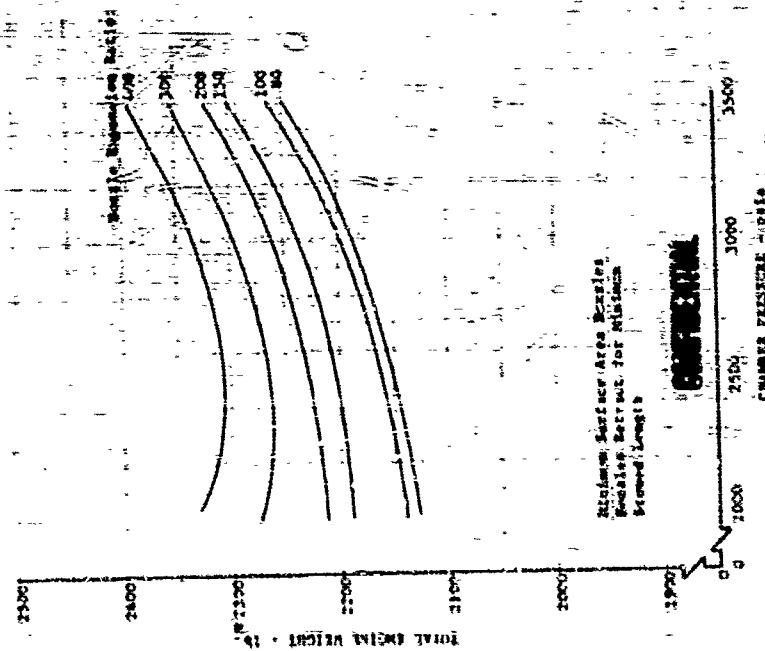
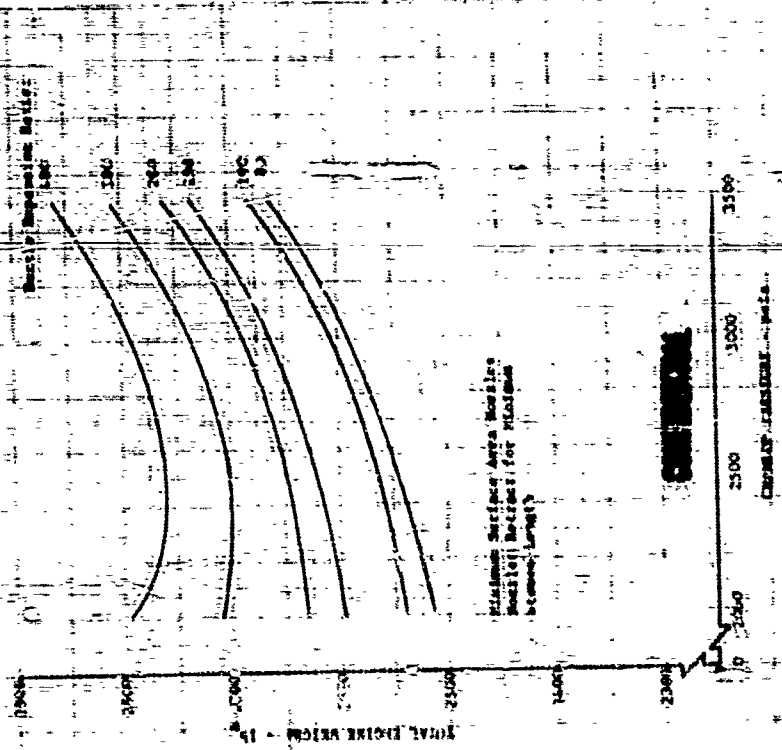


Figure 718. Total Engine Weight vs Chamber Pressure for 200,000-lb Thrust Engine (ϵ_p = Minimum)

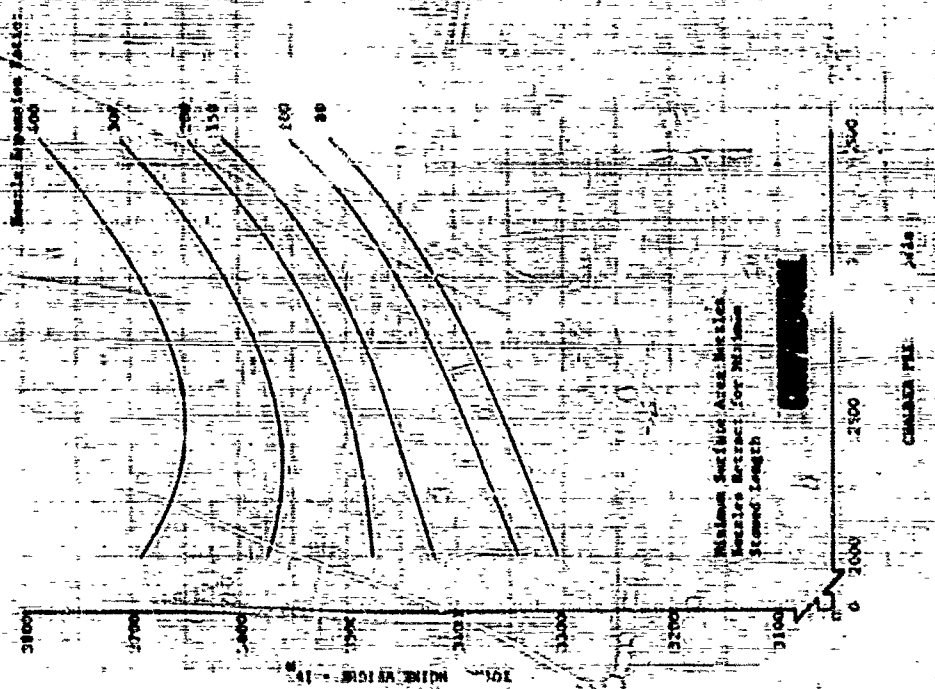
DF 56343



DF 56344

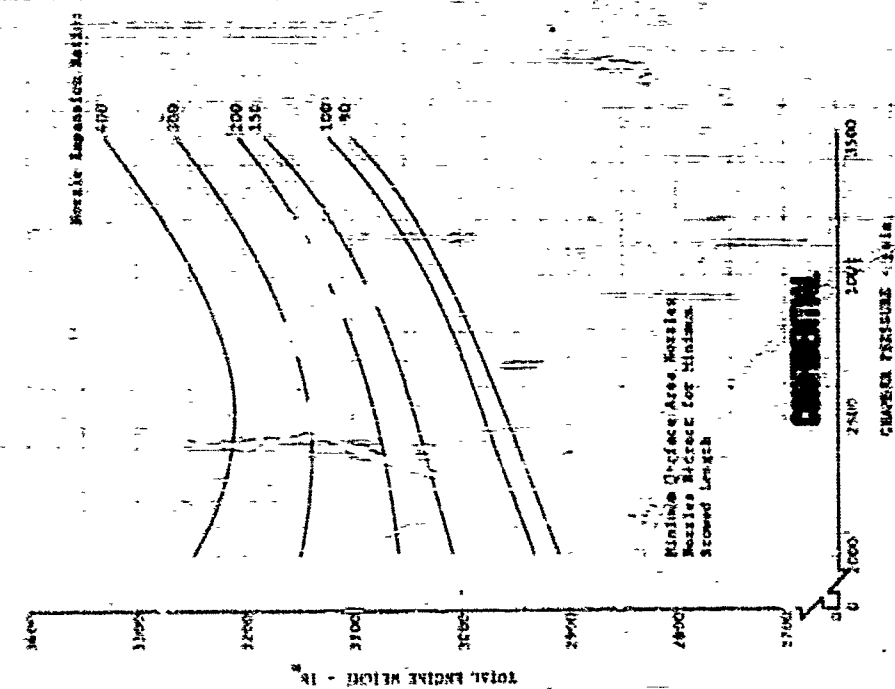
Figure 719. Total Engine Weight vs Chamber Pressure for 250,000-lb Thrust Engine (ϵ_p = Minimum)

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DF 56346

Figure 721. Total Engine Weight vs Chamber Pressure for 350,000-lb Thrust Engine (e_p = Minimum)



DF 56345

Figure 720. Total Engine Weight vs Chamber Pressure for 300,000-lb Thrust Engine (e_p = Minimum)

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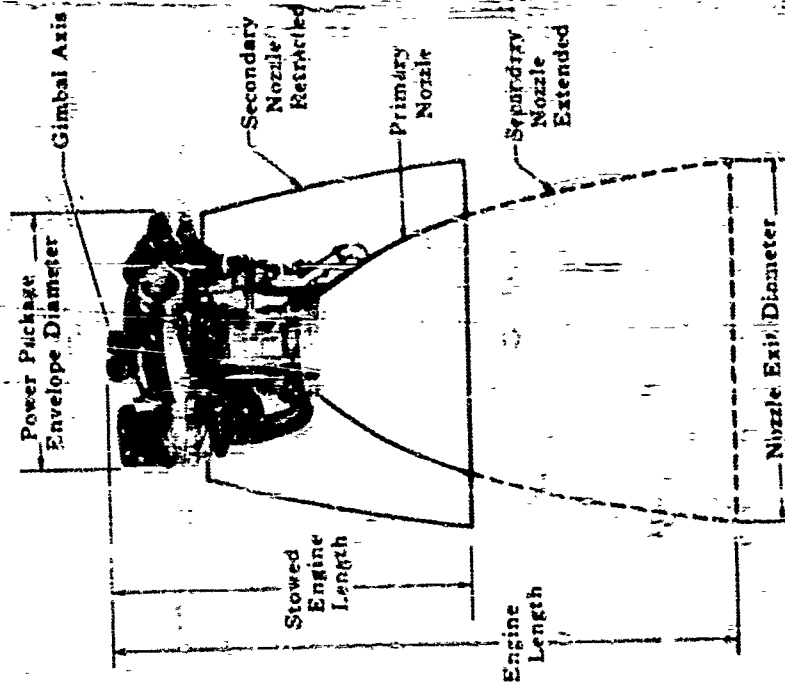


Figure 722. Engine Configuration With Two-Position Nozzle

FD 21152

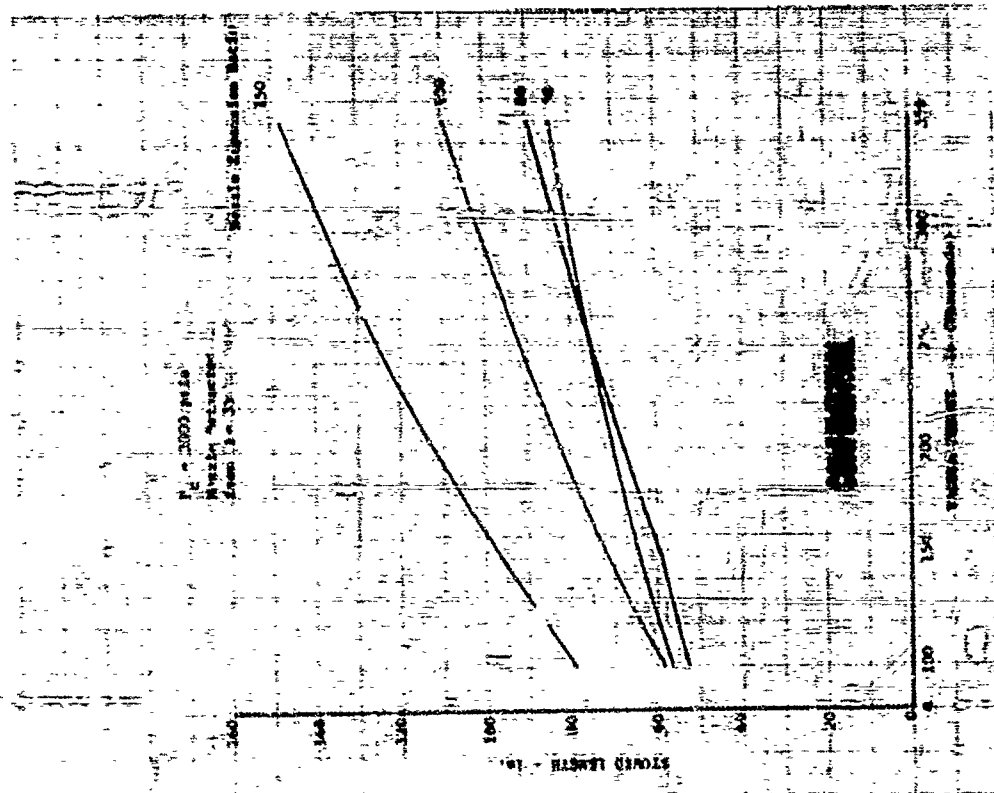


Figure 723. Stowed Length vs Vacuum Thrust for Engine With Base Contour Two-Position Nozzle ($\epsilon = 35$)

DF 56239

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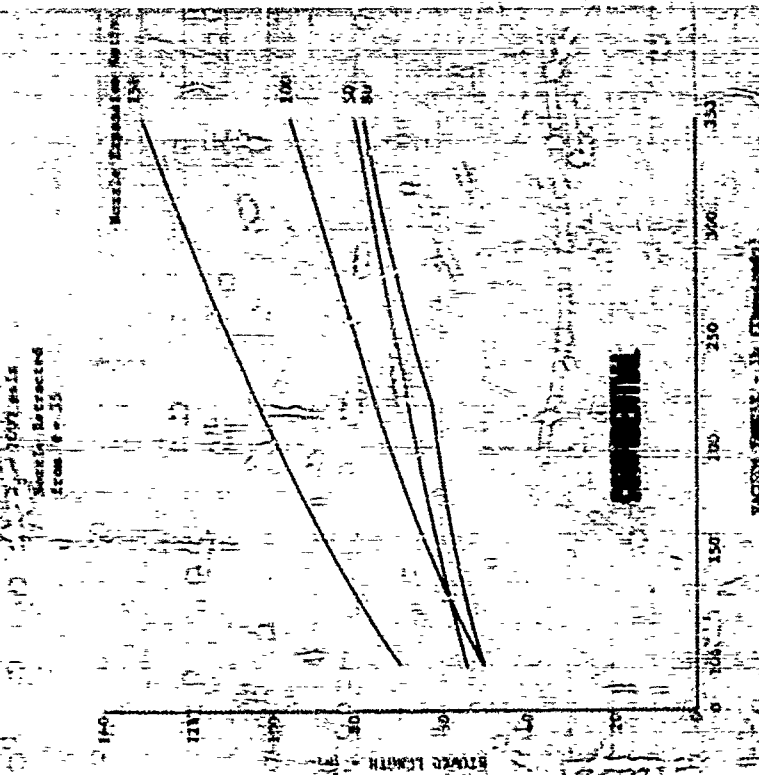


Figure 725. Stowed Length vs V_2 in Thrust for Engine With Minimum Surface Area Contour Two-Position Nozzle ($\epsilon_p = 35$) DF 56237

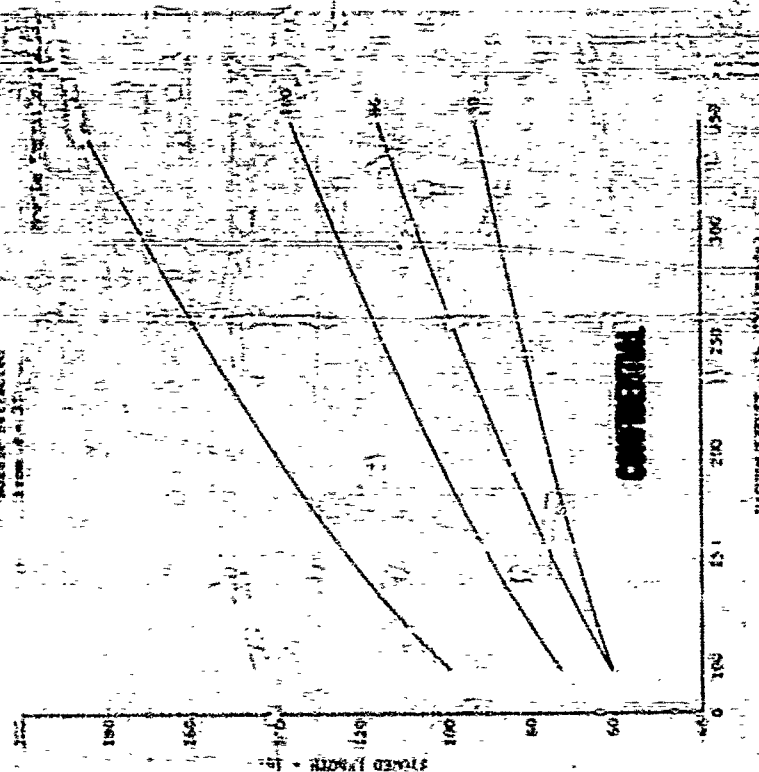


Figure 724. Stowed Length vs Vacuum Thrust for Engine With Maximum Performance Contour Two-Position Nozzle ($\epsilon_p = 35$) DF 56238

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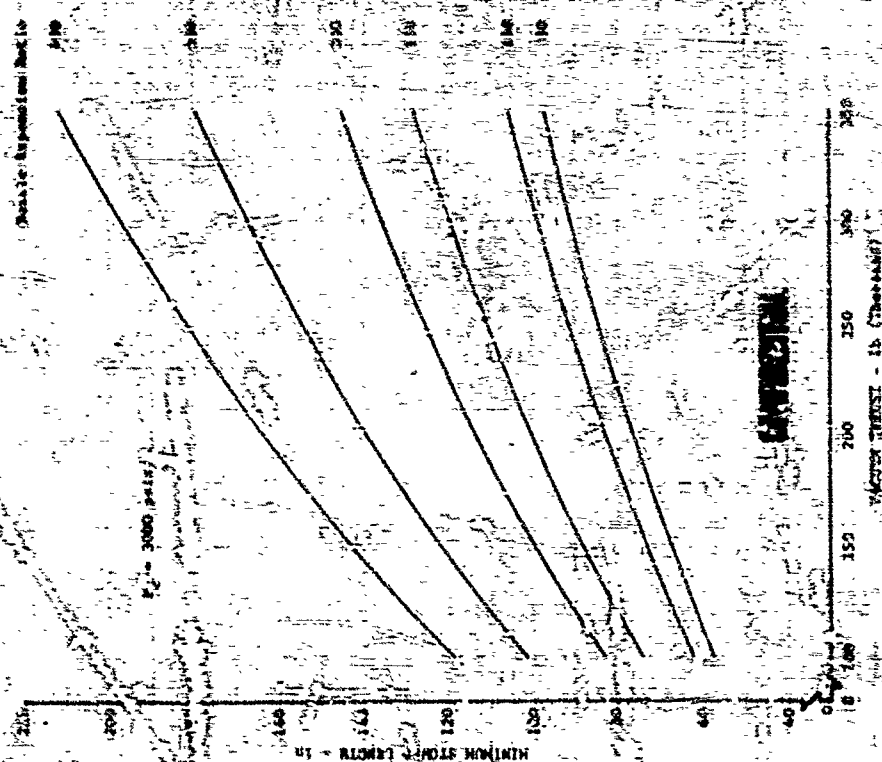


Figure 727. Minimum Stowed Length vs Vacuum Thrust for Engine With Maximum Performance Four Two-Position Nozzle

DF 56259

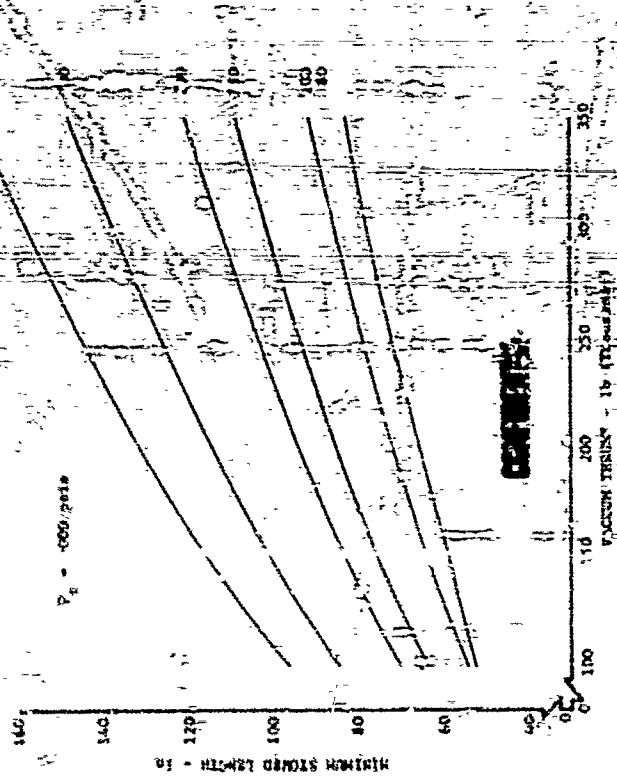
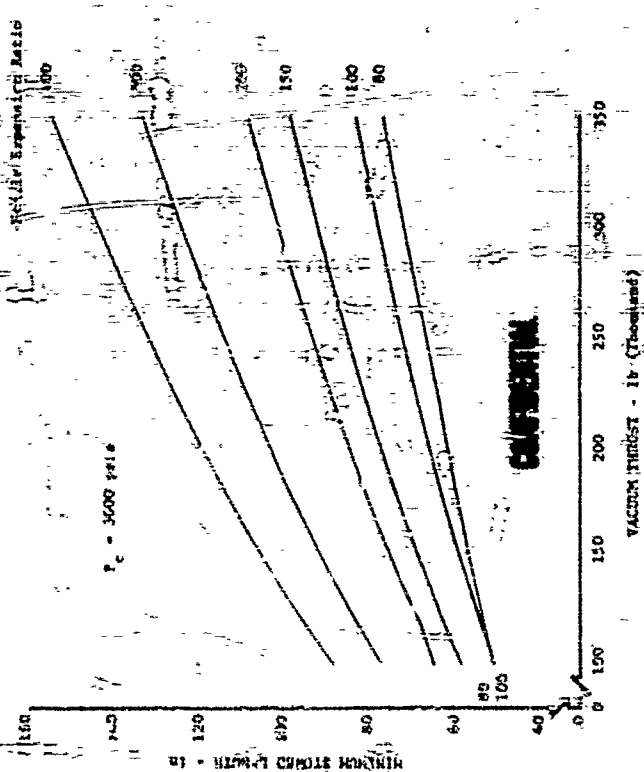
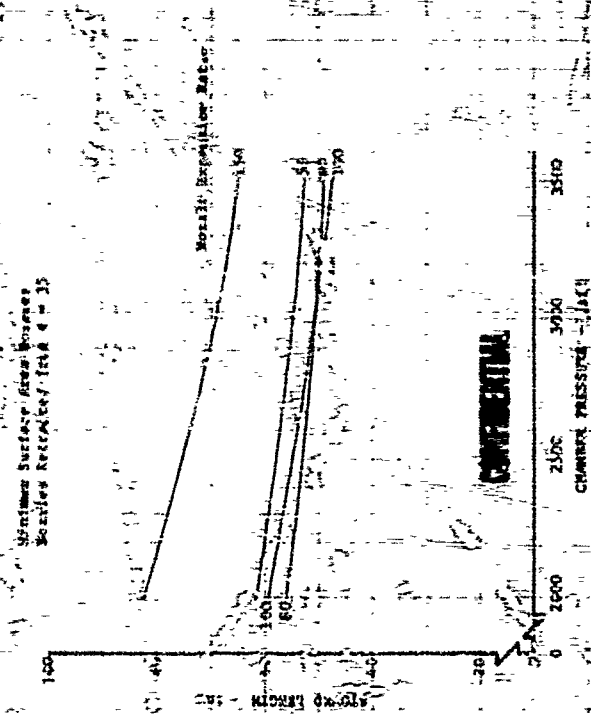


Figure 726. Minimum Stowed Length vs Vacuum Thrust for Engine With Base Contour Two-Position Nozzle

DF 56210

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DF 56230

Figure 729. Minimum Stowed Length vs. Chamber Pressure for 100,000-lb Thrust Engine ($\epsilon_p = 35$)

DF 56258

Figure 728. Minimum Stowed Length vs. Vacuum Thrust for Engine With Minimum Surface Area Contour Two-Position Nozzle

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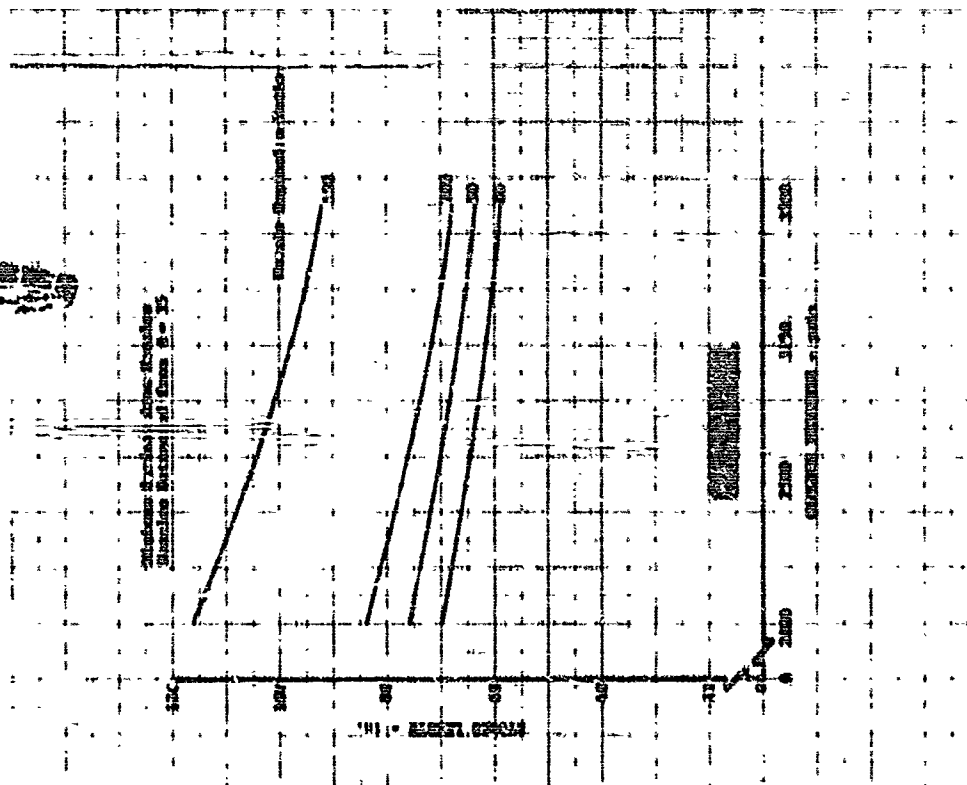


Figure 731. Stowed Length vs Chamber Pressure for 200,000-lb Thrust Engine ($\epsilon_p = 35$)

DF 56229

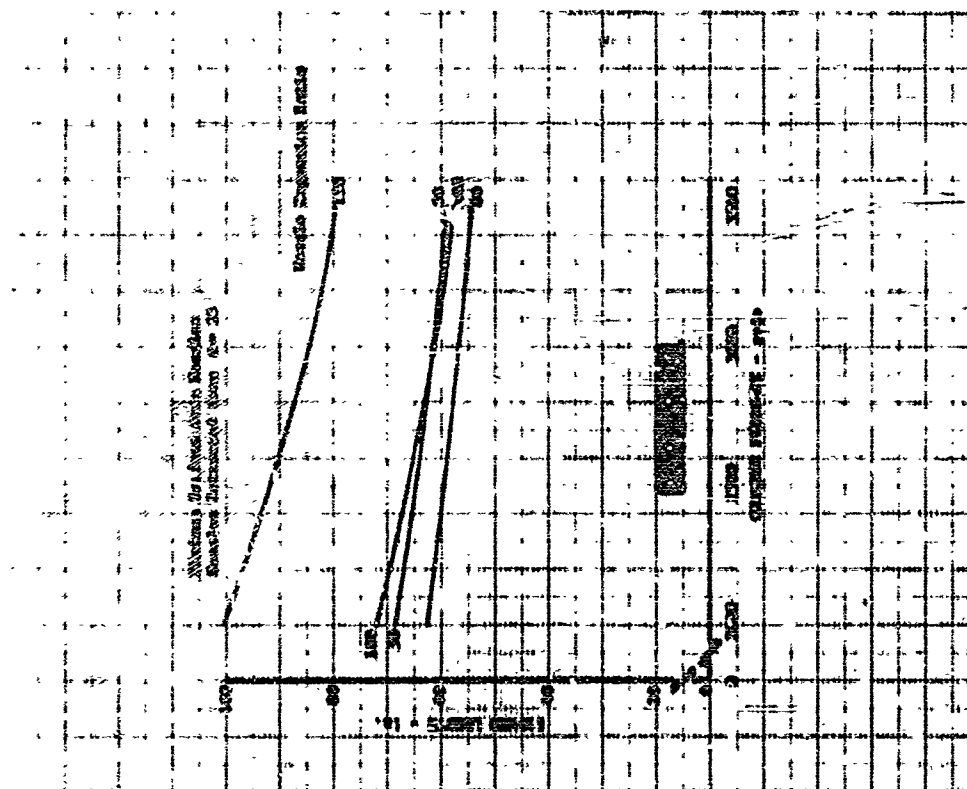


Figure 730. Stowed Length vs Chamber Pressure for 150,000-lb Thrust Engine ($\epsilon_p = 35$)

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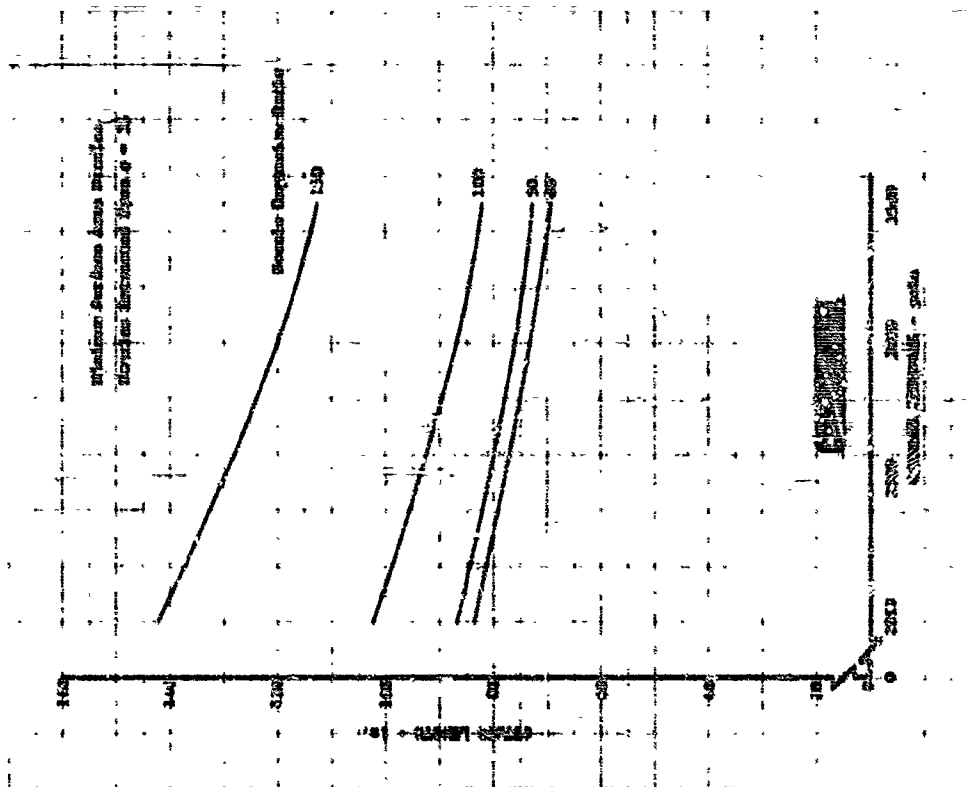


Figure 733. Stowed Length vs Chamber Pressure for 300,000-lb Thrust Engine ($\epsilon_p = 35$) DF 56216

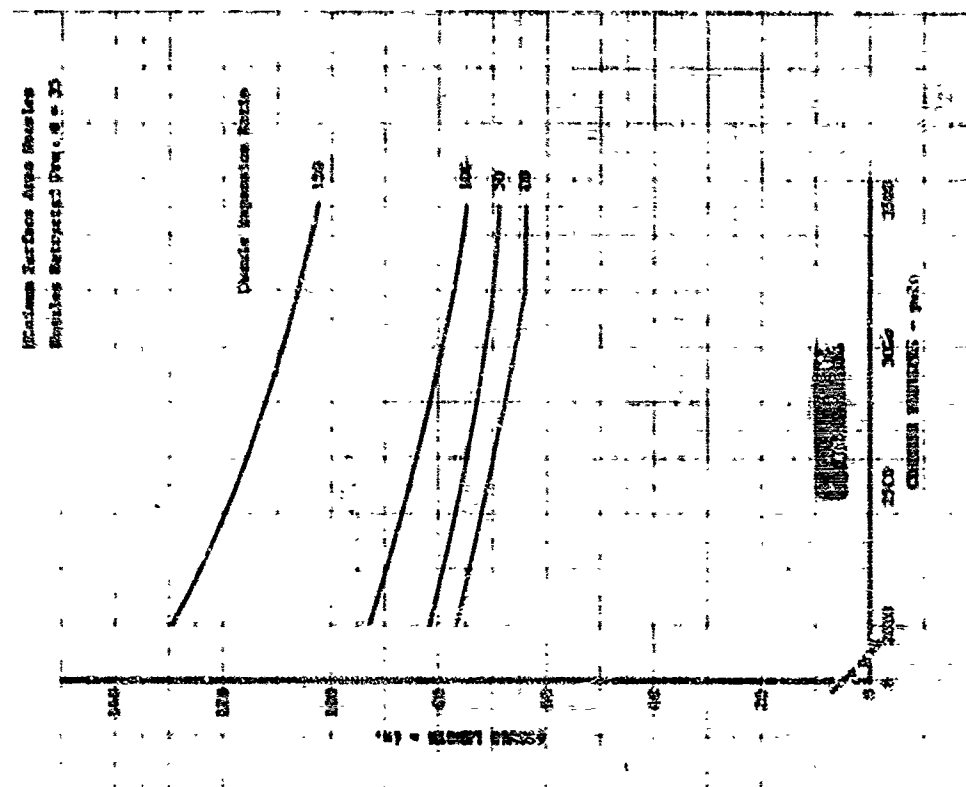


Figure 732. Stowed Length vs Chamber Pressure for 250,000-lb Thrust Engine ($\epsilon_p = 35$) DF 56227

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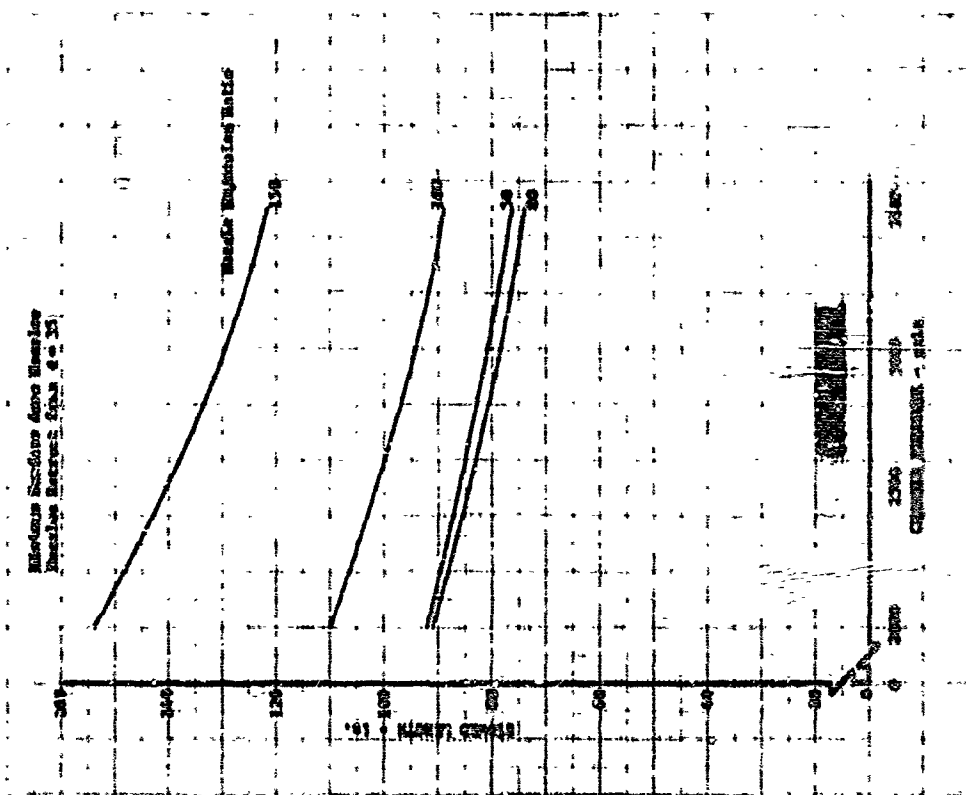


Figure 734. Stowed Length vs Chamber Pressure for 350,000-lb Thrust Engine ($\epsilon_p = 25$)

DF 56225

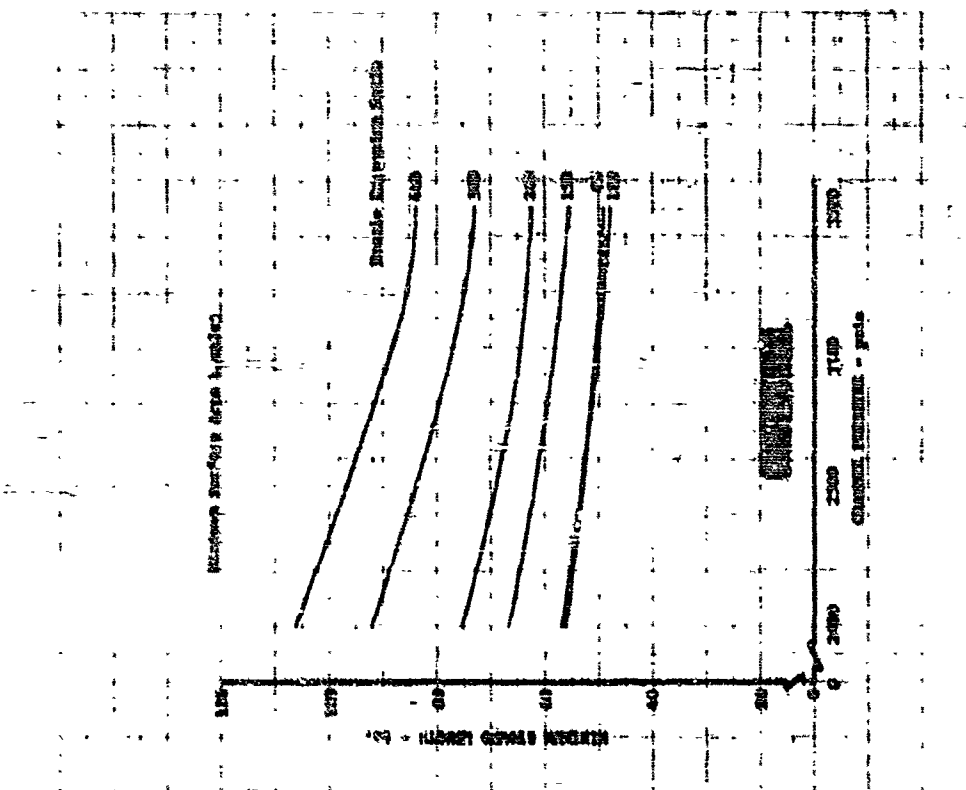
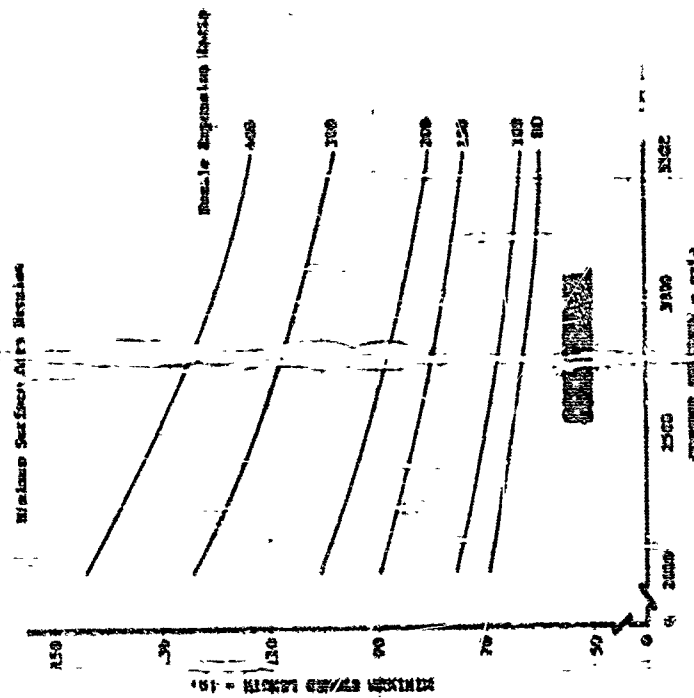


Figure 735. Minimum Stowed Length vs Chamber Pressure for 100,000-lb Thrust Engine

DF 56236

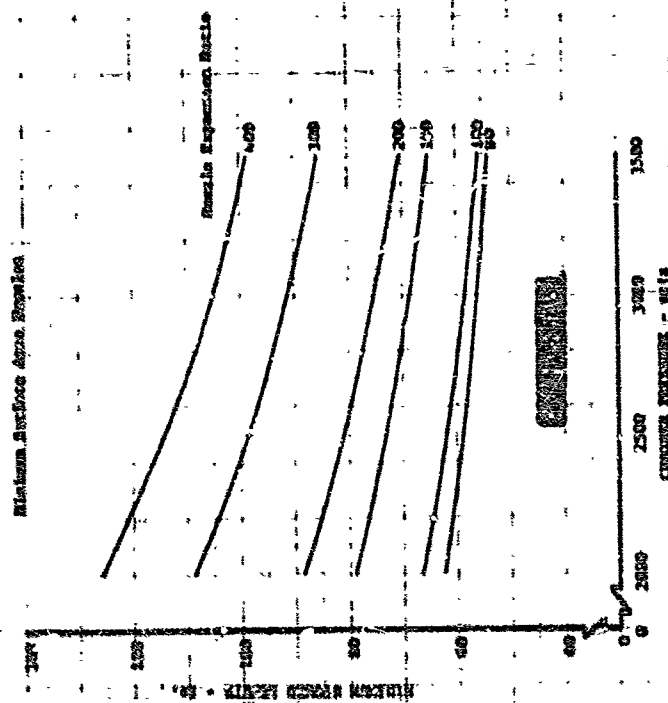
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DP 56234

Figure 737. Minimum Stowed Length vs Chamber Pressure for 200,000-lb Thrust Engine



DP 56235

Figure 736. Minimum Stowed Length vs Chamber Pressure for 150,000-lb Thrust Engine

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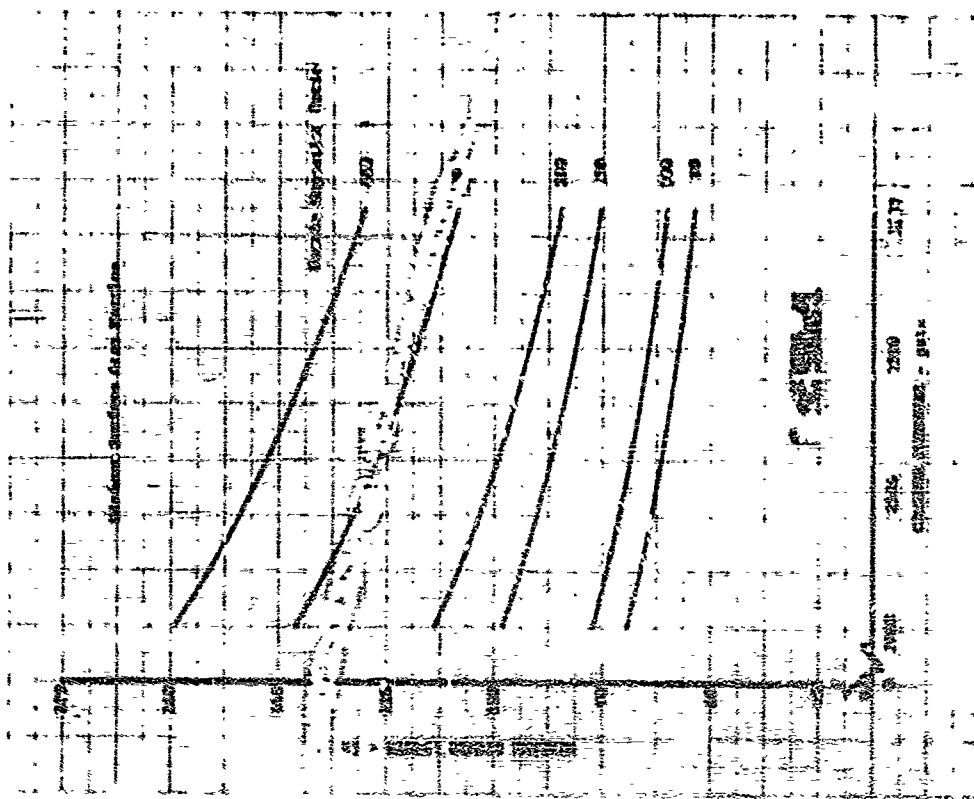


Figure 738. H1 Stowed Length vs Chamber Pressure for 250,000-lb Thrust Engine

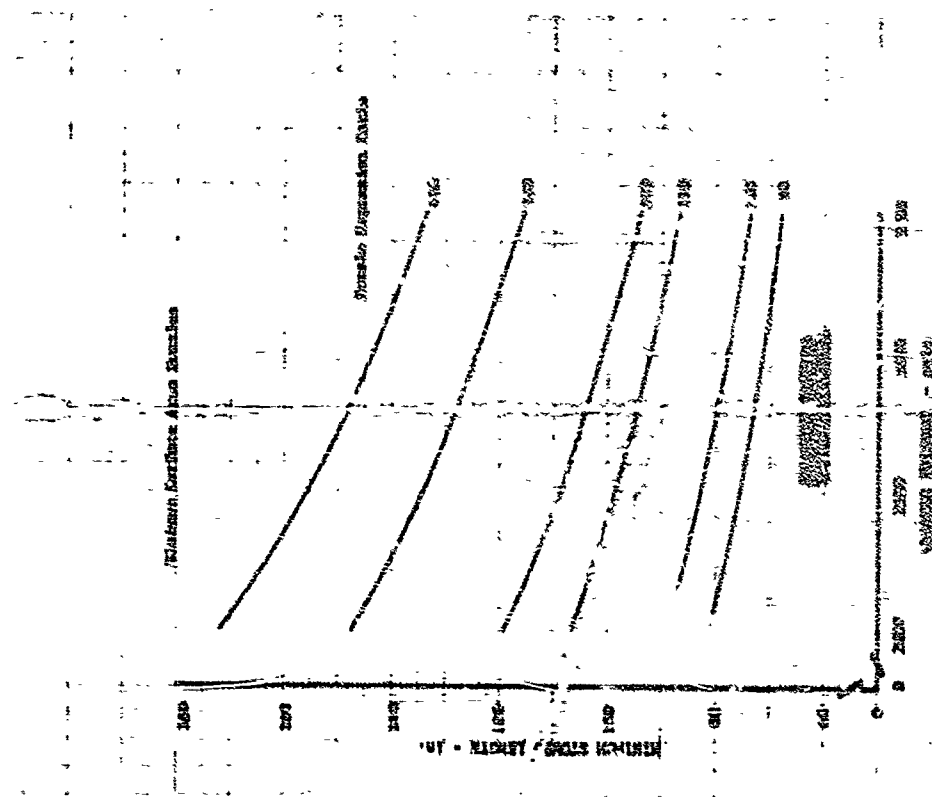


Figure 739. Minimum Stowed Length vs Chamber Pressure for 300,000-lb Thrust Engine

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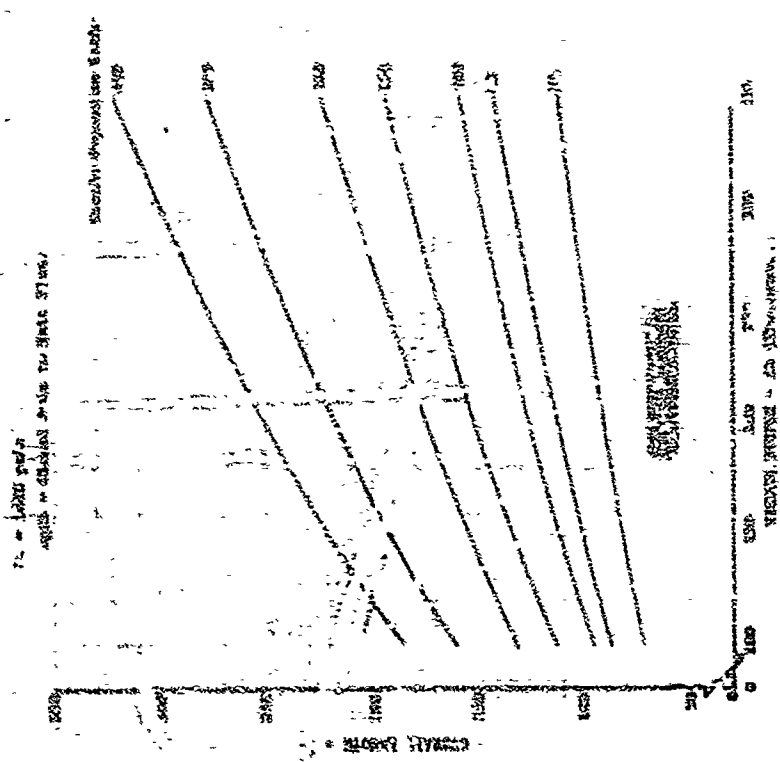


Figure 741. Overall Length vs Vacuum Thrust with Base Dynamic Contour

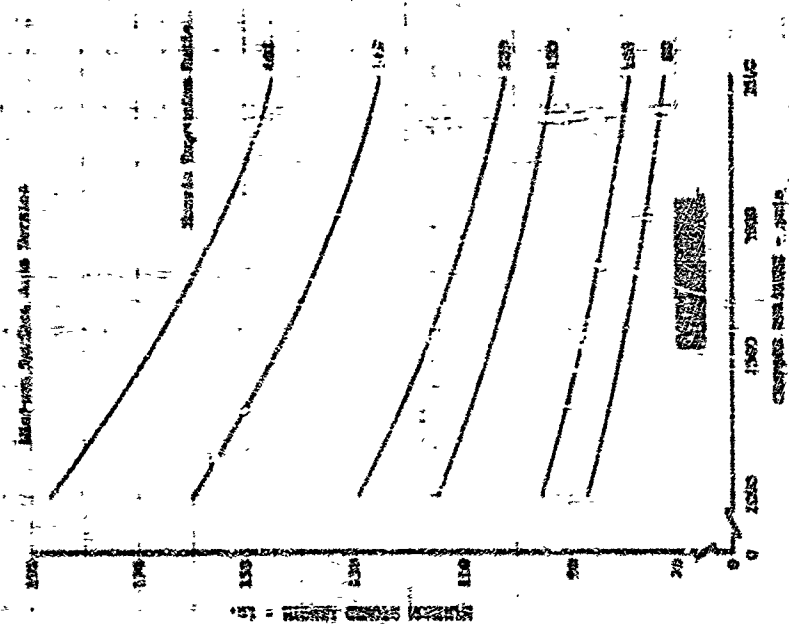


Figure 740. Minimum Stowed Length vs Chamber Pressure for 350,000-lb Thrust Engine

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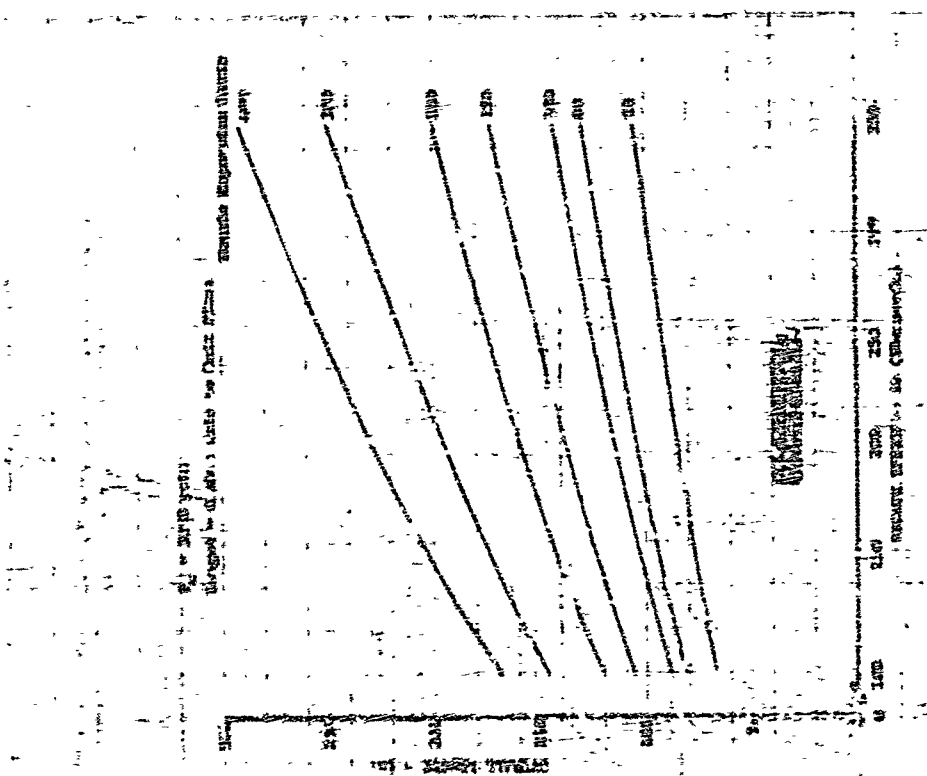


Figure 743. Overall Length vs Vacuum Thrust with Minimum Surface Area Nozzle Contour

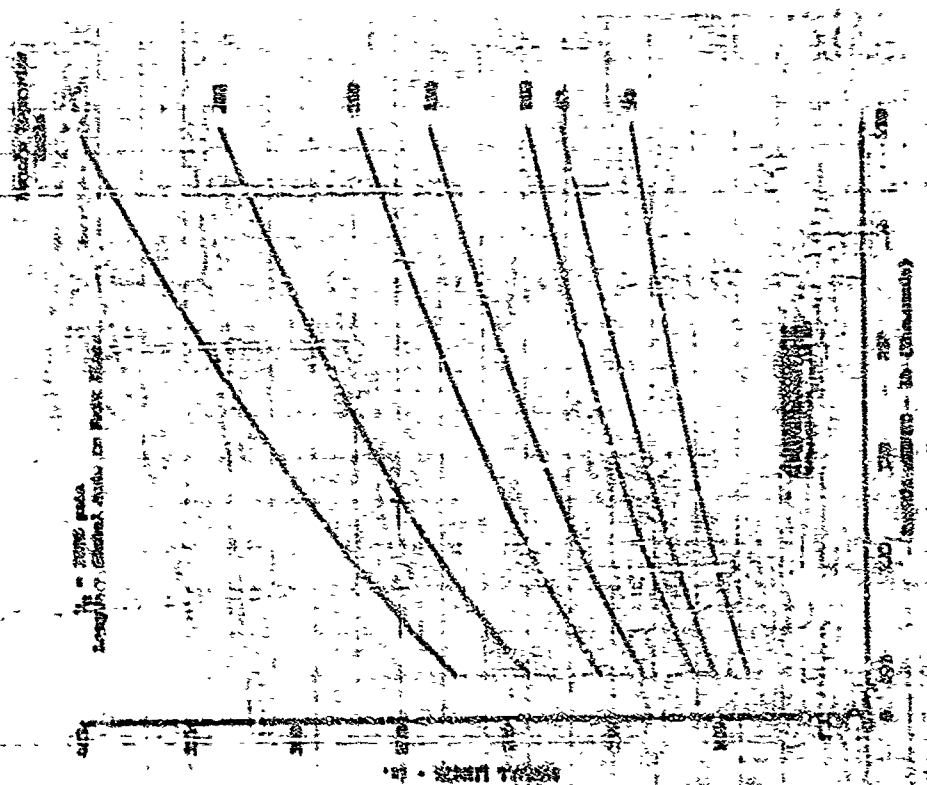


Figure 742. Overall Length vs Vacuum Thrust with Minimum Performance Nozzle Contour

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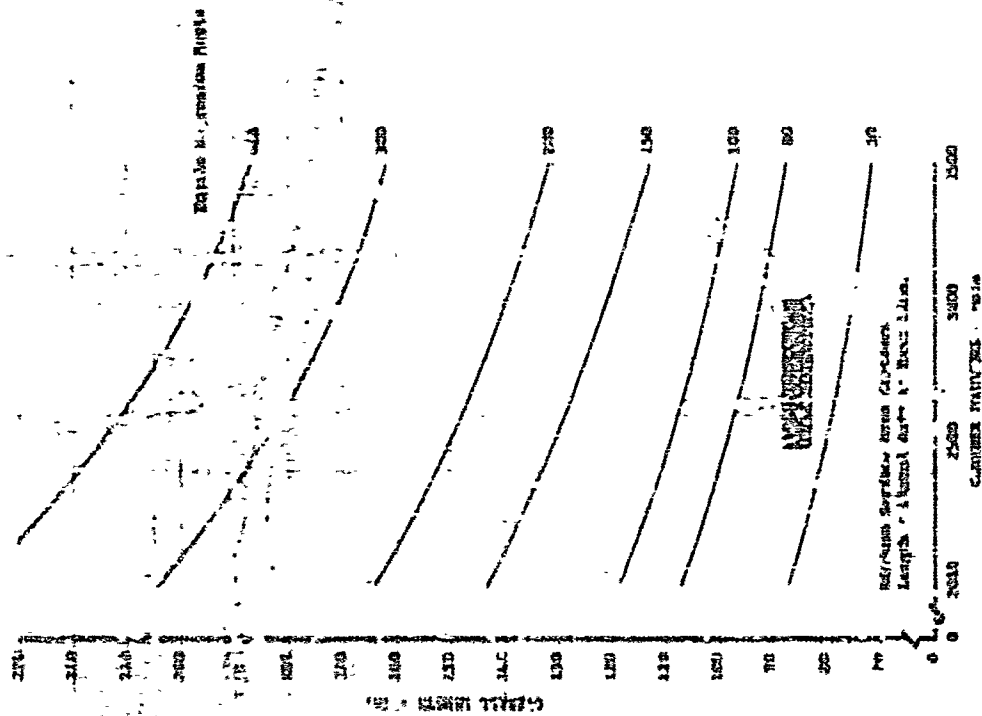


Figure 745. Overall Length vs Chamber Pressure for 150,000 lb Thrust Engine

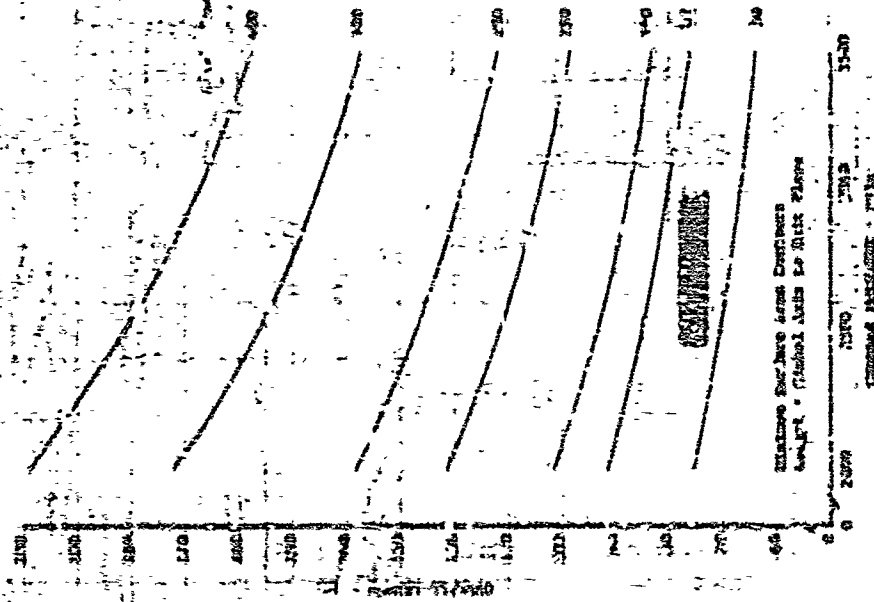


Figure 746. Overall Length vs Chamber Pressure for 150,000 lb Thrust Engine

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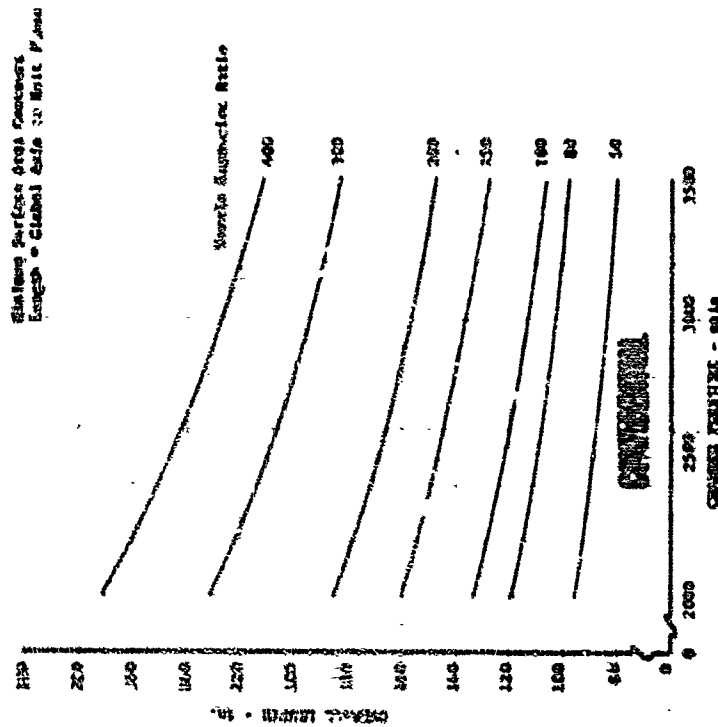
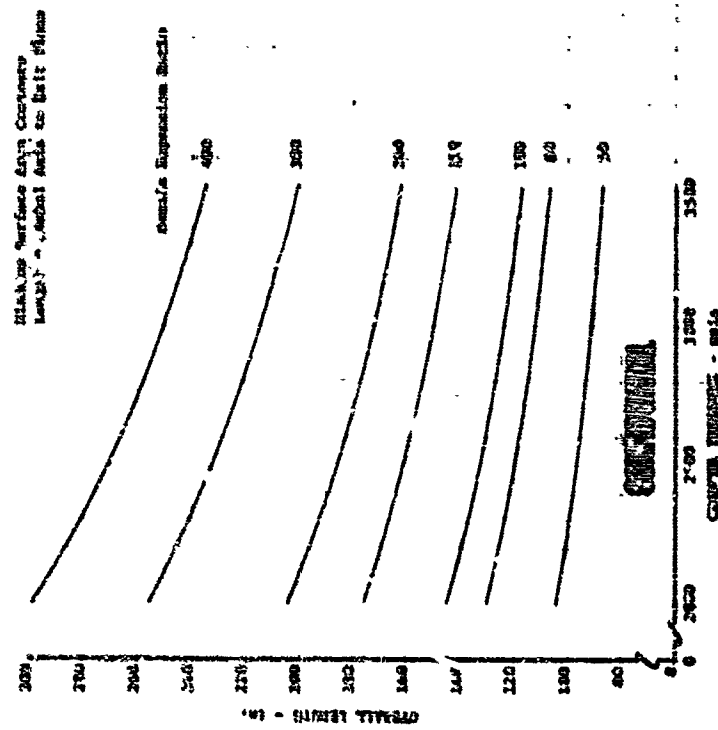


Figure 746. Overall Length vs Chamber Pressure for 200,000-lb Thrust Engine

DR 55307

Figure 747. Overall Length vs Chamber Pressure for 250,000-lb Thrust Engine DF 55800

DP 559.00

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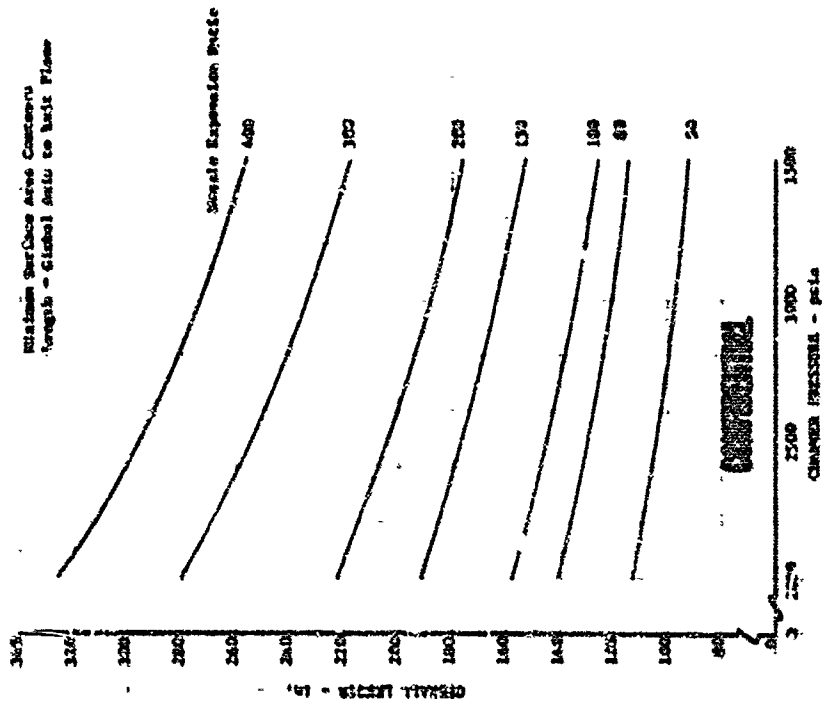


Figure 748. Overall Length vs Chamber Pressure for 300,000-lb Thrust Engine

DP 55799

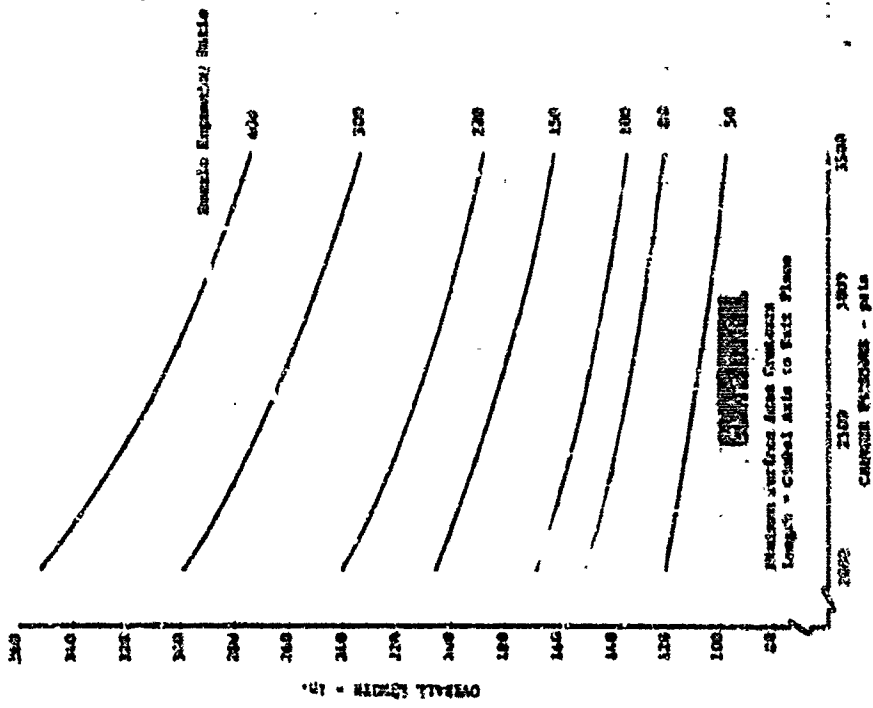
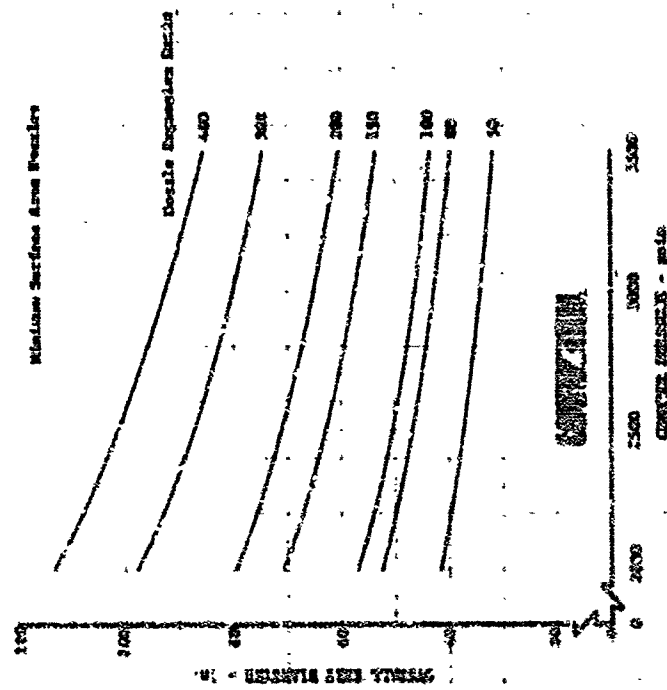


Figure 749. Overall Length vs Chamber Pressure for 350,000-lb Thrust Engine

DP 55798

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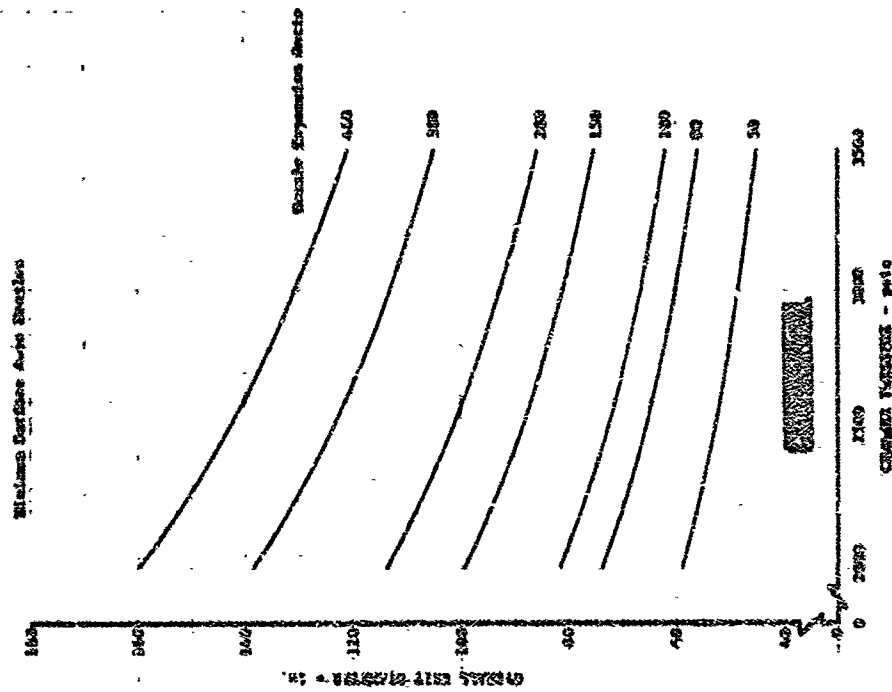


Figure 753. Overall Exit Diameter vs Chamber Pressure for 200,000-lb Thrust Engine

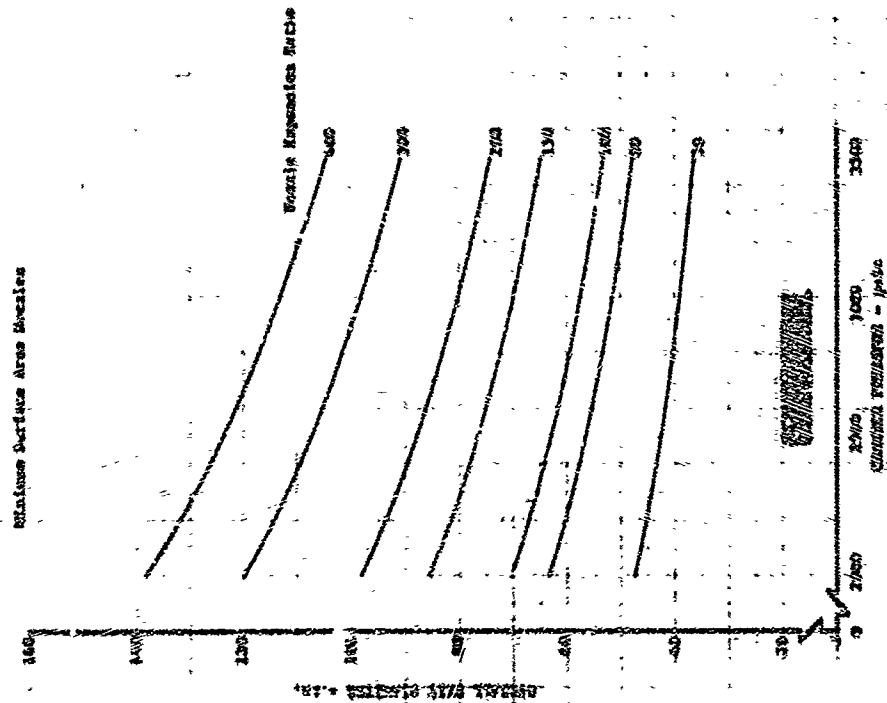


Figure 752. Overall Exit Diameter vs Chamber Pressure for 150,000-lb Thrust Engine

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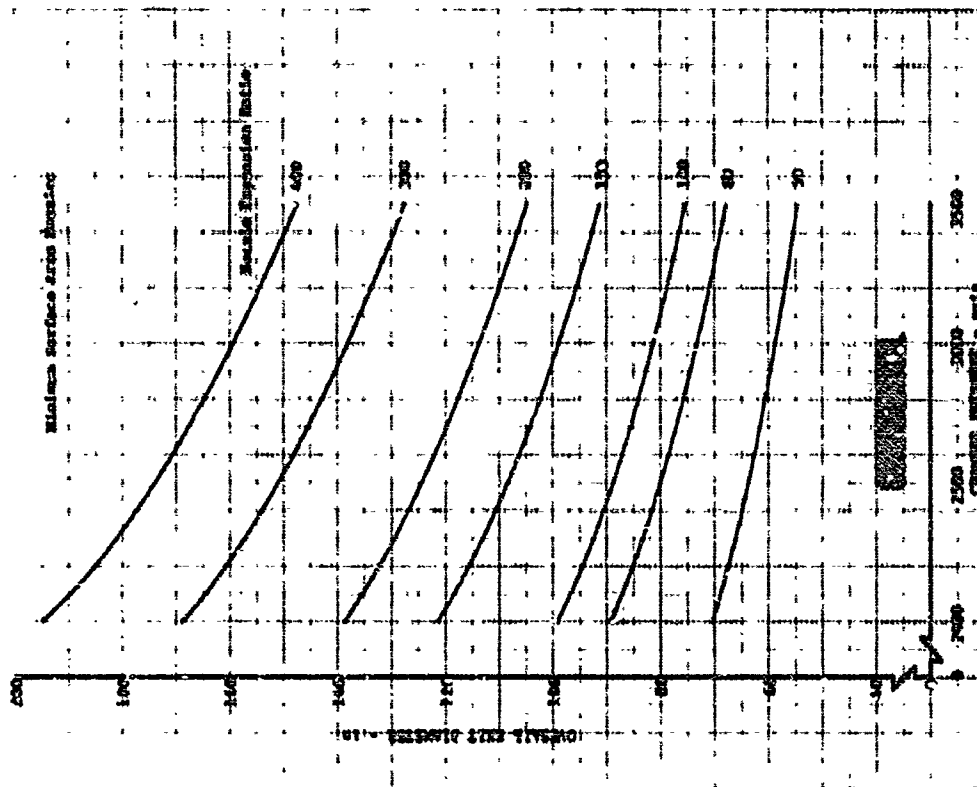


Figure 755. Overall Exit Diameter vs Chamber Pressure for 300,000-lb Thrust Engine

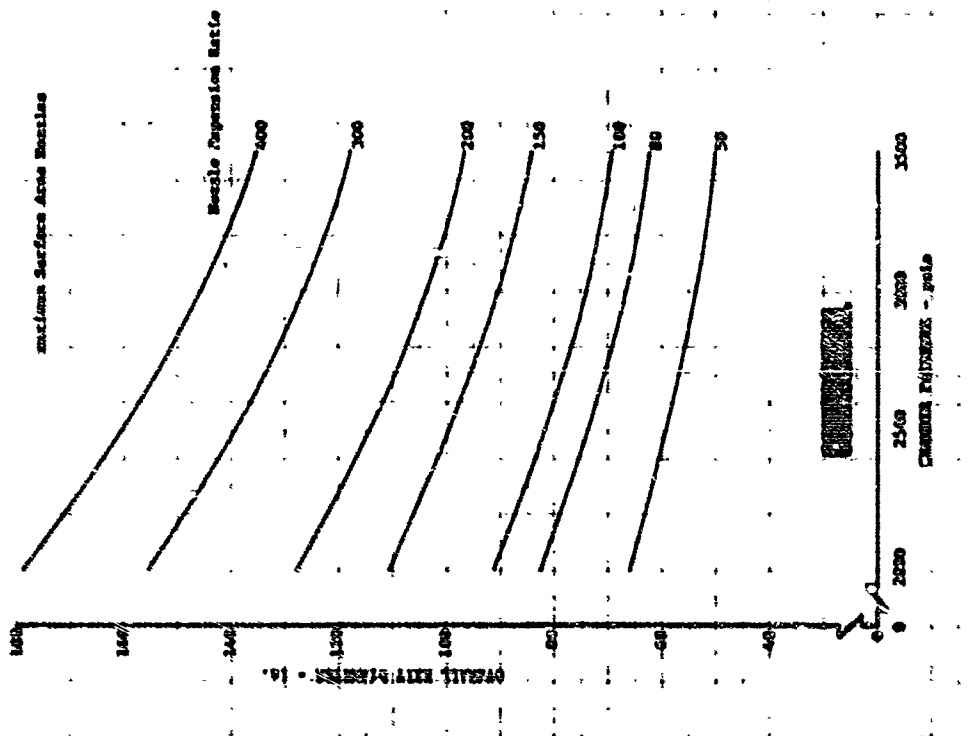


Figure 754. Overall Exit Diameter vs Chamber Pressure for 250,000-lb Thrust Engine

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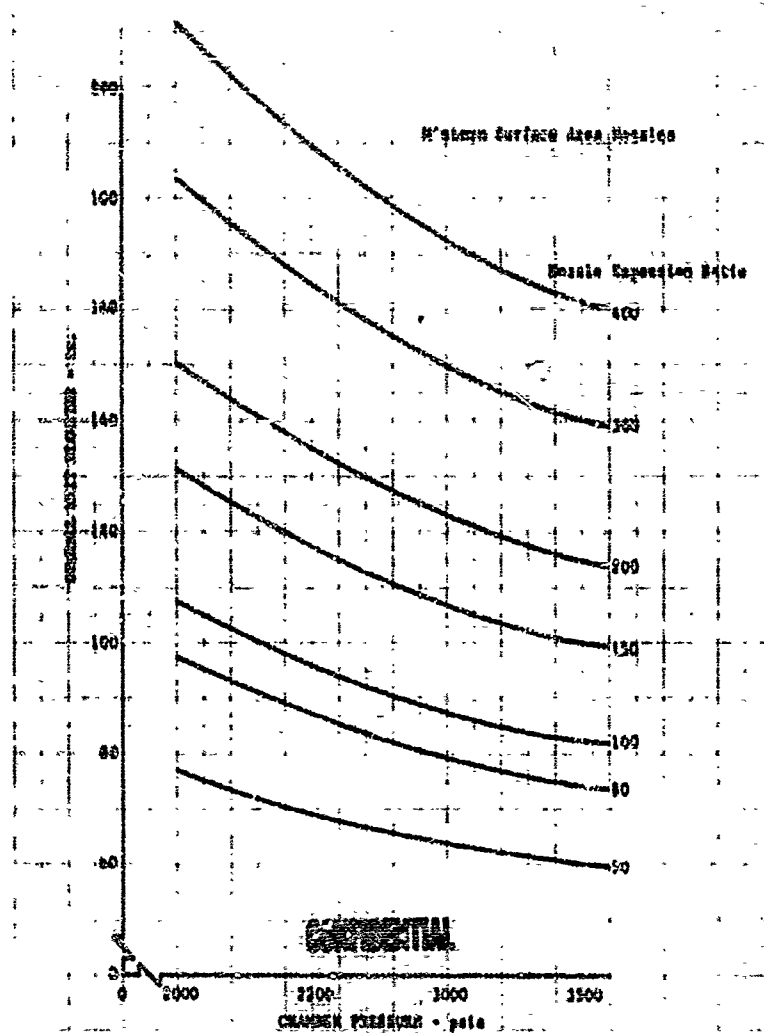


Figure 756. Overall Exit Diameter vs Chamber Pressure for 350,000-lb Thrust Engine

DF 56213

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EXPERIMENTAL DATA REDUCTION

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APPENDIX IV
PERFORMANCE DATA REDUCTION

(U) A treatment of the data, which are used to obtain the experimental performance parameters, is summarized in the following paragraphs.

A. NOZZLE THROAT AREA

1. Uncooled Chamber Tests

(U) Nozzle throat erosion during the uncooled tests requires a method for prorating the total erosion that occurs during a test to that which has occurred up to and during the data point time. Nozzle throat erosion is essentially eliminated except during the data period by using gaseous hydrogen film cooling during the transient conditions. The throat area at the data point was calculated from pre- and post-test throat diameters by assuming a constant diametric erosion rate during the data period when the coolant is turned off.

(U) To check the validity of the constant erosion rate assumption, a second method was used for comparison.

(U) The second method calculates the throat area from the equation:

$$A_x^*/A_0^* = k_1 \dot{W}_p / P_c$$

where: k_1 is chosen to make A_x^*/A_0^* equal to unity immediately after the coolant is turned off.

(U) Because only minor changes in mixture ratio and chamber pressure occur during the period that the nozzle coolant is turned off, c^* and η_c^* are considered as constant. By assuming that the significant nozzle throat erosion occurs while the coolant is off and dividing A_x^* by the original throat area (A_0^*), the rate of change in A_x^* with coolant turned off can be estimated by the equation above.

(U) A comparison of the two methods is shown in figure 757.

(U) An estimate of the nozzle discharge coefficient, C_d , and resilient mechanical area change, ΔA^* , because of pressure and thermal gradients of test conditions were made to determine the aerodynamic throat area. The aerodynamic throat area, A^* , is defined by the following equation.

$$A^* = (A_x^*)(1 + \Delta A^*/A_0^*)(C_d)$$

(U) The physical throat area change of the ATI graphite nozzle insert used for the uncooled tests was estimated by computer analysis of the pressure and thermal gradients in the graphite nozzle throat insert. The nozzle insert was divided into a series of cylinders and the deformations caused by the temperature and pressure gradients were analyzed separately, and then superimposed. The analysis indicated that $\Delta A^*/A^*$ equals -0.009, a 0.9% throat area decrease. The compressive pressure loading on the outside diameter of the nozzle throat insert tended to decrease the throat area by 1.2% while thermal expansion tended to increase throat area by 0.3%.

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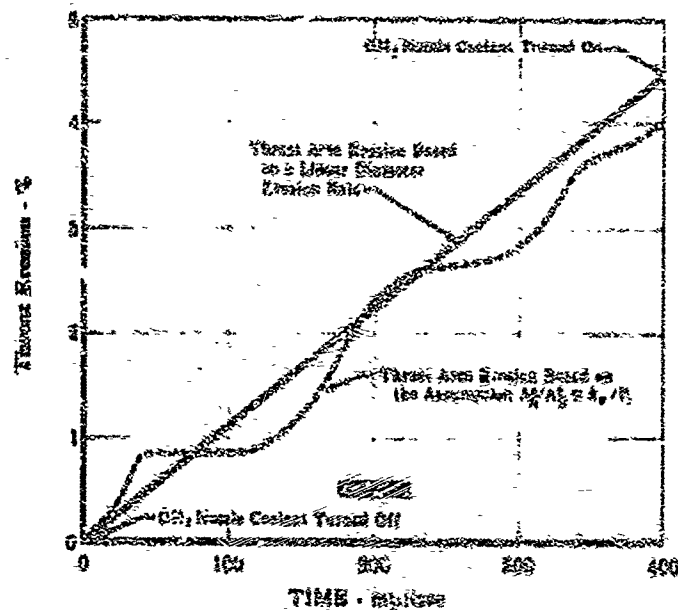


Figure 757. Nozzle Throat Estimation FD 17254A
(U) The discharge coefficients were determined from the nozzle inlet geometry by the relations shown in figure 758.⁸

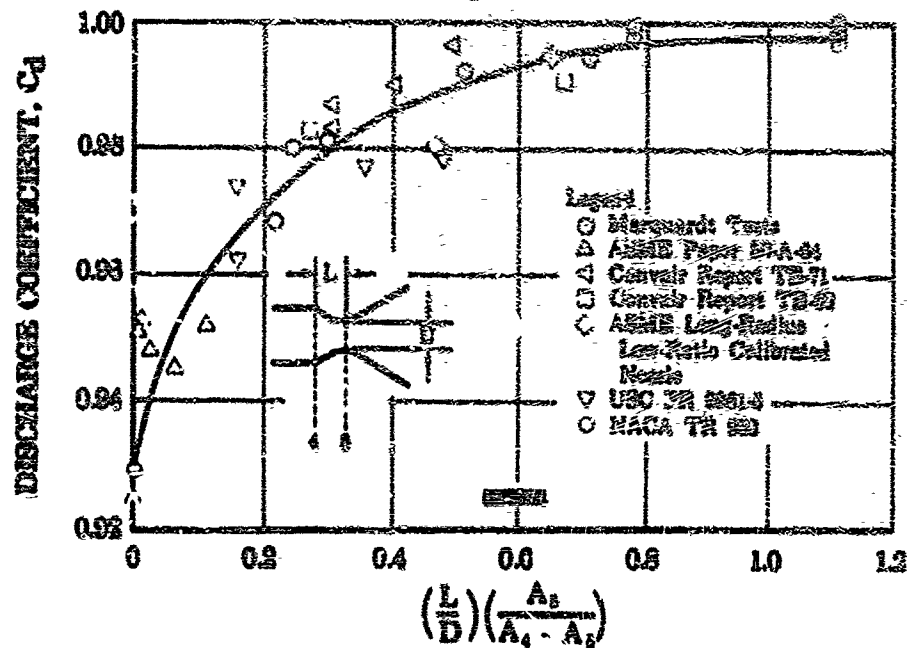


Figure 758. Discharge Coefficient vs Chamber Geometry FD 21118

(C) An estimate of nozzle throat discharge coefficient (C_d) for the contraction ratio of 5 chamber is 0.992.

⁸ Refer to Marquardt Report 3162, 1 August 1951.

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2. Cooled Chamber Tests

(C) The cooled chamber, AAW/Ag , was assumed to be negligible because the cooled copper wafers are subject to less severe thermal gradients than the uncooled graphite nozzle insert.

(C) An estimate of the nozzle throat discharge coefficient for the contraction ratio of 3 chamber is 0.559.

B. NOZZLE EXIT AREA

1. 50K Model Tests

(U) The nozzle exit area was determined by physical measurements after every test.

2. 250K Tests

(U) The measured nozzle exit area $(A_e)_0$ was corrected for thermal expansion based on measured metal temperature.

$$A_e = (A_e)_0 (\text{expansion factor})$$

C. NOZZLE AREA RATIO

(U) The nozzle area ratio was determined from the calculated aerodynamic throat and exit areas.

$$e = A_e / A^*$$

D. STAGNATION CHAMBER PRESSURE

(U) Main chamber pressure was measured as a static pressure at the injector face (P_{ci}) on the cooled chamber tests and by an additional static pressure tap (P_{cn}) at the entrance to the convergent nozzle section for the uncooled thrust chamber tests.

(U) For the uncooled chamber tests, the nozzle throat stagnation chamber pressure was calculated from measured nozzle entrance static pressure and the relation:

$$P_c = P_{cn} \times \text{function}(r, A_e/A^*)$$

(U) For the cooled chamber tests, a momentum balance determined the nozzle entrance static pressure from the injector face readings. The throat stagnation pressure was then calculated as described above.

(U) The momentum balance analysis assumed that the flow is parallel and homogeneous at the nozzle entrance. Neglecting wall friction, the equation is:

$$A_c P_{ci} + z = \dot{w} \frac{V}{g} + A_c P_{cn}$$

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where:

- z = Axial momentum of the propellants at the injector face
- A_c = Combustion chamber area
- P_{c1} = Static pressure at the injector face
- v = Velocity at the nozzle entrance, $f(r, A_c/A^*)$
- P_{c2} = Static pressure at the nozzle entrance.

(U) If each propellant is injected at an injector pressure differential ΔP , with an effective area A_{cd} , and inclined to the chamber axis at an angle the total momentum is:

$$z = (2A_{cd} \Delta P \cos \theta)_f + (2A_{cd} \Delta P \cos \theta)_o$$

(U) For the uncooled runs, the nozzle entrance static pressure was calculated to be approximately 38 psia less than the measured static pressure at the injector face. Corresponding pressure measurements verified this static pressure loss to be 50 ± 5 psia; this verified the momentum balance method used.

E. PROPELLANT FLOW RATE

$$(U) \quad \dot{W}_p = \dot{W}_o + \dot{W}_f$$

where:

$\dot{W}$ = Volumetric flow rate x density

(U) Volumetric propellant flow rates are measured with turbine-type flowmeters or standard orifices. The propellant densities are established by temperature and pressure measurements at the flowmeters.

(U) Oxygen contamination was measured from samples taken immediately before and after each test. The oxidizer flow rate was corrected for the difference in densities of O_2 , N_2 , and A.

(U) Propellant leakages through stand valves and flange seals were measured with standard orifices.

F. MIXTURE RATIO

$$(U) \quad r = \dot{W}_o / \dot{W}_f$$

G. THRUST

1. SOX Model Tests

$$(U) \quad F_{vac} = F_{meas} + P_a (A_g) + (A_o - A_g) P_a$$

where:

A_g = Nozzle area at diffuser seal

P_a = Pressure acting on outside of nozzle skirt downstream of the diffuser seal.

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(U) For test (50SC3C), which was conducted without a diffuser, separation occurred and a separation drag thrust correction was applied by graphically integrating the pressure area parameters downstream of the area ratio at which the separation occurred.

2. 230K Tests

(U) For the retracted nozzle tests the sea level thrust was measured directly.

(U) For the extended nozzle tests:

$$F_{vac} = F_{meas} + A_e \times P_{amb}$$

H. CHAMBER HEAT LOSS

(d) The 50K uncooled tests were conducted with graphite and copper chamber liners. The graphite was uncooled but the copper liners were GH film-cooled except at the data point. A heat transfer computer program was conducted to analyze both the graphite and copper chamber liner heat transfer using film coefficients as calculated by the Bartz⁹ equation.

(C) The total heat flux was estimated to be 5000 and 4200 Btu/sec for the uncooled copper and graphite liners, respectively. The effect of these heat fluxes on characteristic velocity and vacuum impulse was estimated by a theoretical combustion program to be approximately 0.28 and 0.44% respectively.

I. NOZZLE HEAT LOSS

(C) For the uncooled nozzle tests the nozzle skirt downstream of an area ratio of 4.75 was film-cooled with water except at the data point. Analysis of the transient heat transfer was used to obtain a heat loss and heat loss profile.

(C) On the 250K tests (250SC7C through 250SC11C), the primary nozzle was hydrogen cooled from an area ratio of 4.75 to an area ratio of 20. The nozzle heat loss in this section was determined from measured cooling flow rates and inlet and discharge conditions.

(U) The effect of heat loss on performance was estimated by computing the theoretical performance for an equivalent loss of heat at discrete nozzle stations.

⁹ Bartz, D.R., "A Simple Equation for Rapid Estimation of Rocket Nozzle Convective Heat Transfer Coefficients," Jet Propulsion 27 (49), 1957.

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J. PROPELLANT INJECTOR PARAMETERS

1. Fuel Injection Velocity

(U) In the following equations subscript "f" refers to the hydrogen in the preburner and to the preburner combustion products in the main burner.

$$V_f = \sqrt{\frac{P_t}{\rho_f} \frac{2\gamma_f}{\gamma_f - 1} \left[\left(\frac{P_t}{P} \right)_f^{\frac{\gamma_f - 1}{\gamma_f}} - 1 \right]} \left(\frac{P_t}{P} \right)_f^{\frac{\gamma_f + 1}{\gamma_f}} \quad (250K \text{ Tests})$$

$$V_f = \sqrt{\frac{2\gamma_f \Delta P_f}{\rho_f}} \quad (50K \text{ Tests})$$

2. Oxidizer Injection Velocity

$$(U) \quad V_o = \sqrt{\frac{2\gamma_o \Delta P_o}{\rho_o}}$$

3. Injection Velocity Ratio

$$(U) \quad VR = \frac{V_f}{V_o}$$

d. Injection Momentum Ratio

$$(U) \quad MR_{\text{main burner}} = \dot{w}_{fs} V_f / \dot{w}_o V_o = \left(1 - \frac{\dot{w}_r}{\dot{w}_f} \right) \left(\frac{\dot{w}_f V_f}{\dot{w}_o V_o} \right)$$

where:

$\dot{w}_{fs}$ = Flow through the fuel slots around the oxygen elements

$\dot{w}_r / \dot{w}_f = A_{cd} \text{ Rigimesh injector face} / A_{cd} \text{ Fuel total} \times 100 = \% \text{ face cooling}$

$$(U) \quad MR_{\text{preburner}} = \dot{w}_f V_f / \dot{w}_o V_o$$

K. SPECIFIC IMPULSE

$$(U) \quad I_s = F / \dot{w}_p$$

L. IMPULSE EFFICIENCY

1. Nozzle Retracted

$$(U) \quad \eta_{I_{s1}} = 100(I_{s1}^t / I_{s1})$$

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where I_{01}^I is a function of propellant inlet conditions, chamber pressure, exhaust pressure, overall mixture ratio, nozzle area ratio, oxidizer contamination, and nozzle and chamber heat loss; for one-dimensional isentropic flow and shifting equilibrium.

2. Nozzle Extended

$$(U) \quad \eta_{I_{vac}} = 100(I_{vac}/I_{vac}^I)$$

where I_{vac}^I is a function of propellant inlet conditions, chamber pressure, overall mixture ratio, nozzle area ratio, oxidizer contamination, and nozzle and chamber heat loss; for one-dimensional isentropic flow and shifting equilibrium.

M. THRUST COEFFICIENT

$$(U) \quad C_I = F/P_c A^*$$

N. THRUST COEFFICIENT EFFICIENCY

1. Nozzle Retracted

$$(U) \quad \eta_{C_{F_{sl}}} = 100(C_{F_{sl}}/C_{F_{sl}}^I)$$

where $C_{F_{sl}}^I$ is a function of propellant inlet conditions, chamber pressure, exhaust pressure, overall mixture ratio, nozzle area ratio, oxidizer contamination, and nozzle heat loss; for one-dimensional isentropic flow and shifting equilibrium.

2. Nozzle Extended

$$(U) \quad \eta_{C_{F_{vac}}} = 100(C_{F_{vac}}/C_{F_{vac}}^I)$$

where $C_{F_{vac}}^I$ is a function of propellant inlet conditions, chamber pressure, overall mixture ratio, nozzle area ratio, oxidizer contamination, and nozzle heat loss; for one-dimensional isentropic flow and shifting equilibrium.

O. CHARACTERISTIC VELOCITY

$$(U) \quad c^* = P_c A^* g / \dot{w}_p$$

P. CHARACTERISTIC VELOCITY EFFICIENCY

$$(U) \quad \eta_{c^*} = 100(c^*/c^{*I})$$

where c^{*I} is a function of propellant inlet conditions, chamber pressure, throat mixture ratio, oxidizer contamination, and chamber heat loss (for uncooled tests); for one-dimensional isentropic flow and shifting equilibrium.

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Q. TEST DURATION AND DATA POINT TIME

(U) Test duration and the data point times were selected by the techniques shown in figure 759.

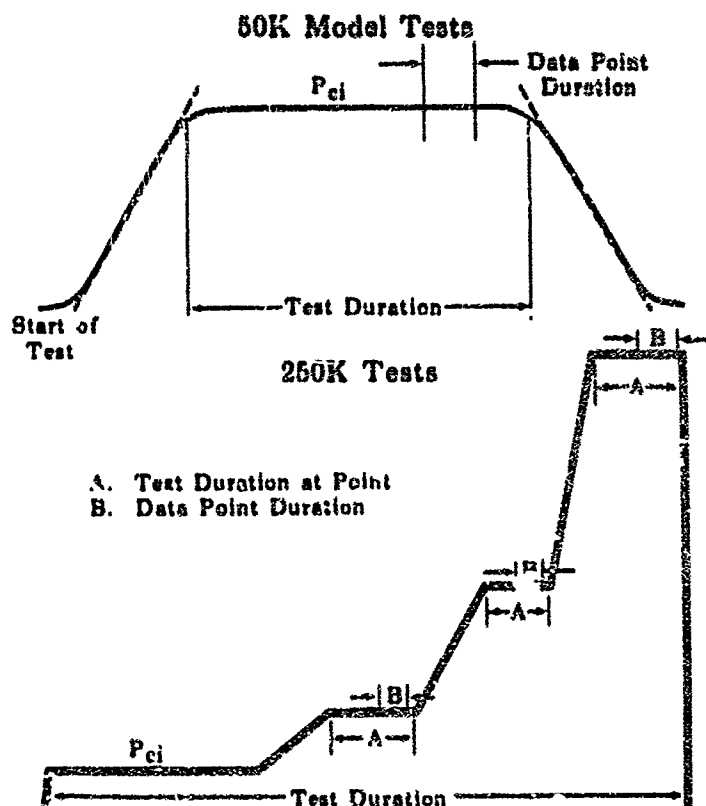


Figure 759. Typical P_c Tracer Showing Base for Firing Duration and Data Period

FD 23065

R. TEST INSTRUMENTATION ERROR ANALYSIS

1. 50K Data

(U) An error analysis was made to estimate the accuracy of the combustion performance data. The method by Dr. J. N. Berretone¹⁰, "Tolerancing by the Statistical or Differential Method," was used. According to Berretone's equation, the error, Δy , in the dependent variable y , is a function of the independent variables, x , and their estimated maximum probable errors Δx as shown by the relation:

$$\Delta y = \left[\sum_{i=1}^N \left(\frac{\partial y}{\partial x_i} \right)^2 (\Delta x_i)^2 + \left(\frac{\partial y}{\partial x_2} \right)^2 (\Delta x_2)^2 + \left(\frac{\partial y}{\partial x_N} \right)^2 (\Delta x_N)^2 \right]^{1/2}$$

where N is the total number of independent variables.

¹⁰Chairman, Department of Statistics, Western Reserve University

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(U) Maximum expected instrumentation errors (two standard deviations) for the 50,000-lbf thrust level tests on the B-26 test facility were:

Parameter	Maximum Expected Instrumentation Error
Chamber pressure*	$\pm 0.41\%$
Flowmeter pressures	$\pm 0.48\%$
LO ₂ volumetric flow rate	$\pm 0.64\%$
LH ₂ volumetric flow rate	$\pm 0.71\%$
GH ₂ volumetric flow rate	$\pm 1.4\%$
LO ₂ flowmeter temperature	$\pm 0.13^{\circ}\text{R}$
LH ₂ flowmeter temperature	$\pm 0.13^{\circ}\text{R}$
GH ₂ flowmeter temperature	$\pm 2.0^{\circ}\text{R}$
Thrust	$\pm 0.36\%$

*Based on redundant pressure readings.

(U) The maximum expected error for nozzle throat area was $\pm 1.0\%$ and $\pm 0.5\%$ for the uncooled and cooled tests, respectively.

(C) Applying the error analysis equation yields the following estimated performance data errors:

Parameter	Uncooled Tests		Cooled Tests	
	Nominal Value	Estimated Error (% of Nominal)	Nominal Value	Estimated Error (% of Nominal)
F _{vac}	50,000 lbf	$\pm 0.42\%$	50,000 lbf	$\pm 0.42\%$
$\dot{w}_o$	96.2 lb/sec	$\pm 1.16\%$	94.5 lb/sec	$\pm 1.15\%$
$\dot{w}_f$	14.8 lb/sec	$\pm 0.87\%$	14.2 lb/sec	$\pm 0.87\%$
$\dot{w}_c$			3.0 lb/sec	$\pm 1.5\%$
r	6.5	$\pm 1.45\%$	5.5	$\pm 1.38\%$
c*	7390 ft/sec	$\pm 1.46\%$	7540 ft/sec	$\pm 1.05\%$
CF _{vac}	1.96	$\pm 1.13\%$	1.92	$\pm 0.58\%$
t _{vac}	449 sec	$\pm 1.10\%$	448 sec	$\pm 1.07\%$
c*'	7540 ft/sec	$\pm 0.29\%$	7775 ft/sec	$\pm 0.21\%$
CF _{vac} '	2.00	$\pm 0.21\%$	1.95	$\pm 0.18\%$
t _{vac} '	468 sec	$\pm 0.10\%$	471 sec	$\pm 0.03\%$
η_c^*	98.0%	$\pm 1.49\%$	97.0%	$\pm 1.07\%$
$\eta_{CF_{vac}}$	98.0%	$\pm 1.15\%$	98.0%	$\pm 0.60\%$
$\eta_{t_{vac}}$	96.0%	$\pm 1.11\%$	95.0%	$\pm 1.07\%$

2. 250K Data

(U) The error analysis made of the 250K data used the method based on the preliminary recommendations of the Interagency Chemical Rocket Propulsion Group, Measurement Uncertainty Committee, and provides four separate numbers. The factors considered are:

1. Precision Error (σ) - A measure of the scatter or nonrepeatability of a measurement. The precision error of a measurement is reported as one sample standard deviation. It is a statistic calculated directly from redundant measurement data or indirectly as a linear combination of variance estimates.

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2. Degrees of Freedom (DF) - The effective sample size used in estimating the precision error. The degrees of freedom associated with the precision error is reported and reflects the sample size and the method used to obtain the precision error estimate. If the estimate is a root sum square of other estimates based on different sample sizes, the Welch-Satterthwaite¹¹ method is used to estimate the degrees of freedom. If the degrees of freedom exceeds 30, it is not reported (except that DF > 30).
3. Bias (B) - The systematic error of the measurement. All known biases are removed; the remaining bias is unknown in both magnitude and sign. The error is reported as an upper limit or upper bound based on a best engineering judgment. These limits are ordinarily in the form $\pm B$. B is not a statistic (i.e., not data).
4. Uncertainty (U) - An arbitrary measure of the system accuracy or closeness of the measurement to the truth. This is calculated from

$$U = \sqrt{B^2 + (t_{95;DF}\sigma)^2}$$

where:

$t_{95;DF}$ = The 95% value based on a t distribution for the degrees of freedom (DF) reported in (2).

B = Bias from (3)

σ = Precision error from (1)

The uncertainty is calculated as a linear combination of bias and precision error. This term does not have statistical validity but is useful as a single number representing accuracy.

(U) The errors of calculated parameters based on measured parameters are calculated using the method of partial derivatives and the statistical theory of propagation of error. The precision error for a calculated value, x, which is a function of two other variables y, z, i.e., $x = f(y,z)$ is calculated using a Taylor series expansion for a function of two variables. Thus

$$\sigma_x = \sqrt{\left(\frac{\partial f}{\partial y} \sigma_y\right)^2 + \left(\frac{\partial f}{\partial z} \sigma_z\right)^2}$$

¹¹ K. A. Brownlee, Statistical Theory and Methodology in Science and Engineering," John Wiley & Sons, New York City, N. Y., 1960 pp 230-240.

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(U) The maximum expected measurement errors for the 220,000-lb thrust level tests on the E-8 test facility were:

Parameter	Precision (1 σ)	Bias (Maximum Estimated)
Chamber pressure	$\pm 0.22\%$	$\pm 0.23\%$
Flowmeter pressure	$\pm 0.05\%$	$\pm 0.18\%$
LO ₂ flow rate (total)	$\pm 0.20\%$	$\pm 1.32\%$
LO ₂ flow rate (preburner)	$\pm 0.21\%$	$\pm 1.52\%$
LO ₂ flow rate (main burner)	$\pm 0.20\%$	$\pm 1.33\%$
LH ₂ flow rate (preburner)	$\pm 0.21\%$	$\pm 1.67\%$
GH ₂ flow rate (preburner)	$\pm 0.33\%$	$\pm 3.00\%$
CH ₂ flow rate (coolant)	$\pm 0.17\%$	$\pm 2.50\%$
LO ₂ flowmeter temperature	$\pm 0.05^\circ\text{R}$	$\pm 0.10^\circ\text{R}$
LH ₂ flowmeter temperature	$\pm 0.02^\circ\text{R}$	$\pm 0.05^\circ\text{R}$
GH ₂ flowmeter temperature	$\pm 0.14^\circ\text{R}$	$\pm 1.31^\circ\text{R}$
Thrust	$\pm 0.19\%$	--
A _t	$\pm 0.07\%$	--
A _c	$\pm 0.09\%$	--

(C) Applying the error analysis yields the following estimated performance data errors:

Parameter	Nominal* Value	Precision (1 σ)	Bias (Maximum Estimated)	Uncertainty (U)
F _{vac}	246,290	$\pm 0.16\%$	--	$\pm 0.31\%$
$\dot{w}_o$	467.7	$\pm 0.14\%$	$\pm 0.89\%$	$\pm 0.93\%$
$\dot{w}_f$	74.3	$\pm 0.18\%$	$\pm 2.20\%$	$\pm 2.22\%$
$\dot{w}_c$	5.3	$\pm 0.17\%$	$\pm 2.50\%$	$\pm 2.52\%$
r	5.88	$\pm 0.22\%$	$\pm 1.67\%$	$\pm 1.72\%$
c*	7556	$\pm 0.26\%$	$\pm 0.82\%$	$\pm 0.97\%$
C _{Fvac}	1.918	$\pm 0.26\%$	$\pm 0.23\%$	$\pm 0.56\%$
I _{vac}	450	$\pm 0.30\%$	$\pm 0.79\%$	$\pm 0.88\%$
c*'	7695	± 0.04	$\pm 0.31\%$	$\pm 0.32\%$
C _{Fvac} '	1.941	± 0.03	$\pm 0.23\%$	$\pm 0.24\%$
I _{vac} '	465	± 0.01	$\pm 0.06\%$	$\pm 0.07\%$
η_c^*	98.2	$\pm 0.26\%$	$\pm 0.87\%$	$\pm 1.02\%$
η_{CFvac}	98.8	$\pm 0.26\%$	$\pm 0.32\%$	$\pm 0.61\%$
η_{Ivac}	96.7	$\pm 0.20\%$	$\pm 0.79\%$	$\pm 0.86\%$

*Based on test No. 250SG7C, 100% thrust data.

Note: The degrees of freedom were greater than 30.

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(U) The above analyses of the 50% and 250% data apply only to instrumentation and other measuring techniques to which maximum expected errors can be estimated with a high degree of confidence. The effect of other factors such as heat loss, contamination effects, imperfect steady-state conditions, and inexact knowledge of the thermodynamic nozzle throat area behavior can only be approximated and increase the actual maximum expected errors over the values given.

S. ANALYTICAL PERFORMANCE MODELS

1. Combustion Gas Profile

(U) A combustion gas profile model was used to determine the effect on performance of combustion chamber mixture ratio profile (incomplete mixing) exclusive of the effect of transpiration cooling.

(U) The propellants, chamber pressure, exhaust nozzle area ratio, and average engine mixture ratio are selected, and the theoretical characteristic velocity and thrust coefficient calculated as reference values. The reaction products are distribution divided into nineteen stream tubes of equal mass fraction. A parabolic mixture ratio about the mean mixture ratio is arbitrarily assumed. A mixture ratio of some arbitrary increment below the average engine mixture ratio is selected, and the required mixture ratio for a corresponding stream tube above average engine mixture ratio is calculated as follows:

(U) The weight fraction of fuel and oxidizer based on the engine mixture ratio are:

$$X_{f_n} = \frac{1}{1 + r_{on}} \text{ and } X_{o_n} = \frac{r_{on}}{1 + r_{on}}$$

(U) The weight fractions of the total fuel and total oxidizer in a lower mixture ratio stream tube were:

$$X_{f_1} = \frac{Y_1}{1 + r_1} \text{ and } X_{o_1} = \frac{Y_1 r_1}{1 + r_1}$$

(U) the weight fraction of total fuel and oxidizer in the higher mixture stream tube is:

$$X_{f_h} = X_{f_n} - X_{f_1} \text{ and } X_{o_h} = X_{o_n} - X_{o_1}$$

and the mixture ratio is:

$$r_h = \frac{X_{o_2}}{X_{f_h}}$$

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(U) Having determined the mixture ratio for all the stream tubes, the performance for the unmixed case is calculated. First, the throat condition for the simultaneous expansion of the stream tubes is found by minimizing the flow area for the sum of the stream tubes.

$$\dot{W} = \rho A V = \text{constant}$$

$$A = \frac{\dot{W}}{\rho V} = \text{minimum at nozzle throat}$$

or

$$A^* = \frac{Y_1}{\rho_1 V_1} + \frac{Y_2}{\rho_2 V_2} + \dots + \frac{Y_{19}}{\rho_{19} V_{19}} = \text{minimum} \quad (12)$$

(U) Gas densities and velocities are found by determining equilibrium performance with expansion to a series of pressures near the estimated throat pressure. An area is calculated and the throat pressure is determined for the minimum value solution of equation (12).

(U) A similar procedure is then used to determine the vacuum specific impulse based on expansion to the area ratio of interest. For a given exit pressure the area ratio is:

$$\frac{A_e}{A^*} = \frac{Y_1 / \rho_1 V_{1e} + Y_2 / \rho_2 V_{2e} + \dots + Y_{19} / \rho_{19} V_{19e}}{Y_1^* / \rho_1^* V_1^* + Y_2^* / \rho_2^* V_2^* + \dots + Y_{19}^* / \rho_{19}^* V_{19}^*} \quad (13)$$

(U) For several pressures near the estimated exit pressure the density and velocity are found with the combustion deck. The exit pressure corresponding to the area ratio of interest is found by interpolation.

(U) I_{vac} and c^* are calculated for each stream tube with the equilibrium performance deck for the determined throat pressure and nozzle exit pressure. The unmixed performance is then calculated by taking weighted averages of the stream tubes.

$$c^* = Y_1 c_1^* + Y_2 c_2^* + \dots + Y_{19} c_{19}^*$$

$$I_{vac} = Y_1 I_{vac 1} + Y_2 I_{vac 2} + \dots + Y_{19} I_{vac 19}$$

(U) The unmixed characteristic velocity and vacuum specific impulse efficiency relative to the completely mixed case are:

$$\eta_c^* = \frac{c^*}{c^*} \text{ and } \eta_{I_{vac}} = \frac{I_{vac}}{I_{vac}^*}$$

where c^* and I_{vac}^* are theoretical values at the average mixture ratio, and the thrust coefficient efficiency is:

$$\eta_{CF_{vac}} = \frac{\eta_{I_{vac}}}{\eta_c^*}$$

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(U) η_c^* , η_{cp} , and η_{vac} , as functions of mixture ratio profile at constant engine mixture ratio can be determined by assuming different increments of mixture ratio, $r_{oa} = r_1$, and repeating the calculations.

2. Transpiration Cooling Mixture Ratio Profile Model

(U) An analytical model that assumes partial mixing of the injector combustion gases and hydrogen coolant was used to estimate the effect of transpiration cooling on performance. A mixture ratio profile for the flow passing through the throat section is assumed as shown in figure 760. The profile is calculated by assuming:

1. The mixture ratio varies linearly with area across a diffusion layer r from the wall to the mainstream.
2. The local gas temperature across the profile is equal to the ideal combustion temperature at the local mixture ratio.
3. The stream tube mass fraction is proportional to tube area fraction.

(U) The profile in the diffusion layer is divided into i concentric stream tubes of equal area. The area of each tube (ΔA_i^*) is determined by summing the weighted average of the percent fuel of each tube (including the mainstream) and equating the sum to the average overall percent fuel value at the throat, as shown in the following equation:

$$\sum_i (\% \text{ fuel})_i \Delta A_i^* = (\text{Overall Throat } \% \text{ fuel}) A^* \quad (14)$$

(U) The predicted characteristic velocity (c^*) is determined by summing the mass fraction contribution of the theoretical characteristic velocities of each stream tube.

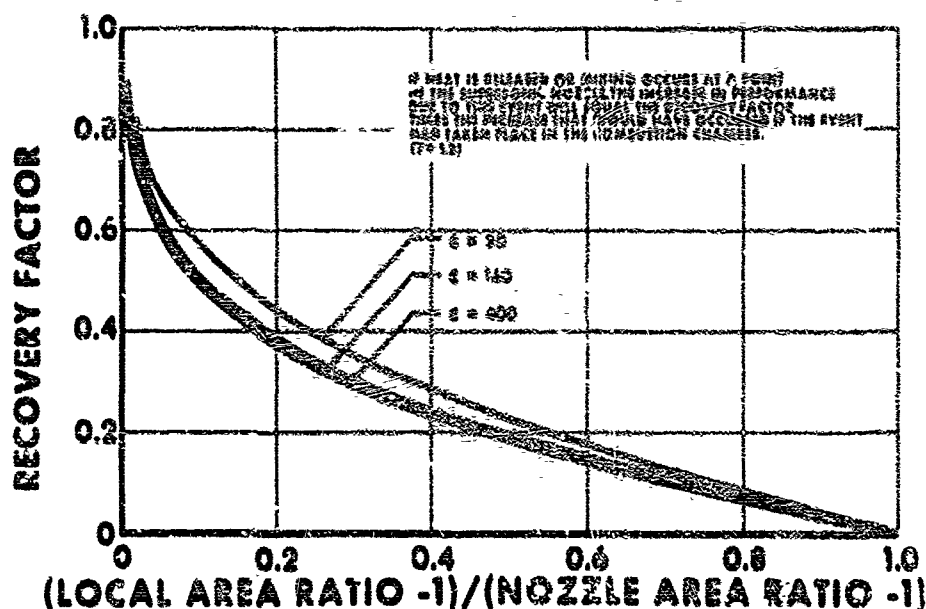


Figure 760. Mixture Ratio Profile

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(U) Base specific impulse is the summation of the specific impulses corresponding to the various stream tube mixture ratios at the throat as expressed by the following equation:

$$I_{S \text{ Throat (Base)}} = \frac{\left[\frac{\text{Total Transpiration Engine - Cooling Down-Flow}}{\text{Total Engine Flow}} \right] \sum_1 (I_S)_1 \left(\frac{\Delta A_1^*}{A^*} \right)}{1} \quad (15)$$

(U) Because a portion of the transpiration cooling flow is injected through the nozzle wall downstream of the throat, the continuing diffusion of the cooling flow into the main flow (burning or releasing of energy) alters this base impulse.

(U) Specific impulse is calculated at the cross section where the transpiration cooling flow terminates in the nozzle by the same procedure as that described for the throat section. Sections in the nozzle downstream of the end of transpiration cooling are examined also. New mixture ratio profiles are constructed and their resulting specific impulses calculated. The new profiles are determined by assuming that the diffusion layer increases at a constant rate for every unit length/unit diameter (L/D) in the nozzle. The difference between the thickness of the diffusion layer at the end of the transpiration cooled section and at the throat divided by the $\sum(\Delta L/D)$ at the end of the cooling section is the radial growth per increment of L/D. Therefore, at any other section in the nozzle the $\sum(\Delta L/D)$ of that section will yield the increase in diffusion layer thickness over the value at the throat. $\sum(\Delta L/D)$ vs expansion ratio for a typical bell nozzle is illustrated in figure 761.

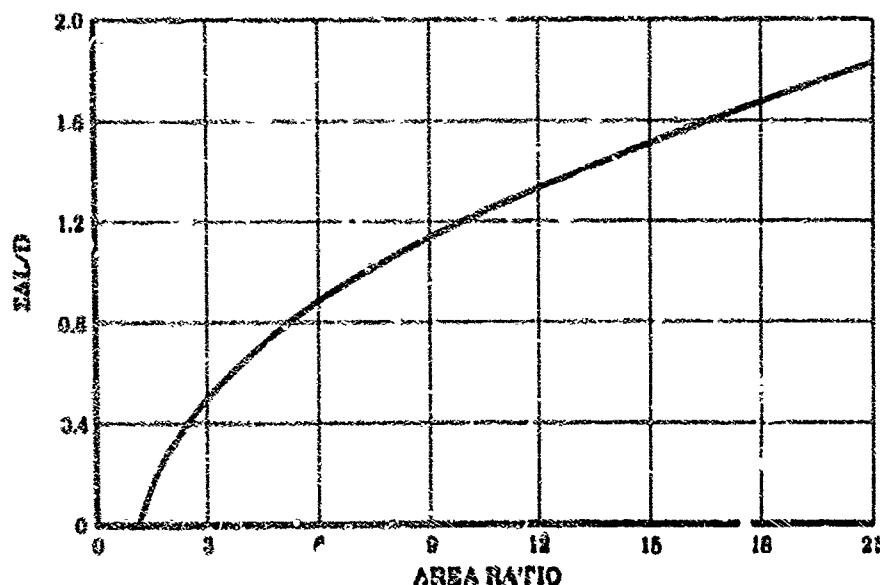


Figure 761. $\sum(\Delta L/D)$ vs Area Ratio

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(U) The relative impulse contribution (ΔI) calculated at each section is correlated by a recovery factor, RF, (refer to figure 762) to obtain an impulse gain associated with the mixing in the nozzle. This increase in impulse added to the base impulse calculated at the throat as shown in equation (16) is the total predicted specific impulse for the nozzle.

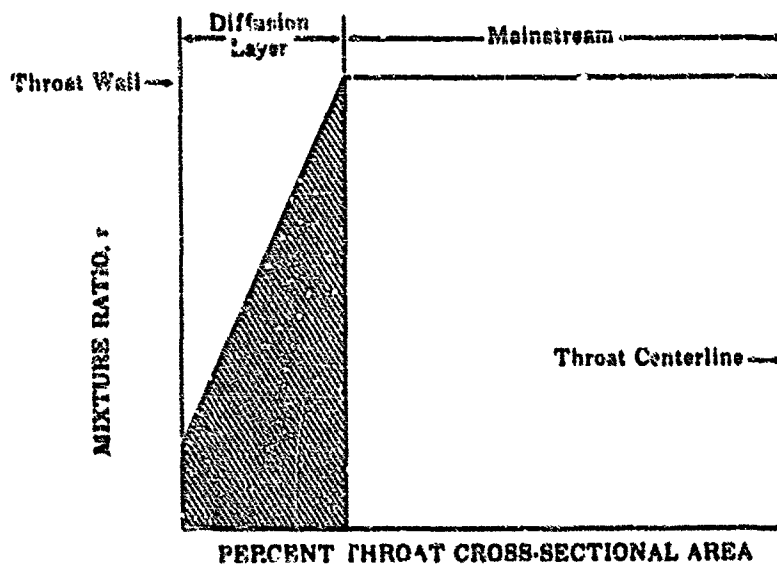


Figure 762. Nozzle Recovery Factor

FD 5204

$$I_S = I_{S \text{ Throat (base)}} + RF_1 \left[\Delta I \right]_{\epsilon=1}^{\epsilon=ET} + RF_2 \left[\Delta I \right]_{\epsilon=ET}^{\epsilon=\epsilon_1} + \dots + RF_n \left[\Delta I \right]_{\epsilon=E_{n-2}}^{\epsilon=\epsilon_x} \quad (16)$$

where:

RF = Recovery factor
ET = Area at end of transpiration cooling
 ϵ_x = Nozzle exit area ratio

(U) The theoretical predicted thrust coefficient (C_F) is found by taking the ratio of specific impulse to the c^* predicted by the model.

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APPENDIX V
POCKET EXHAUST SAMPLING PROBE

(C) Correlation of the high pressure test data with theoretical analytical models such as the combustion gas profile model discussed in Appendix II suggests that a significant source of performance loss (3.0% at $r = 6.5$ based on 50K tests) is attributable to the main chamber combustion gas mixture ratio profile. The assumed profile at $r = 6.5$ is shown in figure 763.

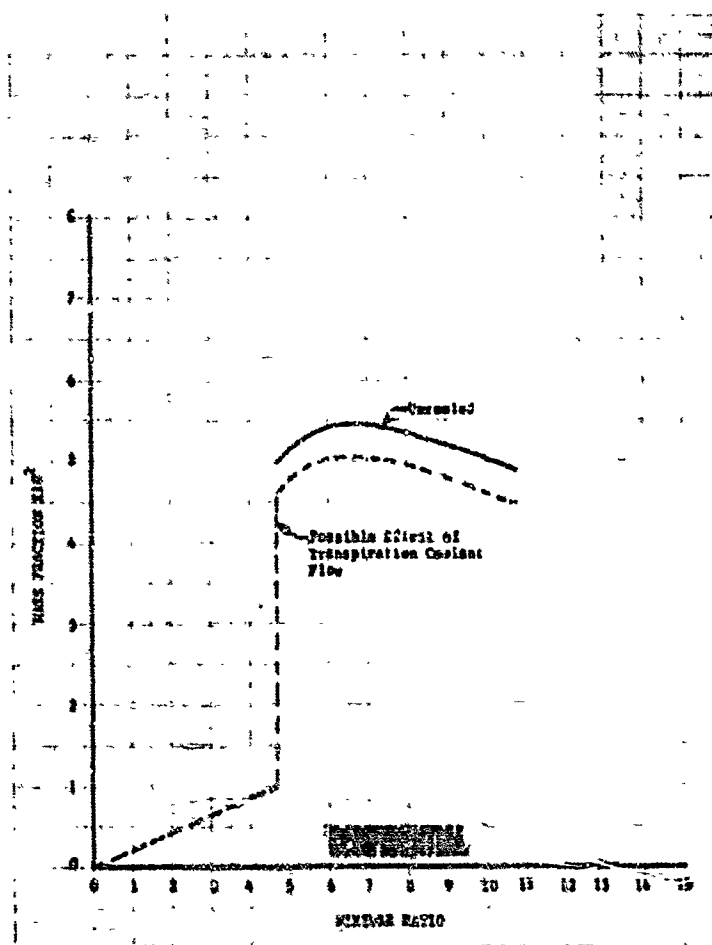


Figure 763. Mixture Ratio Profile Predicted by Combustion Gas Profile Model for $r_{inj} = 6.5$ and $P_c = 3000$ psi DF 59677

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(U) A probe with four radial sampling ports, shown in figure 764, was used to evaluate the mixture ratio profiles at the exit plane of the secondary nozzle.

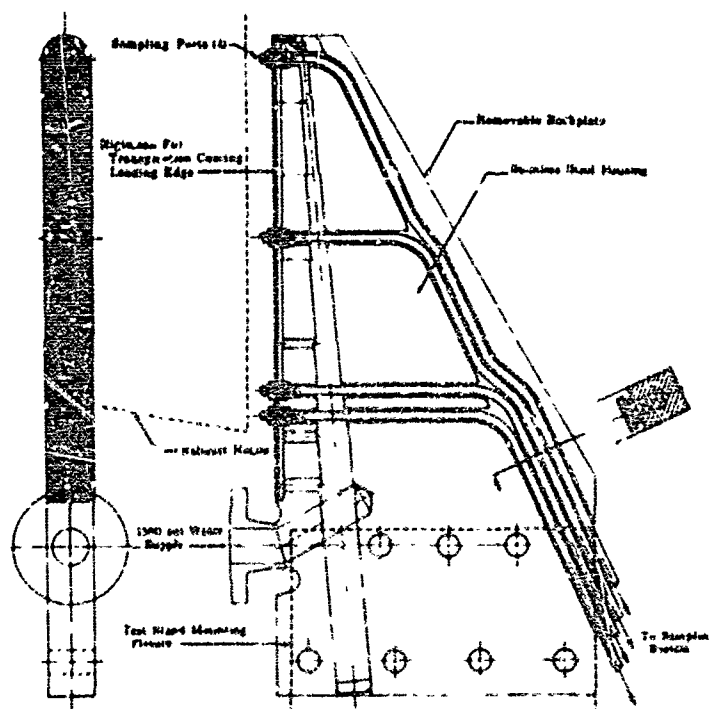


Figure 764. Rocket Exhaust Gas Sampling Probe FD 20173A

(U) The sampling system is shown schematically and installed on the test stand in figures 765 and 766, respectively. Probe tip radial locations are shown in figure 767. Two techniques were employed for profile determination:

1. Continuous monitoring resonator technique
2. Trapped gas sample chromatograph analyses.

(U) The resonator technique is based on the principle that a resonator frequency is proportional to the speed of sound in the media, which is a function of the gas temperature, molecular weight, and specific heat ratio. The gas temperatures and resonator frequencies were measured and used to calculate the molecular weight (mixture ratio). The resonators were frequency-calibrated and the frequency measured by a vibration pickup. The relationship of frequency to mixture ratio and gas temperature is shown for resonator No. 4 in figure 768. In the gas analysis technique, a sample is trapped in the sampling loop of a solenoid-actuated sampling valve and transferred to the chromatograph by a helium carrier, where the constituents are separated by differences in their affinity for a molecular sieve material. The quantity of each constituent is then determined electrically by measuring the changes in the resistance of a heated element, which is affected by the thermal conductivity of the gas. Samples were taken at full thrust with the nozzle extended.

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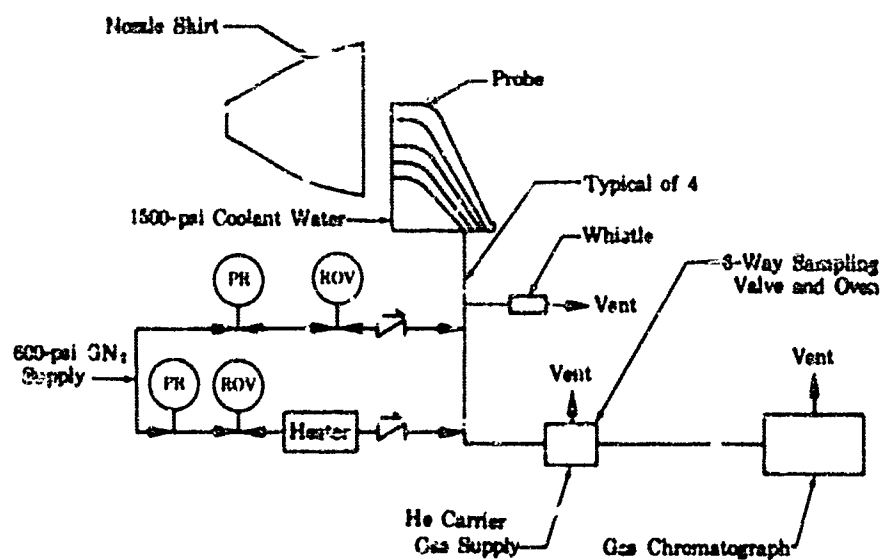


Figure 765. Rocket Exhaust Combustion Gas Sampling System

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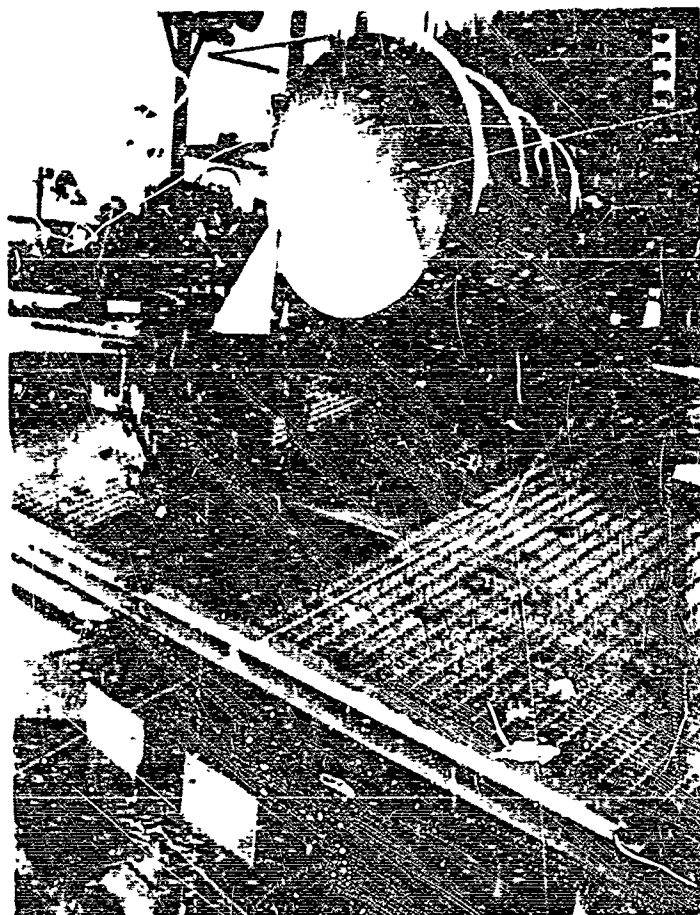


Figure 766. Exhaust Gas Sampling System Installed in Test Stand

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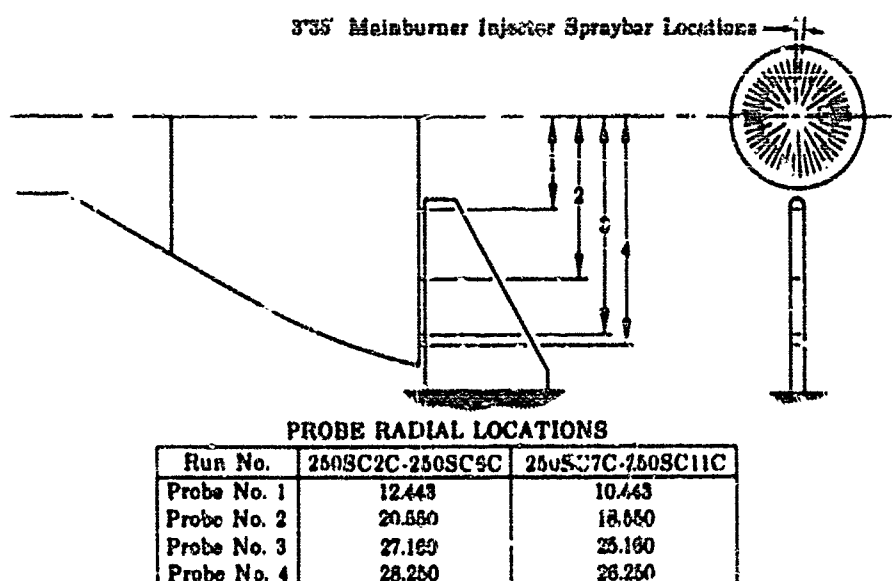


Figure 707. Probe Tip Locations

FD 23012

(C) The probe leading edge was water-cooled Rigiresh and the sampling tubes were water cooled internally in the housing. Samples were taken during seven tests (250SC2C through 250SC6C, 250SC8C, and 250SC11C). Valid data were not obtained for the first five tests because of insufficient hot gas purge time to adequately heat the lines and prevent condensation. The GH_2 heating system did not provide the capacity to preheat the system significantly above ambient (130°F maximum). Purge time was increased from 4 to 7.75 seconds and system temperatures were increased, although not enough to prevent condensation in all cases. Valid resonator data were obtained at location No. 4 during the last two tests and are shown plotted in figure 768. The chromatograph data cannot be used quantitatively because temperatures at the sample valve remained too low to prevent condensation. However, a significant variation in hydrogen and the fact that some oxygen was captured, as shown in figures 769 and 770, substantiates the theory that a significant mixture ratio profile exists.

(U) The hardware was in good condition after testing, as shown in figure 771. Some erosion of the sampling tips and of the Rigiresh face occurred where the shock wave from the sampling ports impinged on the housing. The probe will be modified to allow more coolant in these areas for future tests. The sampling system will be modified to ensure adequately high system temperatures (>300°F) to prevent water condensation.

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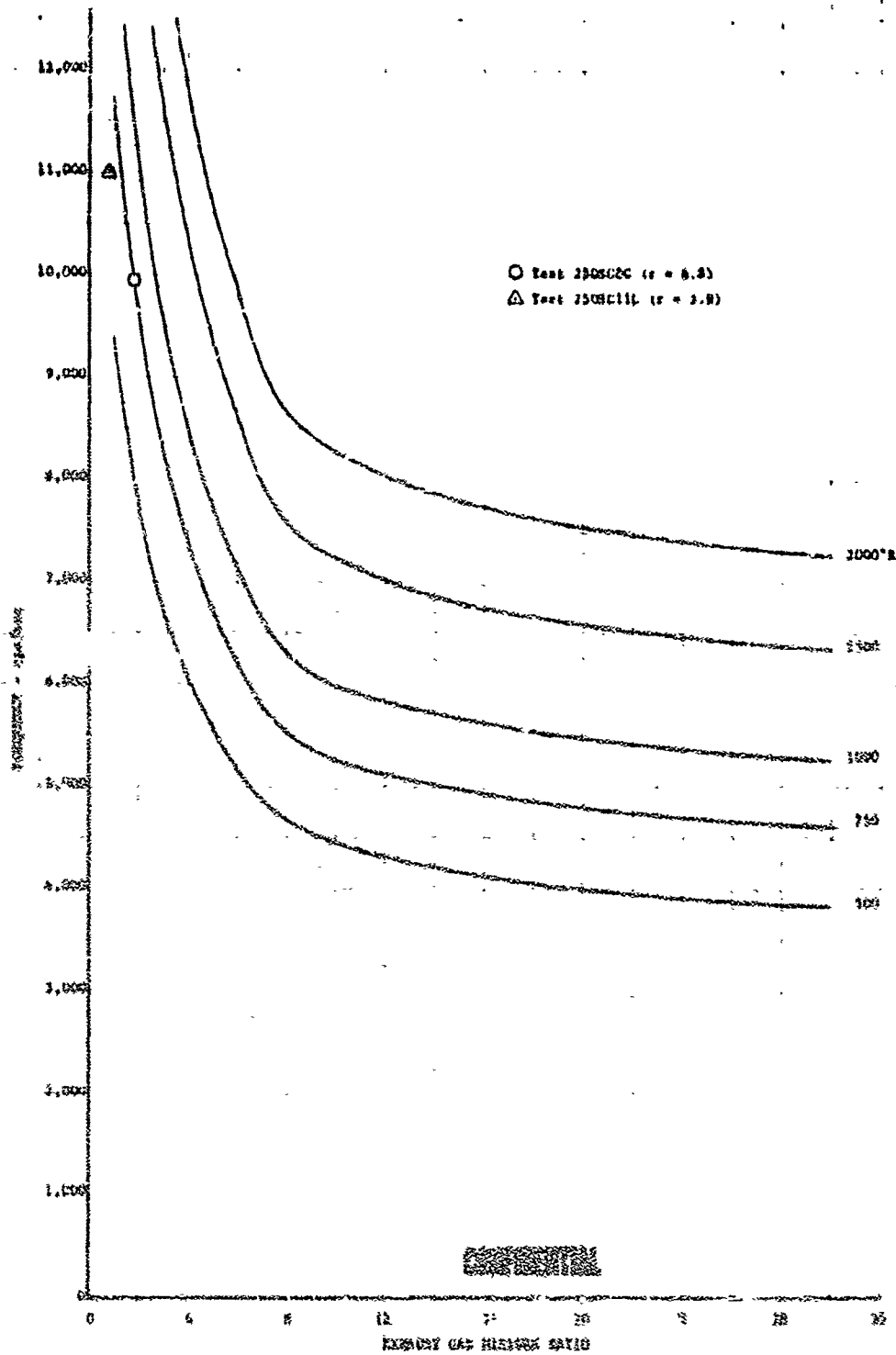


Figure 768. 250K Exhaust Gas Analysis

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△ Test No. 250000C (r = 1.9)
○ Test No. 250000C (r = 6.8)

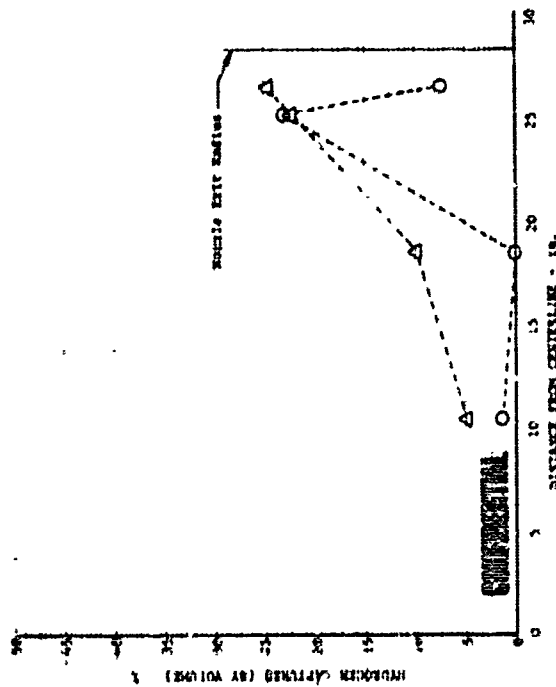


Figure 769. Percent Free Hydrogen (Captured by Sampling Probe) vs Radial Distance From Centerline at Nozzle Exit

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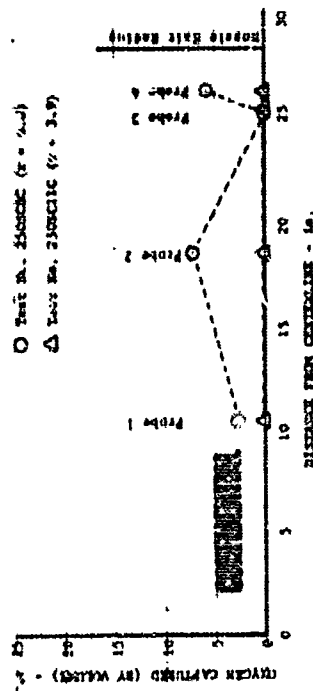


Figure 770. Percent Oxygen (Captured by Sampling Probe) vs Radial Distance From Centerline at Nozzle Exit

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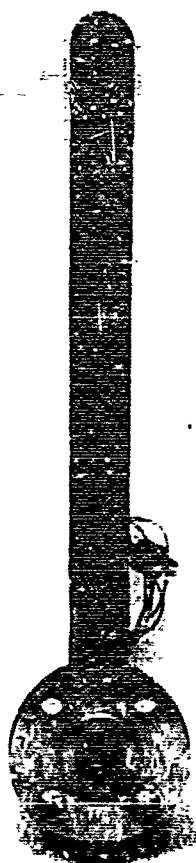


Figure 771. Post-Test Condition of Exhaust
Gas Sampling Probe

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DOCUMENT CONTROL DATA - R&D		
<i>(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)</i>		
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3 REPORT TITLE FINAL REPORT ADVANCED CRYOGENIC ROCKET ENGINE PROGRAM STAGED-COMBUSTION CONCEPT		2b GROUP 4
4 DESCRIPTIVE NOTES (Type of report and inclusive dates) Final Report, 1 March 1966 to 30 September 1967		
5 AUTHOR(S) (Last name, first name, initial) Atherton, Robert R.		
6 REPORT DATE 15 December 1967	7a TOTAL NO OF PAGES 814	7b NO OF REFS 4
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b PROJECT NO c 681C d BPSN 63681C	9b OTHER REPORT NO(S) (Any other numbers that may be assigned this report) AFRPL-TR-67-298	
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11 SUPPLEMENTARY NOTES Patent Secrecy "Special" and "A" permits	12 SPONSORING MILITARY ACTIVITY Propulsion Laboratory Research and Technology Division, Edwards, California Air Force Systems Command, USAF	
13 ABSTRACT Phase I of the Advanced Development Program for a High Performance Oxygen/Hydrogen Rocket Engine, which was sponsored by the Air Force Rocket Propulsion Laboratory at Pratt & Whitney Aircraft, was an evaluation of the critical technology associated with the staged-combustion bell nozzle engine system. Experimental evaluation was conducted in the areas of preburner, main chamber, nozzle, turbopump bearings, and engine controls. In addition, engine system (module) preliminary design and applications studies were conducted. In the Module Design Study, a system cycle balance, steady-state off-design analyses, transient analyses, component and system design studies, a weight study, and a parametric engine study were completed. The Applications Study was completed, and a separate final report, AFRPL-TR-67-270, was issued. Under the Cooling Investigation, 50K staged combustion tests were conducted that demonstrated high impulse efficiency and the two-position translating nozzle. Under the Turbopump Component Investigation, endurance testing of the hydrogen turbopump bearing demonstrated long life is feasible, but roller skewing remains a significant problem. Under the Module Control System Investigation, the oxidizer flow divider valve, the mixture ratio valve, and ignition systems were designed, manufactured, and tested with the preburner and main burner. Continued development is needed to improve seal performance. The Preburner Demonstration investigation was completed, and ignition, control, and dynamic combustion stability were demonstrated, however, additional development is required to reduce the hot gas temperature profile. Under the 250K Staged Combustion investigation, the 250K main chamber was tested, and the performance and feasibility of the full-scale tandem concept, including the two-position nozzle and dynamic combustion stability, was demonstrated at various thrust levels.		

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14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Advanced Development Program						
High Performance Oxygen/Hydrogen Rocket Engine						
250K Demonstrator Engine						
High-Pressure, Staged-Combustion, Bell-Nozzle Engine						
Design and Analysis						
Module Design						
Application Study						
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Mrs. Mary Racovich, FTMKR-5
Directorate of Materiel
Procurement Division
Edwards Air Force Base, California 93523

- Reference: (a) ^{N/R} PWA FP 67-11, 250K High Performance Reusable Oxygen/Hydrogen Rocket Engine, dated 21 August 1967.
- (b) ^{N/R} PWA FR-1810, Component's Design Handbook, Advanced Development Program for a High Performance Oxygen/Hydrogen Rocket Engine, dated 30 June 1966.
- (c) ^{N/R} PWA FR-1911, Quarterly Report No. 1, Advanced Cryogenic Rocket Engine Program Staged - Combustion Concept, dated June 1966.
- (d) ^{N/R} PWA FR-1928, Quarterly Report No. 1, 250K Thrust Chamber Technology Program, dated 30 June 1966.
- (e) ^{N/R} PWA FR-2372, Final Report - Advanced Engine Design Study, Bell, (AEB), dated July 31, 1967.
- (f) PWA FR-2597, Advanced Cryogenic Rocket Engine Program Staged - Combustion Concept - Final Report, dated December 1967.

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Dear Mrs. Racovich:

The U. S. Patent Office has issued a Secrecy Order with a modifying "Security Requirements Permit" against United Aircraft Corporation's U. S. Patent Application No. 725,954, entitled "Dual Slot Swirler Injector Element." This Secrecy Order relates to a single throttleable injection element that provides a wide range of throttleability. You are advised that the referenced documents contain information relating to this concept.

Mrs. Mary Racov'ch

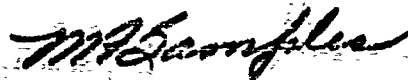
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