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Orbit Transfer Vehicle Engine Study Phase A, Extension I

Contract NAS 8-32999
Alternate Low-Thrust Capability
Task Report 32999E1-T2
April 1980

Prepared For:
George C. Marshall Space Flight Center
National Aeronautics And Space Administration

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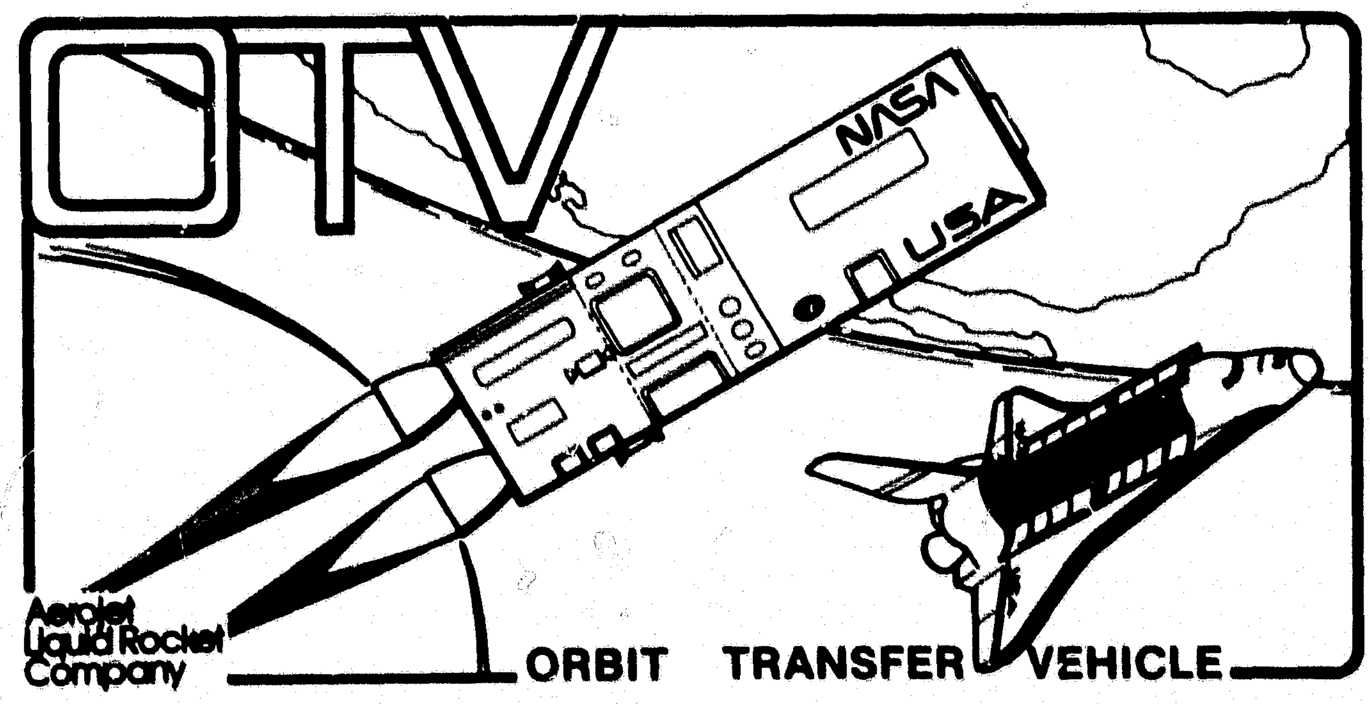

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ORBIT TRANSFER VEHICLE ENGINE STUDY
PHASE "A" EXTENSION 1

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Alternate Low-Thrust Capability

Task Report

Prepared for

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National Aeronautics and Space Administration
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The following ALRC personnel have contributed, significantly, to the data and information contained in this report.

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K. L. Christensen, Systems Analysis

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J. I. Ito, Performance and Stability Analysis

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FOREWORD

This task report is submitted for the Orbit Transfer Vehicle Engine Study, Phase "A", Extension 1 per the requirements of Contract NAS 8-32999. This work is being performed by the Aerojet Liquid Rocket Company for the NASA/Marshall Space Flight Center. The study authority to proceed was received on 20 July 1979.

The study program consists of engine system, programmatic, cost and risk analyses of OTV engine concepts. These evaluations will ultimately lead to the selection and conceptual design of the OTV engine for use by the OTV vehicle contractor.

The NASA/MSFC COR is Mr. D. H. Blount. The alternate COR is Mr. J. F. Thompson. The ALRC Program Manager is Mr. L. B. Bassham and the Study Manager is Mr. J. A. Mellish.

TABLE OF CONTENTS

TABLE LIST

FIGURE LIST

I. INTRODUCTION

A. BACKGROUND

The Space Transportation System (STS) includes an Orbit Transfer Vehicle (OTV) that is carried into low Earth orbit by the Space Shuttle. The primary function of the OTV is to extend the STS operating regime beyond the Shuttle to include orbit plane changes, higher orbits, geosynchronous orbits and beyond. The NASA and the DOD have been studying various types of OTV's in recent years. Data have been accumulated from the analyses of the various concepts, operating modes and projected missions. The foundation formulated by these studies established the desirability and the benefits of a low operating cost, high performance, versatile OTV. The OTV must be reusable to achieve a low operating cost. It is planned that an OTV have an initial Operating Capability (IOC) in 1987.

The OTV has, as a goal, the same basic characteristics as the Space Shuttle, i.e., reusability, operational flexibility, and payload retrieval along with a high reliability and low operating cost. It is necessary to obtain sufficient data, of a depth to assure credibility, from which comparative systems analyses can be made to identify the performance, development, costs, risks and program requirements for OTV concepts. The maximum potential of each concept to satisfy the mission goals will be identified in the OTV systems studies initiated in FY-79.

This program is a continuation of a study of oxygen/hydrogen engines for OTV applications. This study extension will provide preliminary design data, plans and cost information which will complement the data generated to satisfy the original Statement of Work on Contract NAS 8-32999, dated 6 July 1978. This engine data from the original and extension efforts, together with the system studies, will provide the basis to objectively select, define and design the preferred OTV engine.

I, Introduction (cont.)

B. OBJECTIVES

The major objectives of this Phase "A" engine study extension are: (1) optimize an advanced expander cycle engine for OTV applications, (2) investigate the feasibility of providing low-thrust capability within the same expander cycle engine, (3) provide additional safety, reliability, development risk, cost and planning data on OTV engine candidates, and (4) provide design and programmatic parametric data on the OTV engines for use by NASA and OTV system contractors. The original and engine study extension, in conjunction with the system studies, will provide comparative data on engine design alternatives and identify engine requirements, concepts and approaches recommended for further study on a subsequent conceptual design phase.

To accomplish the program objectives, a study program composed of five (5) major tasks and a reporting task is being conducted. The study tasks are:

Task I: Advanced Expander Cycle Engine Optimization
Task II: Alternate Low-Thrust Capability
Task III: Safety, Reliability and Development Risk Comparison
Task IV: Cost and Planning Comparison
Task V: Vehicle Systems Studies Support

This report presents the results of Task II, Alternate Low-Thrust Capability.

II. SUMMARY

The objective of this task was to assess the feasibility and design impact of a requirement for the advanced expander cycle engine to be adaptable to extended low-thrust operation of approximately 1K to 2K lb. The rated full thrust used in this evaluation was 15K lbF. This thrust level was selected to be consistent with Contract NAS 8-33574 (Orbital Transfer Vehicle Advanced Expander Cycle Engine Point Design Study) efforts.

The primary guideline used in this study was to establish a low thrust operating point that did not compromise the engine at rated thrust operation. However, this did not preclude the use of "kits" to obtain the low thrust goal. Kits could take the form of turbine flow area modification, a new replacement low thrust injector, or even new pumps. The engine low-thrust operating point is also a dedicated condition and the engine is not required to operate at both the 15K thrust level and the low thrust level on the same mission. The analyses established the limiting components and the results are summarized herein.

The OTV point design engine can be reduced in thrust with minor injector modifications from 15K to 1K without significantly affecting combustion performance efficiency or injector face/chamber wall thermal compatibility. Likewise, high frequency transverse mode combustion instability is not expected to be detrimentally affected. Primarily, the operational limitations consist of feed system chugging instabilities and potential coupling of the injector response with the chamber longitudinal mode resonances under certain operating conditions which should be determined in future efforts. The recommended injector modification for low-thrust operation is a change in the oxidizer injector element orifice size.

Operation of the baseline 15K OTV chamber liner and nozzle tube bundle at thrust levels as low as 2K was determined to be feasible. For both components, the wall temperature at the coolant inlet was found to increase at low thrust because of poor coolant heat transfer at the reduced coolant inlet temperatures.

This results in a degradation of cycle life for the nozzle, but not the chamber. However, the nozzle can be redesigned to provide the desired cycle life without penalizing the full thrust design. A preliminary flow stability analysis of the chamber indicates it is marginally stable at 2K. Prediction of coolant heat transfer characteristics at the 2K thrust level is very uncertain because the coolant pressure is very close to its critical pressure. Detailed analysis of even lower pressures, which results in two-phase coolant flow, was beyond the scope of the present effort. However, this problem can be overcome by increasing the pump discharge pressure so that the coolant outlet pressure remains above critical pressure. This may make cooling to 1 K lbF possible.

Turbopump operation and stability were investigated to the thrust levels as low as 1K (6.67% of rated thrust). Minimum levels of operation for the fuel and oxidizer turbopumps, without modification, were established as 10% (1.5K lbF) and 13.3% (2K lbF), respectively. These values were established for stable pump operation at the flow rates required for engine operation at a mixture ratio of 6.0. The oxidizer pump problem can be alleviated by increasing the pump flow. This is done by adding a recirculation line and valve between the boost and main pumps. For example, the oxidizer pump flow could be doubled so that it would operate with the flow normally associated with 2K thrust while the discharge flow would be for 1K thrust operation. The cycle is easily power balanced even with the oxidizer pump horsepower increase. There is some concern about adding too much heat to the fluid and dumping this back into the main pump inlet.

The controls system components were not found to be limiting and new valves or a valve "kit" are not required.

The engine is not power balance limited over the full 15K to 1K thrust range.

II, Summary (cont.)

Based upon the analyses and results, the following conclusions have been reached.

- The only "kit" required for operation at a 2K thrust level and a mixture ratio of 6.0 is a different set of oxidizer injection elements.

- Chamber coolant flow stability may be a concern below 2K lbF operation.

- Oxidizer pump stability could be a problem below a 2K thrust level although a recirculation flow could alleviate the problem.

- Operation at a thrust level of at least 10% of rated thrust (1.5K lbF) may be feasible although further analyses and experimental effort are required to verify the low-thrust operation.

III. ALTERNATE LOW-THRUST CAPABILITY ANALYSES

A. INJECTOR MODIFICATION ANALYSIS

1. Analysis

Table I shows some significant injection/combustion parameters which were estimated or calculated at various throttle operating points between thrust levels of 15K to 1K at a mixture ratio of 6.0. Fuel injection velocity either remains constant or increases slightly due to increased coolant bulk temperature rise. Thus, the compressible gH_2 $\Delta P_f/P_c$ ratio remains adequate throughout the entire throttle range and does not adversely affect feed system stability. On the other hand, the incompressible LO_2 injection velocity is directly proportional to thrust. Therefore the oxidizer injection ΔP_{ox} is proportional to Pc^2 and the $\Delta P_{ox}/P_c$ ratio decreases linearly with thrust, aggravating LO_2 induced chug instability.

The oxidizer combustion time lag almost varies inversely with Pc or thrust as shown on Table I. A preliminary chug stability analysis utilized these oxidizer combustion time lags to estimate the necessary $\Delta P_{ox}/P_c$ ratio. The simplified chug analysis solutions are conservative for two reasons: (1) it is based upon assumed liquid/liquid injectors having nearly equal fuel and oxidizer combustion time lags where both are rate limiting and (2) it neglects to account for any momentum exchange efficiency of the axial H_2 injection velocity upon the LO_2 swirl co-axial spray assumed in the analysis. An estimate of the necessary $\Delta P_{ox}/P_c$ as a function of the thrust and associated combustion time lags for 84 swirl co-axial elements having oxidizer orifice diameters of ~.100 in. for the point design and ~.050 in. diameter for the low thrust injector mod is indicated on Table I. These preliminary results indicate chug stable combustion characteristics between full thrust down to approximately half thrust without modification. At progressively lower thrust levels, the required oxidizer ΔP requirements increase dramatically due to reduced oxidizer injection velocities and longer time lags. By 1/4

TABLE I

THROTTLING EFFECT UPON COMBUSTION PARAMETERS

Parameters	Units												
Percent Thrust		100		50		25		25		13.3		6.7	
Thrust	(lbf)	15,000		7500		3750		3750		2000		1000	
Chamber Pressure	(psia)	1200		600		300		300		160		80	
Mixture Ratio, O/F		6.0		6.0		6.0		6.0		6.0		6.0	
Inj. Vel. Ratio, V_f/V_o		8.2		17.9		38.4		9.6		19.2		42.5	
Inj. Mom. Ratio, $\dot{W}_oV_o/\dot{W}_fV_f$		.726		.335		.156		.625		.313		.141	
Result Cone Angle	(deg)	12.6		7.4		3.9		11.5		7.0		3.6	
Fuel Pump Disch. Temp.	(°R)	90		70		55		55		47		40	
Fuel Turbine Inlet Temp.	(°R)	535		581		642		642		713		805	
Fuel Inj. Man. Temp.	(°R)	473		514		568		568		630		712	
Oxid. Elem. Dia.	(in)	.100		.100		.100		.050		.050		.050	
Propellant Circuit		Oxid	Fuel	Oxid	Fuel	Oxid	Fuel	Oxid	Fuel	Oxid	Fuel	Oxid	Fuel
Flowrate	(lbm/sec)	26.94	4.49	13.50	2.25	6.75	1.125	6.75	1.125	3.60	0.60	1.80	0.30
Injector ΔP	(psid)	215	109	54	59	13	32	215	32	61	19	16	11
Injection Velocity	(fps)	168	1385	84	1505	42	1613	168	1613	90	1727	45	1913
$(\Delta P/P_c)_{inj}$		.179	.091	.090	.098	.044	.107	.71[illegible]	.107	.381	.119	.200	.138
$(\Delta P_{ox}/P_c)_{Req'd}$ Chug Stab		.07		.15		.27		.04		.09		.19	
Oxidizer Atom./Vap.													
Mass Median Drop Radius	(in)	.00165		.00216		.00292		.00091		.00128		.00165	
Atom. Wave Length	(in)	.849		1.002		1.215		.411		.526		.713	
ΔX-20% Oxid. Vap.	(in)	.109		.148		.216		.113		.161		.222	
L_{gen} (LO_2, throat)	(in)	124.0		91.2		52.3		119.4		83.6		60.7	
% Oxid. Vap.	%	100.0		100.0		99.9		100.0		100.0		99.8	
Atom. Time Lag	(msec)	.421		.994		2.411		.204		.487		1.320	
20% Vap. Time Lag	(msec)	.054		.147		.429		.056		.149		.411	
Total Oxid. Time Lag	(msec)	.475		1.141		2.840		.260		.636		1.731	

thrust, the LO_2 injection velocity has decreased to 40 fps at which point the LO_2 swirl cone is expected to collapse due to increased surface tension effects with chugging instability becoming highly probable. As a matter of fact, the LO_2 element should be modified to assure at least 40 fps minimum injection velocity regardless of thrust, which results in approximately 20% minimum $\Delta P_{ox}/P_c$ at 1K thrust. This also results in about a .050 in. LO_2 diameter at 1K thrust and is the only component design modification deemed necessary for satisfactory thrust chamber combustion operation.

Therefore, given the freedom to modify the LO_2 element resistance, the injector can be modified to avoid feed system coupled combustion instabilities. Within the range of chug stable operation, the oxidizer combustion time lags can also be adjusted to avoid chamber longitudinal mode resonance frequencies. Therefore, the selection of a low thrust operating point is not dictated by thrust chamber combustion capabilities.

Atomization/vaporization analyses of the point design injector and the low thrust mod indicate the LO_2 vaporization will be adequate at all thrust levels in the 18-in. long chamber. Therefore, combustion efficiency is not expected to vary significantly with thrust, nor will the axial combustion distribution be altered enough to have noticeable effects upon combustion chamber thermal compatibility. The chamber pressure reduction at low thrust, however, will cause a decrease in Kinetic (ODK) performance. Similarly, at low thrust, the mass flowrate reduction in fixed hardware will increase the Boundary Layer performance loss. This will result in somewhat lower performance than an engine designed only for low-thrust operation.

2. Summary and Conclusions

The OTV point design engine can be reduced in thrust with minor modification from 15K to 1K without significantly affecting combustion performance efficiency or injector face/chamber wall thermal compatibility. Likewise, high frequency transverse mode combustion instability is not expected to be detrimentally affected. Primarily, the operational limitations consist of feed system chugging instabilities and potential coupling of the injector response with the chamber longitudinal mode resonances under certain operating conditions which still have to be determined.

It has been analytically estimated that the point design engine can be conservatively throttled down to at least 50% of rated thrust, but even by the most optimistic assumptions, cannot be throttled below 25% (3750 lbf) thrust before encountering chug instabilities. However, the only limitation is due to inadequate oxidizer injection element ΔP which can be readily corrected by allowing for two alternate dash no. swirl co-axial elements which differ only in their oxidizer port injection diameters (e.g., .100 in. dia for 15K point design and .050-in. diameter for 1K Low Thrust Engine). The injector manifolding, combustion chamber and nozzle geometry, co-axial element length and O.D., and fuel orifice areas can all be kept identical. Because an injection element substitution (kit) is required for any thrust level below 4K, it makes no difference from a combustion standpoint whether the low thrust value of 2K, 1K, or any other value is selected.

In order to achieve an adequate H_2 bulk temperature rise to drive the expander cycle turbines, a relatively long (L' = 18 in.) chamber length has been selected. This causes the chamber longitudinal mode acoustic resonance frequencies to fall into a frequency regime susceptible to injection element responses. The longitudinal mode chamber impedance remains essentially fixed at its acoustic mode sensitive frequencies, independent of engine throttling but the injector response varies widely with the injection flowrates (velocity). Thus, even though the point design engine will be nominally optimized to be free of longitudinal mode instabilities at full thrust, it should be anticipated that the engine can fall in and out of sensitive resonance modes within various throttle ranges which have not been established and were beyond the scope of this initial assessment.

3. Recommendations

a. Assume that by replacing the oxidizer co-axial injection elements, that the point design engine can provide acceptable combustion characteristics down to 1K thrust.

b. Conduct more detailed chug stability margin analysis and longitudinal mode combustion stability analyses at the above selected operating point(s).

c. Determine optimum element design modifications which are required to stabilize combustion at the selected low thrust operating condition.

B. LOW-THRUST THERMAL ANALYSIS

This subtask addressed the thermal design feasibility of low thrust operation with an engine designed for normal operation at 15,000 lbf with a chamber pressure of 1200 psia. The desired reduction in thrust to 1000-2000 lbf presents a problem from a thrust chamber cooling standpoint, because the coolant pressure is only slightly supercritical at 2000 lbf and is below the critical pressure at 1000 lbf. In these regions, heat transfer characteristics are very poorly defined, and previous experience has indicated the possibility of flow instability in this regime. These analyses considered low thrust operation at supercritical coolant pressures, assuming the hydrogen heat transfer correlation (Ref. 1) applicable for high supercritical pressures to be valid in the near critical region. Two-phase analyses at subcritical coolant pressures would require extensive computer program development and thus, were beyond the scope of the present effort. Therefore, the lowest thrust analyzed was 2000 lbf. Estimated coolant inlet conditions as a function of thrust are shown on Figure 1. At 2000 lbf, the inlet pressure is 235 psia; the critical pressure of hydrogen is 188 psia.

Low thrust analyses were conducted for both the chamber and nozzle cooling systems. In the baseline design these components are cooled in parallel, with 85 percent of the hydrogen flowing through the chamber. The interface area ratio is 8:1. A preliminary flow stability analysis is included for the chamber based on a modification of Ref. 2 to account for axial variations in channel design and heat flux.

1. <u>Chamber Analyses</u>

The baseline chamber (ε_c = 3.66, L' = 18 in.) was designed during previous Phase A study efforts. This chamber employs rectangular channels in a zirconium-copper liner. Low-thrust results compared with full thrust operation are summarized in Table II. Wall temperatures in most of the chamber are reduced at low thrust due to higher coolant bulk temperatures. Heat transfer coefficients for hydrogen improve with heating and can more than offset the increased sink temperature seen by the wall. This is shown on Table II by the reduction in wall temperature at the outlet compared with the increased coolant bulk temperature rise. At the coolant inlet at area ratio 8:1, where the hydrogen temperature is reduced at low-thrust, the wall temperature increases significantly due to the degradation in heat transfer characteristics of very cold hydrogen. Because the inlet region wall temperatures are very low at full thrust, this increase is not of concern. Of greater importance is the temperature

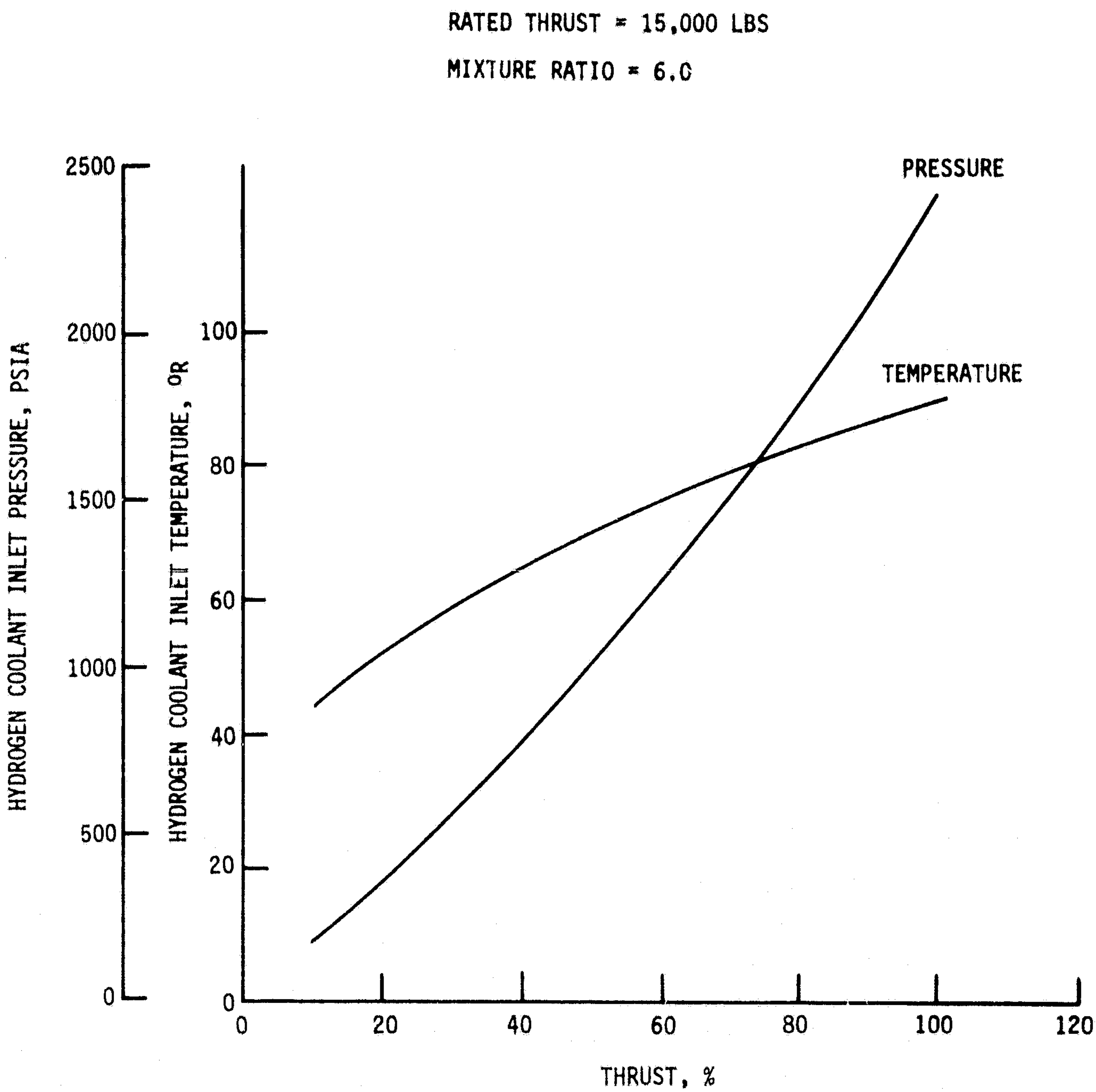


Figure 1. Estimated Coolant Jacket Propellant Inlet Conditions During Throttling

TABLE II

CHAMBER LOW THRUST ANALYSES

Thrust Klbf	Coolant T_{in}, °R	Coolant ΔT_b, °F	Coolant ΔP, psi	Wall Temperature, °F		$T_{wg}-T_{NI}$, °F		300 Cycle $T_{wg}-T_{Ni}$, °F		Stability Number
				Inlet	Outlet	Inlet	Outlet	Inlet	Outlet	
15	90	407	93	17	842	283	659	860	665	85
4	57	544	33	237	671	116	291	690	580	-7
3	52	581	26	330	663	92	238	640	560	-10
2	47	631	19	425	660	66	178	590	518	-11

differential between the gas-side surface and the electroformed nickel closure, because it controls thermal strain and thus, cycle life. Table II indicates this differential is significantly lower at reduced thrust at all chamber locations due to the lower heat fluxes and excellent thermal conductance through the lands to the nickel outer wall. The allowable differentials for 300 cycles are included on Table II; note that the inlet region is overcooled at full thrust.

The stability number is shown in the last column of Table II. Theoretically, if this number is greater than zero, stability is predicted. However, comparison of the predictions of Ref. 2 with test data indicate a model bias, such that the inception of instability was generally observed at numbers less than -15. Based on this bias, the low thrust cases of Table II are seen to be stable by a small margin.

2. Nozzle Analyses

A nozzle tube bundle was also designed in previous Phase A study efforts. In this design, 15 percent of the cold hydrogen is used in a two-pass A286 tube bundle. The number of tubes (326) was selected to provide a cycle life of 300 cycles. At all thrust levels considered in this analysis, the coolant inlet exhibits the highest thermal strain as well as the maximum wall temperature.

Results for low thrust operation are given in Table III and compared with full thrust operation. Maximum wall temperature and the maximum wall temperature differential (T_{wg} - T_b) controlling thermal strain increase as thrust is reduced; therefore, cycle life is reduced. This results from the lower estimated coolant inlet temperatures (Figure 1) associated with reduced thrust and the corresponding degradation in coolant heat transfer coefficient noted previously. To illustrate this result, alternate low thrust analyses were conducted with a fixed inlet temperature equal to that at full thrust (90°R);

TABLE III

NOZZLE LOW-THRUST ANALYSES

Thrust Klbf	Coolant ΔT_b, °F	Wall Temperature, °F			$T_{wg}-T_b$, °F		
		Inlet	Turn	Outlet	Inlet	Turn	Outlet
15	661	1011	722	826	1381	777	535
4	812	1244	689	831	1647	708	422
3	-	1285	-	-	1693	-	-
2	866	1319	668	838	1732	668	386

in this case the wall temperatures at the inlet for reduced thrust were well below that for full thrust.

It should be noted that the nozzle design can be modified to accommodate low thrust operation without penalizing full thrust operation. The nozzle is cooled in parallel with the chamber and must be orificed because it has a much lower pressure drop than the chamber. By overcooling at full thrust, the desired cycle life can be obtained at low thrust. The increased pressure drop required merely reduces the orifice pressure drop.

3. Thermal Analysis Conclusions

Operation of the baseline 15K OTV chamber liner and nozzle tube bundle at thrust levels as low as 2K was determined to be feasible. For both components, the wall temperature at the coolant inlet was found to increase at low thrust because of poor coolant heat transfer at the reduced coolant inlet temperatures. This results in a degradation of cycle life for the nozzle, but not the chamber. However, the nozzle can be redesigned to provide the desired cycle life without penalizing the full thrust design. A preliminary flow stability analysis of the chamber indicates it is marginally stable at 2K.

With regard to the near critical or subcritical pressure operation, the chamber pressure for 1K and 2K lbf are about 80 and 160 psia, respectively. Figure 1 shows the LH_2 coolant jacket inlet temperature is between 40 and 47°R. The critical pressure of LH_2 is 188 psia and its critical temperature is 59°R. Therefore, if the H_2 critical conditions occur within a high heat flux section of the chamber, it will be extremely difficult to reliably cool it for long life capability due to the wide variations in coolant heat capacity (C_p), density (ρ), thermal conductivity (k) and viscosity (μ) near the critical point. Some design flexibility exists such as installing an orifice in the fuel line between the cooled chamber and fuel injector manifold to keep the H_2 pressure above the critical pressure in the coolant passages. This places the burden upon the fuel pump but the engine can be power balanced (see Section III,E) because of the higher turbine inlet temperatures at low thrust which are shown on Figure 2. The orifice would constitute part of the low-thrust "kit" design modification.

C. TURBOPUMP ANALYSES

Analyses were also undertaken to predict the performance of the OTV turbopumps to operate at low engine thrust levels. The purpose of this subtask was to examine the low-thrust off-design performance of the pumps and turbines and identity operating regions where actual and potential instabilities can exist. Low-thrust off-design performance was based on pump and turbine designs which conform to the performance requirements of the OTV engine operating at 100% thrust and design mixture ratio. Tables IV and V contain the operating specifications for the turbopumps and engine at both the nominal MR of 6.0 and the off-design mixture ratio operating point of 7.0.

1. Flow Schematic

Figure 3 is the flow schematic showing the significant groupings of the turbomachinery components which was analyzed. On the fuel side, liquid hydrogen (LH_2) flows from the tankage through the boost pump to the main stage. After the main stage, the fuel flows through the thrust chamber and absorbs heat, changing phase from liquid to gaseous. A diverter valve allows a portion of the flow to bypass both the fuel and oxidizer turbines. Consequently, this valve controls engine thrust level. The second valve in the system allows flow to bypass the fuel turbine. This second valve adjusts the power of the fuel turbine and causes adjustments in engine mixture ratio.

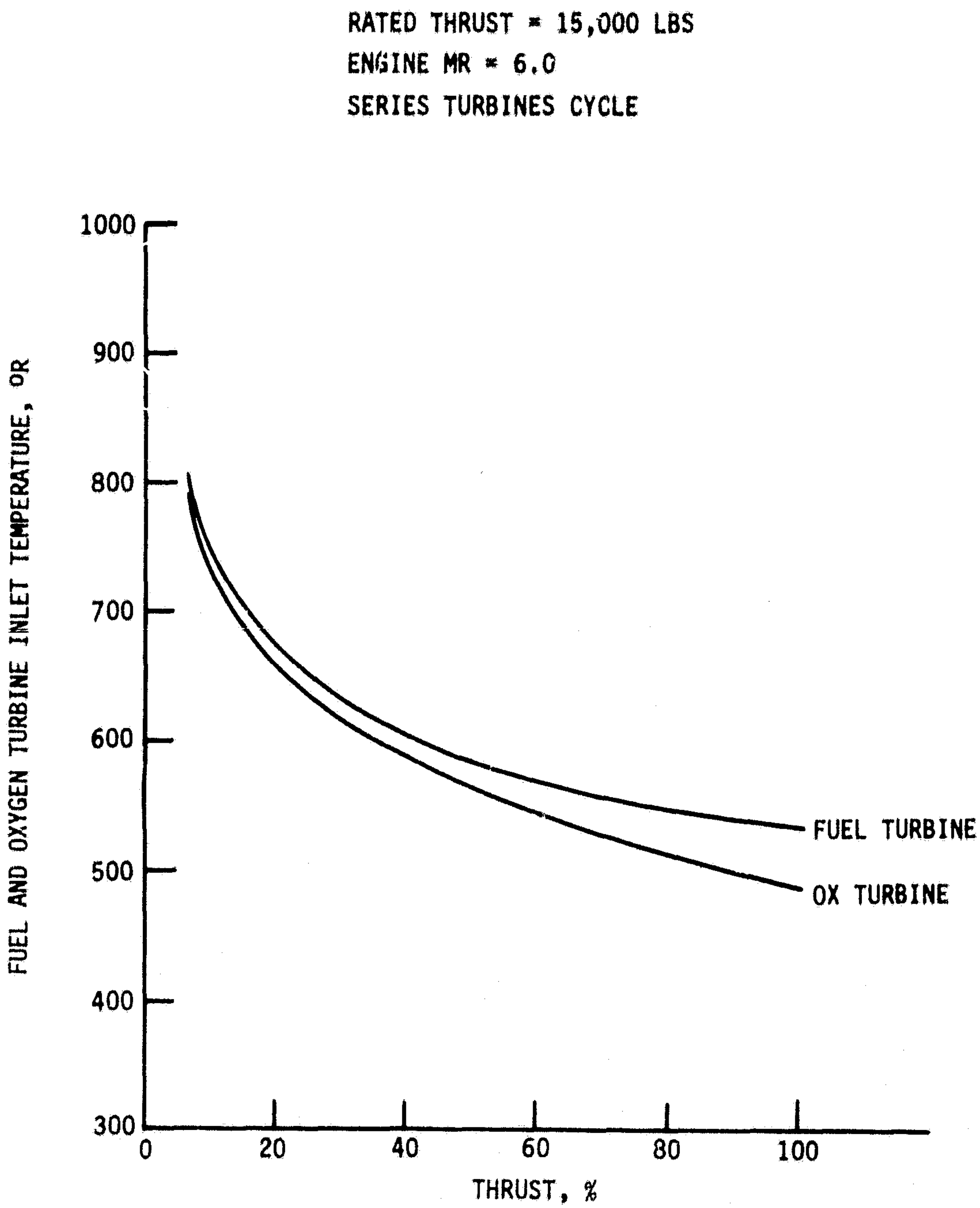


Figure 2. Fuel and Oxidizer Turbine Inlet Temperature vs % Thrust

TABLE IV

MAIN PUMP DESIGN PARAMETERS

Rated Thrust 15,000 lb

	ENGINE MIXTURE RATIO 6.0		ENGINE MIXTURE RATIO 7.0	
	LOX	LH_2	LOX	LH_2
INLET TEMPERATURE, °R	162.7	37.8	162.7	37.8
INLET PRESSURE, PSIA	46.6	51.0	46.6	51.0
VAPOR PRESSURE, PSIA	15.0	18.0	15.0	18.0
NET POSITIVE SUCTION HEAD, FT	64.1	1075	64.1	1075
VOLUMETRIC FLOWRATE, GPM	170.3	456.0	174.9	401.1
SUCTION SPECIFIC SPEED, $(RPM)(GPM)^{1/2}/(FT)^{3/4}$	20000	10240	20000	9226
SPEED, RPM	34720	90000	34260	86490
DISCHARGE PRESSURE, PSIA	1487	2473	1463	2290
HEAD RISE, FT	2921	78910	2872	72945
NUMBER OF STAGES	1	3	1	3
SPECIFIC SPEED, $(RPM)(GPM)^{1/2}/(FT)^{3/4}$	1140	931	1155	890

TABLE V

MAIN TURBINE DESIGN PARAMETERS

Rated Thrust = 15,000 lb

	ENGINE MIXTURE RATIO			
	6.0		7.0	
	LOX	LH_2	LOX	LH_2
INLET PRESSURE, PSIA	1471	2286	1430	2138
INLET TEMPERATURE, °R	488.5	535	513.3	557
FLOWRATE, LB/SEC	4.22	4.22	3.71	3.71
GAS PROPERTIES				
C_p, SPECIFIC HEAT AT CONSTANT PRESSURE, BTU/LB-°R	3.652	3.652	3.652	3.652
γ, RATIO OF SPECIFIC HEATS	1.395	1.395	1.395	1.395
SHAFT HORSEPOWER	235.6 (1)	1013 (2)	239 (1)	837 (2)
PRESSURE RATIO (TOTAL TO STATIC)	1.109	1.544	1.122	1.485
TURBINE BYPASS FLOWRATE, LB/SEC	0.27 (3)		0.24 (3)	

(1) Includes 5% horsepower penalty for boost pump drive flow.

(2) Includes 3% horsepower penalty for boost pump drive flow.

(3) 6% of total available hydrogen flow.

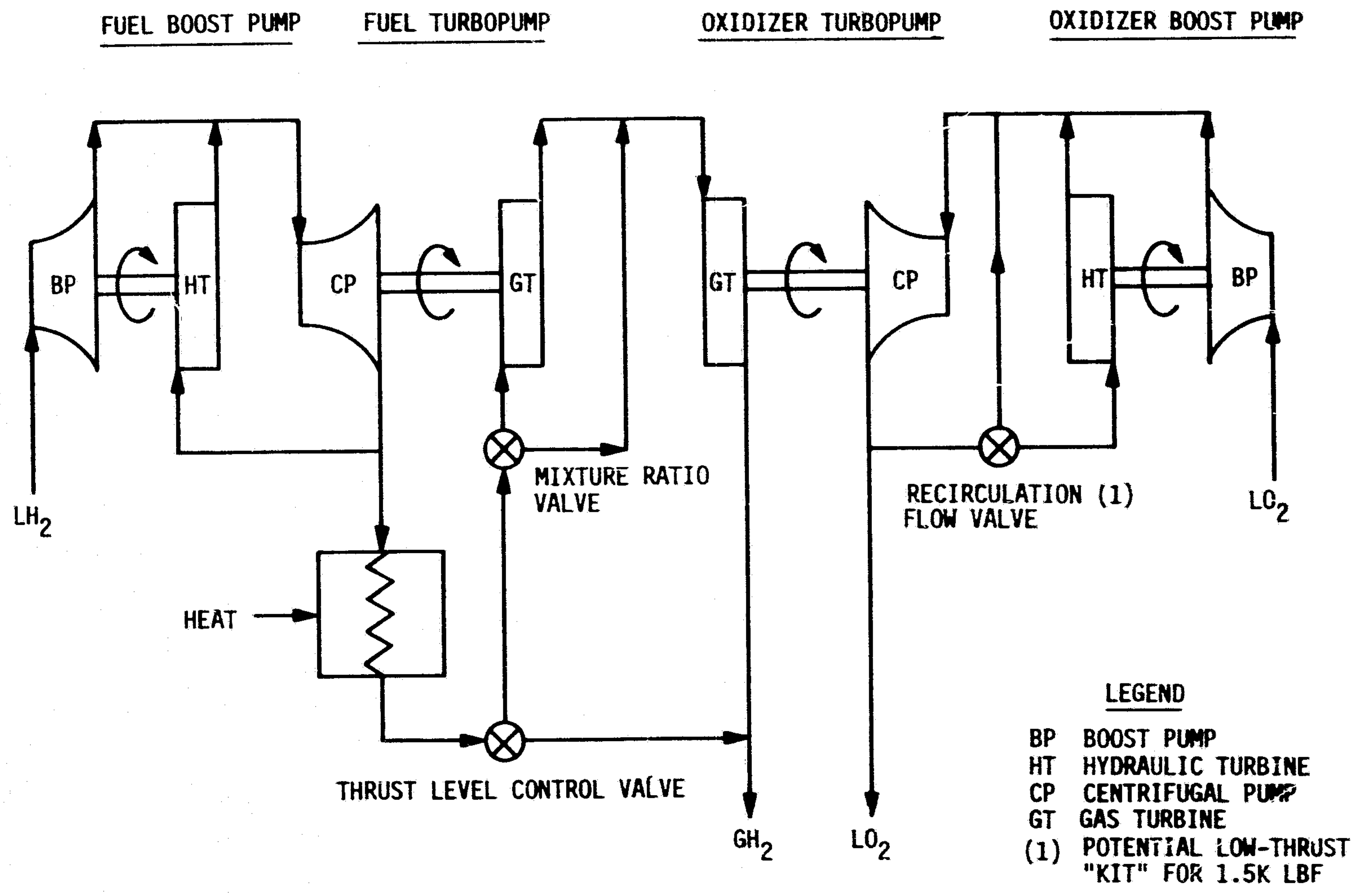


Figure 3. Flow Schematic Showing Turbomachinery Grouping

Exhaust gas from the turbines is subsequently combined with the flow from the turbine bypass valve and delivered to the engine combustion apparatus. Liquid oxygen flows from the tankage through boost and main stage pumps and subsequently to the engine thrust chamber.

Each boost pump is driven by a hydraulic turbine. Flow for each of these turbines comes from the discharge of the respective main stage pumps and is returned to the main stages through recombination with the boost pump discharge.

2. Oxidizer Turbopump Preliminary Design Analysis

The performance of the oxidizer pump is predicted from the design point definition and a preliminary impeller design. Table IV shows the design point parameters and corresponding values for the OTV oxidizer pump. Table VI reflects the preliminary main oxidizer pump preliminary design characteristics resulting from the design point operating specification and the criteria of Ref. 3.

Based on the information on Table IV and the above calculated values, the prediction of pump characteristics in terms of head rise, flow and shaft speed were completed. In addition, overall pump efficiency with respect to flow was also predicted and is shown in Section III.E.

Figure 4 reflects the oxidizer discharge head with respect to flow and shaft speed. Also shown in this figure are the required engine characteristic for operation at a mixture ratio of six and the stable operating limit. As shown, the engine operating characteristic is always in the region of negative slope for all of the shaft speed curves shown. Accordingly, the oxidizer pump has stable operating characteristics from 10,000 to 34,720 rpm. This speed range corresponds to an engine thrust level from 13.3 to 100% of design thrust.

TABLE VI

OTV OXIDIZER PUMP PRELIMINARY DESIGN CHARACTERISTICS

Shaft Speed	34720 rpm
Impeller Inlet Diameter	1.47 in.
Impeller Discharge Diameter	2.94 in.
Inlet Meridional Velocity	47.9 ft/sec
Discharge Meridional Velocity	67.3 ft/sec
Number of Blades	4 partial, 4 full
Blade Discharge Angle	28.5°
Overall Efficiency	63.6%
Hydraulic Efficiency	77.3%
Head Coefficient	.472
Number of Stages	1

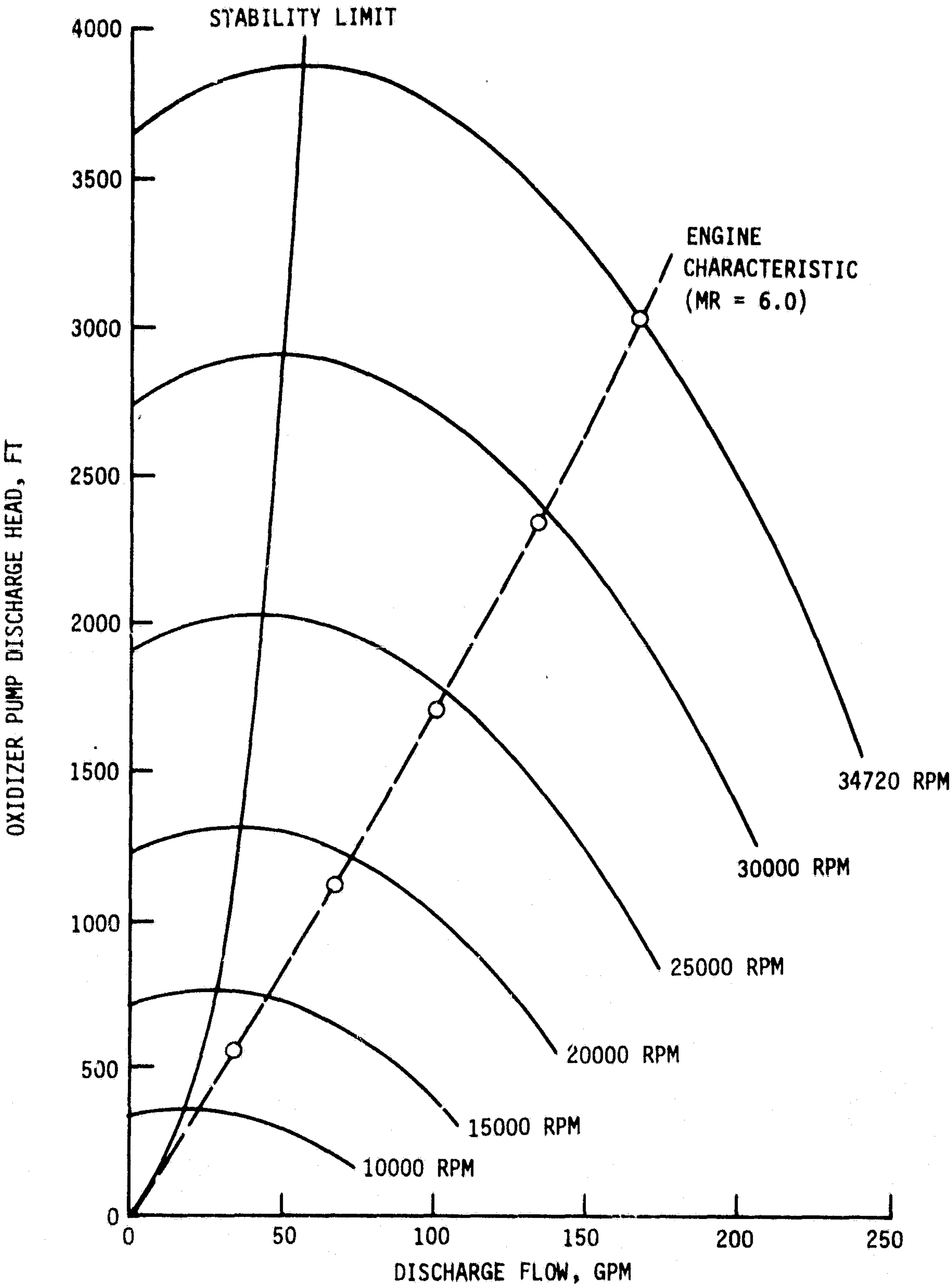


Figure 4. Oxidizer Pump Characteristics

Figure 5 illustrates the required shaft power of the oxidizer turbine in terms of pump mass flowrate. It was assumed that the boost pump recirculating turbine drive flow would absorb 5% of the power required by the main stage pump. As shown in the figure, the oxidizer turbine power varies between 236 HP at 34,720 rpm and 5 HP at 10,000 rpm. These two points correspond to engine thrust levels of 100% and 13.3% respectively.

Based on the power requirements of the turbine and predicted inlet temperatures, pressures and mass flow, a preliminary turbine design was formed. This turbine has single stage with a mean diameter of 5.545 inches. Figure 6 is a sketch of the flow passage and identifies the pertinent design point parameters and values.

Generalized performance of the oxidizer turbine is reflected in Figure 7. The top curve shows the estimated efficiency with respect to the blade speed ratio. At the bottom, the variation of normalized weight flow is shown for corresponding turbine pressure ratios.

3. Fuel Turbopump Preliminary Design Analysis

As with the oxidizer turbopump, the fuel turbopump preliminary design is based on the design point parameters in Tables IV and V and the criteria of References 3 and 4. Table VII shows the results of the preliminary design calculations.

Figure 8 shows the fuel pump discharge head-capacity relationship for shaft speeds varying between 90,000 and 20,000 rpm. Figure 9 shows the required power of the fuel turbine with respect to the fuel pump flowrate for various shaft speeds. As illustrated on Figure 8, the engine characteristic is located in the negatively sloped region of the discharge head-

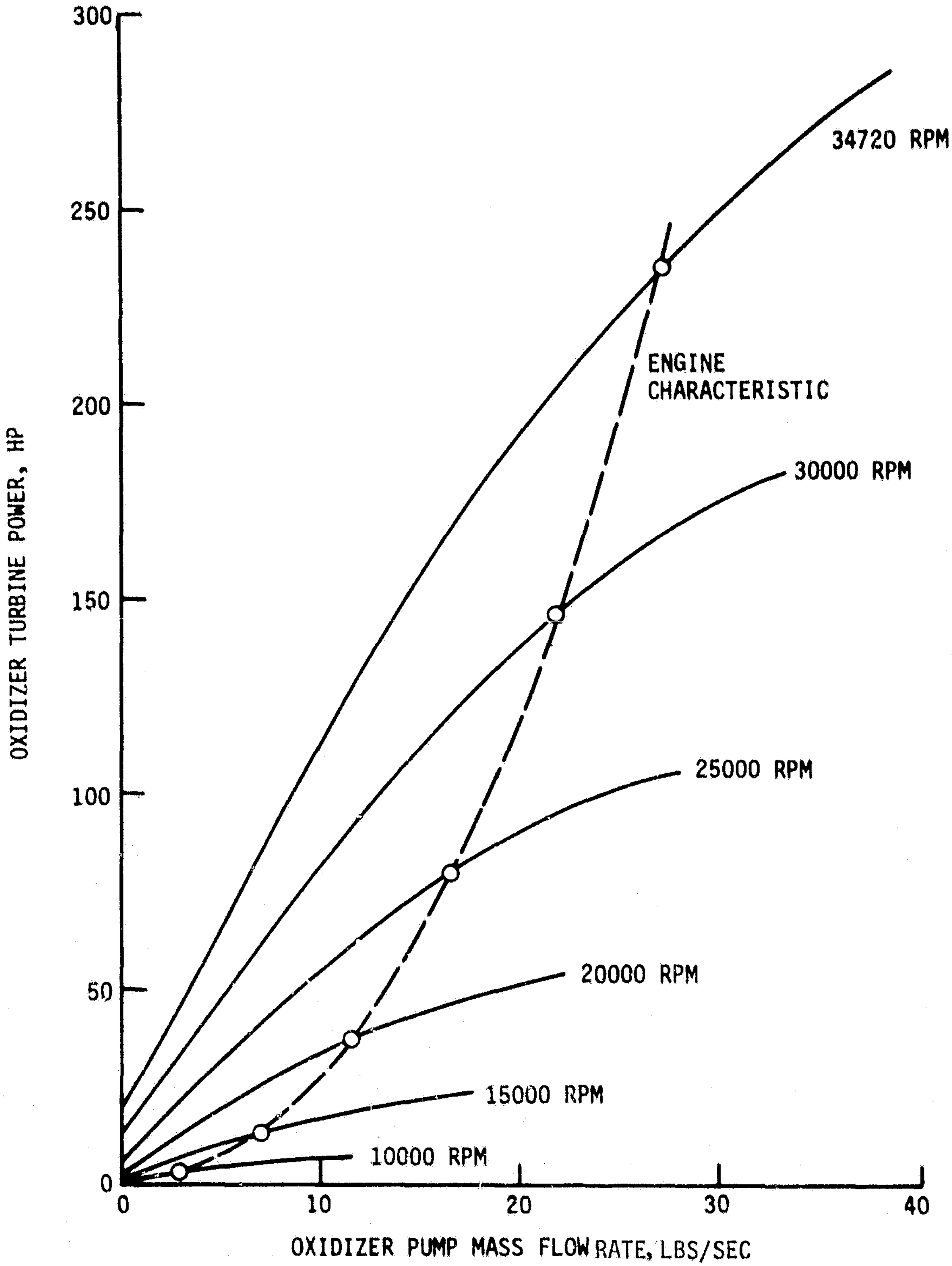


Figure 5. Oxidizer Turbine Power With Respect to Oxidizer Pump Flowrate

DESIGN POINT

P_o = 1471 PSIA

T_o = 488.5 °R

P_e = 1326 PSIA

$\dot{W}$ = 4.22 LB/SEC

N = 34,720 RPM

FLUID: GH_2

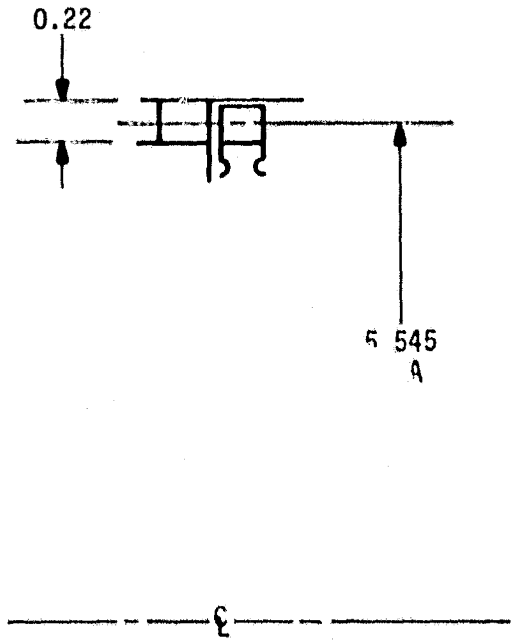


Figure 6. Oxygen Pump Turbine Design Characteristics

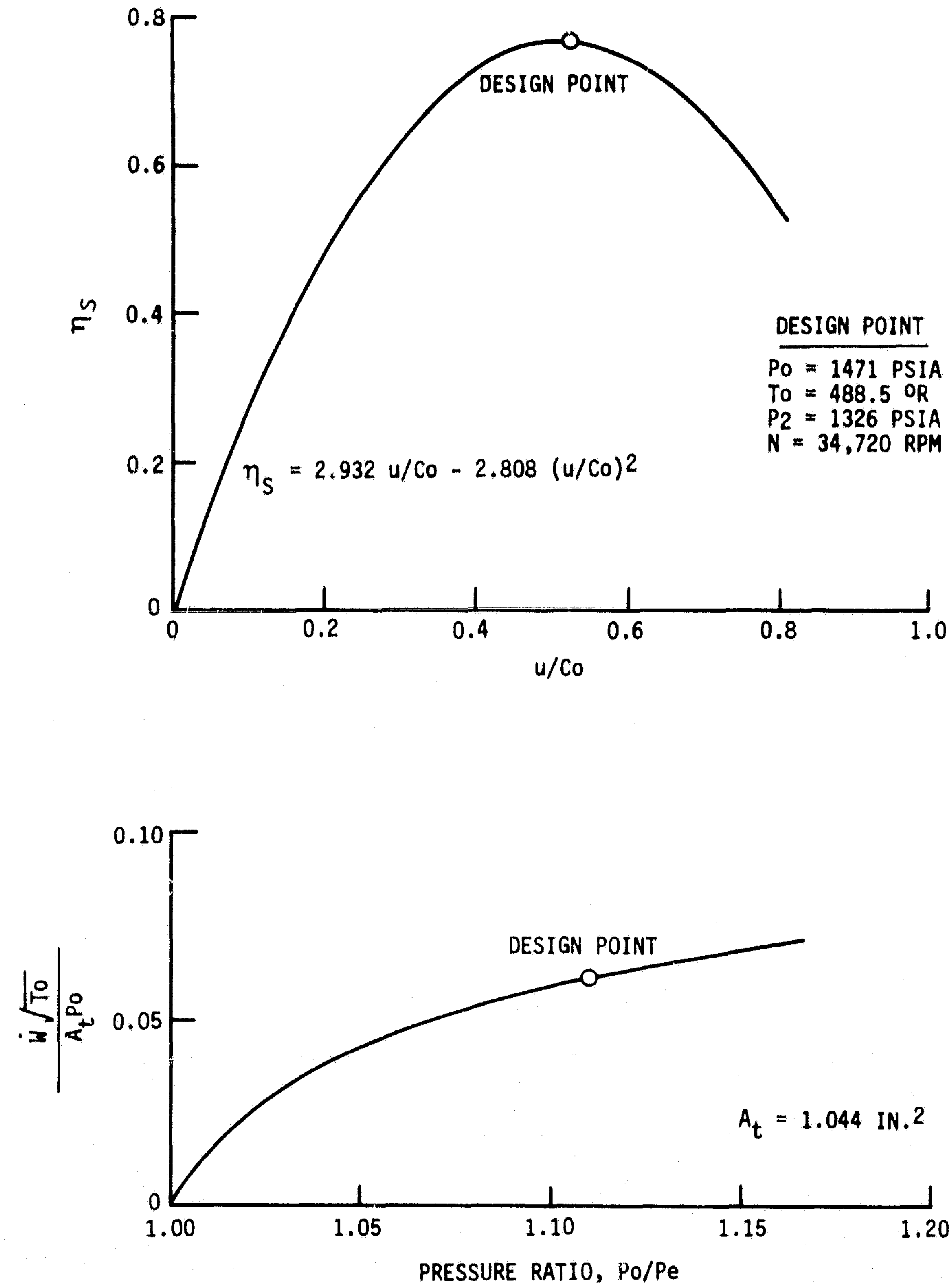


Figure 7. LOX Turbine Performance

TABLE VII

OTV FUEL PUMP PRELIMINARY DESIGN

Shaft Speed	90000 rpm
Impeller Inlet Diameter	1.68 in.
Impeller Discharge Diameter	3.36 in.
Inlet Meridional Velocity	263.1 ft/sec
Discharge Meridional Velocity	289.4 ft/sec
Number of Blades	6 full, 6 partial
Blade Discharge Angle	40.3°
Overall Efficiency	65.5%
Hydraulic Efficiency	78.7%
Stage Head Coefficient	0.486
Number of States	3

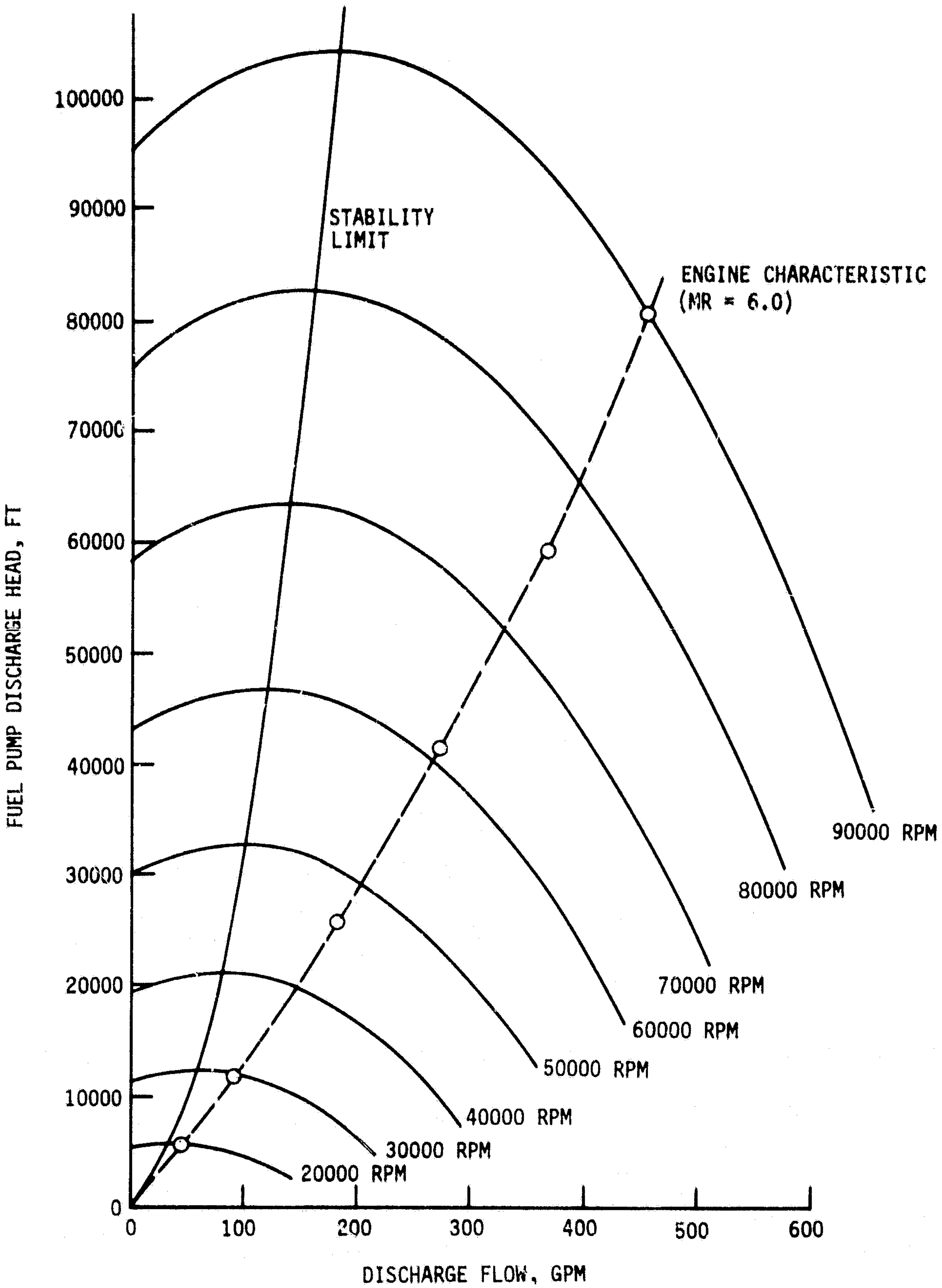


Figure 8. OTV Fuel Pump Characteristics

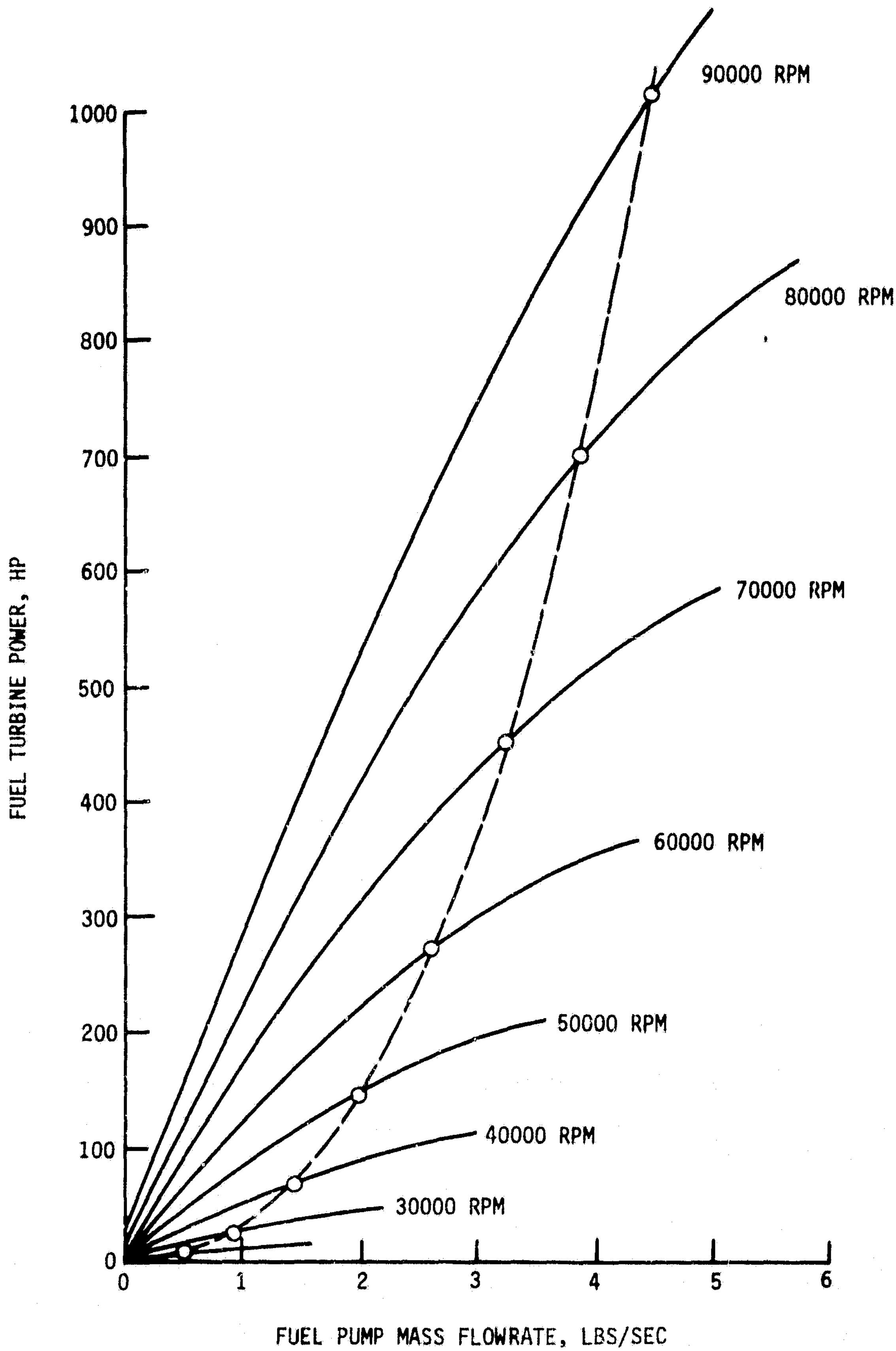


Figure 9. Fuel Turbine Power With Respect to Fuel Pump Flowrate

capacity curves for the various shaft speeds. Consequently the pump should operate between 10 and 100% of full thrust without any instabilities.

A two-stage turbine was selected to drive the fuel pump. Figure 10 illustrates the basic flow passage configuration associated with this turbine. Figure 11 shows the performance of this turbine in terms of efficiency, blade speed ratio, normalized weight flow and pressure ratio.

4. Overall Turbopump Performance

Turbopump efficiency is defined as the product of pump and turbine efficiencies. Figure 12 illustrates this composite efficiency for both the fuel and oxidizer turbopumps. Also shown is the minimum required efficiency which was calculated by assuming all of the fuel flow, including that bypassing the turbines through the thrust level control valve, was available for energy conversion. The figure shows that there is sufficient margin between the preliminary design estimate and the minimum requirement so that system power balancing at the low-thrust levels should not present a problem. This was verified by the cycle analysis results presented in Section III,E.

5. Turbopump Analyses Conclusions

The study results indicate that the turbopumps will operate without instabilities to 13.3% (2K) of rated thrust for the oxidizer turbopump and 10% (1.5K) of rated thrust for the fuel turbopump without modification. This operating range for the pumps is shown on Table VIII.

The study results also show that there is sufficient cycle power balance margin at low-thrust so that pump flows can be increased at low-thrust to avoid instability at even lower thrust levels. This can be accomplished by adding a recirculation line and valve between the main pump

DESIGN POINT

P_0 = 2286 PSIA

T_0 = 535°R

P_e = 1481 PSIA

$\dot{W}$ = 4.22 LB/SEC

N = 90,000 RPM

FLUID - GH_2

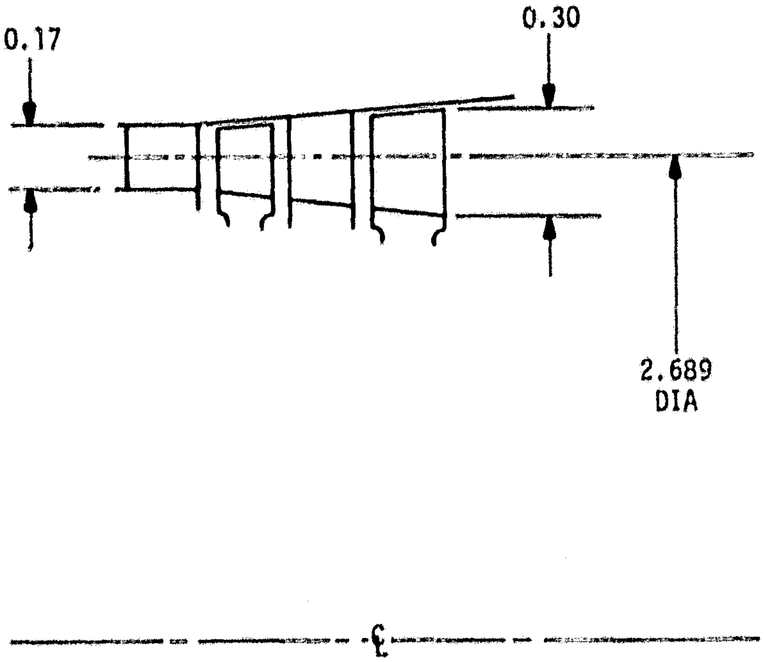


Figure 10. Hydrogen Pump Turbine Design Characteristics

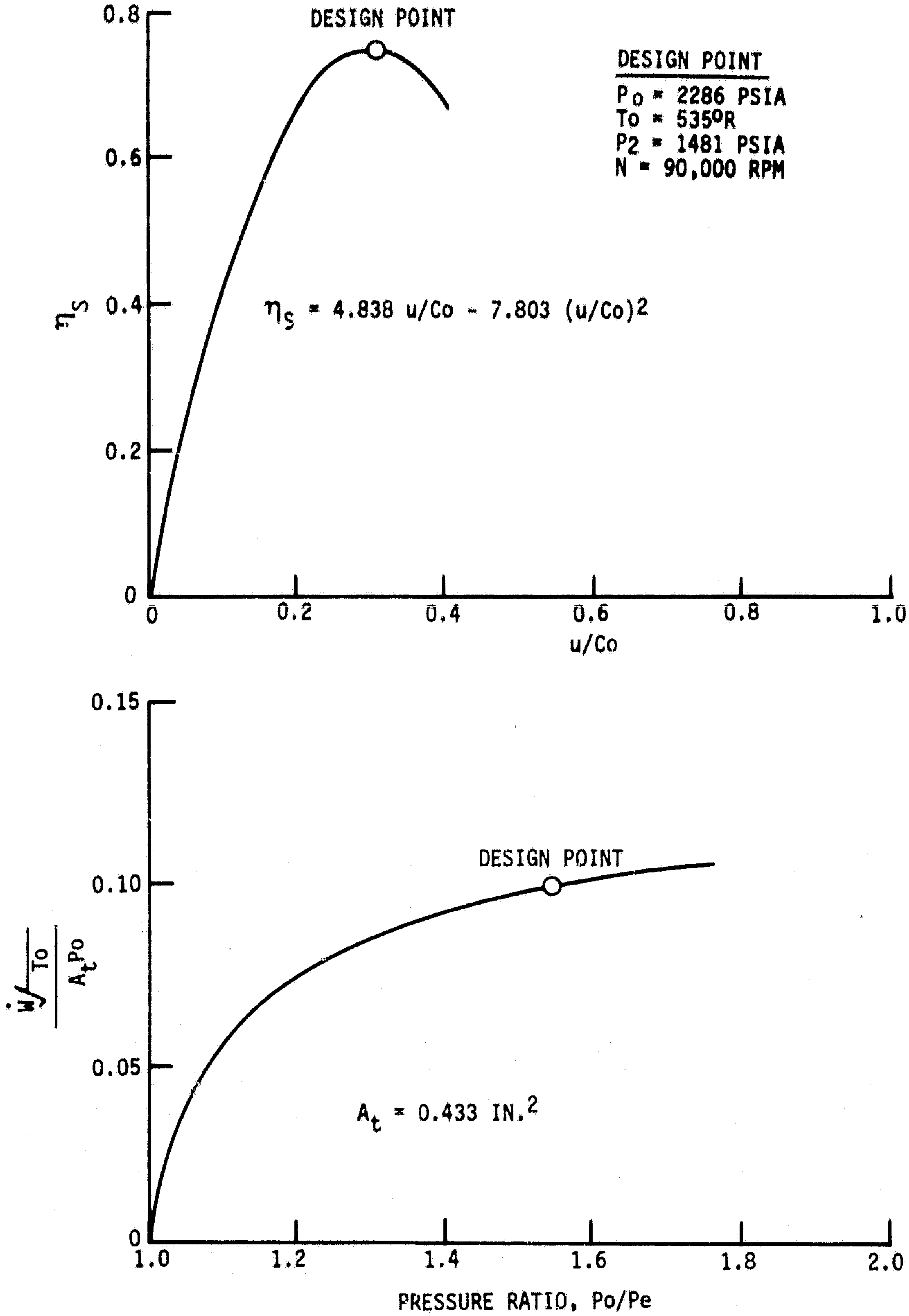
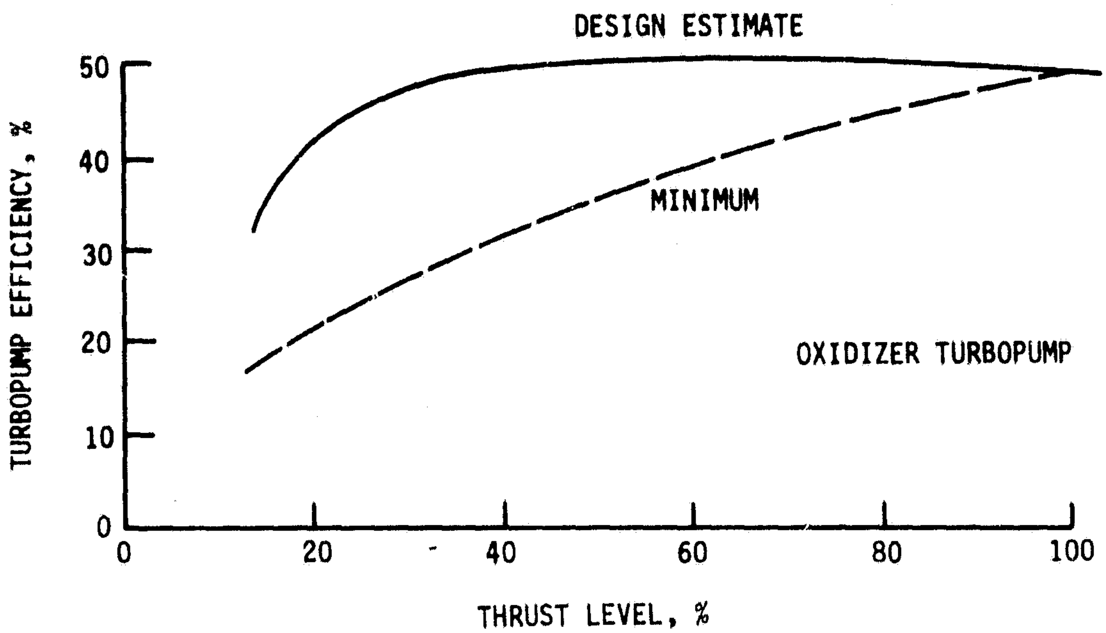


Figure 11. LH_2 Turbine Performance



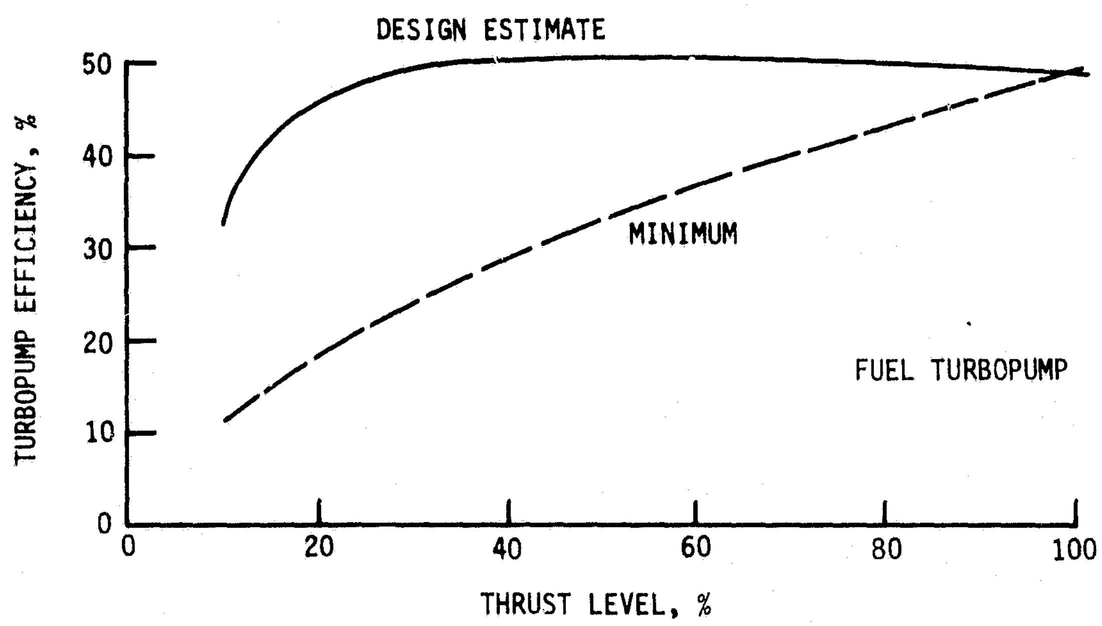


Figure 12. Comparison of Estimated and Minimum Required Turbopump Efficiencies

TABLE VIII

RANGE OF TURBOPUMP VARIABLES

	OXIDIZER PUMP		FUEL PUMP	
	MINIMUM*	MAXIMUM	MINIMUM	MAXIMUM
Thrust Level (%)	13.3	100.0	10.0	100.0
Flowrate (gpm)	23	170	47	456
Shaft Speed (rpm)	10000	34720	20000	90000
Discharge Heat (ft)	352	3015	5794	80572

*NO MODIFICATIONS

and boost pump as shown on Figure 3 for the oxidizer pumping system. This oxidizer recirculation "kit" would get the thrust level to a value at least as low as the fuel side and "kitting" the fuel side might make even lower thrust operation feasible. The concern here is that the recirculation flow may get too hot to dump back into the main pump inlet. More detailed analyses and/or experimental data is required to firmly fix the minimum thrust level.

D. LOW-THRUST CONTROLS

An active control system was selected during Task I, Advanced Expander Cycle Engine Optimization, conducted for this study. The cycle schematic, showing the location of the controls, is presented on Figure 13. These controls were also analyzed for low-thrust operation. It has been concluded that new valves or a valve "kit" are not required. The rationale for this conclusion is discussed in the following paragraphs.

Because the engine is required to operate in the pumped-idle mode, the turbine bypass valve used for the full-thrust mode (Valve #3, Figure 13) can also be modulated to control the required steady-state low thrust level. Equal percentage flow characteristics can be designed into the valve so that a given percent change in opening at the pumped-idle thrust level would have the same percentage effect on flow as it would have at the full-thrust level. This same design philosophy can be applied to the turbine flow control valve (Valve #4 of Figure 13) which is utilized to control engine mixture ratio. These flow control valves can be designed to accommodate any selected low thrust level.

No functional problems are anticipated for the remaining valves in the engine system. If engine weight is a problem, the full-thrust engine valves could be replaced by a smaller set of valves designed to operate at a discrete low thrust point. This would save about 25 lbs of engine weight but is probably not worth the additional cost.

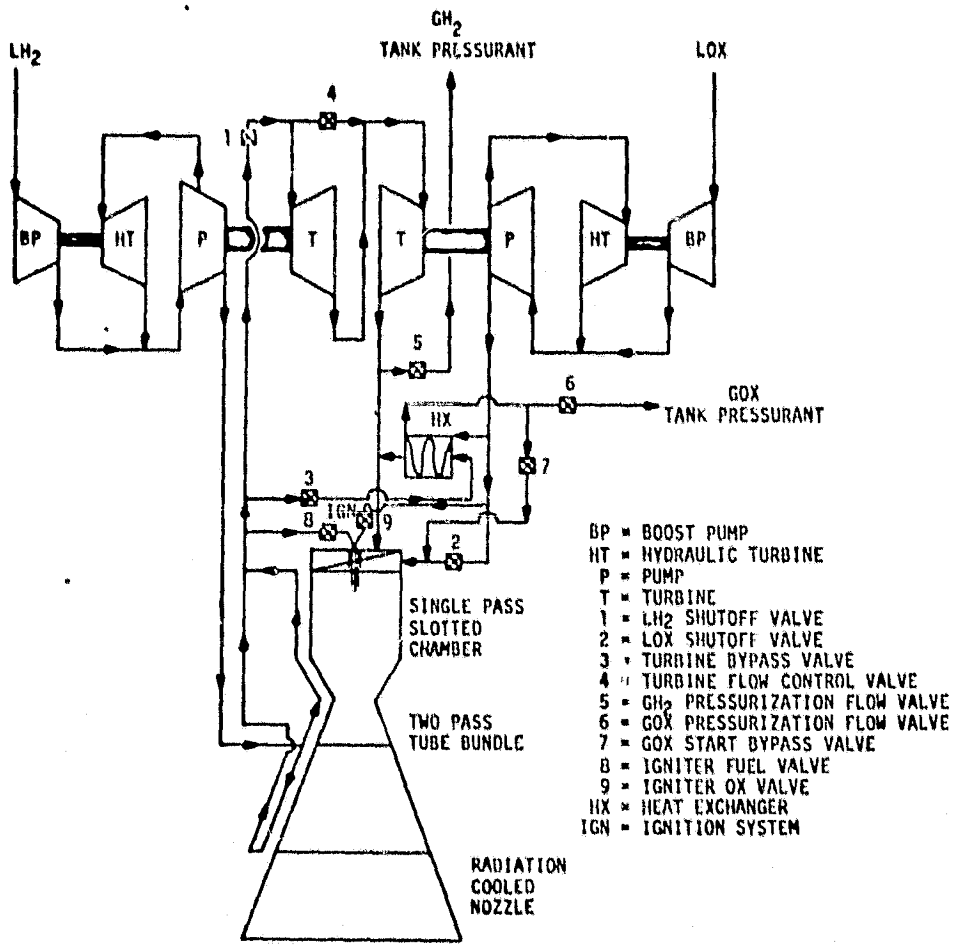


Figure 13. Advanced Expander Cycle Engine Flow Schematic

E. ENGINE SYSTEM ANALYSES

The objectives of this subtask were to establish engine operating conditions at low-thrust operating points and to determine engine power balance limits, if any. The thermal, turbopump and injector design analyses results were used to evaluate the expander cycle power balance over a thrust range of 7K lbf to 1K lbf or, 50% to 6.7% of rated thrust, respectively.

Coolant jacket pressure drop data used in this analysis is shown on Figure 14. The turbine inlet temperatures were previously shown on Figure 2. Pump design and off-design efficiency data are shown on Figures 15 and 16 for the main oxygen and hydrogen pumps, respectively. The turbine performance curves were shown on Figure 7 for the oxidizer turbine and Figure 11 for the hydrogen turbine.

The injector pressure drop criteria used in the analysis are shown on Figure 17. These criteria are based upon the results of the injector modification analysis. Below 50% of rated thrust, the oxidizer elements were assumed to be sized to meet a minimum injector stiffness ($\Delta P/P_c$) requirements of 15% until the injection velocity reached 40 fps at about 8% thrust. Two typical orifice sizes in this range are shown on the figure. An orifice size of .050" was selected for the 1K lbf operation (6.7% thrust).

The results of the engine power balance analysis are shown on Figures 18 and 19. Figure 18 shows the pump discharge pressure requirements as a function of thrust. The hydrogen pump discharge pressure is purposely kept relatively high in the very low thrust region to maintain the coolant jacket exit pressure above the critical pressure of hydrogen. This avoids the problems associated to two-phase cooling.

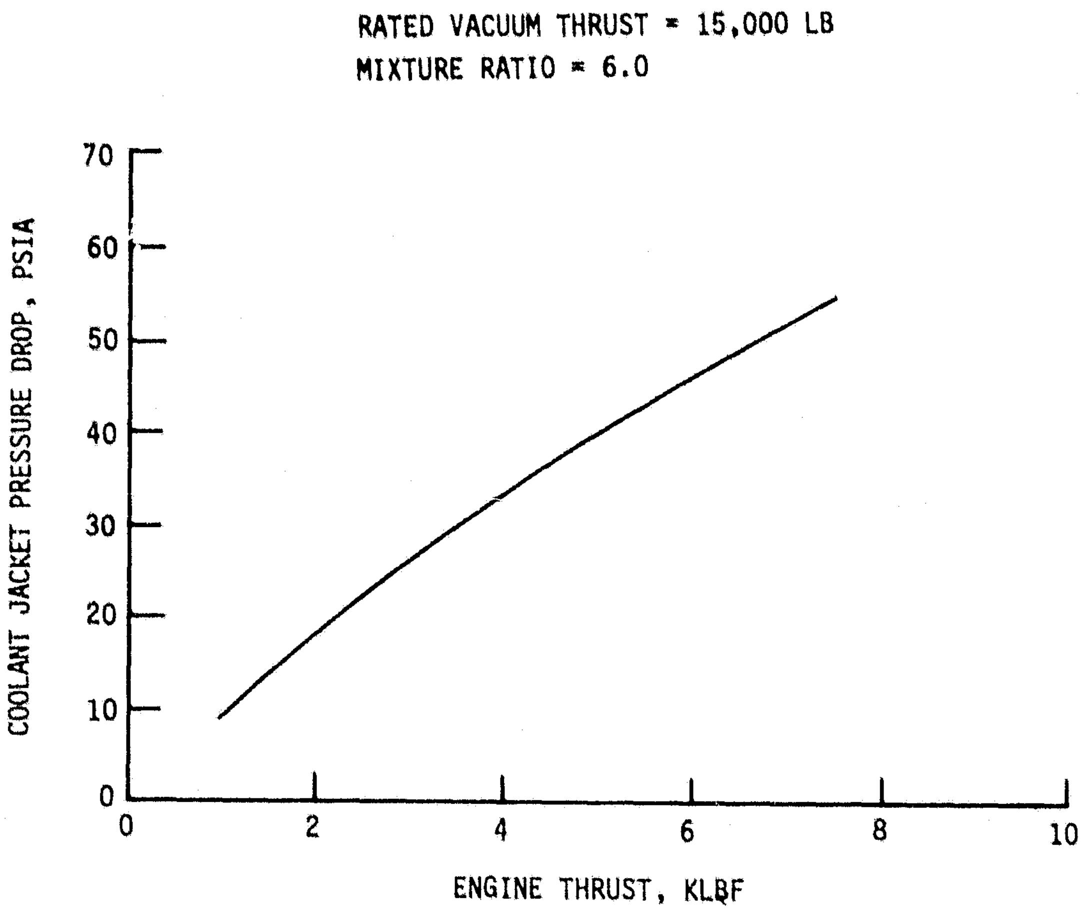


Figure 14. Coolant Jacket Pressure Drop vs Thrust

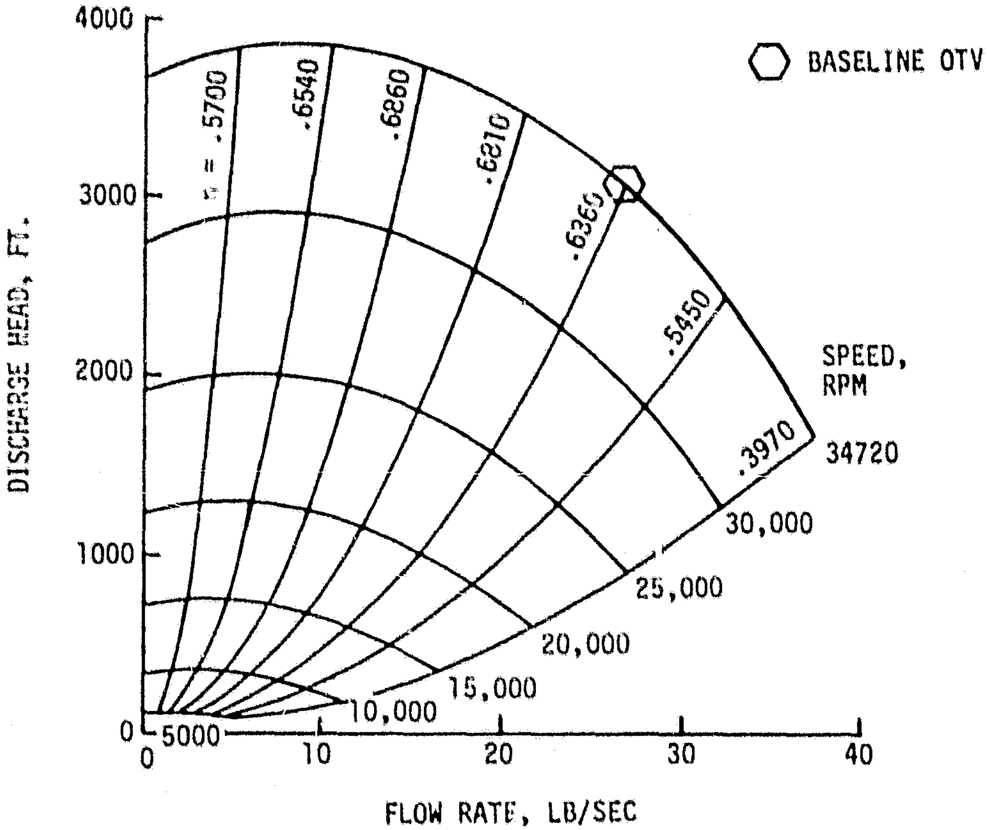


Figure 15. Main Oxygen Pump Performance Map

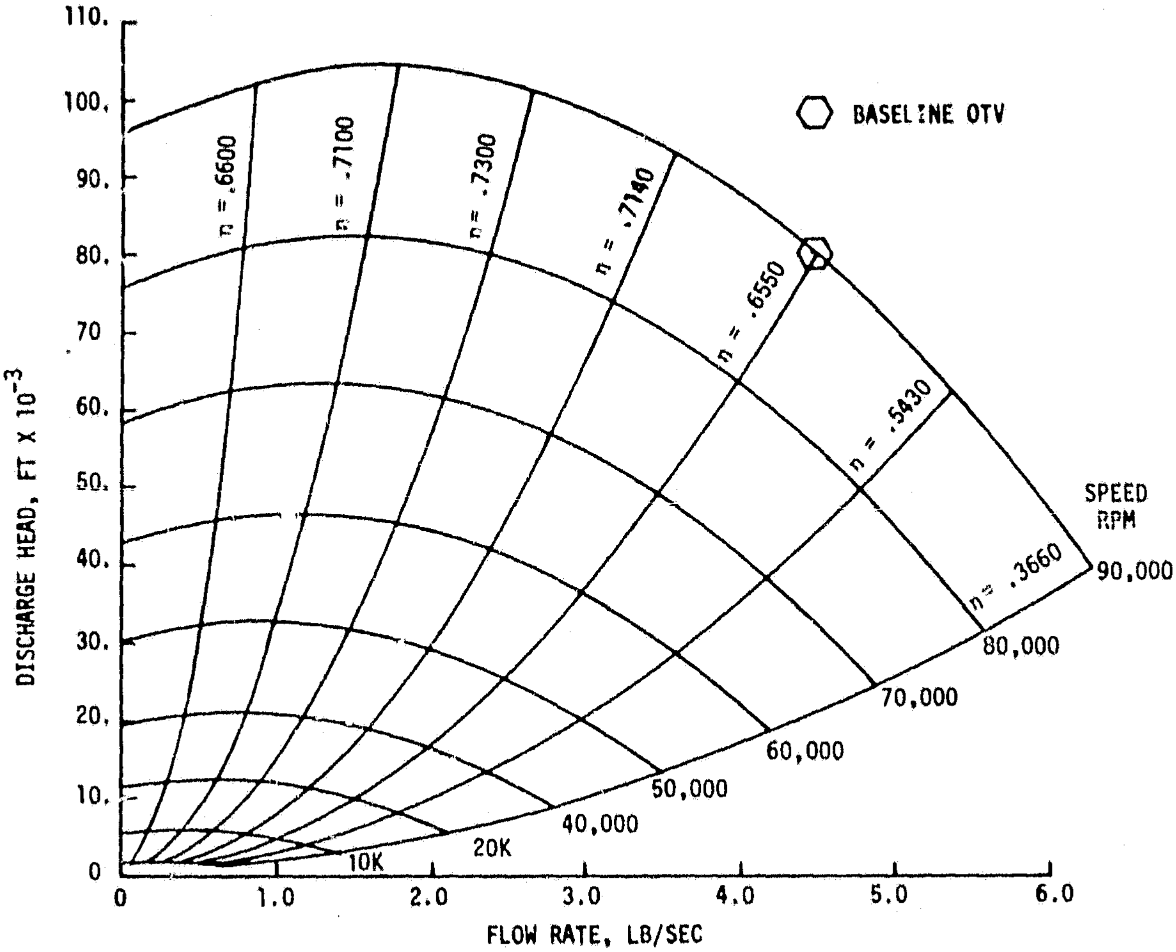


Figure 16. Main Hydrogen Pump Performance Map

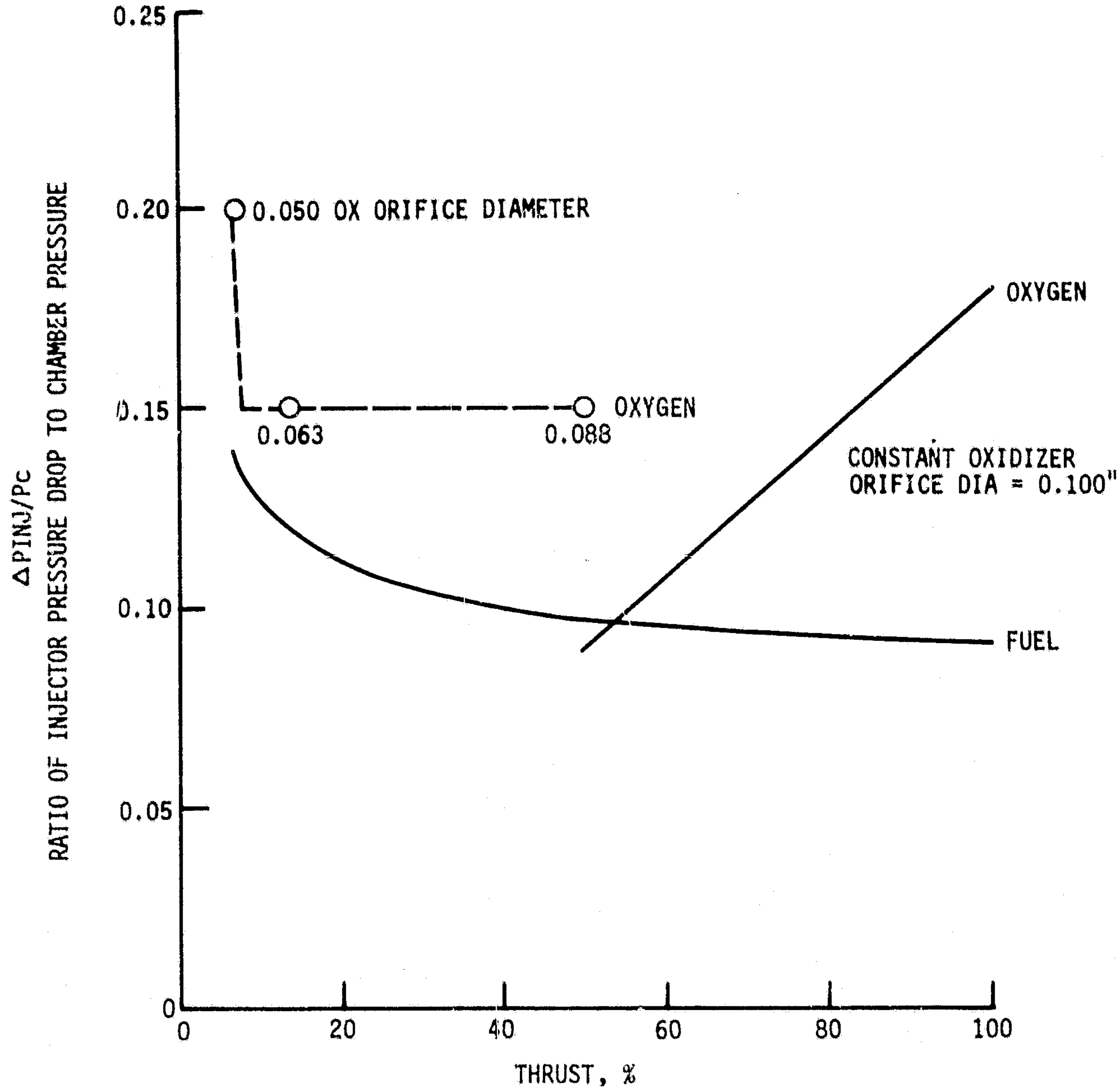


Figure 17. Injector Pressure Drop Criteria

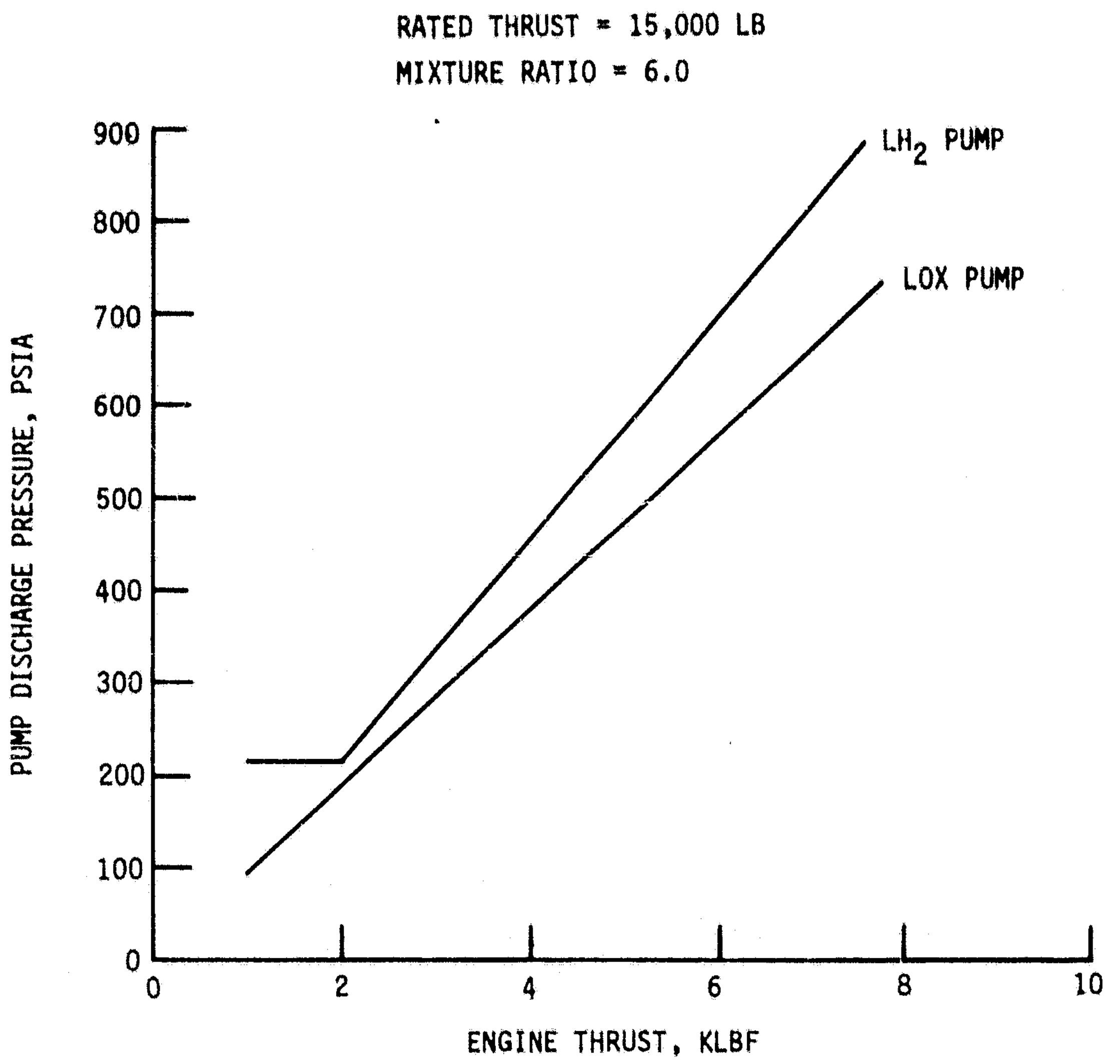


Figure 18. Pump Discharge Pressure Requirements at Low-Thrust

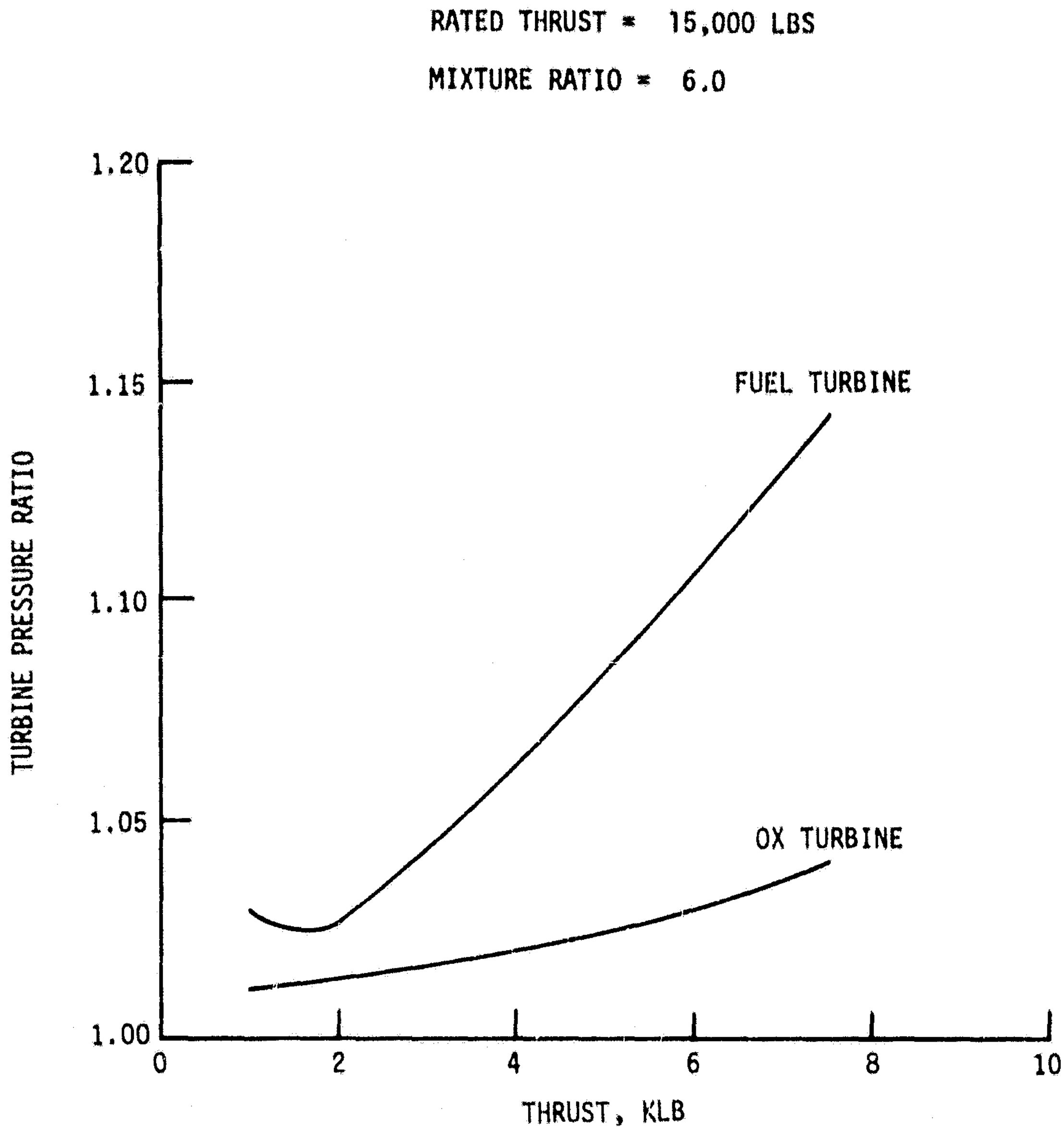


Figure 19. Turbine Pressure Ratio Requirements at Low-Thrust

The turbine pressure ratio requirements as a function of thrust are shown on Figure 19. The very low pressure ratios are evidence of the ease with which the cycle is power balanced at low-thrust.

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