P54

NASA CR 187215
RI/RD 90-180

FINAL REPORT
ORBIT TRANSFER ROCKET ENGINE TECHNOLOGY PROGRAM

ADVANCED ENGINE STUDY
TASK D.6

Prepared By:

C. M. ERICKSON
ROCKWELL INTERNATIONAL CORPORATION
Rocketdyne Division

Prepared For:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

February 1992

NASA-Lewis Research Center

Contract NAS3-23773

G. P. Richter, Program Manager

ROCKETDYNE DIVISION OF ROCKWELL INTERNATIONAL CORPORATION
6633 Canoga Avenue; Canoga Park, CA 91303

(NASA-CR-187215) ORBIT TRANSFER ROCKET ENGINE TECHNOLOGY PROGRAM: ADVANCED ENGINE STUDY Final Report (Rockwell International Corp.) 84 p N92-22592

CSCL 21H Unclas

G3/20 0083747

1. Report No. NASA 187215	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle ORBIT TRANSFER ROCKET ENGINE TECHNOLOGY PROGRAM-FINAL REPORT, ADVANCED ENGINE STUDY, TASK D.6		5. Report Date February 28, 1992 6. Performing Organization Code
7. Author (s) C. M. ERICKSON		8. Performing Organization Report No. RI/RD 90-180
9. Performing Organization Name and Address ROCKETDYNE DIVISION, ROCKWELL INTERNATIONAL 6633 Canoga Avenue Canoga Park, CA 91303		10. Work Unit No. RTOP 506-42-21, TASK YOS2582 11. Contract or Grant No. NAS3-23773
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS & SPACE ADMINISTRATION Washington, DC 20546		13. Type of Report and Period Covered Final report 14. Sponsoring Agency Code

15. Supplementary Notes

Program Manager: G. P. Richter, NASA-Lewis Research Center; Cleveland, OH

16. Abstract

In Task D.6 of the Advanced Engine Study, three primary subtasks were accomplished: 1) Design and Parametric Data, 2) Engine Requirement Variation Studies, and 3) Vehicle Study/Engine Study Coordination.

Parametric data were generated for vacuum thrusts ranging from 7500 lbf to 50000 lbf, nozzle expansion ratios from 600 to 1200, and engine mixture ratios from 5:1 to 7:1. the Failure Modes and Effects Analysis (FMEA) driven baseline design generated in Tasks D.4 and D.5 was used as a departure point for these parametric analyses. These data are intended to assist in vehicle definition and trade studies.

In the Engine Requirements Variation Studies, the individual effects of increasing the throttling ratio from 10:1 to 20:1 and requiring the engine to operate at a maximum mixture ratio of 12:1 were determined. Off-design engine balances were generated at these extreme conditions and individual component operating requirements analyzed in detail. Potential problems were identified and possible solutions generated.

In the Vehicle Study/Engine Study coordination subtask, vehicle contractor support was provided as needed, addressing a variety of issues uncovered during vehicle trade studies. This support was primarily provided during Technical Interchange Meetings (TIM) in which Space Exploration Initiative (SEI) studies were addressed.

17. Key words (Suggested by author (s))	18. Distribution Statement
Hydrogen/Oxygen Engine Deep Throttling Hydrogen/Oxygen Technology High Mixture Ratio High Pressure Pumps/combustion Orbit Transfer Vehicle High Area Ratio Nozzles Expander Cycle Engine	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	76	

* For sale by the National Technical Information Service, Springfield, Virginia 22151

CONTENTS

TABLES

FIGURES

FIGURES (Continued)

FOREWORD

The work reported herein was conducted by the Advanced Programs and Engineering personnel of Rocketdyne, a Division of Rockwell International Corporation, under Contract NAS3-23773 from November 1988 to September 1990. G. P. Richter, Lewis Research Center, was the NASA Program Manager. Mr. R. Pauckert was the Rocketdyne Project Manager, and T. Harmon Project Engineer. C. Erickson and A. Martinez were responsible for technical direction of the effort while D. Bhatt made important technical contributions to the program. Secretarial support was provided by D. Senit.

INTRODUCTION

The Advanced Engine Study has been outlined as a four year effort in which the Orbit Transfer Vehicle Engine (OTVE) design is iterated to allow resolution of vehicle/engine integration issues as well as advanced engine performance, operation and maintenance issues. In tasks D.1/D.3 an engine design was developed which was driven by space maintenance requirements and by a Failure Modes and Effects Analysis (FMEA). In Task D.4 this design was updated based on revised vehicle requirements. In addition, a preliminary maintenance plan and a concept for space operable disconnects were developed in that task. In Task D.5 a complete engine layout was prepared for the advanced engine at a thrust level of 7500 lbf.

Task D.6 is an extension of this earlier work generating parametric and operational data using the D.4/D.5 baseline engine design as a departure point. These parametric data are intended to assist in vehicle definition and trade studies. In addition, the most recent requirements for a Space Transfer Engine were incorporated into the analysis.

Objectives

The specific objectives of Task D.6 as defined by five separate subtasks in the statement of work (SOW) are summarized in Table 1. Upon submission of this final report, all subtasks with the exception of Subtask 4 - Vehicle Study / Engine Study Coordination will have been completed. Subtask 4 has been conducted on a level of effort approach with support provided as needed.

Approach

The approach through which Task D.6 was completed is presented in the schedules shown in Figures 1 and 2. The first schedule was originally generated for the work plan required in Subtask 1. Completion of this study was protracted due to a delay in the selection of the thrust level at which the variation studies were conducted. The schedule for Subtask 9, Engine Variability Studies, was generated after the thrust level for these studies was selected by NASA LeRC.

Table 1

Task D.6 Objectives

Subtask 1 - Work Plan:

Generate Task Order Work Plan defining planned activities, schedules, milestones, and resource utilization.

Subtask 2 - Design and Parametric Analysis:

Generate on-design engine parametric data over a thrust range from 7500 lbf to 50000 lbf. These data are to include engine delivered specific impulse, mass, and dimensional envelope. Balance data to be provided at on-design mixture ratio (MR) and at off-design MR's of 5.0 and 7.0. On-design parametric data re also to be provided for a range of nozzle area ratios from the terminus of the regeneratively cooled nozzle section to 1200.

Subtask 3 - Engine Requirement Variation Studies:

Determine the individual effects of increasing the throttling ratio requirement from 10:1 to 20:1 and requiring the engine to operate a maximum MR of 12.0:1 at a thrust level determined by NASA LeRC.

Subtask 4 - Vehicle Study/Engine Study Coordination:

Provide support to vehicle/mission studies as needed.

Subtask 5 - Final Report

Months from Go-Ahead	1	2	3	4	5	6	7	8	9	10	11	12	13	14
1. Verify current baseline	38													
2. Establish groundrules	28													
3. Generate Q & Delta press. data	108	108		54		108								
4. Modify mass property/wt. code	76	119												
5. Modify cost code	76	119												
6. Run parametric on-design balances		114	152	152	152			133						
7. Run off-design balances				76	76	57	36							
8. Des. Anal. @ 50 klbf			71	71	71	71	71	71	71	71				
9. Engine req. var. studies		61	61	61	61	61	61	61	61	61	61	61	61	
10. Reports (√ mo. / ∇ final)	15	15	15	15	15	15	15	15	15	15	15	15	46	42 ∇
11. Reviews							▼							∇
Total Hours	341	536	299	429	375	312	183	280	147	147	76	76	107	42
Mo. Ave. E.P.	2.0	3.1	1.7	2.5	2.2	1.8	1.1	1.6	0.8	0.9	0.4	0.4	0.6	0.2

Orbit Transfer Rocket Engine Technology Program

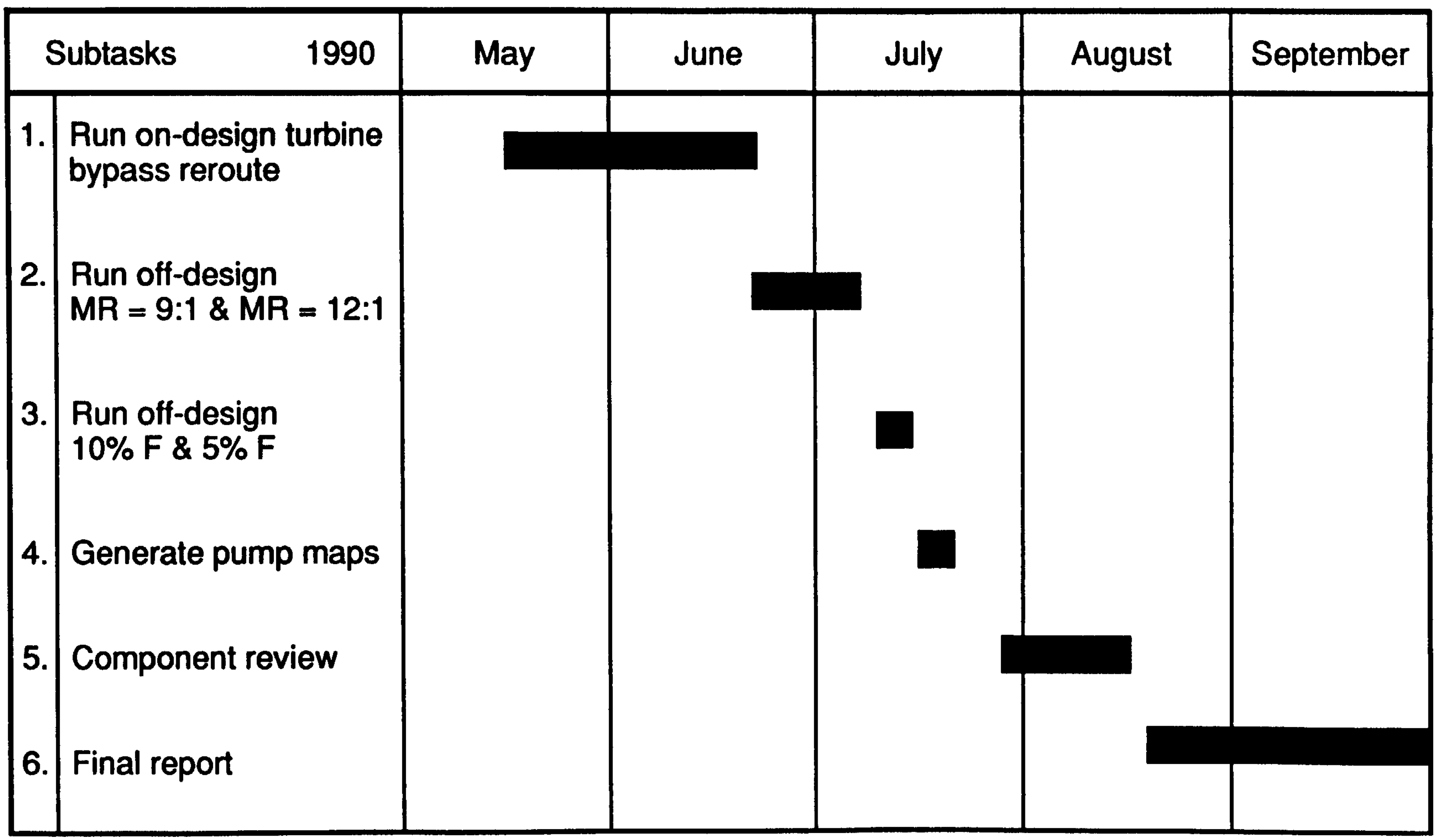
Advanced Engine Study - Task D.6 Schedule

Milestones: 1 Codes & Properties Updates 2 Prelim. Parametrics 3 Final Parametrics 4 50 klbf Analysis

Figure 1

Figure 2

Task D.6 - Subtask 9
Engine Variations Schedule



SUMMARY OF ACCOMPLISHMENTS

Thrust parametric data was generated for the advanced engine configuration over a range from 7.5 klbf to 50 klbf. Engine mixture ratio was held constant at 6.0:1 for this scan with engine cycle balances being generated at intermediate thrust levels of 15, 20, 25, and 35 klbf. Detailed heat transfer analysis was conducted at each thrust level for the combustor and nozzle. This was necessary to properly determine the impact of thrust upon attainable chamber pressure and resulting performance due to the sensitivity of the expander cycle to heat loads. Photographic scaling with a constant length/diameter ratio for the combustor was employed for these parametrics.

In addition, a parametric scan of on-design mixture ratio was conducted. Mixture ratios of 5:1 and 7:1 were investigated at thrust levels of 7.5, 15, 35, 50 klbf.

Off-design cycle balances were also generated at MR's of 5.0:1 and 7.0:1 at each of the thrust levels.

On-design parametric data were also generated over a range of nozzle area ratios from the end of the regeneratively cooled nozzle section to an area ratio of 1200. These parametrics were generated at each of the thrust levels addressed above. Output data for these parametric scans include engine performance, envelope, and weight.

A thrust level of 20 klbf was then chosen by NASA LeRC for engine requirement variation studies in which the effects of increasing the throttling requirement from 10:1 to 20:1, and requiring the engine to operate at a maximum MR of 12:1 were evaluated.

Initial studies revealed that the baseline configuration which evolved out of the D.1 through D.5 Advanced Engine Studies was incapable of operating at MR's above 9:1. This situation was remedied by a flow circuit change in which the fuel turbine bypass was rerouted and by incorporating additional LOX turbine bypass reserve at the on-design operating point.

This revised configuration was then capable of operating at the desired maximum MR of 12:1. In additon to the off-design engine cycle balances generated at the extreme conditions, individual components analyses were conducted to identify potential problems encountered at the high MR and deep throttled operating points.

Several possible complications were observed to which potential solutions were identified.

Several miscellaneous tasks were conducted as vehicle studies support under Subtask 4 - Vehicle Study/Engine Study Coordination.

TECHNICAL DISCUSSION

Design and Parametric Analysis

Groundrules

A set of groundrules was established for use during the generation of the parametric engine cycle balances. These groundrules reflect the technology level which will be incorporated into the advanced engine and primarily consist of turbomachinery operating limits. A review of current state of the art (SOA) materials and advanced materials expected to be available for use in 1995 was made. From the material properties, operating limits were derived for various turbopump parameters. Based on this effort, it was determined that the maximum allowable fuel pump impeller tip speed was 2300 ft/s.

Since hydrostatic bearings are to be used, no upper limit was placed on bearing DN (bearing bore diameter x speed) or pump speed. A maximum turbine pitchline velocity of 1800 ft/s and A x N^2 (annulus area x speed squared) of 10.0 x 10^{10} in^2 x RPM^2 was used as limits for both main turbines.

Heat Transfer Analysis

The power used to drive the turbines in an expander cycle is extracted from the combustor and nozzle through regenerative cooling with hydrogen propellant. In order to properly evaluate the effect of thrust level upon attainable chamber pressure, it was necessary to accurately define the coolant heat loads and pressure drops in the combustor and nozzle coolant circuits at each thrust level. The combustor designs for which the heat transfer analyses were conducted reflected advanced manufacturing techniques enabling low pressure drop. By using a maximum channel height/width of 8.0:1 (previous limit = 4.0:1) significant reductions in coolant pressure drop were realized.

An iterative approach was necessary to define the heat loads and pressure drops in the main combustion chamber (MCC) and nozzle coolant circuits. An estimate of the maximum attainable chamber pressure was first made and combustor/nozzle geometry estimated. This preliminary geometry was used in detailed heat transfer analysis in which heat loads and pressure drops were calculated. These data were used to generate an engine cycle balance in which chamber pressure is maximized and combustor/nozzle geometry redefined. The revised Pc and geometry data were used to repeat the heat transfer analysis, the output of which is used to rebalance the engine again. One analysis cycle as described above usually resulted in a convergence of Pc, geometries and heat loads. This process was repeated at each of the thrust levels addressed.

Engine Weight Code

An existing engine weight code for low thrust (15 klbf) upper stage engines was updated to accommodate the thrust range of these parametrics. Advanced low weight materials were assumed for these engines. This primarily entailed the use of composite materials and results in a reduction in overall engine weight of 20% relative to present day conventional materials.

Parametric Data

Thrust Scan. The approach adopted for generation of the thrust parametrics was to maximize vacuum specific impulse (Isv) within a fixed engine length while varying nine pertinent engine parameters. These optimization variables were: (1) chamber pressure, (2) nozzle epsilon, (3) nozzle percent length, (4) fuel T/P speed, (5) oxidizer T/P speed, (6) fuel turbine pressure ratio (PR), (7) oxidizer turbine PR, (8) fuel turbine pitchline velocity (PLV), and (9) oxidizer turbine PLV.

Photographic scaling, in which combustor length/throat diameter and combustor length/engine length were held constant, was assumed for these parametrics. Throat area for a given thrust was primarily determined by attainable chamber pressure. The combustor and engine length were then determined through a fixed combustor length/throat diameter (L/D) ratio set by a 15,000 lbf reference engine with a 20 in. long combustor and overall engine length of 146 in. The engine configuration (flow paths, T/P staging, etc.) was fixed for the parametric scan.

The primary parameter impacting engine maximum attainable performance, weight, and envelope is chamber pressure. A plot of chamber pressure versus thrust is provided in Figure 3 . A relatively sharp increase in Pc is observed between the minimum thrust level of 7500 lbf and 15,000 lbf. The chamber pressure then reaches a relatively stable level with a moderate increase through 50,000 lbf.

The shape of this curve is influenced primarily by three effects: (1) heat extraction per pound of fuel, (2) turbomachinery efficiency, and coolant circuit pressure drop. The heat extraction per pound of fuel varies with thrust and has a direct bearing on the power available to drive the turbines and in turn strongly influences chamber pressure. The measure of heat extraction is reflected in the turbine inlet temperature. A plot of fuel turbine inlet temperature versus thrust is presented in Figure 4 . The shape of this curve is impacted by the ground rule of photographic scaling, since combustor geometry affects the surface area through which the heat is transferred.

Turbomachinery efficiency also has a direct bearing on the attainable chamber pressures. A plot of hydrogen pump efficiency versus thrust level is provided in Figure 5 . As the thrust level is decreased, the required pump impeller diameter decreases to handle the lower flowrate. The corresponding clearances within the pump also decrease until a minimum is reached. At this point, further decreases in impeller diameter result in loss of pump efficiency. This is due to increased internal parasitic leakage losses as the clearance/diameter ratio increases. As pump efficiencies decrease for the smaller pumps (lower thrust), turbine pressure ratios must increase to provide additional power. These increases in turbine pressure ratios then reduce attainable chamber pressure.

A third parameter strongly influencing chamber pressure is the coolant circuit pressure drop. Aside from line losses and valving, the primary pressure losses in the engine are in the combustion chamber and nozzle cooling passages. Of these two, the nozzle pressure drops are relatively small and are essentially constant with thrust. The combustor pressure losses, on the other hand, are large and increase with increasing thrust. A plot of Main Combustion Chamber (MCC) coolant pressure drop versus thrust is provided in Figure 6 . Increases in coolant pressure drop negatively impact the attainable chamber pressure.

It is the combined effect of the heat extraction rates, turbopump efficiencies, and coolant circuit pressure drops that shaped the Pc versus thrust curve.

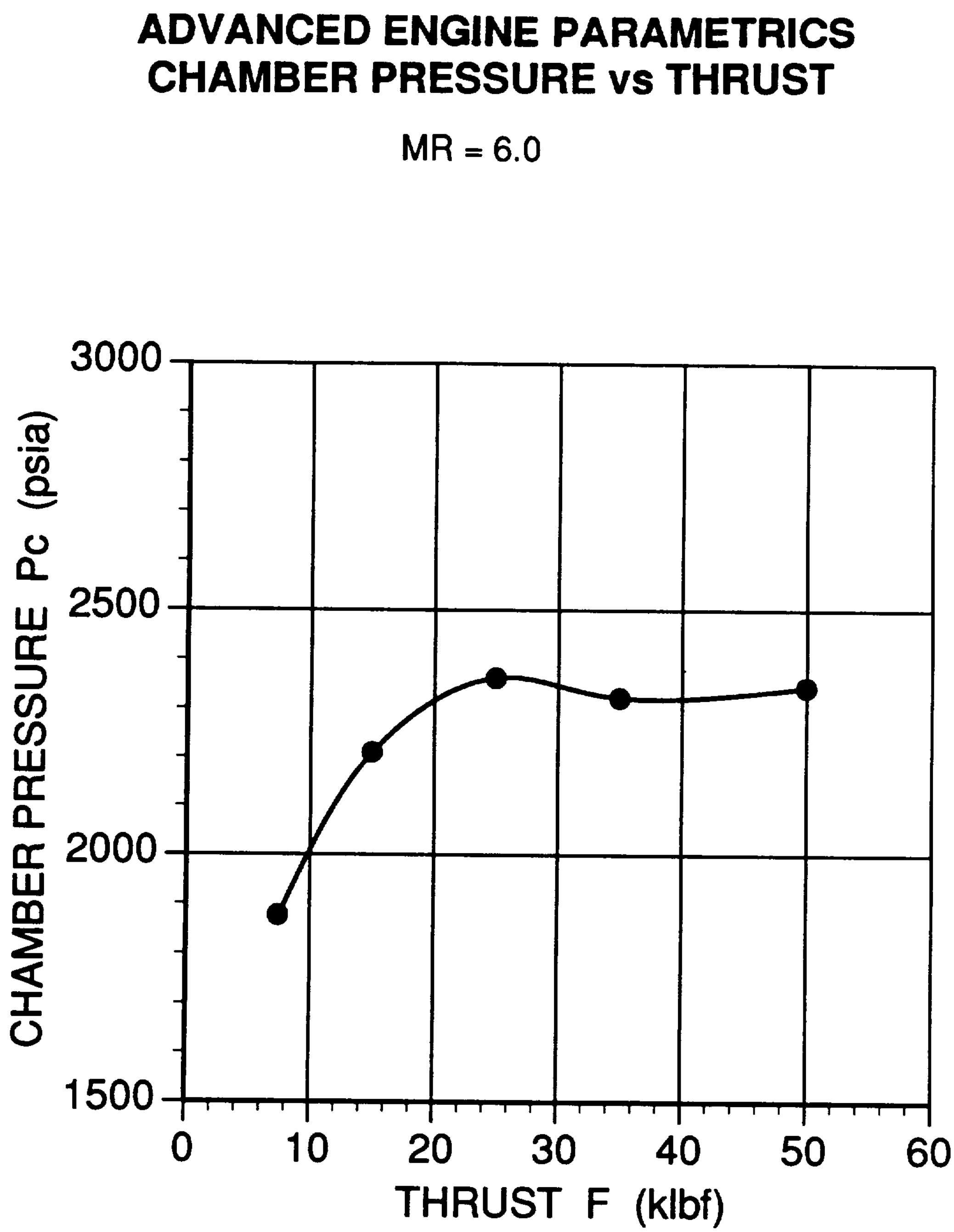


Figure 3. Advanced Engine Parametrics Chamber Pressure versus Thrust

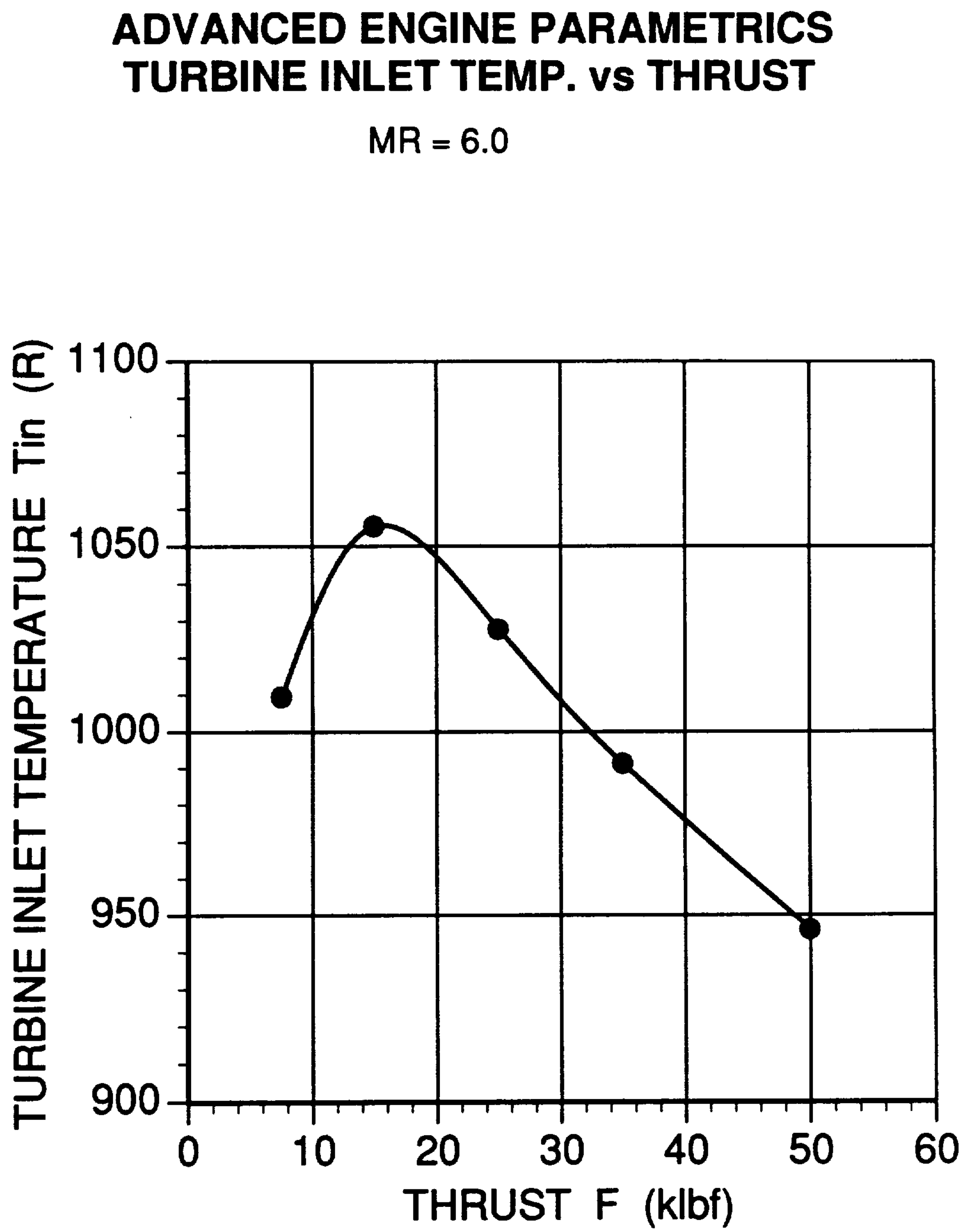


Figure 4. Advanced Engine Parametrics Turbine Inlet Temp versus Thrust

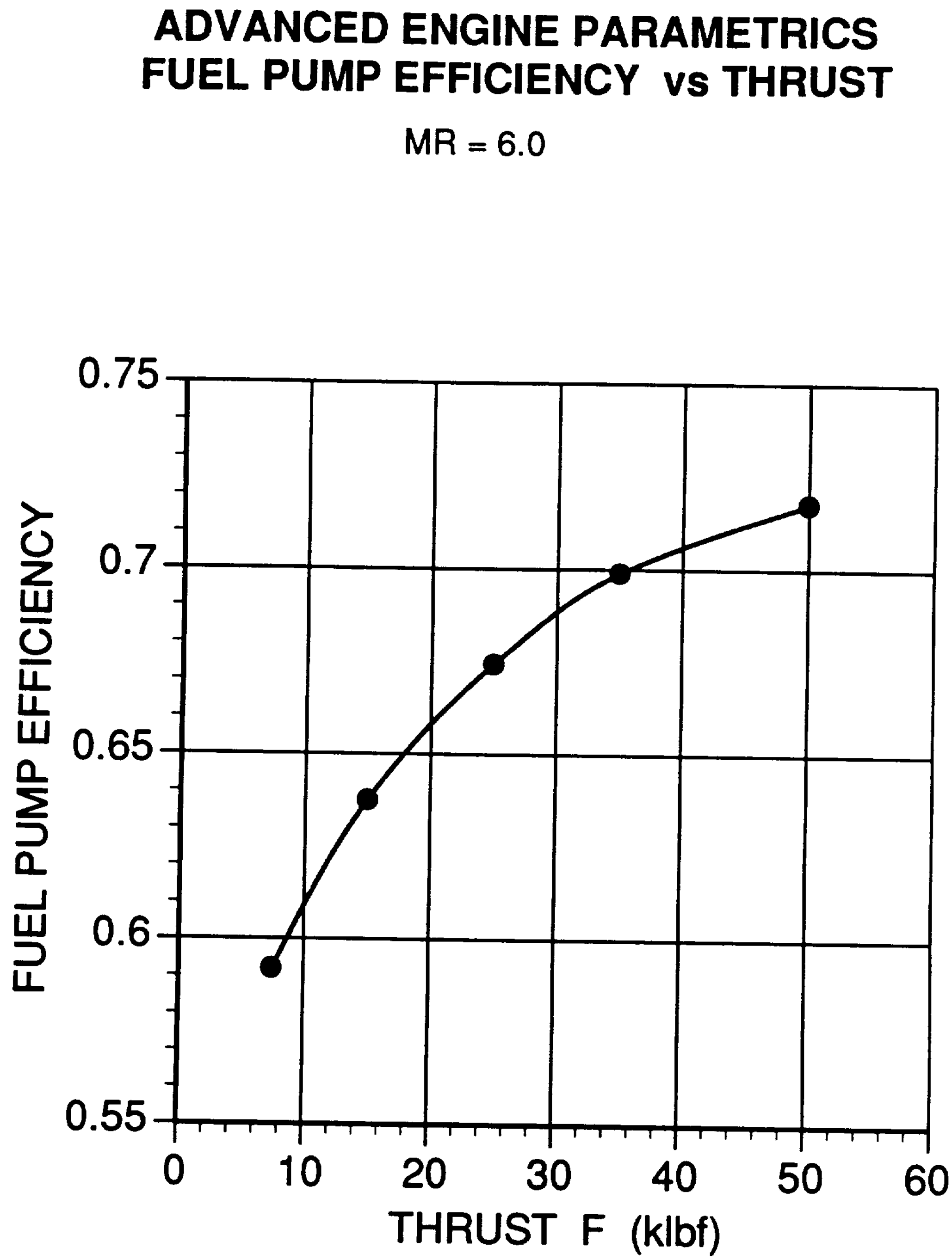


Figure 5. Advanced Engine Parametrics Fuel Pump Efficiency versus Thrust

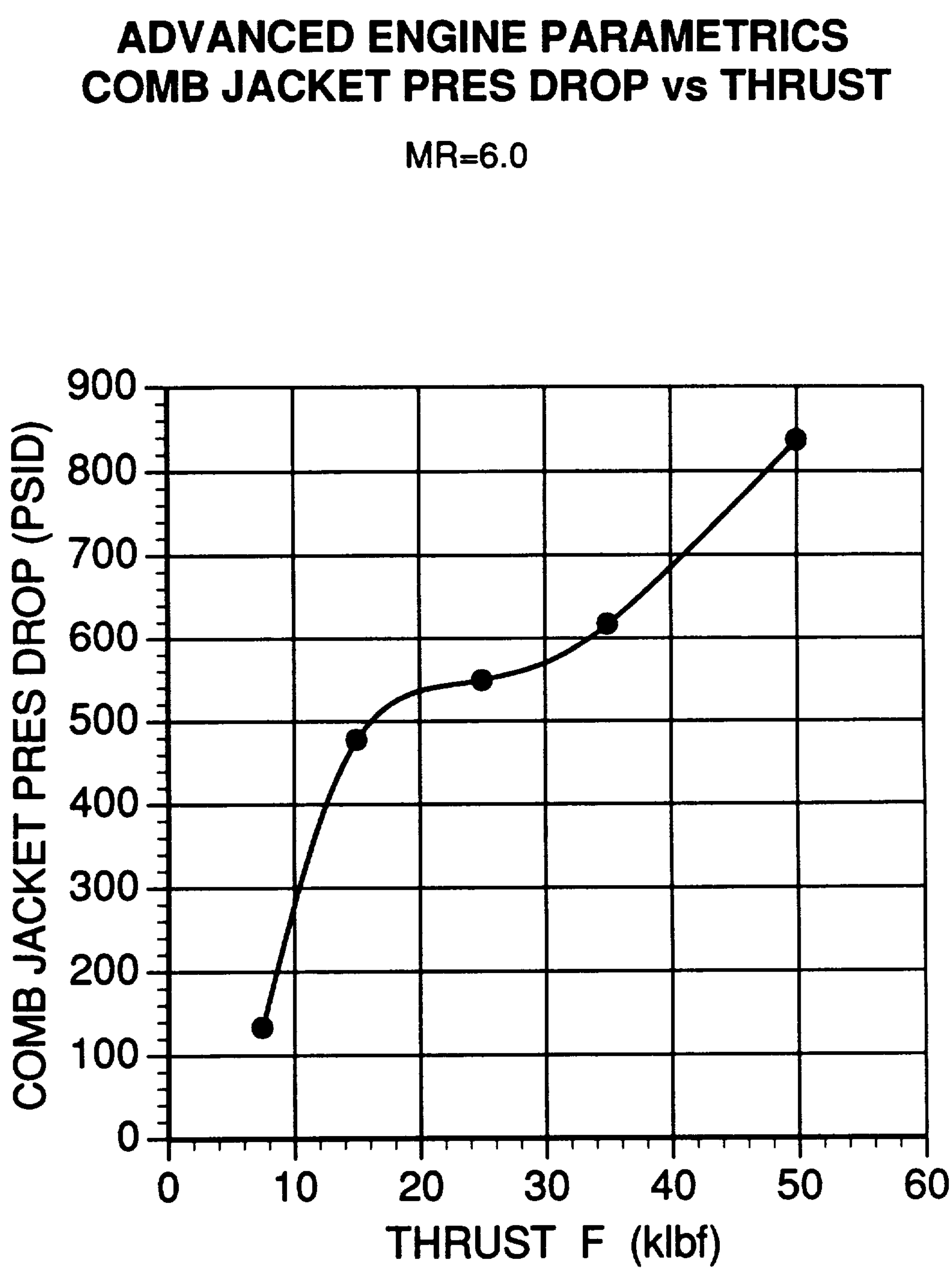


Figure 6. Advanced Engine Parametrics Combustor Jacket Pressure Drop versus Thrust

The engine performance closely follows the chamber pressure trend with thrust since the nozzle expansion ratios are relatively constant. Chamber pressure is another parameter affected by the engine scaling method used. A plot of vacuum specific impulse versus engine thrust is also presented in Figure 7 . As with the chamber pressure, the values reach a relatively stable level after an initial sharp increase. The high chamber pressures coupled with large expansion ratios provide impressive engine performances with specific impulses in the 490 sec to 493 sec range.

Photographic scaling results in nearly linear increases in engine length and nozzle exit diameter with increasing thrust. Plots of engine length, combustor length, and nozzle exit diameter versus thrust are presented in Figures 8 through 10 .

Mixture Ratio Scan. Additional on-design engine cycle balances were also generated in a mixture ratio scan between 5:1 and 7:1 at thrust levels of 7.5, 15, and 50 klbf. For this task, the same envelopes which were arrived at through photographic scaling in the thrust parametric study were assumed. A summary of these results is presented in Table 2 and in plots of Isp, chamber pressure, nozzle expansion ratio versus on-design mixture ratio in Figures 11 through 13 respectively.

Off-design engine cycle balances were also generated at MR's of 5.0:1 and 7.0:1 for each of the five thrust levels addressed in the parametric scan. For this effort the main oxidizer valve was used in conjunction with the oxidizer turbine bypass valve for mixture ratio control. The results of this effort are summarized in Table 3 .

Nozzle Expansion Ratio Scan. On-design engine cycle balances were generated for the parametric scan of nozzle expansion ratio (ϵ). For this effort, epsilons of 600, 900, and 1200 were investigated for each of the five thrust levels addressed in the thrust parametrics.

For the expansion ratio of 600 it was assumed that the nozzles were full regeneratively cooled and no extensions were used. For epsilon of 900 it was assumed that regenerative cooling was used out to an expansion ratio of 600. Between 600 and 900 an extendable/retractable radiation cooled section was incorporated. In the engines with overall expansion ratios of 1200, regenerative cooling was also incorporated to an epsilon of 600, followed by a radiation cooled extendable/retractable section to 1200.

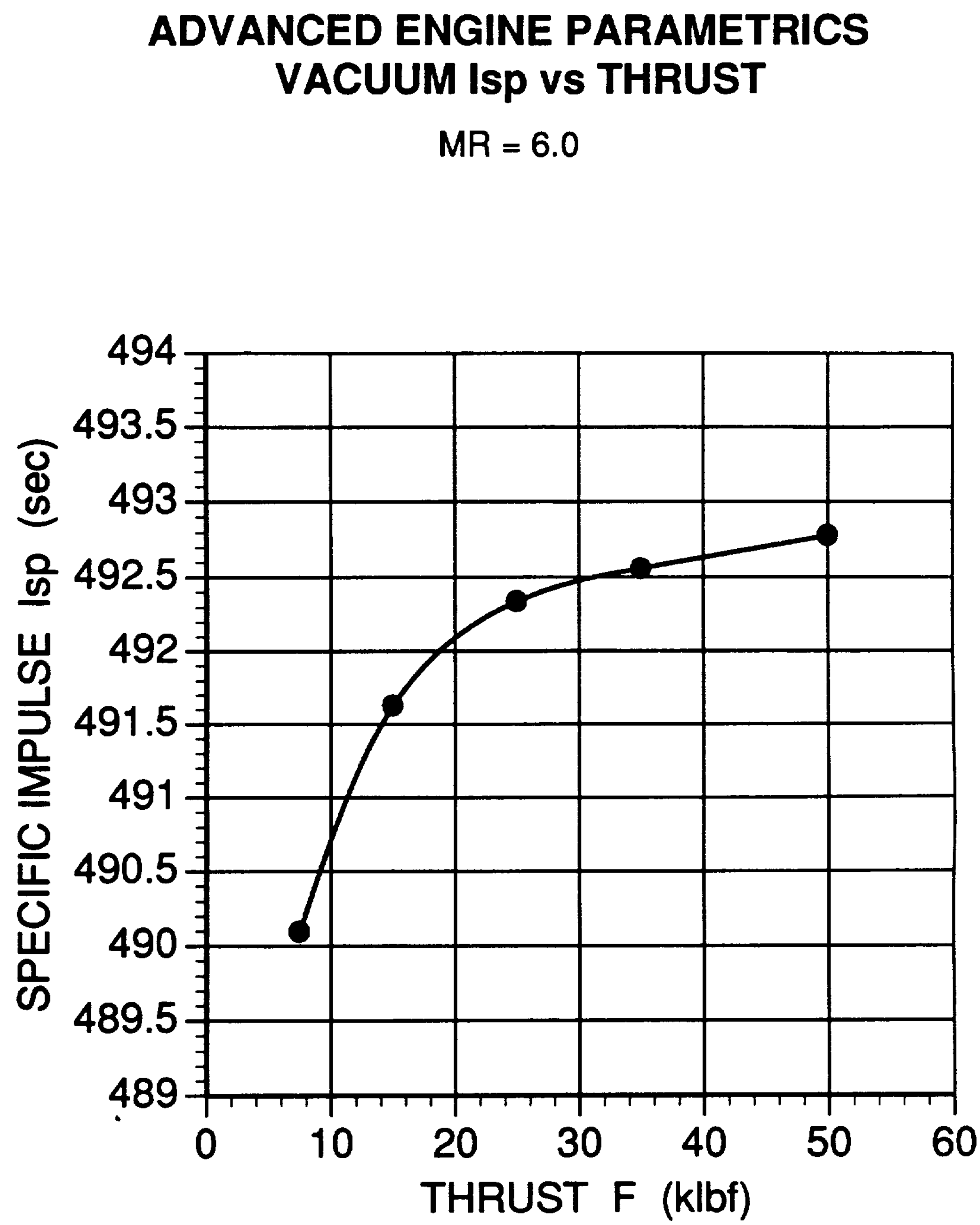


Figure 7. Advanced Engine Parametrics Vacuum Isp versus Thrust

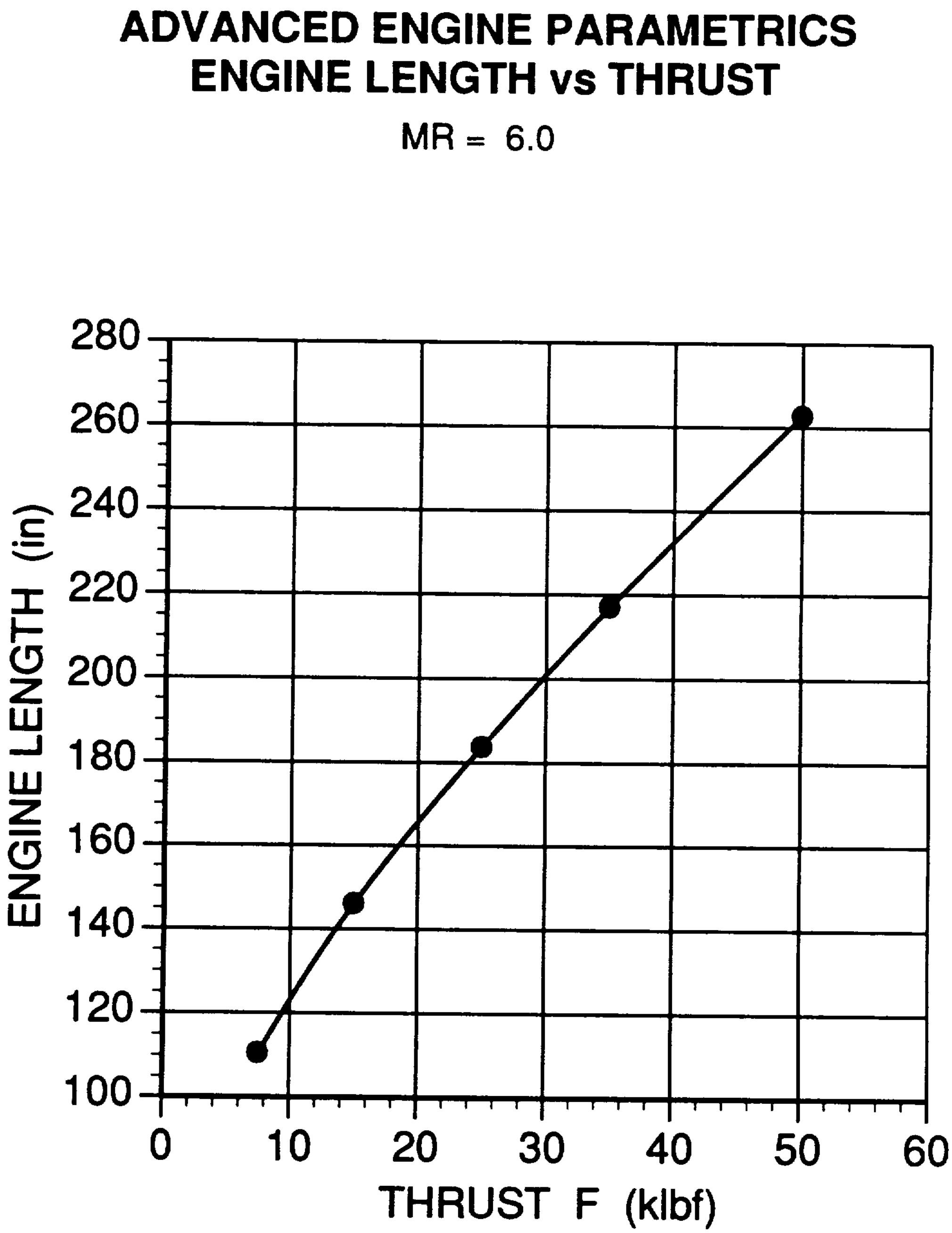


Figure 8. Advanced Engine Parametrics Engine Length versus Thrust

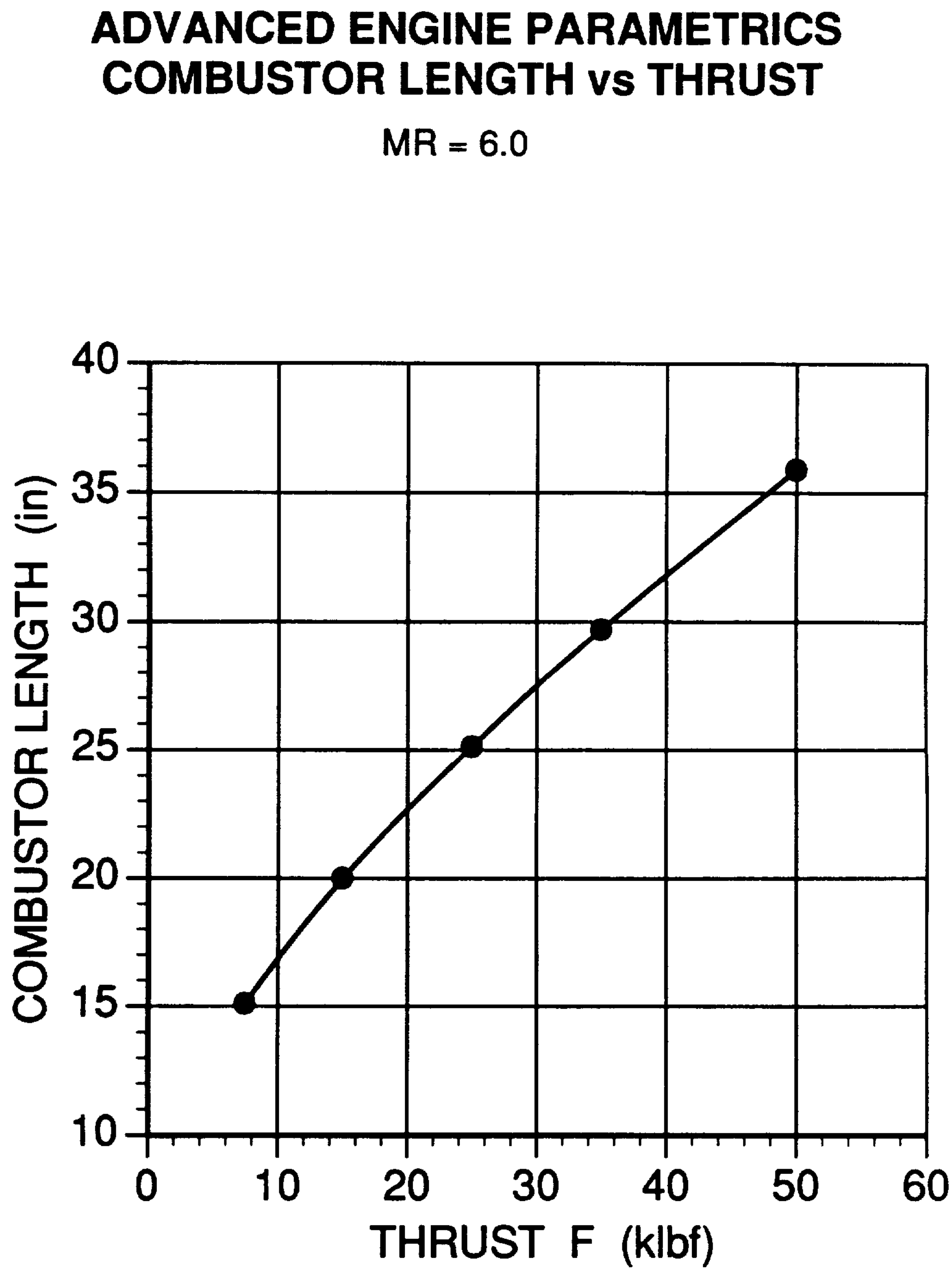


Figure 9. Advanced Engine Parametrics Combustor Length versus Thrust

ADVANCED ENGINE PARAMETRICS
ENGINE DIAMETER vs THRUST

MR = 6.0

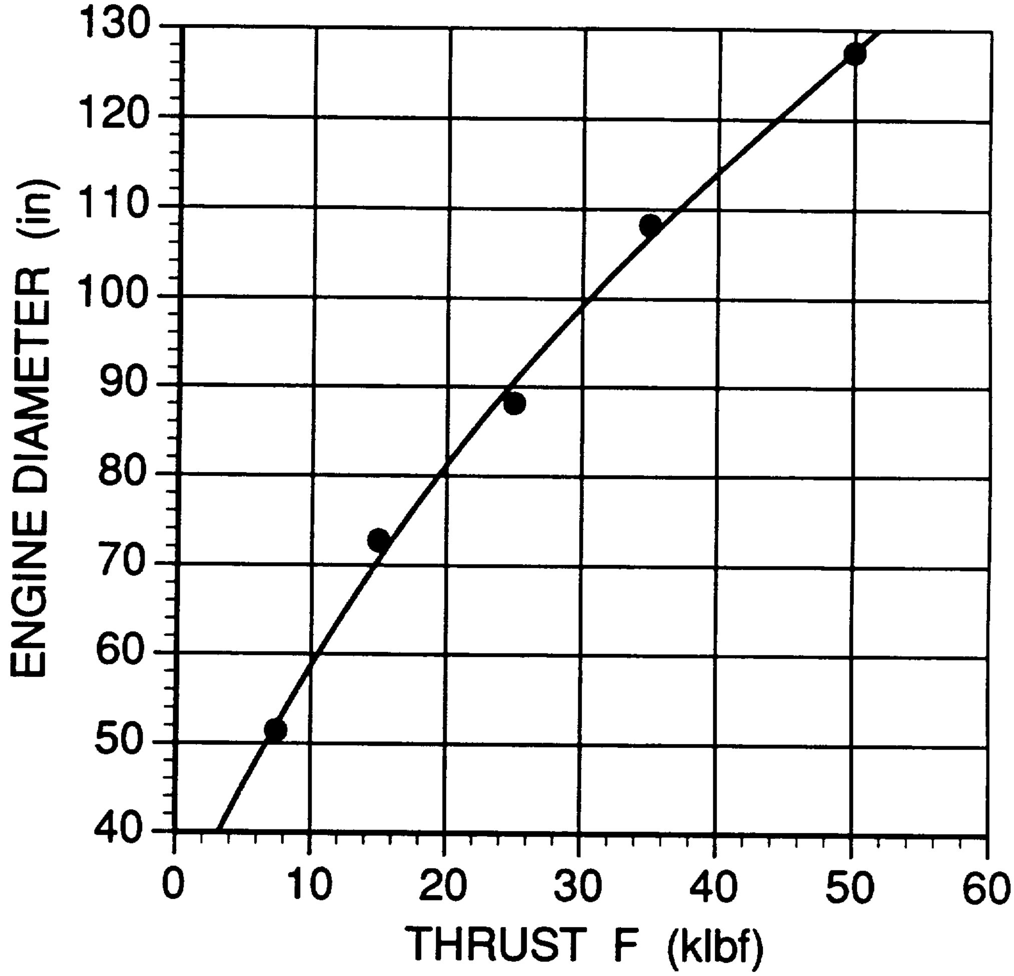


Figure 10. Advanced Engine Parametrics Engine Diameter versus Thrust

Table 2. On-Design Mixture Ratio Parametrics

Thrust, Klb. (vac)	Mixture Ratio	P_c, psia	I_s, vac(sec)	ϵ Overall	L_{ext}, in	D_{ext}, in
7.5	5.0	1738	488.34	934	110.6	51.1
7.5	6.0	1876	490.10	1060	110.6	51.5
7.5	7.0	1977	487.84	1488	119.2	58.2
15	5.0	2023	489.38	967	146.2	68.1
15	6.0	2210	491.63	1246	146.2	72.8
15	7.0	2286	488.85	1296	146.2	71.9
50	5.0	2169	490.05	1082	262.8	126.8
50	6.0	2343	492.78	1223	262.8	127.5
50	7.0	2406	490.88	1330	262.8	129.2

ADVANCED ENGINE PARAMETRICS
Isp versus MIXTURE RATIO

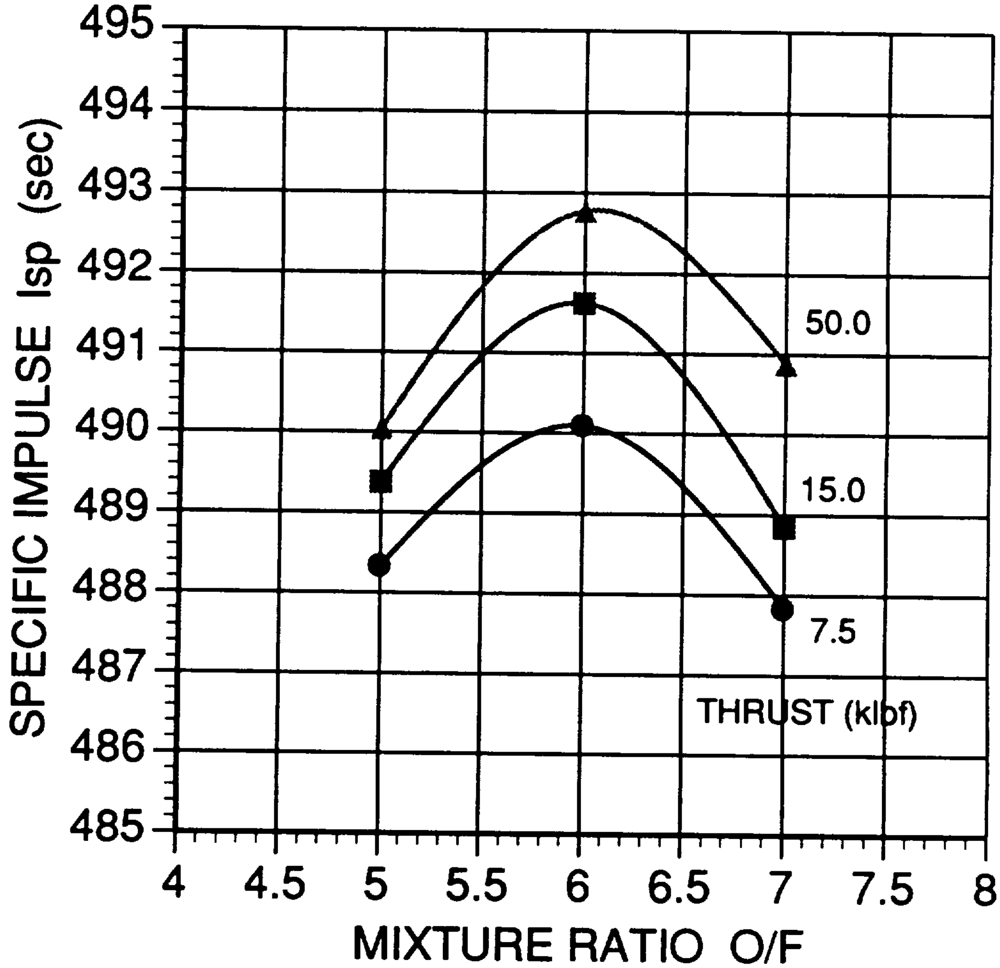


Figure 11. Advanced Engine Parametrics Isp versus Mixture Ratio

ADVANCED ENGINE PARAMETICS
CHAMBER PRESSURE versus MIXTURE RATIO

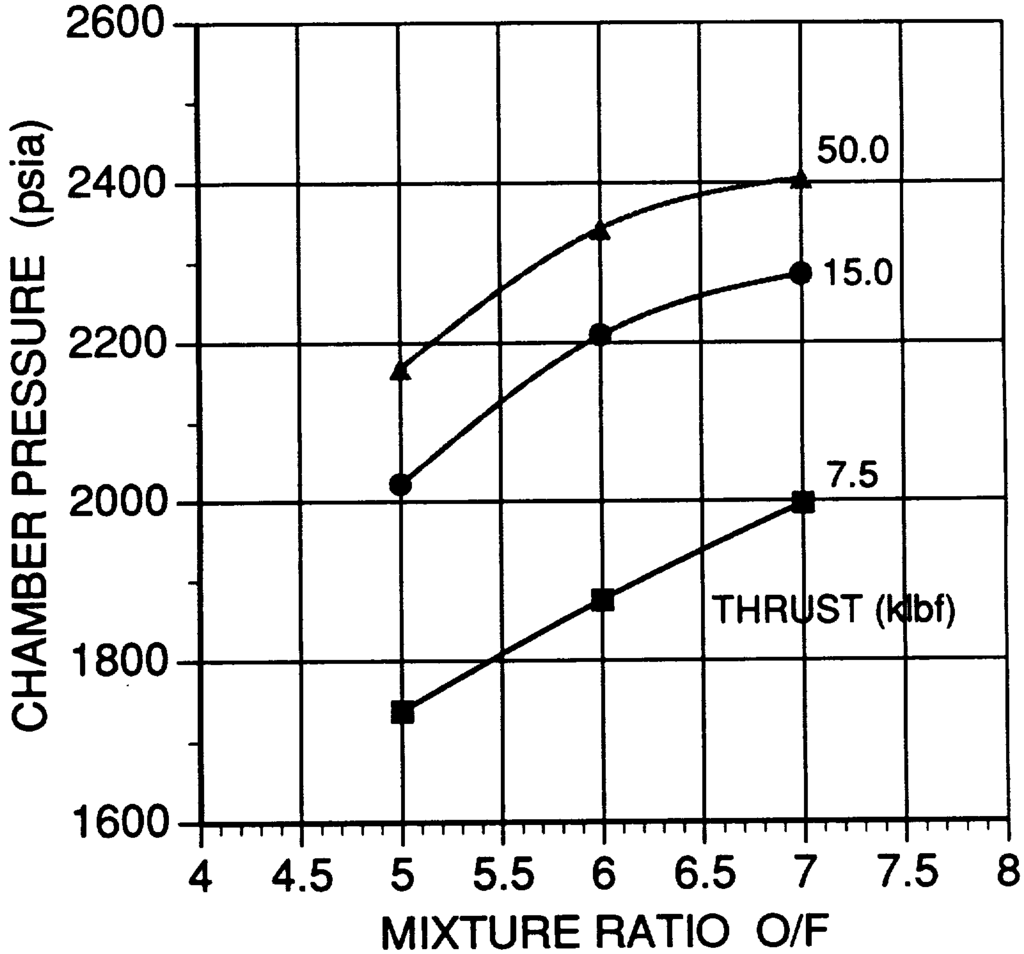


Figure 12. Advanced Engine Parametrics
Chamber Pressure versus Thrust

ADVANCED ENGINE PARAMETRICS
NOZZLE AREA RATIO versus MIXTURE RATIO

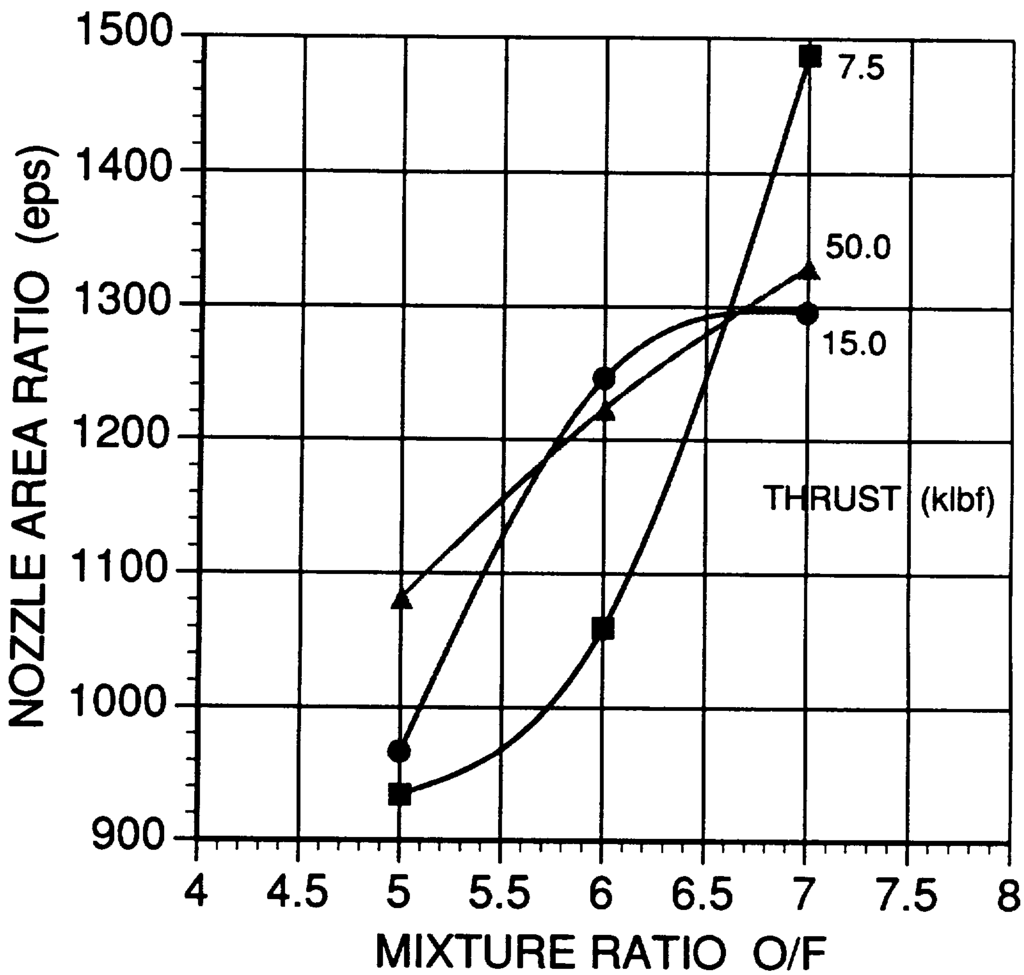


Figure 13. Advanced Engine Parametrics
Nozzle Area Ratio versus Mixture Ratio

Table 3. Thrust Parametrics Off-Design Mixture Ratio Scans

F (klbf)	MR (None)	Pc (psia)	Is (sec)	Lext (in)	Lret (in)	Diam. (in)
7.5	5	1941	489.82			
7.5	6	1876	490.10	110.6	56.8	51.5
7.5	7	1828	485.30			
15	5	2286	490.64			
15	6	2009	491.63	146.2	74.6	72.8
15	7	2147	487.83			
25	5	2442	491.21			
25	6	2360	492.34	183.8	93.4	88.2
25	7	2292	488.93			
35	5	2403	491.25			
35	6	2322	492.56	217.2	110.1	108.3
35	7	2253	489.36			
50	5	2425	491.39			
50	6	2343	492.78	262.8	132.9	127.5
50	7	2273	489.81			

The chamber pressures arrived at for each of the thrust levels in the thrust parametric scan were used in the nozzle epsilon scan. A summary of this data is presented in Table 4 In addition, plots of engine extended length, exit diameter, and vacuum specific impulse are provided in Figures 14 through 16 , respectively. Included in these graphs are the data from the thrust parametrics. The slight variations in trends observed in the Isp versus epsilon plot at the high expansion ratios are due to differences in nozzle percent lengths and the expanded vertical scale of the plot.

Engine Requirement Variation Studies

Introduction

The engine requirement variation subtask consisted of two separate studies: 1) deep throttling (5% of full thrust), and 2) high mixture ratio operation (MR=12:1). Logic flow diagrams of how these studies were conducted are presented in Figures 17 and 18 .

Starting with the on-design baseline engine cycle balance, separate off-design engine cycle balances were generated for the deep throttled and the high mixture ratio operating points. These balances were then analyzed in depth, including generation of pump operating maps, tracking of propellant thermodynamic properties on temperature-entropy (T-S) plots, evaluation of injection pressure drops and combustor cooling, and evaluation of control valve resistance range requirements.

From these analyses, problems were identified and potential solutions were generated. A flow circuit configuration change was required to reach the high mixture ratio operating point at full thrust.

Engine System

Based ondirection from NASA LeRC, a reference baseline engine at a vacuum thrust level of 20 klbf was used for the engine requirement variation studies. The first step in this effort was to run a baseline cycle balance at the 20 klbf thrust level with ground rules and technology limits consistent with the subtask 1 parametric scans. A chamber pressure of 2401 psia was achieved at this thrust level.

Table 4. Expansion Area Ratio Parametrics

Thrust, Klb. (vac)	ϵ Overall/ break	P_c, psia	I_s, vac (sec)	L_{ext}, in	L_{ret}, in	D_{ext}, in	Weight lb.
7.5	1200/600	1876	490.79	114.1	66.3	54.8	291
7.5	900/600	1876	488.90	101.6	73.0	47.5	278
7.5	600/600	1876	488.49	86.7	86.7	38.8	248
15	1200/600	2209	492.04	146.7	84.4	71.4	414
15	900/600	2209	490.14	130.5	93.1	61.9	389
15	600/600	2209	489.65	111.0	111.0	50.6	334
25	1200/600	2360	492.38	181.4	103.7	89.1	588
25	900/600	2360	490.51	161.1	114.5	77.3	548
25	600/600	2360	489.80	136.8	136.8	63.2	469

Table 4. Expansion Area Ratio Parametrics (contd.)

Thrust, Klb. (vac)	$\in$ Overall/ break	P_c, psia	I_s, vac (sec)	L_{ext}, in	L_{ret}, in	D_{ext}, in	Weight lb.
35	1200/600	2322	492.37	214.4	121.7	106.3	812
35	900/600	2322	490.50	190.2	134.7	92.2	736
35	600/600	2322	489.72	161.3	161.3	75.4	636
50	1200/600	2343	492.29	254.3	144.1	126.4	1168
50	900/600	2343	490.44	225.6	159.5	109.7	1040
50	600/600	2343	489.61	191.2	191.2	89.6	906

ADVANCED ENGINE PARAMETRICS
ENGINE LENGTH versus AREA RATIO

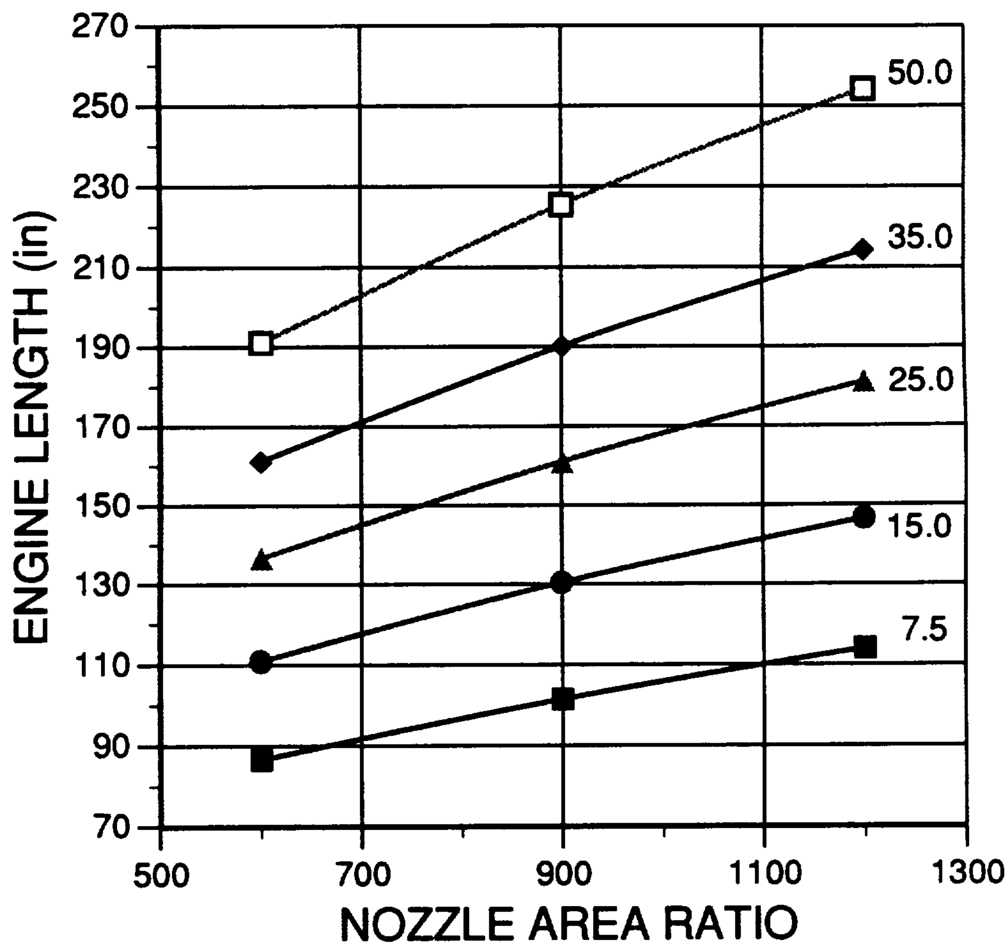


Figure 14. Advanced Engine Parametrics
Engine Length versus Area Ratio

ADVANCED ENGINE PARAMETRICS
ENGINE DIAMETER versus AREA RATIO

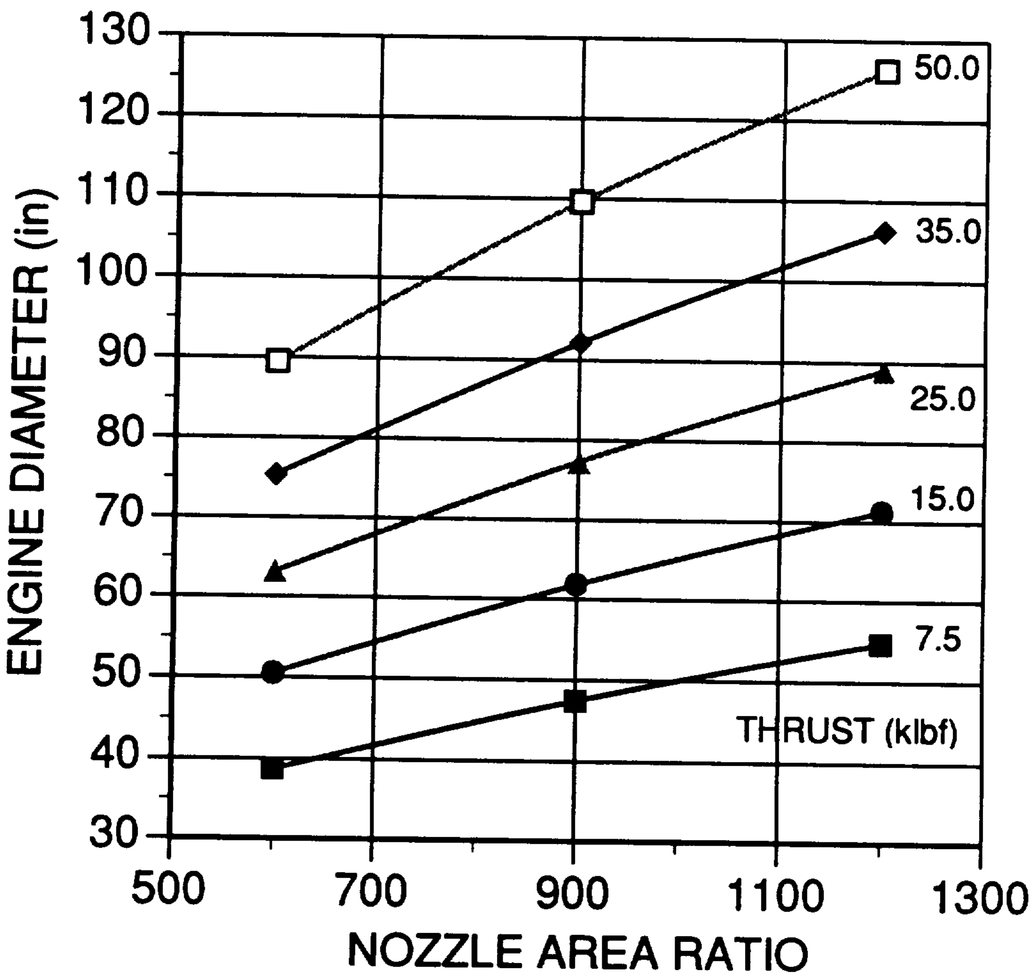


Figure 15. Advanced Engine Parametrics
Engine Diameter versus Area Ratio

ADVANCED ENGINE PARAMETRICS
SPECIFIC IMPULSE versus AREA RATIO

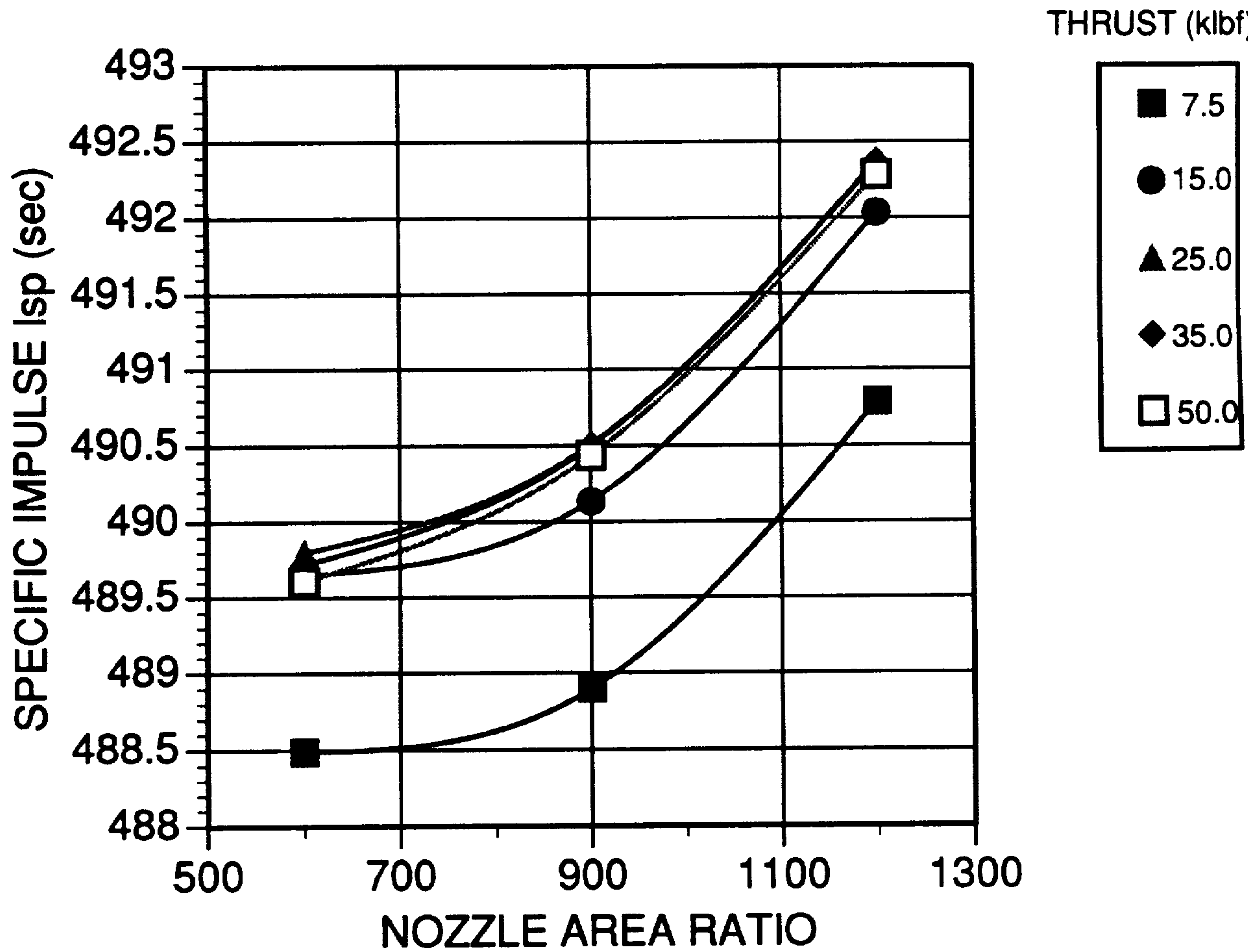


Figure 16. Advanced Engine Parametrics
Isp versus Area Ratio

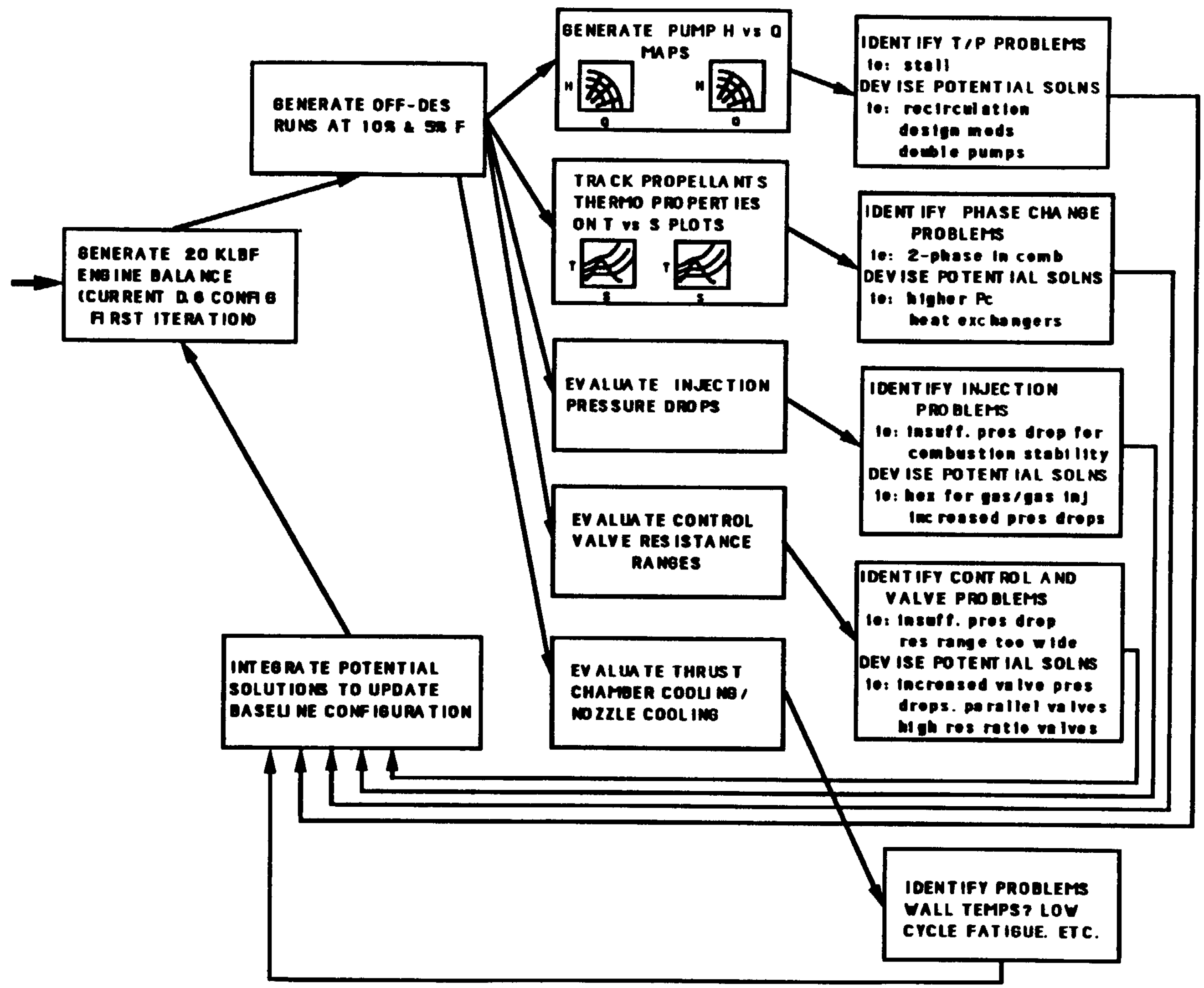


FIGURE 17. Advanced Engine Study Task D.6 Engine Variation Studies 20:1 Throttling

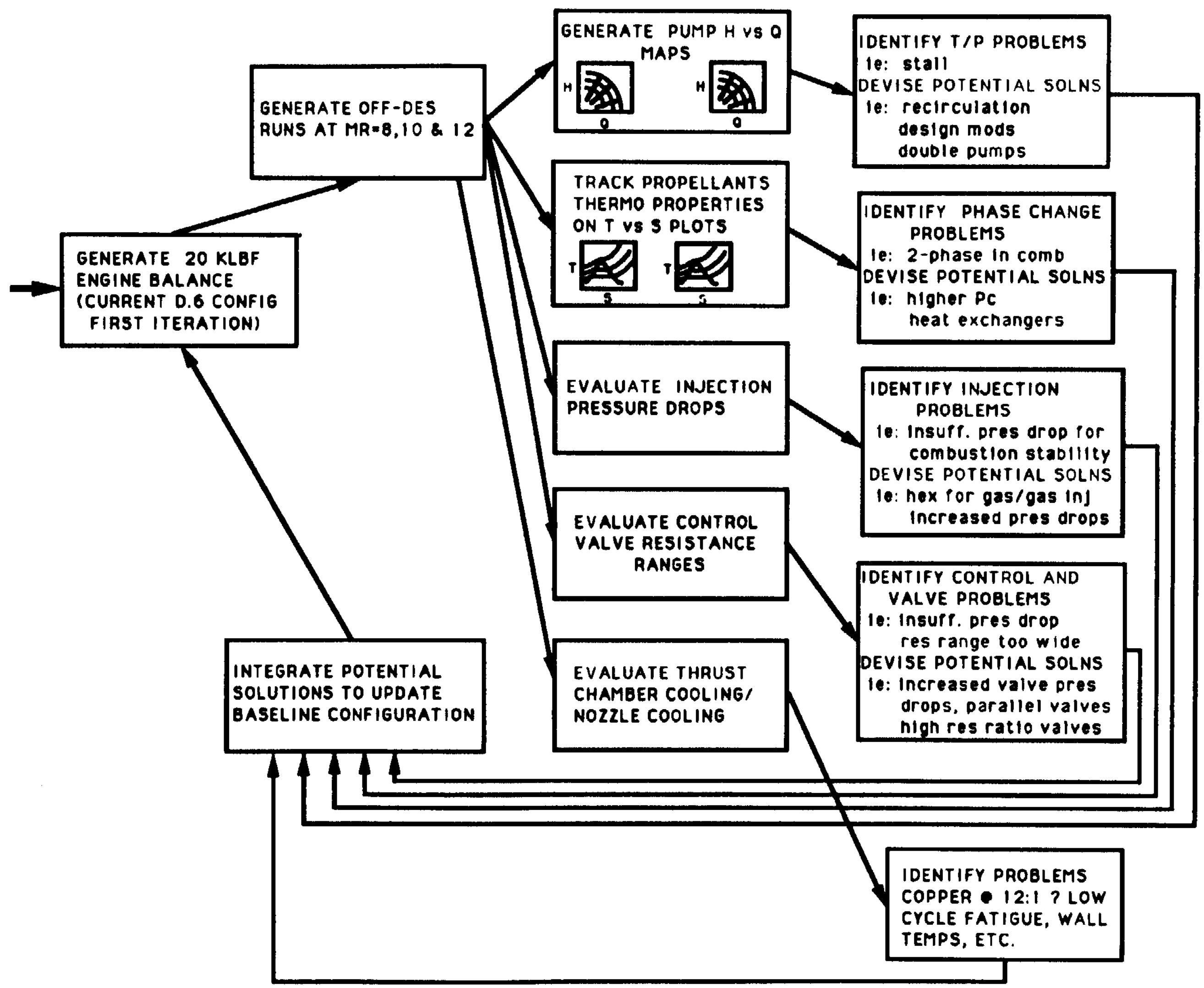


FIGURE 18. Advanced Engine Study Task D.6 Engine Variation Studies 6:1 & 12:1 MIXTURE RATIO

Off-design engine cycle balances were run at full thrust for mixture ratios of 5:1 and 7:1. These are consistent with the baseline advanced engine requirements. Balances were then run at both 10% and 5% of full thrust. This was followed by an attempt to operate the engine at full thrust and a mixture ratio of 12:1. This attempt was unsuccessful and lower mixture ratios were investigated. It was established that the initial baseline configuration was unable to operate at mixture ratios over 9:1 at full thrust due to power limitations caused by the shift in power requirements from the fuel turbopump to the oxidizer turbopump as the mixture ratio was increased.

With the initial engine configuration using series-connected turbines, all of the flow of drive gas must first pass through the fuel turbine before reaching the LOX turbine. Therefore, as the MR is increased, the fuel T/P spins up faster than required, with the extra LH2 pump discharge pressure being dropped across the main fuel valve. Thus as the MR is increased, the overall horsepower requirement of the engine rises until a power limit was reached at MR = 9:1.

This situation was alleviated by rerouting the fuel turbine bypass to discharge into the inlet of the LOX turbine, thus allowing a shift in power from the fuel T/P to the LOX T/P. Analysis of this configuration is discussed in detail in a subsequent section of this report.

Pump Operation

Pump head versus flow (H vs. Q) operating maps were generated for the main hydrogen and main oxygen pumps. The operating points for the on-design and all of the off-design conditions were plotted on these maps. These plots are presented full scale in Figures 19 and 20 and in expanded scale for the low flow conditions in Figures 21 and 22 . Acceptable operating regions have been defined delineating the various limits within which the pumps must operate. These include impeller tip speed limits, cavitation limits, system stability requirements, boilout limits, and bearing load limits.

Throttled operation at 5% of full thrust requires the main fuel pump to operate in the positive slope region of constant speed lines on the head versus flow map. This could result in system coupled flow instabilities. A potential solution to this problem is the recirculation of a portion of the fuel back into the inlet of the pump. This effectively increases the flow through the pump thus shifting the operation into the desirable negative slope region of the pump map.

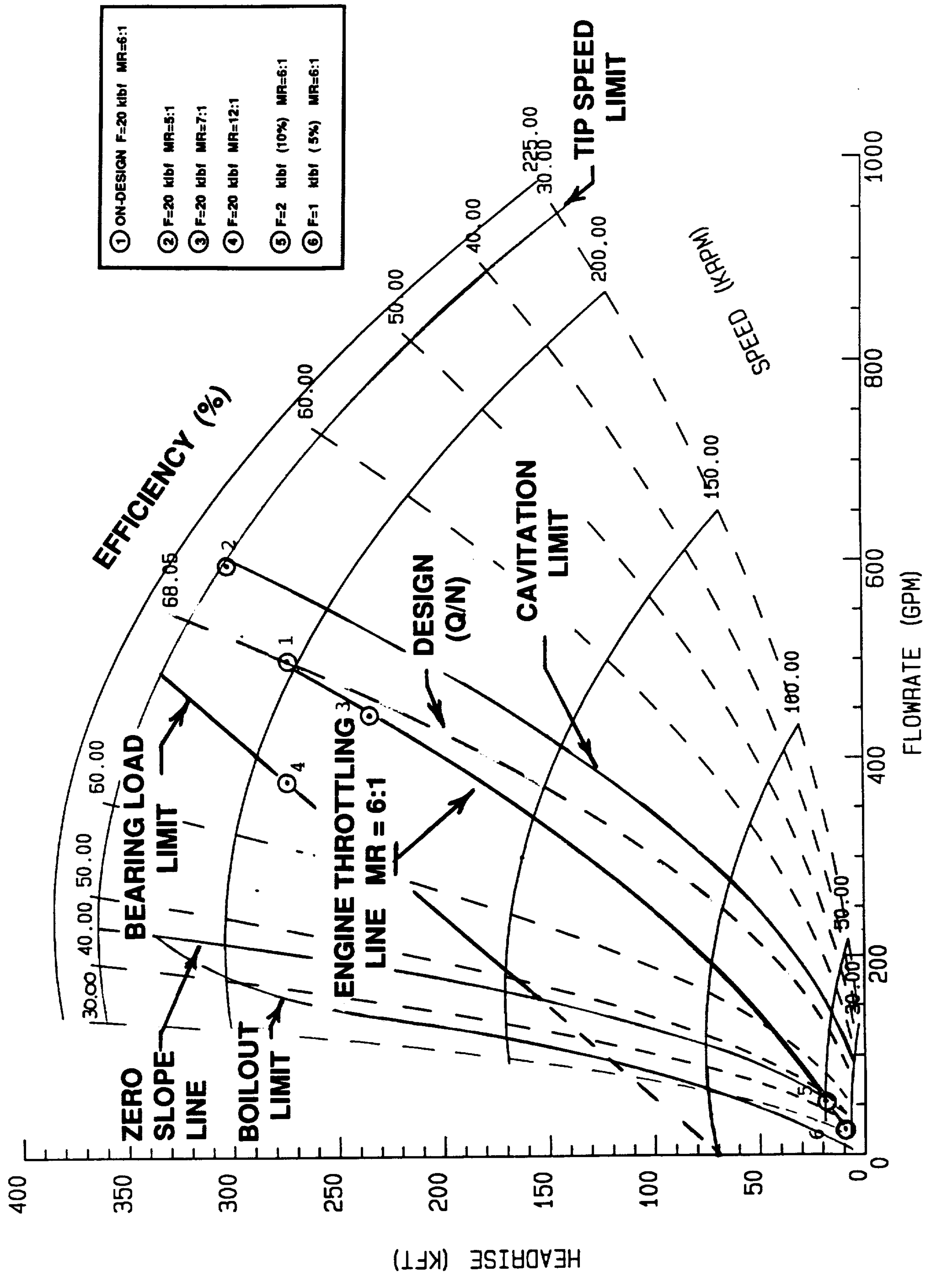


Figure 19. OTV 20 klbf LH2 PUMP MAP

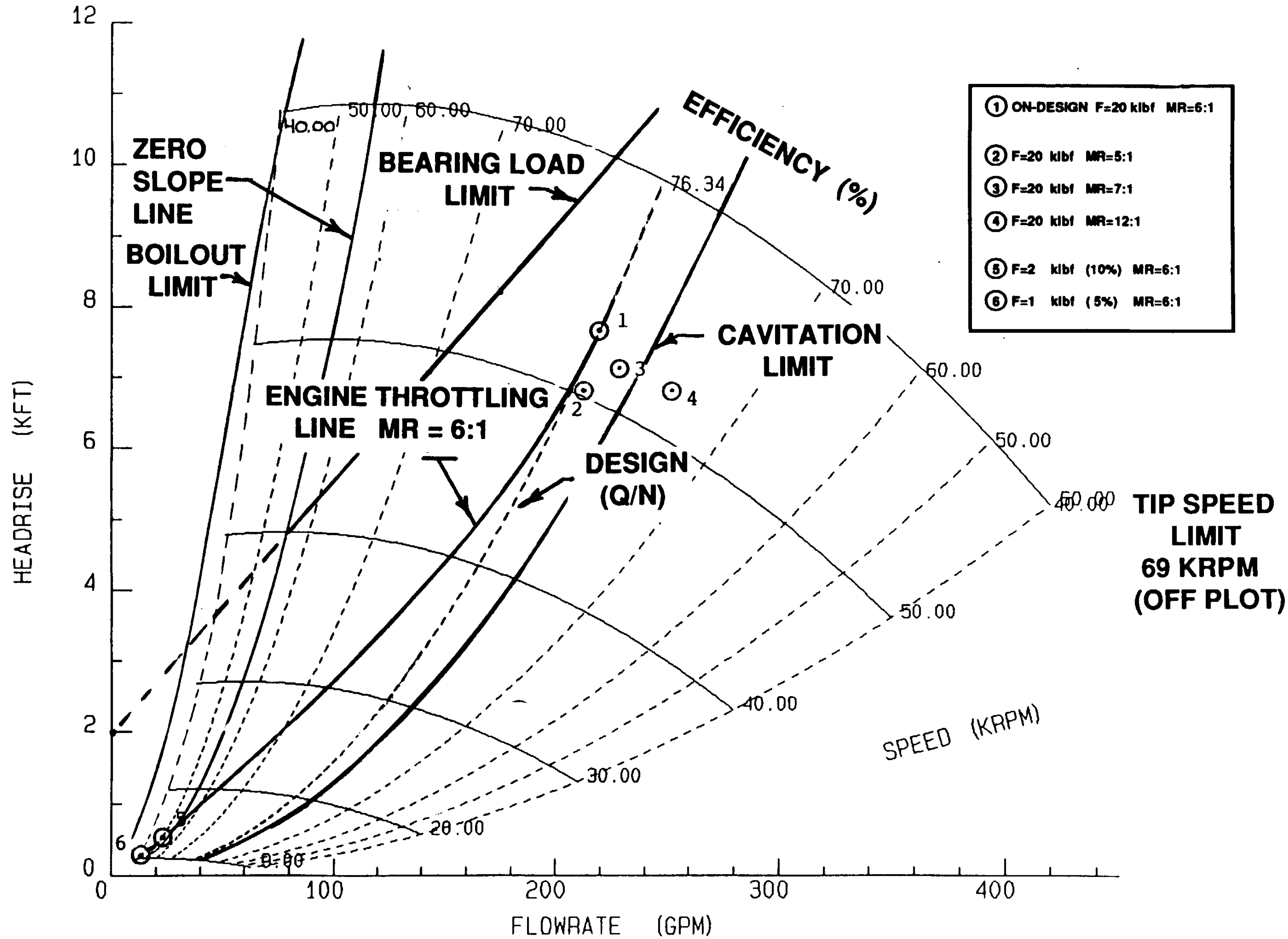


Figure 20. OTV 20 klbf LOX PUMP MAP

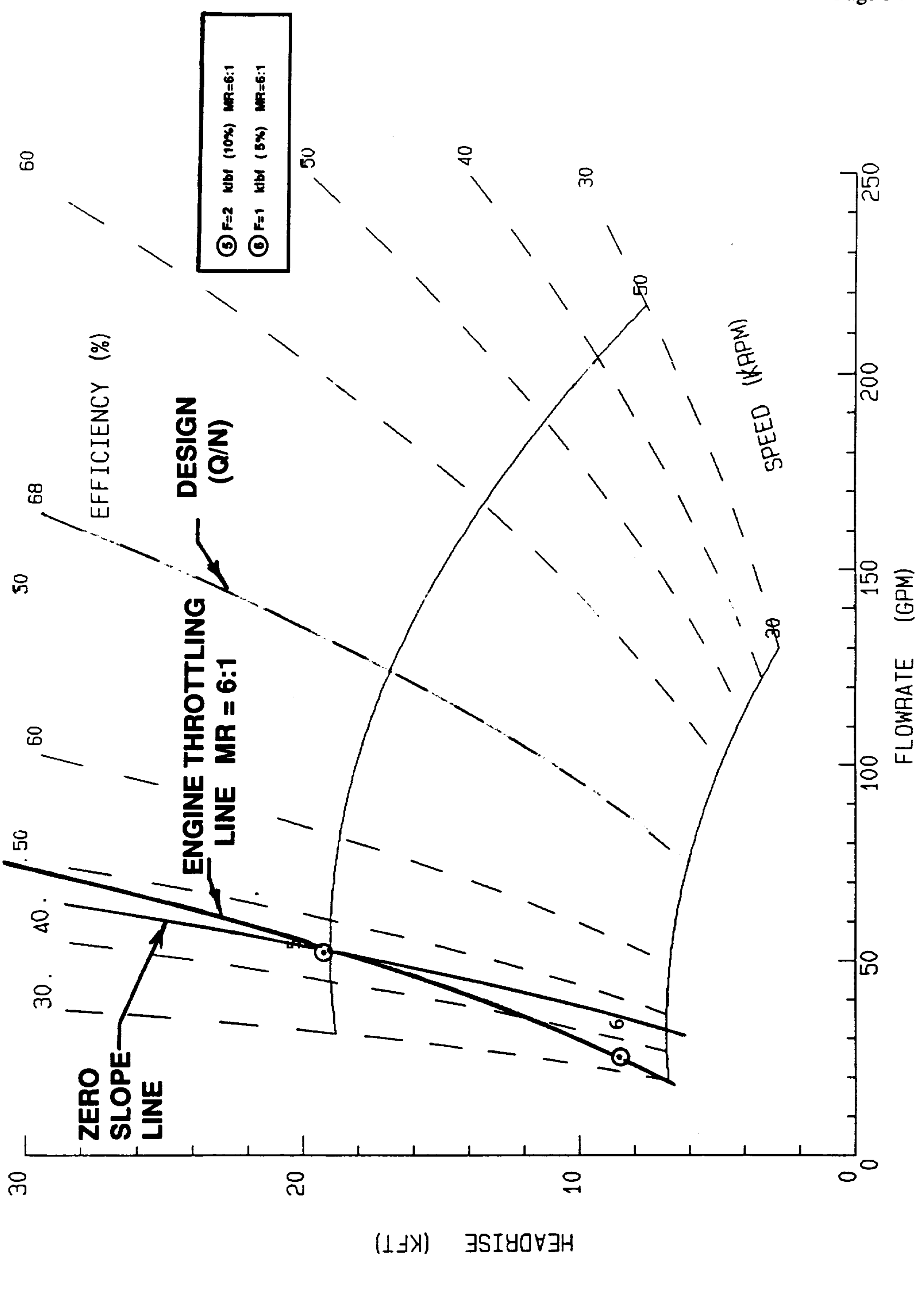


Figure 21. OTV 20 klbf LH2 PUMP MAP (low flow)

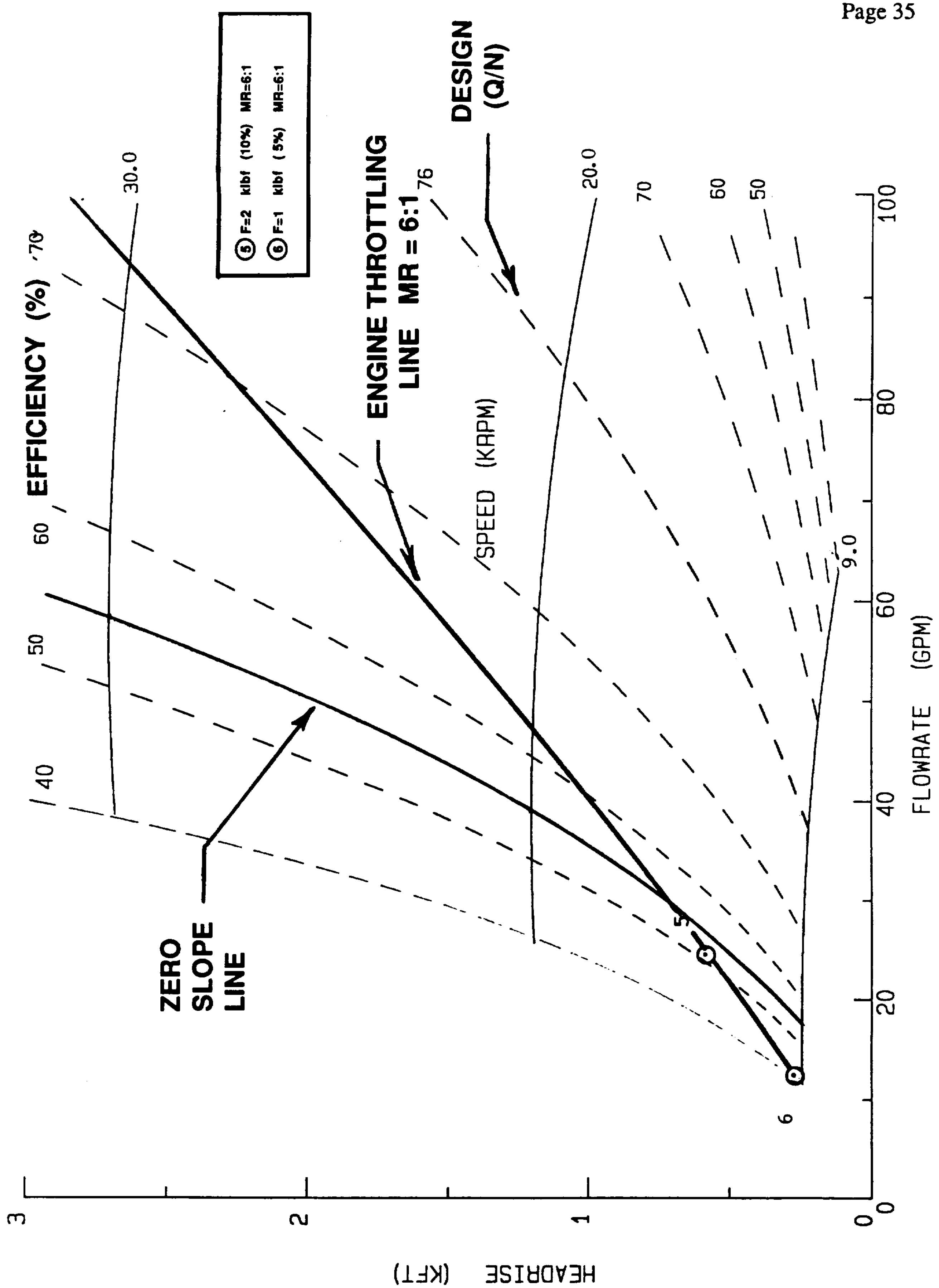


Figure 22. OTV 20 klbf LOX PUMP MAP (low flow)

Another potential solution is to increase the impeller exit flow coefficient by decreasing the blade tip widths. This effectively shifts the zero slope surge line to the left on the pump performance map. The originally selected design was chosen to maximize performance at the design point.

If the power was available to operate above a MR of 9 at full thrust with the initial baseline configuration, the pump would exceed the allowable bearing load limit. This problem could probably be corrected by the use of a double exit volute to more precisely balance radial loads. In addition, refined bearing and seal designs coupled with the strengthening of load supporting areas would also help alleviate this problem.

Rerouting of the turbine bypass to achieve the MR = 12 operating point resulted in lower LH2 pump discharge pressure thus eliminating this problem all together. Details of this are discussed in a subsequent section of this report.

No problems with cavitation or boilout are expected with the main fuel pump during off-design operation.

Throttled operation at both the 10% and 5% thrust levels requires the main LOX pump to operate in the positive slope region of constant speed lines on the head versus flow map. As with the fuel pump, the potential problem of system coupled instabilities could be solved by propellant recirculation or increase of the impeller exit flow coefficient.

Operation at mixture ratios above 9:1 at full thrust will result in cavitation of the main LOX pump. This problem can be alleviated by increasing the blade angle on the inducer which would broaden the allowable operating region.

No problems with bearing load limit or boilout are expected with the main LOX pump during the off-design operation.

Injector Chugging Stability

Injector chugging instability is caused by insufficient pressure drop across the injector. If the system is too "soft", perturbations in the chamber pressure migrate upstream into the feed system and could result in unstable oscillations leading to hardware damage.

A summary of the combustion chamber injection pressure drops is presented in Table 5 for the on-design operating point in addition to all of the off-design points for both the hydrogen and oxygen. Review of this data indicates that the fuel and oxidizer pressure drops are acceptably high at all operating conditions except the LOX pressure drop at 10% and 5% of full thrust. These are marginal, but by close coupling the main oxidizer valve to the injector this potential problem could possibly be avoided. The additional pressure drop provided by the MOV may stabilize the system.

Valve Throttling Ranges

During off-design operation, the engine control valves are required to modulate the propellant flowrates by varying their hydraulic resistances through position changes. The maximum allowable range over which the resistance can vary and still maintain the required sensitivity at the extremes is approximately 100:1.

A summary of the data for the four engine control valves is presented in Table 6 through 9 . These data include propellant flowrates, pressure drops, hydraulic resistances, and resistance ratios relative to the on-design positions. Review of these data revealed that although this valve is normally a non-modulating valve, a modulating main fuel valve (MFV) is required for high mixture ratio (12:1) operation for the initial baseline configuration. Addition of this valve to the control logic should not pose a problem. The resistance ranges required of the main oxidizer valve (MOV) are acceptable for all of the off-design conditions.

Unacceptably wide resistance ranges are required for the fuel turbine bypass valve (FTBV) for the operation at both 5% of full thrust and at high mixture ratio (9:1) operation. This problem may be solved through the use of dual valve packages. With this approach the wide resistance range is accommodated by combining two valves in a parallel flow configuration, one for low flow and one for the high flow requirements. Another possible alternative is a valve design which has a significantly extended control range.

The resistance range required for the oxidizer turbine valve (OTBV) is also unacceptably wide for operation at a mixture ratio of 9:1 at full thrust with the initial flow configuration. As with the FTBV, the solution to this problem may be the use of a parallel valve arrangement.

Table 5

OTV - 20 klbf Engine

Off-Design Injector Pressure Drops

F klb	MR	P_c psia	ΔP H2 psid	$\frac{\Delta P_{H_2}}{P_c}$	ΔP Lox psid	$\frac{\Delta P_{Lox}}{P_c}$	ΔP MOV psid	$\frac{(\Delta P\ Inj + \Delta PMOV)}{P_c}$
20	6	2401	387	0.161	910	0.379	587	0.623
20	5	2483	445	0.185	864	0.348	161	0.413
20	7	2330	337	0.145	965	0.414	349	0.564
20	9	2319	261	0.112	1176	0.507	9	0.511
2	6	245	58	0.237	10.5	0.043	43	0.218
1	6	124	33	0.266	2.7	0.022	25	0.225

Table 6

OTV - Off Design Valve Summary

MFV - H2

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2401	5.827	25	3.778	1.00
20	5.00	2483	6.852	33	3.749	1.00
20	7.00	2330	5.102	20	3.762	1.00
20	9.00	2319	4.386	1721	433.7	114.8
2	6.00	245	0.600	0.32	3.752	1.00
1	6.00	124	0.305	0.09	3.793	1.00

R_{max}/R_{min} = 114.8 > 100 (range unacceptable for 9:1 MR @ 100%F
F = 20klbf M = 9:1/F = 20klbf M = 6:1)

Table 7

OTV - Off Design Valve Summary

MOV - Lox

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2401	34.97	666	34.68	1.00
20	5.00	2483	33.90	161	10.03	0.289
20	7.00	2330	36.12	349	19.47	0.561
20	9.00	2319	39.93	8.5	0.387	0.0112
2	6.00	245	3.642	43	232.3	6.697
1	6.00	124	1.852	25	530.4	15.29

R_{max}/R_{min} = 89.3 < 100 (range acceptable for 9:1 MR @ 100%F
F = 20klbf M = 6:1/F = 20klbf M = 9:1)

R_{max}/R_{min} = 15.3 < 100 (range acceptable for 20:1 F Engine @ MR = 6:1)

Table 8

OTV - Off Design Valve Summary

FTBV - H2

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2401	0.616	4875	10262	1.00
20	5.00	2483	0.321	6026	56130	5.73
20	7.00	2330	0.686	4183	6337	0.617
20	9.00	2319	0.147	4143	1.27 E06	12.38
2	6.00	245	0.347	187	88.6	0.00863
1	6.00	124	0.192	84	58.9	0.00574

R_{max}/R_{min} = 12.4 << 100 (range acceptable for 9:1 MR @ 100%F
F = 20klbf MR = 9:1/F = 20klbf M = 6:1)

R_{max}/R_{min} = 174.2 > 100 (range unacceptable for 20:1 F Engine
F = 20klbf MR = 6:1/F = 1klbf MR = 6:1)

Table 9

OTV - Off Design Valve Summary

OTBV - H2

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2401	0.541	588	1160	1.00
20	5.00	2483	1.495	569	168.1	0.145
20	7.00	2330	0.128	569	18310	15.78
20	9.00	2319	0.049	585	.1175 E06	101.3
2	6.00	245	0.031	19	941.8	0.812
1	6.00	124	0.014	9	952.8	0.822

R_{max}/R_{min} = 101.3 > 100 (range unacceptable for 9:1 MR @ 100%F
F = 20klbf MR = 9:1/F = 20klbf MR = 6:1)

Combustion and Feed System Stability

Combustion or feed system instabilities can be caused by either of the propellants passing through a two-phase region. Temperature versus entropy (T-S) plots are provided for propellants in Figures 23 and 24 . The engine inlet through main pump outlet thermodynamic states are tracked on these figures for the on-design operating point and the 10% and 5% thrust points. Potential two-phase induced instabilities in the cooling circuits and the oxygen injectors have been evaluated.

Two-phase flow conditions are not encountered at the 10% or 5% thrust levels for the hydrogen, due to the high on-design operating pressure (Pc= 2400 psia). This high chamber pressure results in relatively high system pressures even at the low thrust operating points. Therefore, the operating pressures in the fuel circuit are never below the critical pressure. With the propellant heating occurring above the two-phase dome, the density changes are continuous, resulting in no system instabilities.

There is a potential for the LOX to vaporize in the injectors at the low thrust operating points. The oxidizer system pressures are below critical pressure for oxygen and if enough heat is transferred to the LOX, two-phase flow could occur in the injectors. A preliminary heat transfer analysis, in which only the heat transferred from the injector face was considered, indicated that insufficient heating of the LOX occurs from that source to result in two-phase flow. Additional heat from the warm GH2 will also be transferred in the actual injector elements through the LOX posts. This additional heat must be considered. A more detailed heat transfer analysis would be necessary to resolve the question of LOX vaporization.

Cooling Capabilities and Limits - Nozzle and Combustor

As the engine thrust is reduced, the coolant flowrate decreases at a faster rate than the decrease in heat load. Consequently, the combustor and nozzle wall temperatures and coolant bulk temperatures increase. Similarly, when the mixture ratio is increased, the coolant flow decreases causing temperature rises.

Plots of coolant bulk temperature versus thrust and coolant bulk exit temperature versus mixture ratio for the nozzle and combustor are presented in Figures 25 and 26 . These data

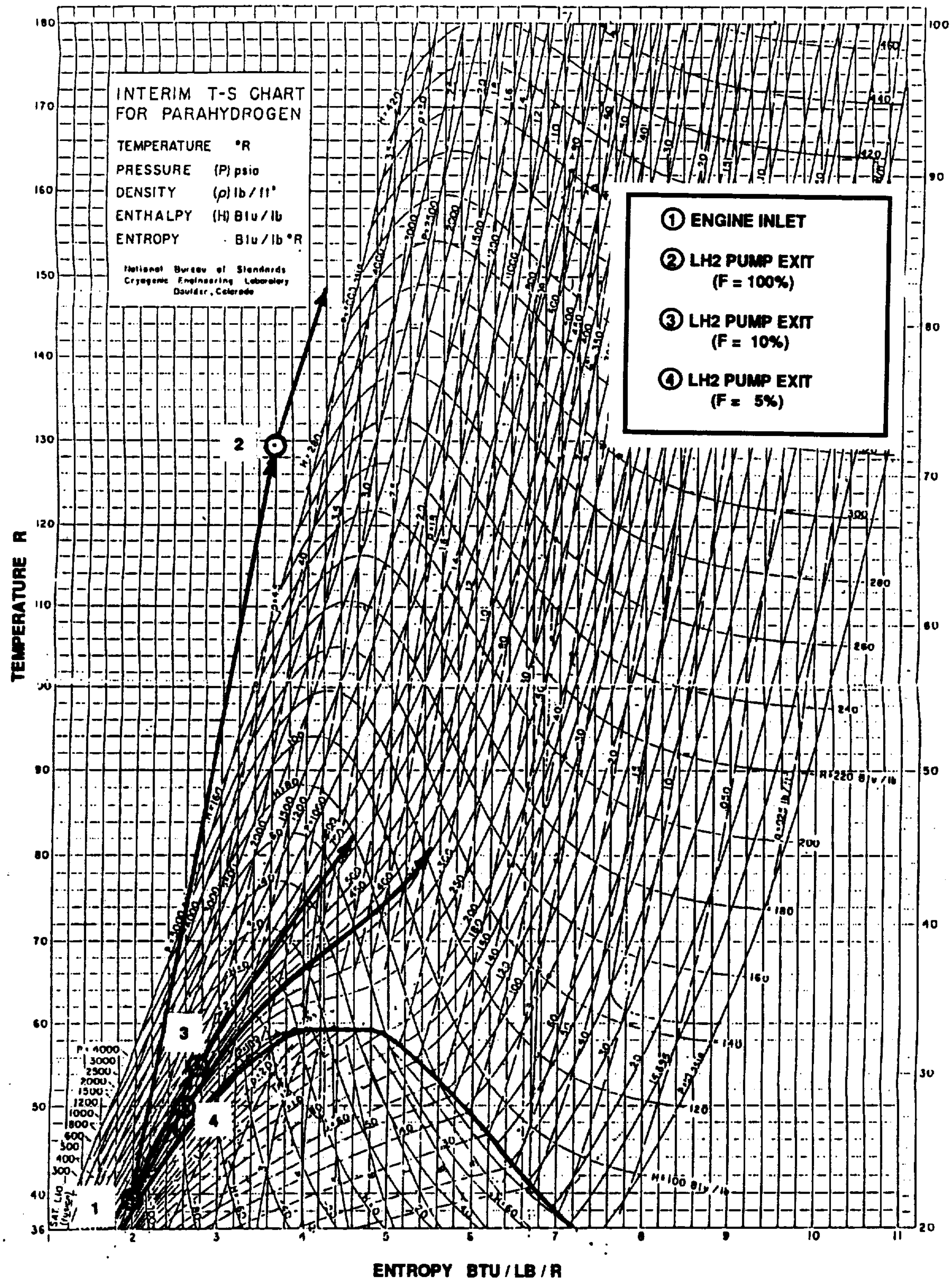


Figure 23. 20 klbf ENGINE H2 PROPERTIES

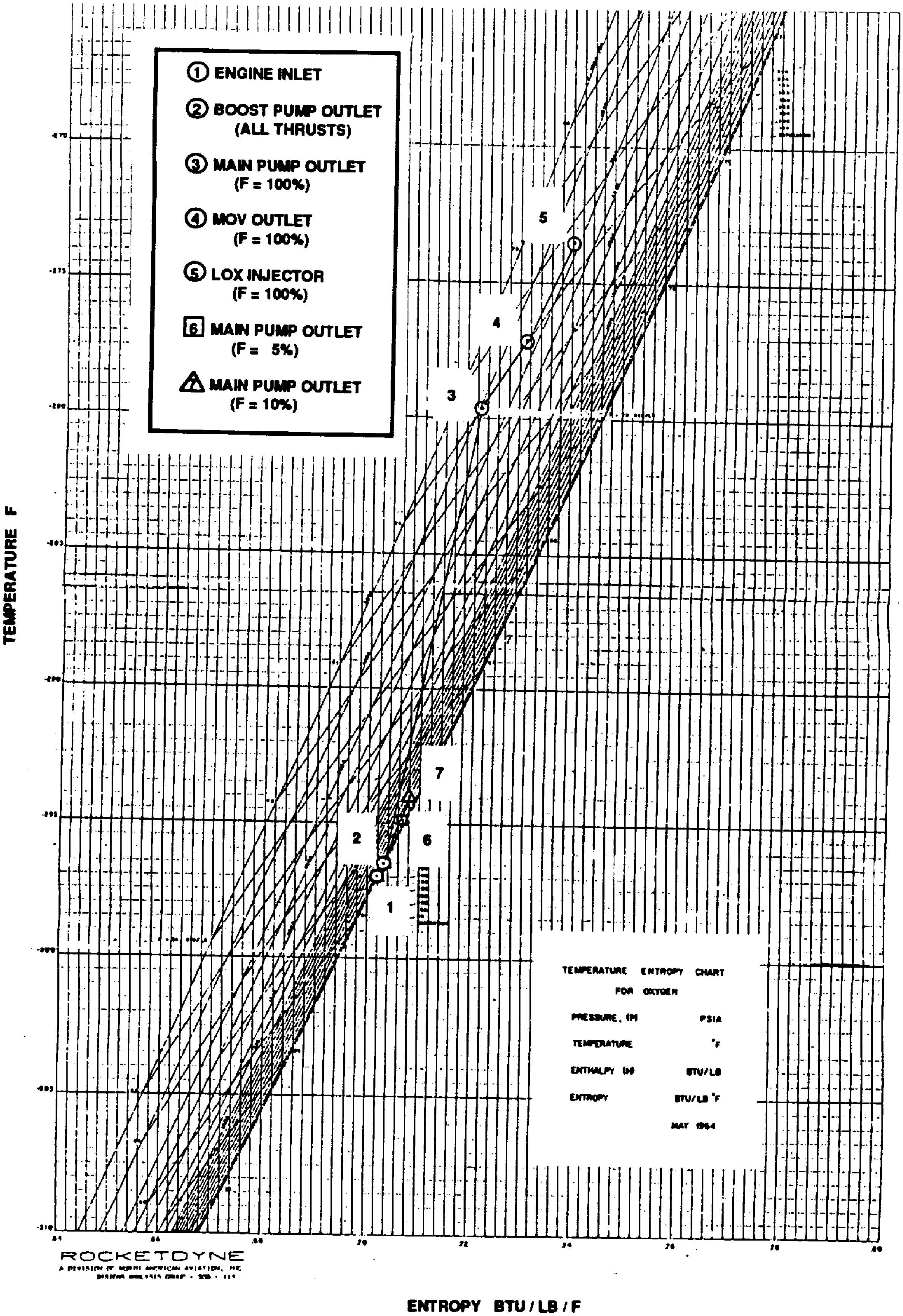


Figure 24. 20 klbf ENGINE O2 PROPERTIES

ENGINE VARIATION STUDIES
COOLANT BULK TEMPERATURE versus THRUST

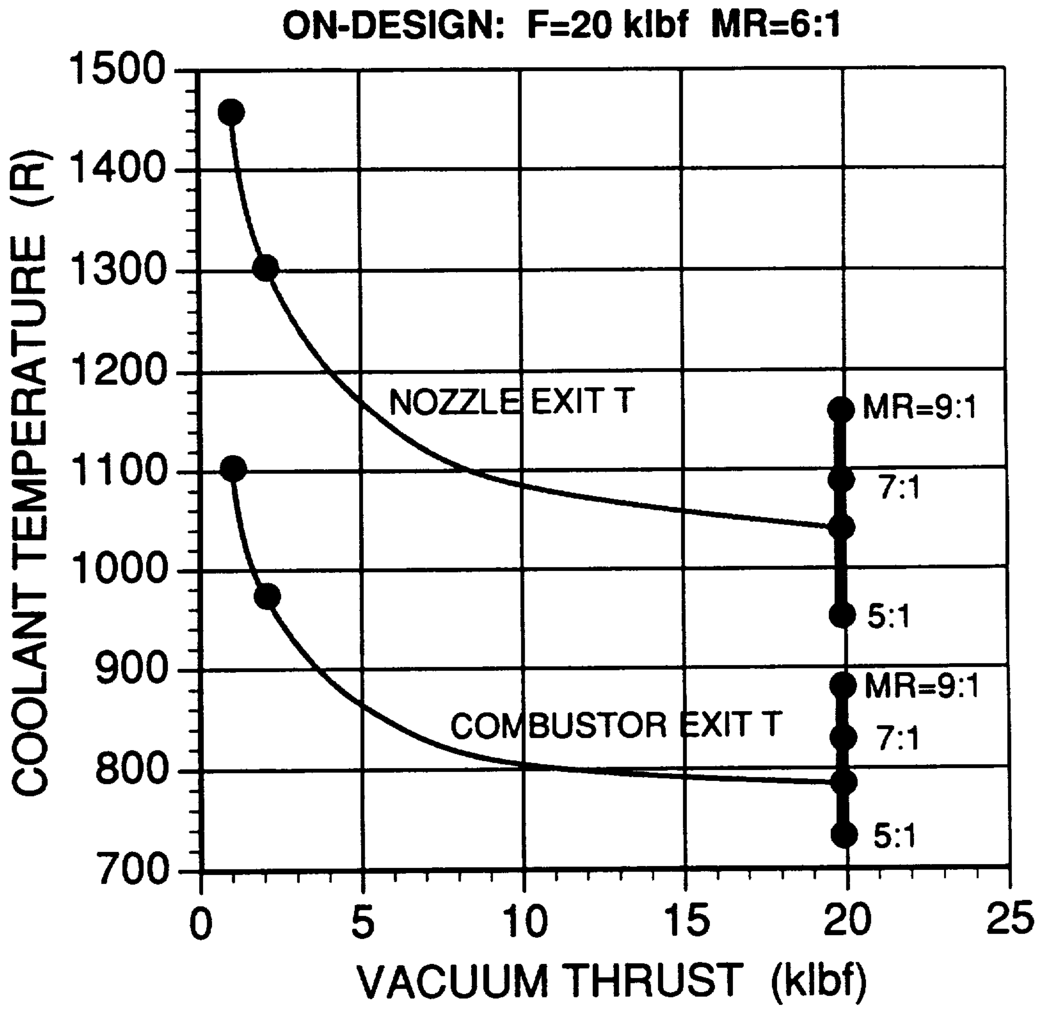


Figure 25. Engine Variation Studies Coolant Temperature versus Thrust

ENGINE VARIATION STUDIES
COOLANT BULK TEMP versus MIXTURE RATIO

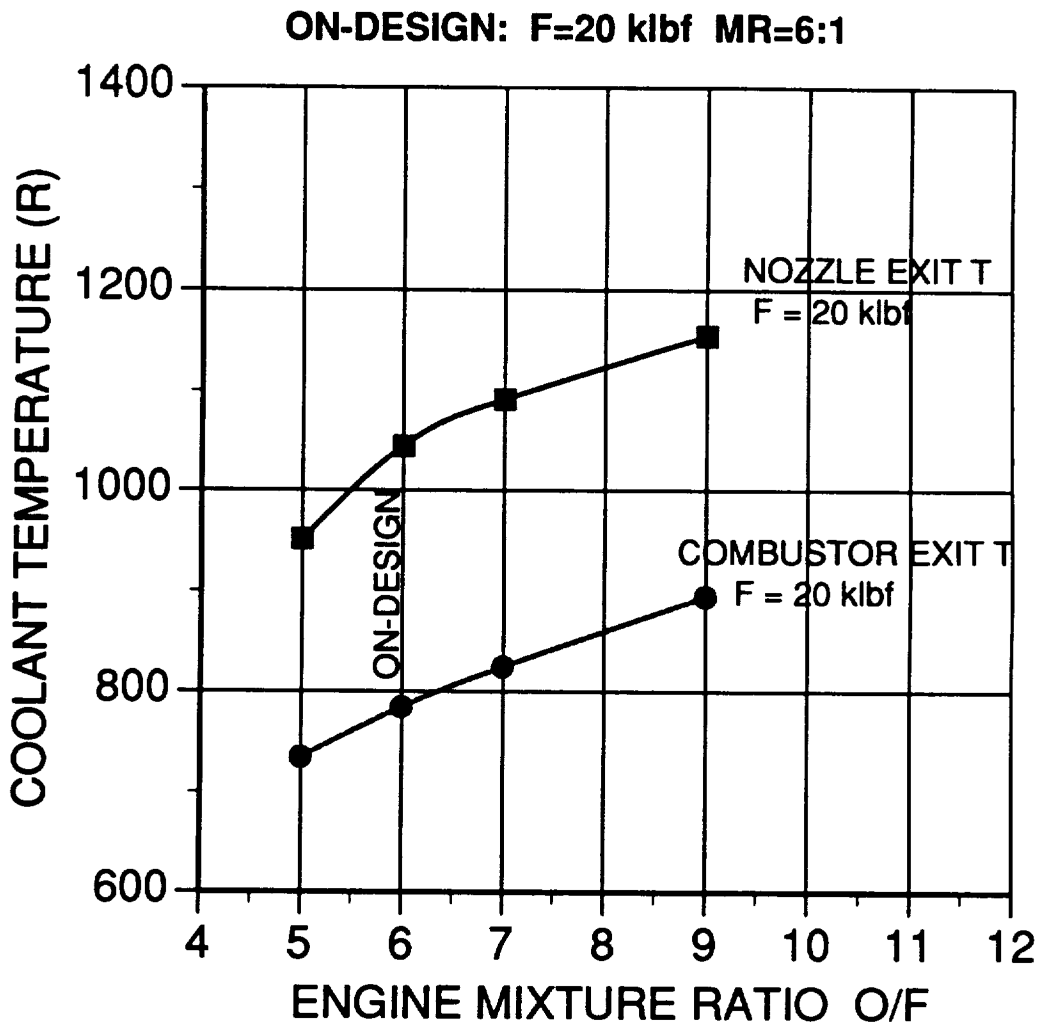


Figure 26. Engine Variation Studies Coolant Temperature versus Mixture Ratio

have been evaluated for potential combustion chamber assembly cooling problems. Analyses indicate that the combustor wall and nozzle wall temperatures are always below the established operating limits for the materials chosen. The combustor would be composed of Narloy Z copper alloy and nozzle of stainless steel A 286 tubes.

A summary of the potential problems uncovered in the engine variations study for the initial baseline configuration and proposed solutions are presented in Table 10 .

Baseline Modification

Engine System. The initial baseline engine configuration was incapable of operating off-design at an engine mixture ratio of 12:1 at full thrust. A power limit was encountered at MR=9:1, the reasons for which were outlined above.

A modification was made to the engine configuration in which the fuel turbine bypass flow was routed upstream of the LOX turbine. This allowed a transfer of power to the oxygen side of the cycle as the engine mixture ratio is increased. The schematics for the initial baseline configuration and the modified flow circuit are presented in Figures 27 and 28 respectively.

Additional power was also reserved for the oxygen T/P by increasing the LOX turbine bypass at the on-design MR = 6:1 condition. An engine cycle balance was generated for the modified configuration with additional LOX T/P bypass. This new baseline engine achieved a chamber pressure of 2215 psia and vacuum Isp of 491.6 sec. A schematic with pressure, temperature, and flowrate schedule is presented for this engine cycle balance in Figure 29 .

The impact upon on-design performance relative to the original flow configuration was a decrease in chamber pressure of 186 psi (2215 psia vs 2401 psia) and a decrease in vacuum Isp of 0.7 sec (491.6 sec vs 492.3 sec). The configurational change had only a small detrimental effect upon chamber pressure and performance because the oxygen turbopump horsepower requirements are much lower than for the hydrogen turbopump. With a low fraction of the total horsepower on the LOX side of the cycle, the extra bypass results in only a small increase in the LOX turbine pressure ratio and resultant decrease in chamber pressure.

Table 10
Engine Variation Studies Initial Baseline Problem Summary

Component	Condition	Problem	Solution
Engine System	O/F > 9:1	• Power Limit	• Parallel Turbines • Additional Fuel Turbine Bypass
Injector	5% F	• Chugging Stability	• ΔP Provided On-Design • ΔP Coupling with MOV
Valve Throttle Range	5%F, O/F ≧ 9.1 O/F ≧ 9.1	• FTBV Control Sensitivity Loss • OTBV Control Sensitivity Loss	• Parallel Valves • Sophisticated Ball Design • 3-Way TBV and OTBV
Fuel Turbopump	5% F	• Positive Slope Speed Line Instability	• Recirculated LH_2 • Actual Pump Less a Problem
	O/F ≧ 9:1	• Excessive Bearing Load	• Double Exit Volute • Refined Bearing and Seals Design • Strengthen Areas Supporting Loads
Oxidizer Turbopump	5%F, 10%F	• Positive Slope Speed Line Instability	• Recirculate LO_2
	O/F > 7:1	• Cavitation Limit Exceeded	• Increase Blade Angle Design
Lox Injector Internals	5%F, 10%F	• Flow Instabilities	• Damper, Cavitating Venturi • Depends on Heat Transfer TBD

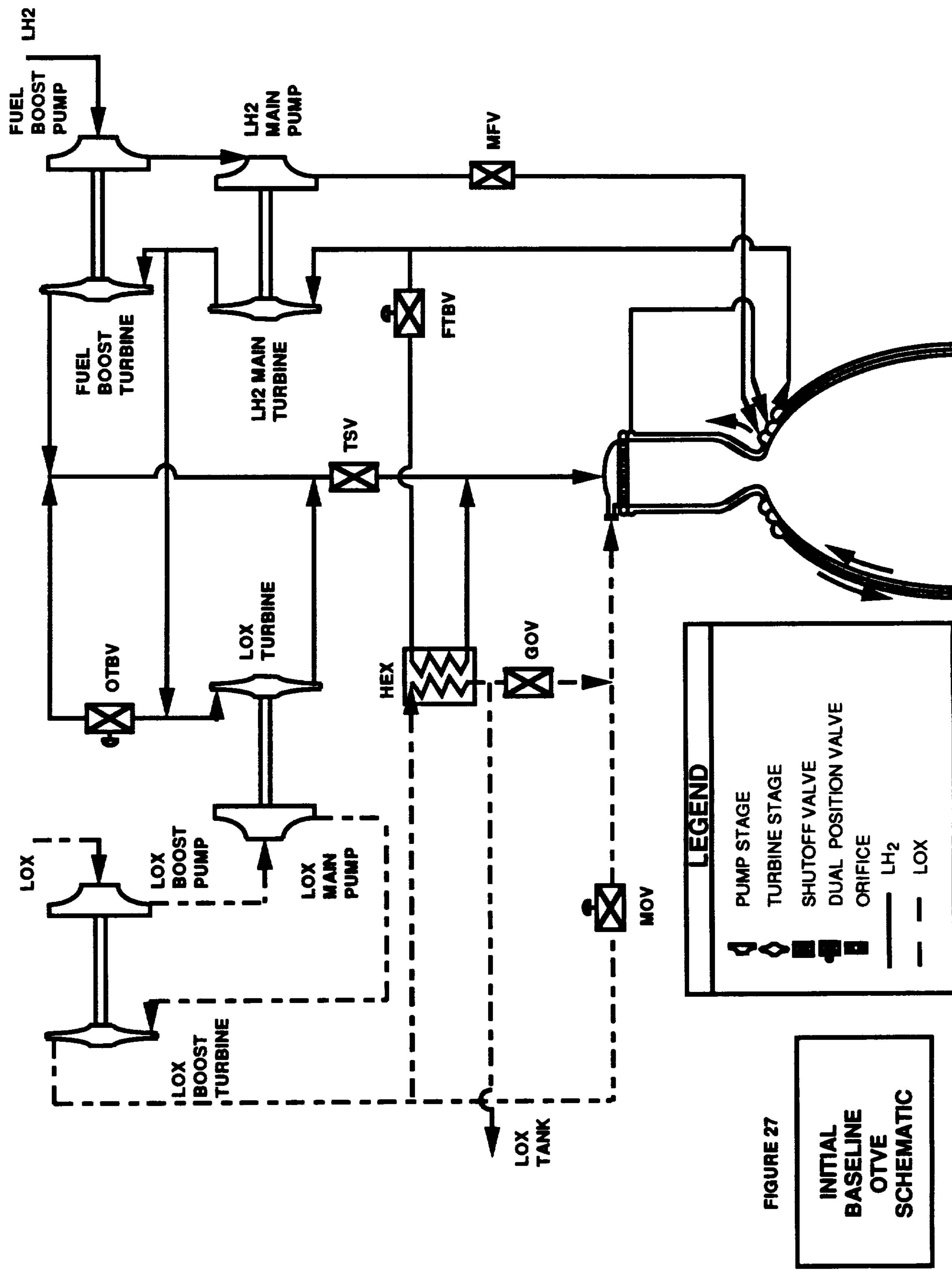


FIGURE 27

INITIAL BASELINE OTVE SCHEMATIC

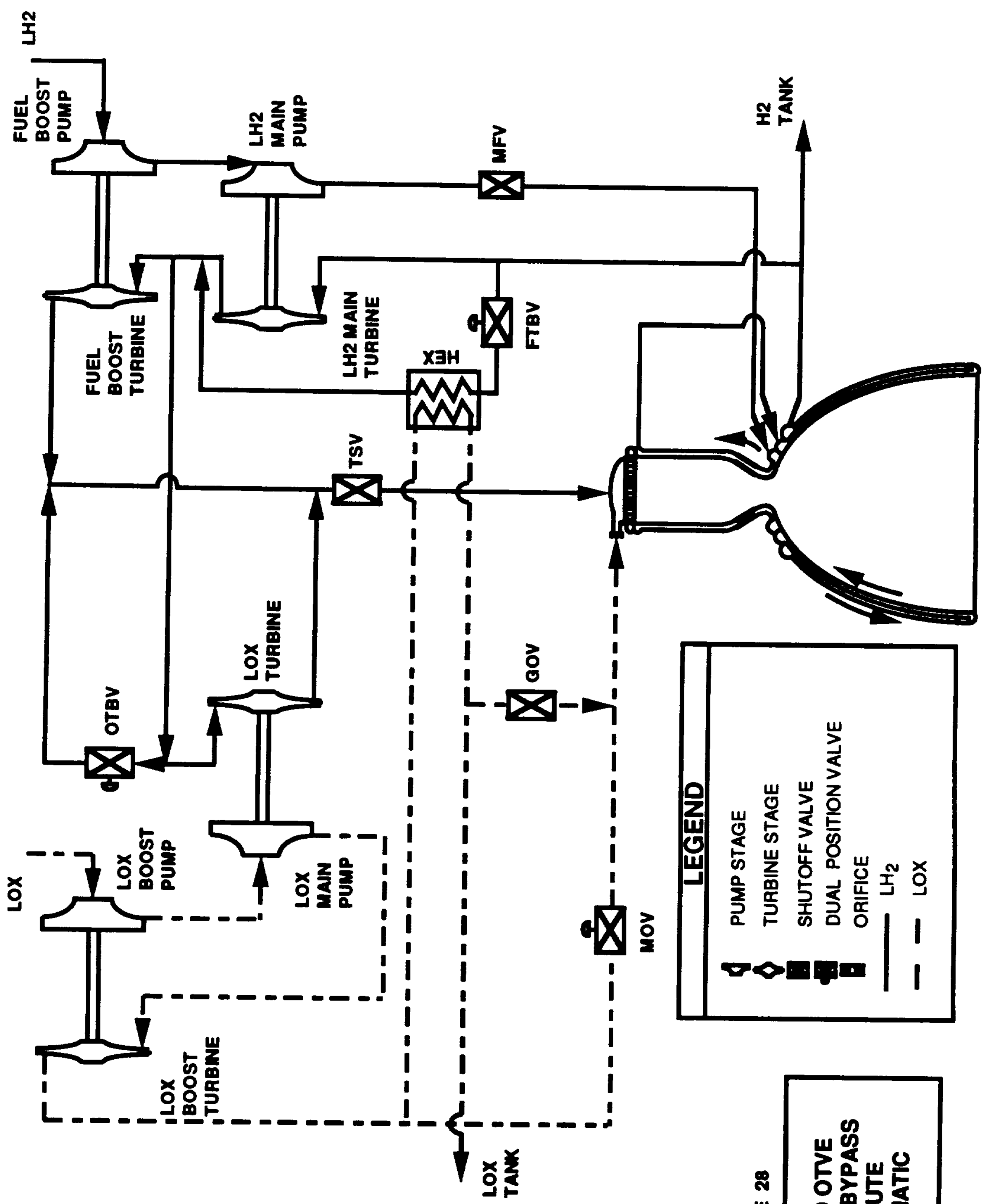


FIGURE 28

REVISED OTVE TURBINE BYPASS REROUTE SCHEMATIC

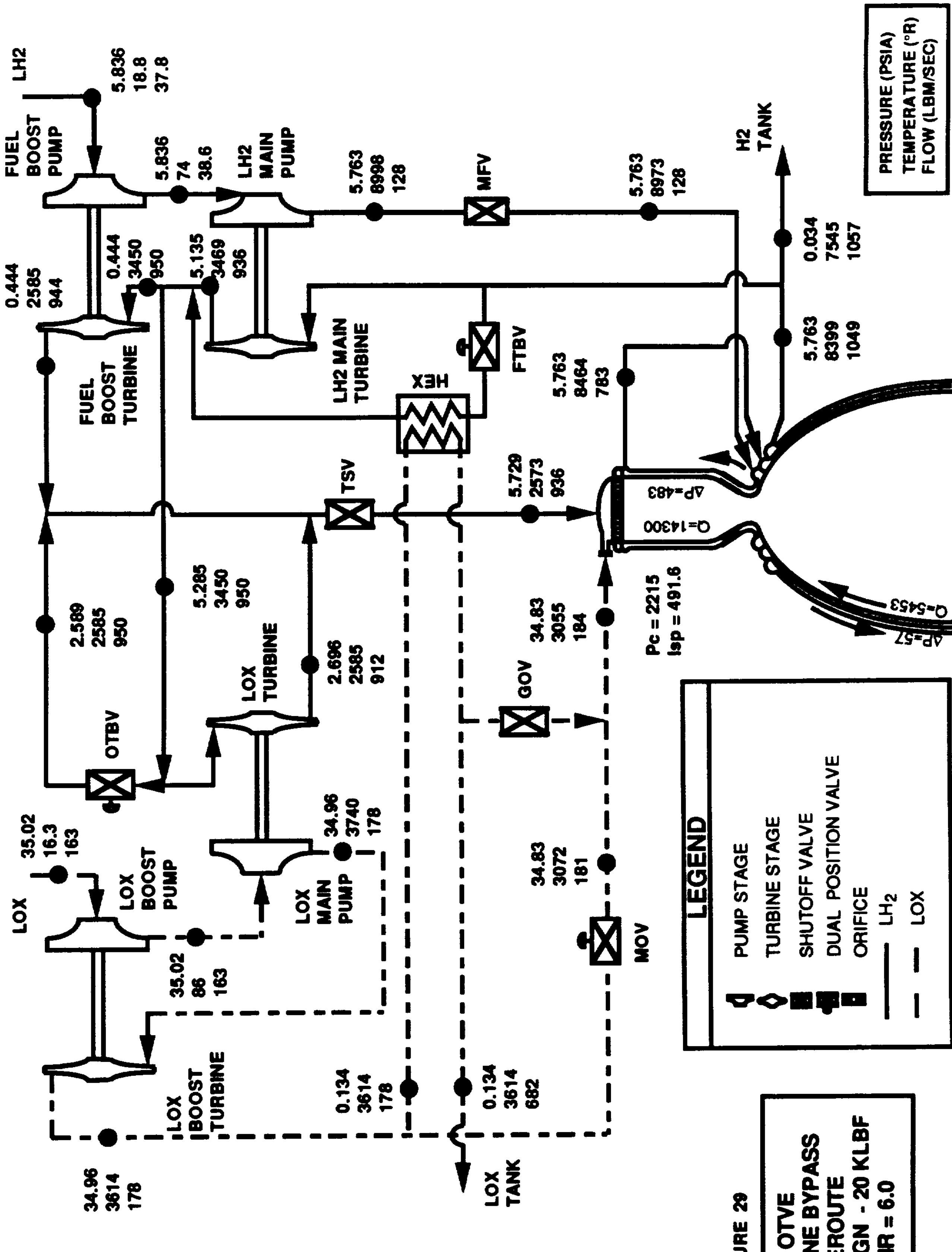


FIGURE 29

OTVE
TURBINE BYPASS
REROUTE
ON DESIGN - 20 KLBF
MR = 6.0

Off-design engine cycle balances were run for this configuration at full thrust with MR's of 5;1, 7:1, and 12:1. The new flow configuration with additional bypass flow enabled a successful power balance at the maximum MR requirement of 12:1. Off-design engine cycle balances were also generated for this engine at 10% and 5% of full thrust with a MR of 6:1. The revised flow configuration did not introduce any complications in deep throttling.

Pump Operation. Pump operating maps depicting head rise versus flowrate plots at various speeds and efficiencies were generated for the LH2 and LOX main pumps. The individual operating points were then plotted on these maps for the on-design and all off-design points.

Allowable operating limits including tip speed, cavitation, surge, boilout, and bearing load limits were also sketched on these pump maps, thus defining the allowable operating regions. These maps are provided in Figures 30 and 31 (expanded scale for deep throttling in Figures 32 and 33).

All operating points for the LH2 main pump except at 5% thrust fall in the acceptable operating region. At 5% thrust level the operating point has shifted to the left of the surge (zero slope constant speed line). This introduces the possibility of pump instability. This same circumstance was encountered in the earlier off-design studies for the initial baseline configuration. As stated before, potential solutions to this problem are pump recirculation or increase of the impeller flow coefficient.

With the baseline engine configuration a bearing load limit was reached in the LH2 pump at MR = 9:1 at full thrust. With the new turbine flow bypass schematic this problem is eliminated since the LH2 pump operates at lower discharge pressures during elevated MR conditions.

Two operating points fall outside of the acceptable region for the LOX main pump. At MR = 12:1 and full thrust, the LOX flowrate has increased to the point where the cavitation limit has been exceeded. A possible solution to this problem is to increase the inducer blade angle. This effectively shifts the cavitation limit line to the right on the pump map. A slight decrease in head coefficient may result, but this could be offset by a higher pump speed.

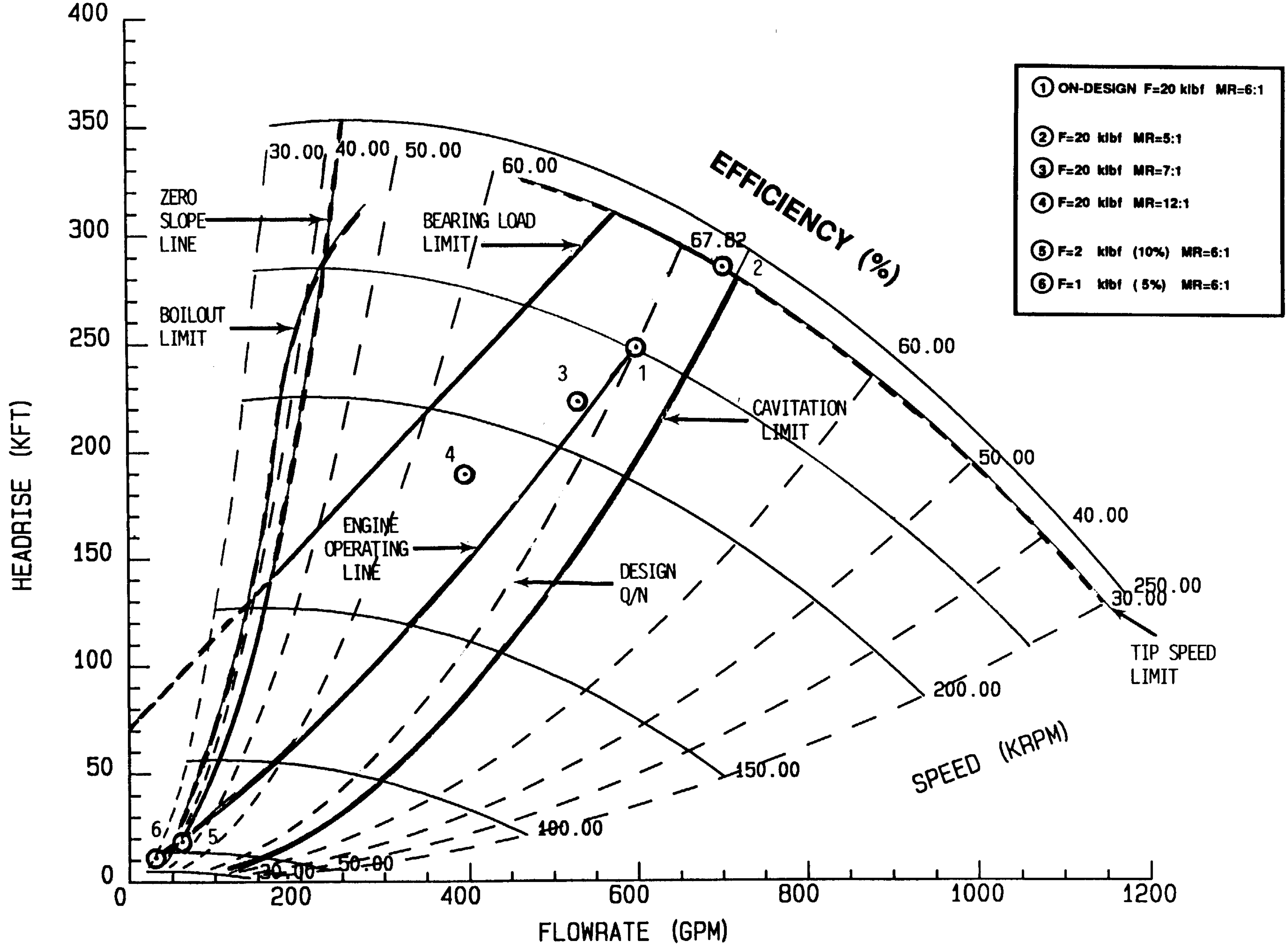


Figure 30. REROUTE 20 klbf LH2 PUMP MAP

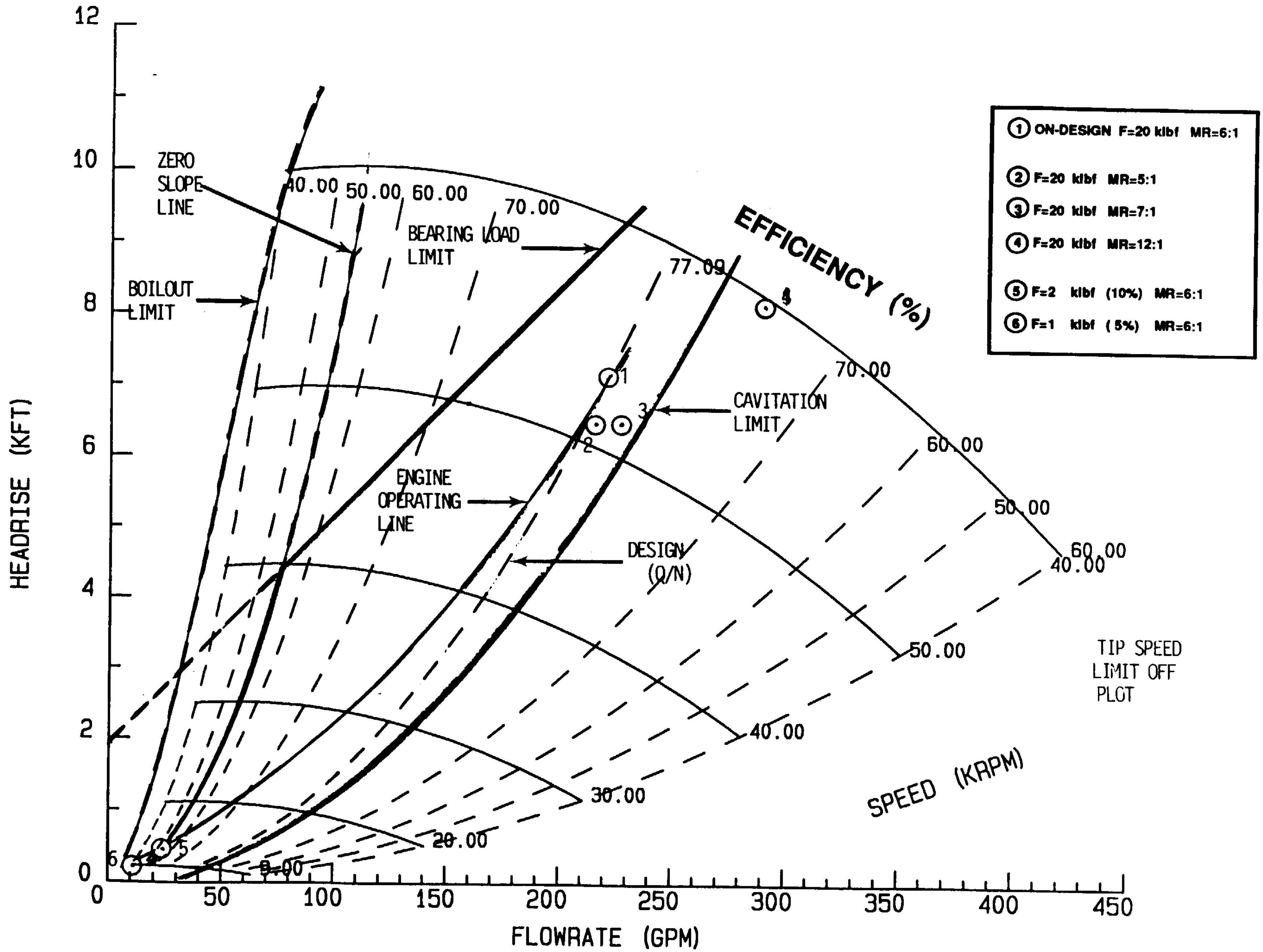


Figure 31. REROUTE 20 klbf LOX PUMP MAP

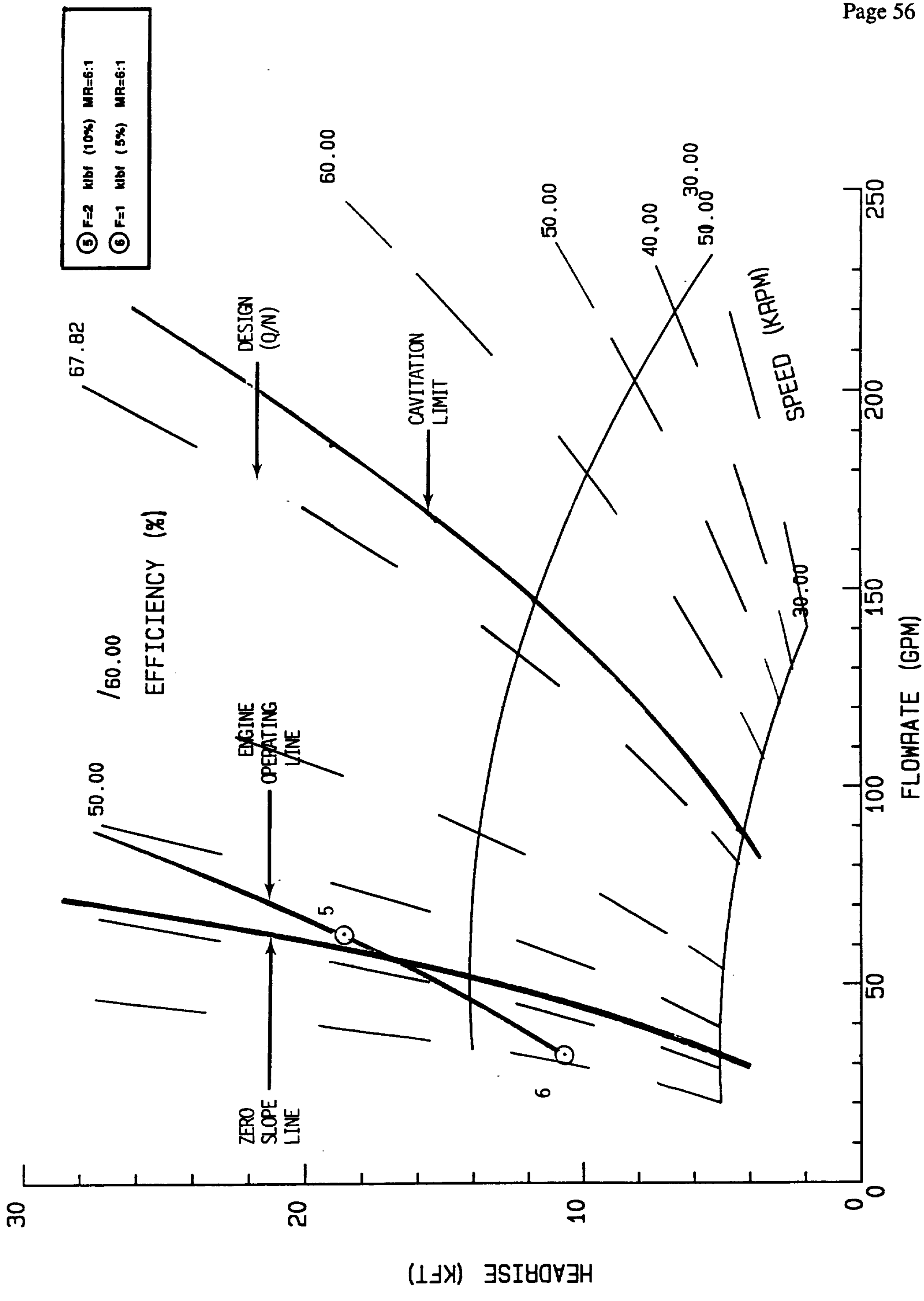


Figure 32. REROUTE 20 klbf LH2 PUMP MAP (low flow)

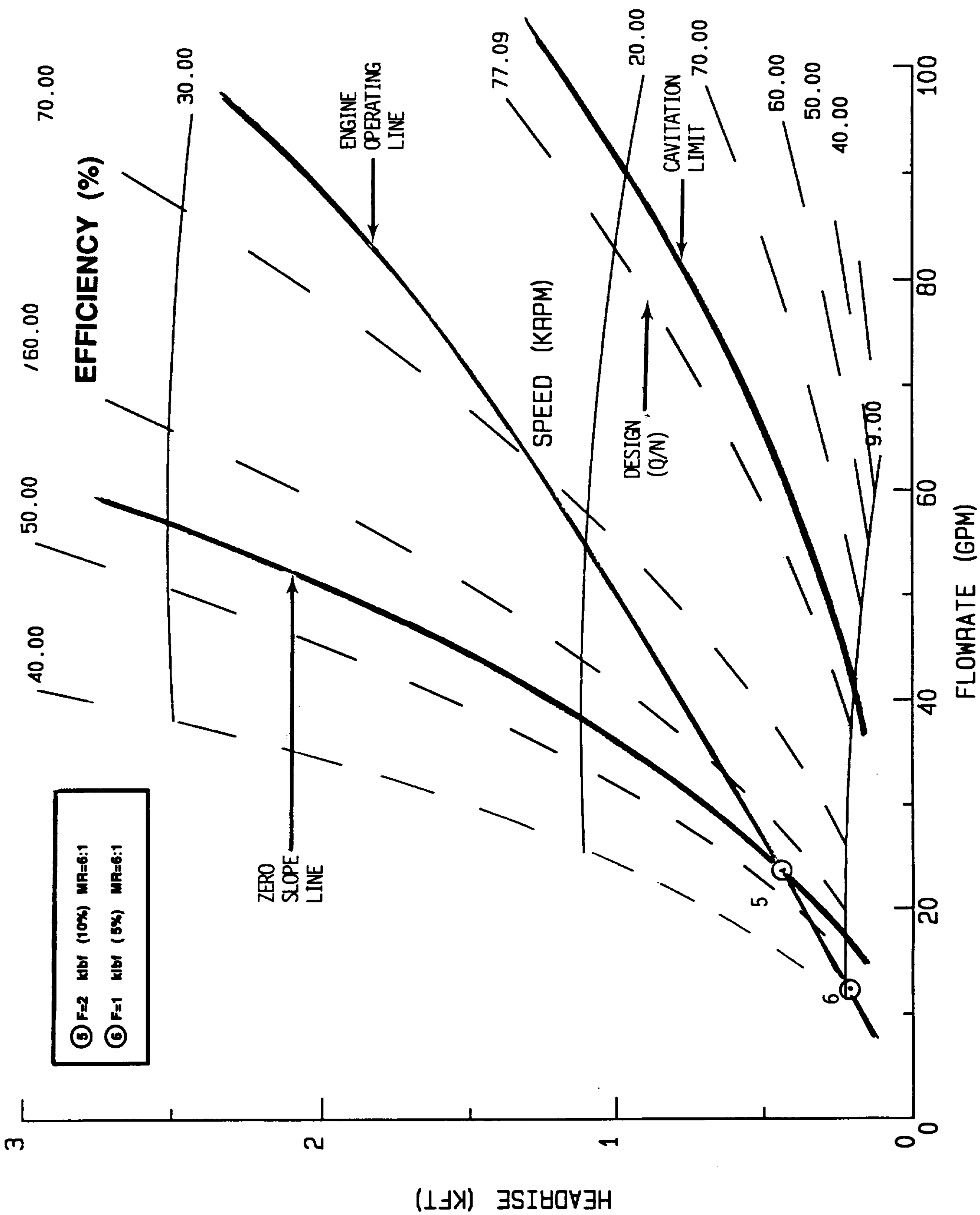


Figure 33. REROUTE 20 klbf LOX PUMP MAP (low flow)

In addition, as with the LH2 pump, throttling down to 5% thrust requires the LOX pump to operate in the potentially unstable positive slope region to the left of the surge line. The flow circuit modification has somewhat improved the throttling situation for the LOX pump in that in the original configuration the pump was operating in the potentially unstable region at 10 % thrust. The same potential solutions mentioned above for the LH2 pump apply here for the LOX pump.

Injector Chugging Stability. The on-design and off-design propellant injection pressure drops are very close to those observed in the initial baseline flow configuration, with the exception of the MR = 12:1 operating point. The LOX injection pressure drop relative to chamber pressure is the highest (most stable situation) for that additional operating point. The conclusion that sufficient pressure drop exists at all operating points (MOV pressure drop credited) to preclude injector chugging stability can also be drawn for the revised baseline configuration.

Valve Throttling Ranges. Valve summary tables are provided for the four valves in Tables 11 through 14 . These data include flowrates, pressure drops, resistances, and resistance ratios relative to the on-design position.

With the revised flow configuration the MFV remains a non-modulating valve and is not needed as an additional control element as is required with the initial baseline for high MR operation.

Unacceptably wide resistance ranges are still required for the fuel turbine bypass valve and the oxidizer turbine bypass valves, as in the original configuration. One of the potential solutions to this problem, as reviewed earlier, is the use of parallel valve arrangements.

Combustion and Feed System Stability. The operating conditions in the feed system and combustor for the revised flow configuration at the throttled conditions are essentially identical to those exhibited in the initial baseline both at on-design and all off-design operating points. Operation at full thrust and MR=12 would result in the least unstable condition since the heat transferred to the LOX in the injectors would be the minimum of all the operating points. The same observations and conclusions presented above for the initial baseline engine apply here.

Table 11 OTV - Off Design Valve Summary Revised Bypass Configuration MFV - H2						
Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2215	5.763	25	3.754	1.00
20	5.00	2291	6.860	33	3.754	1.00
20	7.00	2150	5.180	19	3.754	1.00
20	12.00	2183	3.870	10	3.754	1.00
2	6.00	226	0.610	0.33	3.754	1.00
1	6.00	115	0.306	0.08	3.754	1.00

R_{max}/R_{min} = 1.00 (Non-throttling)

Table 12
OTV - Off Design Valve Summary
Revised Bypass Configuration
MOV - Lox

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2215	34.83	542	32.10	1.00
20	5.00	2291	34.12	44	2.72	0.085
20	7.00	2150	36.04	72	3.98	0.125
20	12.00	2183	46.12	280	9.48	0.297
2	6.00	226	3.760	8	45.7	1.43
1	6.00	115	1.980	1	28.1	0.881

R_{max}/R_{min} = 16.8 < 100 (Range acceptable)

Table 13
OTV - Off Design Valve Summary
Revised Bypass Configuration
FTBV - H2

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2215	0.59	4091	9713	1.00
20	5.00	2291	0.35	5166	41829	4.31
20	7.00	2150	0.68	3440	5560	0.572
20	12.00	2183	0.09	2620	0.254 E06	26.20
2	6.00	226	0.35	156	69.7	0.00718
1	6.00	115	0.20	69	43.5	0.00448

R_{max}/R_{min} = 962 >> 100 (Range unacceptable for 20:1 F Engine
F = 20Klb MR = 5:1/F = 1Klb MR = 6:1)

R_{max}/R_{min} = 3649 >> 100 (Range unacceptable for 12:1 MR @ 100%F
F = 20Klb MR = 12:1/F = 2Klb MR = 6:1)

R_{max}/R_{min} = 5848 >> 100 (Range unacceptable for high MR/deep throttling engine)

Table 14
OTV - Off Design Valve Summary
Revised Bypass Configuration
OTBV - H2

Thrust klb	Mixture Ratio	Chamber Pressure psia	Flowrate lb/sec	Pressure Drop psid	Resistance lbf*s**2/ lbm*ft**3*in**2	Roff/ Ron
20	6.00	2215	2.568	865	68.5	1.00
20	5.00	2291	3.409	808	43.7	0.638
20	7.00	2150	2.21	806	82.4	1.138
20	12.00	2183	0.40	1153	3544	48.5
2	6.00	226	0.44	29	6.47	0.079
1	6.00	115	0.25	9	2.83	0.034

R_{max}/R_{min} = 34 < 100 (Range acceptable for 20:1 F Engine
F = 20Klbf MR = 7:1/F = 1Klbf MR = 6:1)

R_{max}/R_{min} = 614 >> 100 (Range unacceptable for 12:1 MR @ 100%F
F = 20Klbf MR = 12:1/F = 2Klbf MR = 6:1)

R_{max}/R_{min} = 1426 >> 100 (Range unacceptable for high MR/deep throttling engine)

<u>Cooling Capabilities and Limits - Nozzle and Combustor</u>. The operating conditions are also essentially identical for the combustor and nozzle cooling circuits of the two configurations, with the exception of the additional operating point at MR = 12:1 which was unattainable with the initial baseline. The combustor and nozzle wall temperatures are still below the established operating limits at the maximum MR condition. The conclusion that no cooling problems will be encountered still applies for the revised flow configuration baseline.

Vehicle Study/Engine Study Coordination

The purpose of this subtask was to provide support to studies being conducted on the vehicle and mission level by NASA directly or by their contractors. The parametric data generated in Subtask 2 was used for this purpose. Other areas addressed include technology requirements, engine cycle life requirements, and engine startup.

<u>Technology Assessment</u>

Evaluations were made of the current state of the art for space engines which resulted in definition of technologies which are needed. These technologies are categorized in Table 15 according to general mission requirements and a particular engine feature required to satisfy the mission requirement. In some cases a feature can satisfy more than one mission requirement. In some instances a feature which enhances one requirement may tend to degrade another (e.g. maintainability features may detract from reliability).

The most economical method of initially demonstrating the technology is listed. Eventually, the technologies must be demonstrated on the engine (cost effectively, in a test bed engine) and in the full system.

Many of the 'required' features are not absolute requirements at this time but may become such as the result of vehicle or mission level trade studies. The criteria to be considered in the trade studies for each feature are shown in Table 15. An assessment of the relative importance of each feature is indicated in the table.

Table 15. Required Engine Technologies

Required	Feature	Technology	Initial Demonstration	Trade Criteria	Weight
Life	5 missions (10 hr/100 CY)	T/C, T/P Life	Component test	Life, size, performance, cost	2
Life	Alt. engines for some mission phases	Alt. engine life	Component test	Life, performance	2
Life	5-year storage	Space storage of material, components	Space storage demonstration	Storability, weight, cost	1
Cap	High performance in small envelope	High P_C, area ratio	Component test	Cost, schedule	1
Cap	Deep throttling	Wide operating range components, engine	Component test	Cost, performance, reliability	1

Mission requirement (req.): Life = long life; Maint = maintainable; Rel = reliable/safe; Cap = capability
Weight: 1 = very important; 2 = significant; 3 = enhancing

Table 15. Required Engine Technologies (contd.)

Required	Feature	Technology	Initial Demonstration	Trade Criteria	Weight
Maint	Engine replacement	Low Pr. Cryo disconnects	Cryotest joint	Rel., weight, cost	2
Maint	Component Replacement	Hi press Cryo disconnects	Cryotest joint	Rel., weight, cost	1
Maint	Minimum purges	Ox. turbopump, injector	Define limits	Rel./safety, weight, cost	2
Maint	No purges	Ox. turbopump, injector	Vacuum test	Rel./safety, weight, cost	2
Maint	Maintenance ICHM	Sensors, maintenance algorithms, expert systems	Sensor/component/ engine	Rel./safety, cost	1

Mission requirement (req.): Life = long life; Maint = maintainable; Rel = reliable/safe; Cap = capability
Weight: 1 = very important; 2 = significant; 3 = enhancing

Table 15. Required Engine Technologies (contd.)

Required	Feature	Technology	Initial Demonstration	Trade Criteria	Weight
Rel.	Simplicity	Purge elimination	Comp. vacuum test	Rel., safety, weight	3
Rel.	Benign failure	Damage isolation Expander cycle	Comp. destruct. test System test	Safety, weight	1
Rel.	Redundancy Engine out Comp. out	Transient control Transient control	System test Component test	Rel., safety	1
Rel.	Life prediction	ICHM capability Sensor reliability	Algorithms	Safety, cost	2
Rel.	Alternate oper. points	ICHM controllability	Sensors, algor. Component test Life extension	Safety, cost, perf.	3

Mission requirement (req.): Life = long life; Maint = maintainable; Rel = reliable/safe; Cap = capability
Weight: 1 = very important; 2 = significant; 3 = enhancing

Table 15. Required Engine Technologies (contd.)

Required	Feature	Technology	Initial Demonstration	Trade Criteria	Weight
Maint	Previous flight data	Sensors, health prognostics, algorithms, expert syst.	Sensor/component failure tests, algorithm devel.	Rel., safety, cost	1
Maint	Robotic inspection	Miniature maneuverable TV	Earth based Zero-G simulation	Rel., safety, cost	2
Maint	Cryo leak checks	Remote leak detection	Earth based simulation	Rel., safety, cost	2
Maint	Early flight data	Sensors, health algorithms, expert syst.	Sensor/component failure tests, algorithm devel.	Rel., safety, cost	1

Mission requirement (req.): Life = long life; Maint = maintainable; Rel = reliable/safe; Cap = capability
Weight: 1 = very important; 2 = significant; 3 = enhancing

Manrating

The impacts of the requirement to be 'manrated' were assessed from design thru operation. Two areas appear to be most significant: control and safety. The astronaut ability to control the spacecraft better than ground control will be limited to those maneuvers where the crew has a better sense of the requirements than ground control. Examples are final docking (for rendezvous) and landing maneuvers. These could impact the required thrust level, thrust range, thrust sensitivity to valve positions, and minimum impulse bit (for impulse controlled engines). However, these requirements cannot be established at this time and must be evaluated when more specific and detailed designs are in place.

The other area affected by the manrating requirement is the safety issue. Safety must be distinguished from reliability because the high cost of unmanned cargo is such that reliability requirements of manned and unmanned missions may be the same. Reliability is the requirement to complete the mission function as designed. Safety is the requirement to avoid loss of crew or vehicle . For propulsion systems which feature engine-out capability, the engine requirement for safety is to avoid a catastrophic failure.

The relationships between features which enhance safety and/or reliability are as follows: 1) Features which enhance reliability generally also enhance safety, 2) A safety feature which validly shuts an engine down will improve reliability if the mission can be completed without the engine and failure to shut the engine down would have resulted in a catastrophic failure, 3) A safety feature which validly shuts an engine down will not change the reliability if the planned mission cannot be completed (although a safe return is possible), 4) A safety feature which invalidly shuts an engine down such that the planned mission cannot be completed will degrade the system reliability. This places a strong requirement on a safety system to avoid invalid shutdowns. Rugged, redundant, self-checking systems tend to satisfy this requirement.

Reliability is enhanced during the design phase by incorporating such features as simplicity, large margins, and redundancy. Failure Mode And Effects Analyses (FMEA) disclose unreliability issues which are then eliminated or minimized, as much as possible, by redesign. The use of total quality management (TQM) helps to ensure that reliability features: 1) are not overlooked, 2) can be built, 3) can be inspected, and 4) are operable.

Demonstration of reliability can be accomplished analytically or experimentally. The analytical demonstrations are not as convincing as the physical methods but are considerably less expensive. One analytic method is to utilize historic reliability data for components which are similar to those in the subject engine. These data are statistically combined to yield the engine reliability. Another, more recent, approach uses probabilistic analysis techniques.

The experimental demonstration of reliability consists of running hundreds of mission-simulating tests. The number of tests becomes very large when high reliabilities and confidence levels are required. An experimental method related to reliability demonstration is called "limits testing" which is used to demonstrate either actual failure points or successful operation at conditions well beyond the design point. The results indicate the available margins and provide an indication of the reliability at the design point. Relatively few tests are required for this demonstration but significant hardware costs may be incurred. The corresponding approach to demonstrating life of reusable engines is called the "fleet leader" method and consists of successfully operating two or more engines for twice the life of the flight engines.

Safety analyses (Hazard Analysis) build on the FMEA to define and attempt to mitigate safety issues. Methods of enhancing safety, besides reliability improvement, include failure containment and safe shutdown techniques. Safety demonstrations follow along the same lines as reliability demonstrations.

Redundancy is an approach to improve reliability and safety. A propulsion system configuration incorporating engine redundancy would consist of two engines for maximum reliability. However, a short stage, with a low center of gravity, would require excessive gimballing for the single surviving engine to direct its thrust through the vehicle center of gravity. An alternate configuration incorporating engine out capability would be a four engine square configuration. If one engine fails the opposing engine is also shut down so that the remaining two engines can fire parallel to the vehicle axis thereby avoiding excessive gimbal angles. The vehicle or trajectory must be configured such that the mission can be accomplished with only two engines.

Another alternative arrangement of three thrust chamber assemblies in an in-line arrangement fed by common manifolds but having individual propellant control valves would be advantageous, reference Figure 34. These advantages are improved reliability,

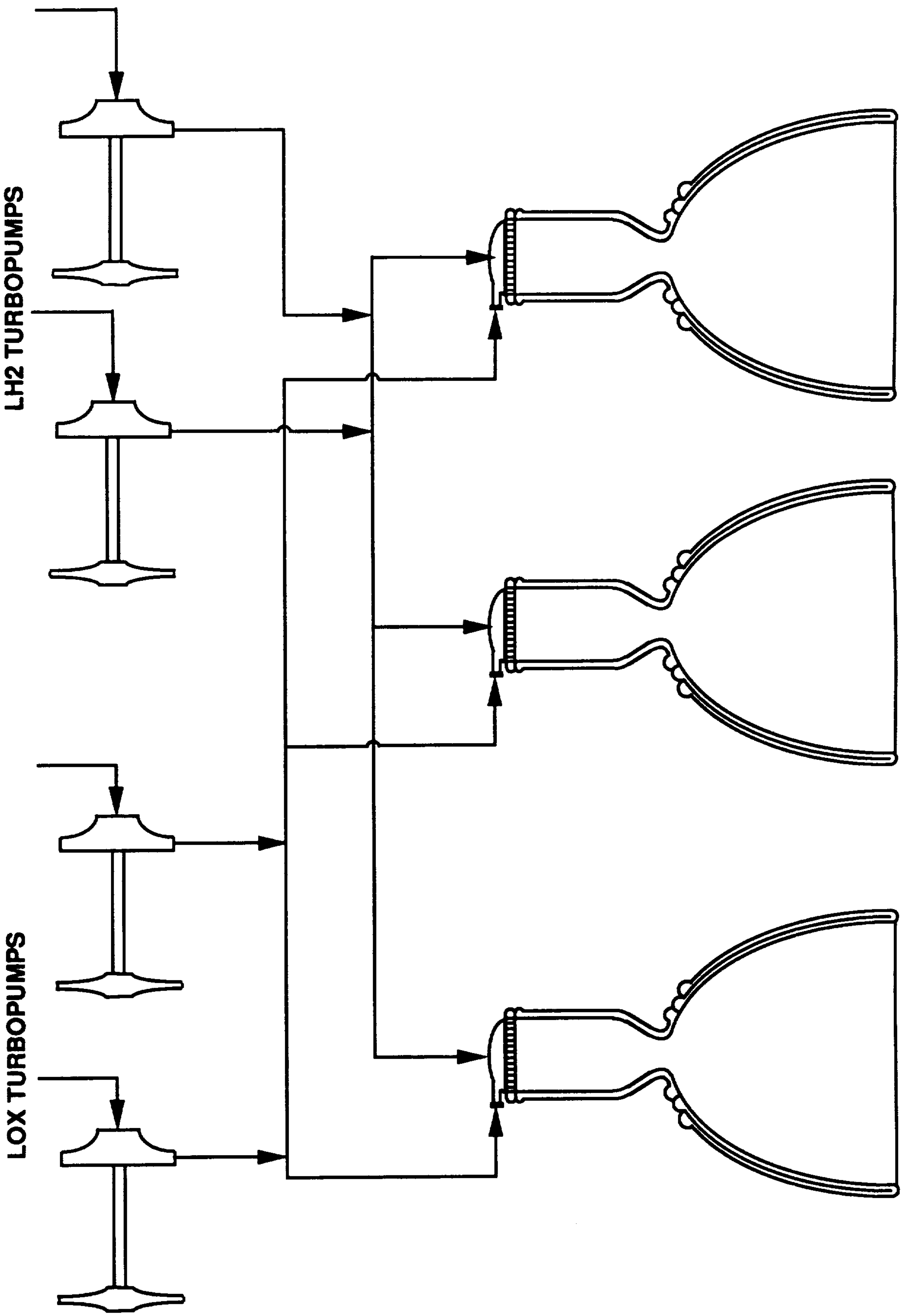


FIGURE 34
INTEGRATED PROPULSION SYSTEM - SIMPLIFIED SCHEMATIC
THREE THRUST CHAMBER / TWO TURBOPUMP SET CONFIGURATION

reduced weight and cost, higher turbopump efficiency, and perhaps improved throttling. If the center chamber were to fail, the two outer chambers would provide power. If one of the outer chambers were to fail, it and the other outer chamber would be shut down, leaving the center chamber to power the vehicle. This alternative requires a combination of up-thrust engine capability and/or ability of the vehicle to complete the mission using the single chamber in its high thrust mode.

Two or three turbopumps would be operated in parallel such that a failure would result in operation with the remaining turbopump(s). The two pump design requires a doubling of the pump flows to compensate for a failure. The three pump configuration requires a 50% increase in flows if full thrust is maintained.

Reliability would be improved (relative to the four engine configuration) by: 1) having fewer components in the nominal configuration which are subject to failure, 2) by the major components operating nominally at conditions (pressures, speeds, and flows) which are significantly below their design points, 3) allowing for multiple failures of major components (i.e. a thrust chamber and a turbopump could both fail and the propulsion system continue to operate).

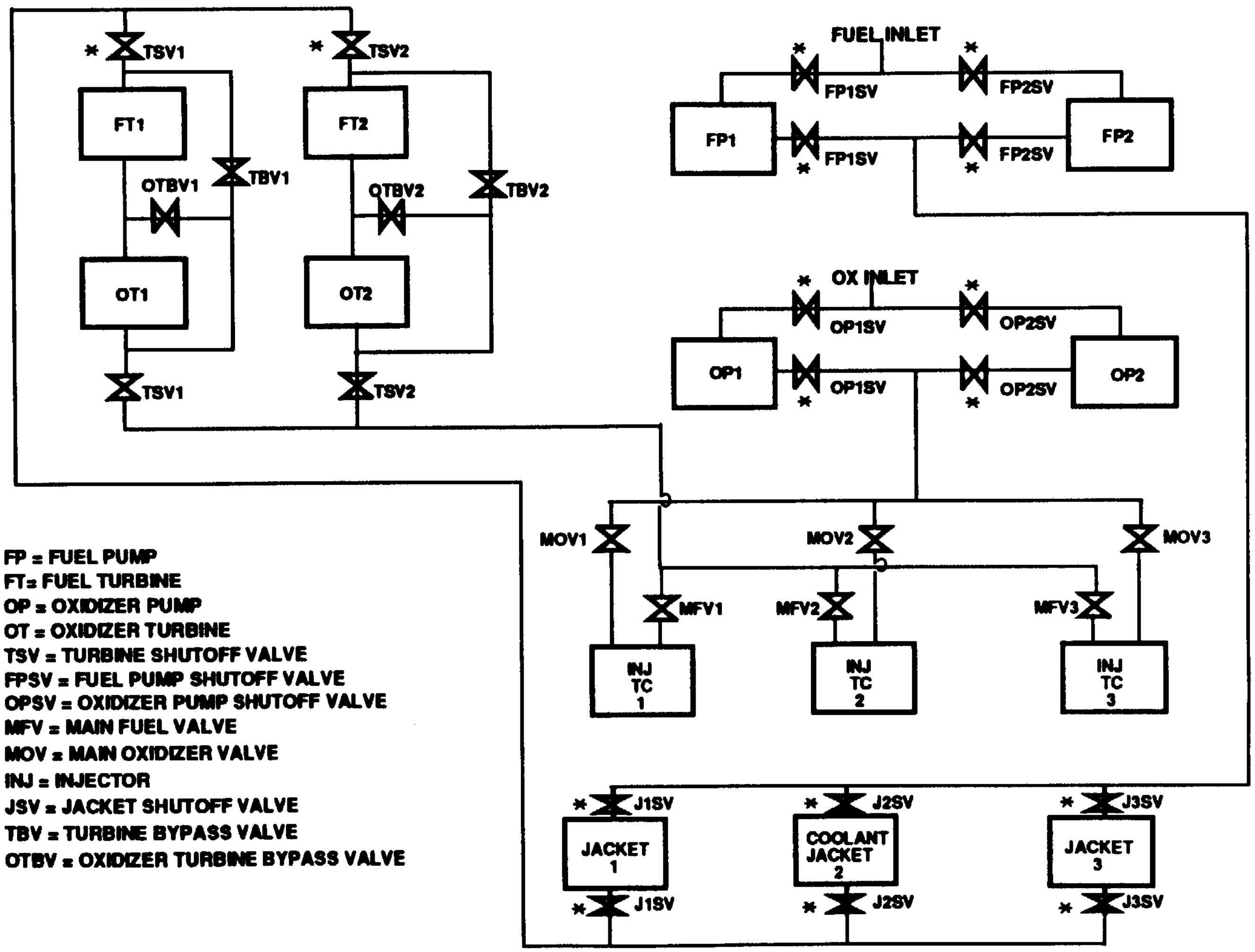
The use of three thrust chambers and two or three turbopumps instead of four of each component in the four engine square configuration, would result in cost savings and improved efficiencies because of the larger sizes. Additional studies are needed to quantify these benefits and possibly reveal potential problem areas. A schematic showing the pumps, turbines, injector-thrust chamber, and coolant jackets is presented in Figure 35. Several additional valves are required to isolate failed components. Squib actuated valves would be used because of their high reliability and low cost.

Engine Cycle Life Requirements

The various maneuvers of a typical Lunar mission for which use of the main propulsion system has been suggested are listed in Table 16. A cluster of four 20Klbf thrust engines which have the capability of operating in pumped idle and tank head idle modes is assumed. Typical velocity increments for each maneuver are shown in the table. The velocity increments, vehicle weight during the maneuver, and total applied thrust define the burn times for each of the maneuvers.

FIGURE 35

INTEGRATED PROPULSION SYSTEM
TWO TURBOPUMPS, THREE THRUST CHAMBERS



FP = FUEL PUMP
FT= FUEL TURBINE
OP = OXIDIZER PUMP
OT = OXIDIZER TURBINE
TSV = TURBINE SHUTOFF VALVE
FPSV = FUEL PUMP SHUTOFF VALVE
OPSV = OXIDIZER PUMP SHUTOFF VALVE
MFV = MAIN FUEL VALVE
MOV = MAIN OXIDIZER VALVE
INJ = INJECTOR
JSV = JACKET SHUTOFF VALVE
TBV = TURBINE BYPASS VALVE
OTBV = OXIDIZER TURBINE BYPASS VALVE

* NEW VALVES

Table 16

Lunar Transfer Vehicle
Engine Duration Requirements

Maneuver	Delta V, fps	Oper. Mode	No. of Eng.	F, Klbf	Dur., sec.
Trans lunar injection	10824	Nom.*	4	80	1085
Midcourse corrections (2)	32.8 Total	P.I.*	2	5	35
		T.H.I.	4	.24	750
Lunar orbit insertion	3608	Nom.*	4	80	218
Trans Earth injection	3608	Nom.*	2	40	113
		P.I.	4	10	900
Midcourse corrections (2)	32.8 Total	P.I.	2	5	7.0
		T.H.I.*	4	.24	150
Pre-entry correction	19.7	P.I.	2	5	4.1
		T.H.I.*	4	.24	85
Leo circularization	1016	Nom.	2	40	13.5
		P.I.*	2	5	100

Notes:

Nom. = Full Thrust = 20Klbf/Eng.
P.I. = Pumped Idle = 2.5K/Eng.
T.H.I. = Tank Head Idle = 0.6K/Eng.

* = Preliminary Recommended Mode

Some of the maneuvers (i.e. the correction maneuvers) have velocity requirements which are so small that impulse variations in the main propulsion system preclude using it for these maneuvers. Using the reaction control system or only two of the four main engines in one of the idle modes would be a better approach. Operating the engines in the idle modes would significantly extend the life of the engines. Using half of the engines at a time would double the life of the propulsion system. The LEO circularization (after aerobraking) maneuver could use two engines in pumped idle or full thrust modes.

The transearth injection maneuver could use four engines in pumped idle mode or two engines at full thrust. Four engines operating at full power are most suited for providing the power for the translunar injection and lunar orbit insertion maneuvers.

The conclusion is that the number of full power life cycles required of the main engines can be fairly small due to use of: 1) other engines (e.g. RCS), 2) low power on the main engines, or 3) only two of the four main engines.

Engine Start

Engine startup comparisons were made for space and lunar surface conditions and for engines with and without zero-NPSH or tank head idle mode capabilities.

Space Start. Autogenous pressurization was assumed for all cases. Propellants would be transferred to the engines to begin the chilling process. The Tank Head Idle (THI) operating mode would be used, if available, to chill the feed system and engine. In this mode the pumps are not rotating and the engine is pressure fed from the tanks. Chamber pressure is low enough and mixture ratio may be biased such that the engine can operate safely with liquid or vapor propellants while chilling the system. If the THI mode is not available then another propulsion system (e.g. the reaction control system) must be used to settle the propellants. Furthermore, the system must be chilled by recirculating propellants (recirculation pumps and power required) or by venting propellants thru the engine which represents a loss of propellants.

After chilling, the pumps would begin to operate at low speed (pumped idle mode) to allow the engine to operate at a sufficiently high pressure to provide gaseous propellants to the tanks for pressurization. An evaluation of operating parameters during the pumped idle mode can be used as a condition monitoring check to help determine the flight readiness of the system. If zero-NPSH pumps are used, the engine can proceed rapidly to mainstage. Pumps which require a positive NPSH will be lighter but result in a longer transient to mainstage operation.

Lunar Surface Start. The positive aspect of starting on the Lunar surface is that the gravity force is available to position the propellants and provide a slight NPSH increment to the pumps. This introduces the negative aspect of starting on the Lunar surface. Initial missions will land and take off from unprepared sites. Therefore, there is a risk of damage to the spacecraft by debris thrown up by the rocket exhaust while the vehicle is on or near the ground. The risk is reduced by minimizing the duration of the start transient. The start transient duration can be reduced by using helium to pressurize the tanks to a high enough pressure to avoid having to rely on the autogenous tank pressurization during the start sequence. Alternatively, zero-NPSH pumps with an attendant weight penalty can be used to accomplish the same objective.

The same concern about damage from eject may preclude using the tank head idle mode for chilling the feed system and pumps. Studies are needed to determine the debris carrying capability of the exhaust during the low pressure THI mode. Recirculation and bleeding propellants from downstream of the pumps is a possible solution. However, this solution entails the potential problem of accumulation of an explosive propellant mixture in the vicinity of the vehicle. Combining and burning the bleeds non-propulsively may be required.

RECOMMENDATIONS

Engine Requdirement Variation Studies

The baseline engine was selected to optimize engine performance and weight at the design point. Operation of the engine at the extreme off-design points investigated disclosed potential problems with some of the components. Modifications to the components or the engine configuration were suggested to alleviate these problems. One such configuration change (rerouting the turbine bypass line) was made to enable operation at MR=12.

It is recommended that the other suggested engine and component modifications be incorporated into the engine model to assess the operability and performance of the engine at the on-design and off-design conditions. Dynamic, as well as static modelling, is recommended to permit determination of system response and stability of the baseline and modified systems.

Vehicle Support

Mission and vehicle options are being studied by NASA for Lunar and Mars missions. These options include assessments and tradeoffs involving various engine configuration and operating point options. Engine analyses should be continued to assure that the options are evaluated using accurate engine data.

1. Report No. NASA 187215	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle ORBIT TRANSFER ROCKET ENGINE TECHNOLOGY PROGRAM-FINAL REPORT, ADVANCED ENGINE STUDY, TASK D.6		5. Report Date February 28, 1992 6. Performing Organization Code
7. Author (s) C. M. ERICKSON		8. Performing Organization Report No. RI/RD 90-180
9. Performing Organization Name and Address ROCKETDYNE DIVISION, ROCKWELL INTERNATIONAL 6633 Canoga Avenue Canoga Park, CA 91303		10. Work Unit No. RTOP 506-42-21, TASK YOS2582 11. Contract or Grant No. NAS3-23773
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS & SPACE ADMINISTRATION Washington, DC 20546		13. Type of Report and Period Covered Final report 14. Sponsoring Agency Code

15. Supplementary Notes

Program Manager: G. P. Richter, NASA-Lewis Research Center; Cleveland, OH

16. Abstract

In Task D.6 of the Advanced Engine Study, three primary subtasks were accomplished: 1) Design and Parametric Data, 2) Engine Requirement Variation Studies, and 3) Vehicle Study/Engine Study Coordination.

Parametric data were generated for vacuum thrusts ranging from 7500 lbf to 50000 lbf, nozzle expansion ratios from 600 to 1200, and engine mixture ratios from 5:1 to 7:1. the Failure Modes and Effects Analysis (FMEA) driven baseline design generated in Tasks D.4 and D.5 was used as a departure point for these parametric analyses. These data are intended to assist in vehicle definition and trade studies.

In the Engine Requirements Variation Studies, the individual effects of increasing the throttling ratio from 10:1 to 20:1 and requiring the engine to operate at a maximum mixture ratio of 12:1 were determined. Off-design engine balances were generated at these extreme conditions and individual component operating requirements analyzed in detail. Potential problems were identified and possible solutions generated.

In the Vehicle Study/Engine Study coordination subtask, vehicle contractor support was provided as needed, addressing a variety of issues uncovered during vehicle trade studies. This support was primarily provided during Technical Interchange Meetings (TIM) in which Space Exploration Initiative (SEI) studies were addressed.

17. Key words (Suggested by author (s))	18. Distribution Statement
Hydrogen/Oxygen Engine Deep Throttling Hydrogen/Oxygen Technology High Mixture Ratio High Pressure Pumps/combustion Orbit Transfer Vehicle High Area Ratio Nozzles Expander Cycle Engine	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	76	

* For sale by the National Technical Information Service, Springfield, Virginia 22151