Report 32999-F

ORBIT TRANSFER VEHICLE (OTV) ENGINE

PHASE "A" STUDY

FINAL REPORT

VOLUME II: STUDY RESULTS

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ORBIT TRANSFER VEHICLE (OTV) ENGINE PHASE "A" STUDY

FINAL REPORT
VOLUME II: STUDY REPORTS

Contract NAS 8-32999

Prepared for

National Aeronautics and Space Administration
George C. Marshall Space Flight Center
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FOREWORD

This final report is submitted for the Orbit Transfer Vehicle (OTV) Engine Phase "A" Study per the requirements of Contract NAS 8-32999, Data Procurement Document No. 559, Data Requirement No. MA-05. This work was performed by the Aerojet Liquid Rocket Company for the NASA-Marshall Space Flight Center with Mr. Dale H. Blount, NASA/MSFC, as the Contracting Officer Representative (COR). The ALRC Program Manager was Mr. Larry B. Bassham and the Study Manager was Mr. Joseph A. Mellish.

The study program consisted of parametric trades and system analysis which will lead to conceptual designs of the OTV engine for use by the OTV systems contractor.

The technical period of performance for this study was from 10 July 1978 to 4 June 1979.

The following Aerojet personnel contributed significantly to the study effort and the final report:

ALRC

K. L. Christensen - Systems Analysis
R. L. Ewen - Heat Transfer Analysis
J. E. Jellison - Structural Analysis
A. V. Lundback - Controls Analysis
R. B. Lundgreen - Performance Analysis
J. C. Mahon - Cost Analysis
R. L. Sabiers - Turbomachinery Analysis
H. W. Valler - Reliability and Safety Analysis

ASC

L. F. Field - Maintenance and Refurbishment Planning

FOREWORD (cont.)

The final report is submitted in three volumes:

- Volume I - Executive Summary
- Volume II - Study Results
- Volume III - Study Cost Estimates

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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 this 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 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 development, costs, 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 1979.

An assessment of the above factors will be made by the NASA to determine the candidate approaches for matching the OTV concepts to mission options within resource and schedule requirements. This study provides the necessary data on OTV engine concept(s) based upon 1980 technology which is required to objectively select, define, and design the preferred OTV engine, and was conducted in very close concert with the NASA.

I, Introduction (cont.)

B. ORBIT TRANSFER VEHICLE (OTV) CHARACTERISTICS

The Orbit Transfer Vehicle (OTV) is planned to be a manned, reusable cryogenic upper stage to be used with the Space Transportation System. Initial Operational Capability (IOC) is 1987 and the design mission is a four-man, 30-day sortie to geosynchronous orbit.

The required round trip payload to geosynchronous orbit is 13,000 lbm, and the weight of the OTV, with propellants and payload, cannot exceed 97,300 lbm. An Orbiter of 100,000 lbm payload capability is assumed, however, the OTV must be capable of interim operation with the present 65,000 lbm Orbiter. The cargo bay dimensions of the 100,000 lbm-Orbiter are assumed to be the same as the 65,000 lbm Orbiter, i.e., a cylinder 15-feet in diameter and 60-feet in length. The OTV cannot exceed 34 feet in length. The OTV is to be Earth-based and will return from geosynchronous orbit for rendezevous with the Orbiter. Both Aeromaneuvering Orbit Transfer Vehicles (AMOTV) and All-Propulsive Orbit Transfer Vehicles (APOTV) are considered. These vehicles are described in NASA Technical Memorandum TMX-73394 (Reference 1).

C, GUIDELINES

The following study guidelines were provided by NASA/MSFC in the contract Statement of Work (SOW) and were used in the conduct of the study program.

1. All engine designs and characteristics will be compatible with requirements and schedules contained in the SOW, and will be based on 1980 technology.

2. Dimensional allowance will be within Shuttle payload bay specifications including dynamic envelope limits. (This does not preclude extendible nozzles.)

3. The engine and OTV will be designed to be returned to Earth in the Shuttle and reused; reusability with minimum maintenance/cost for both unmanned and manned missions is a design objective.

4. The OTV engine shall be designed to meet all of the necessary safety and environmental criteria of being carried in the Shuttle payload bay and operating in the vicinity of the manned Shuttle.

4. Cost, unless otherwise specified, shall be expressed in FY 1979 dollars.

6. Structural Design Criteria.

   The following minimum safety and fatigue life factors shall be utilized. It is important to note that these factors are only applicable to designs whose structural integrity has been verified by comprehensive structural testing which demonstrates adherence to the factors specified below. Where structural testing is not feasible more conservative structural design factors will be supplied by the procuring agency.

   a. The structure shall not experience gross (total net section) yielding at 1.1 times the limit load nor shall failure be experienced at 1.4 times the limit load. For pressure containing components, failure shall not occur at 1.5 times the limit pressure.

b. Limit load is the maximum predicted external load, pressure, or combination thereof expected during the design life.

c. Limit life is maximum expected usefulness of the structure expressed in time and/or cycles of loading.

d. The structure shall be capable of withstanding at least four times the limit life based on lower bound fatigue property data.

7. Components which contain pressure shall be pressure tested at 1.2 times the limit pressure at the design environment, or appropriately adjusted to simulate the design environment, as a quality acceptance criteria for each production component prior to service use. A higher proof test factor shall be used if required by fracture mechanics analysis (see 8.b.).

8. Fracture mechanics analysis shall be accomplished to:

a. Verify that the maximum defect that is possible after final inspection and/or proof testing will not grow to critical size in 4 times the design life of the engine.

b. Establish the proof test pressure/load factor necessary to analytically guarantee 4 times the engine design life.

c. Establish a list of fracture critical parts. A part is fracture critical if unusual (non-routine) processing must be applied to insure that the requirement described in 8.a. is met.

9. The engine effects on OTV stage performance and weight will be considered in trade studies and systems analysis. A ΔV margin of 3% and an inert weight margin of 10% will be used in determining the OTV performance. The mission velocity requirements are contained in NASA TMX-73394.

10. The nominal program mission model contained in NASA TMX-73394 shall be used for both the APOTV and AMOTV to perform the engine program cost analysis.

D. METHOD OF APPROACH

The study was composed of seven major technical tasks and a reporting task. The overall study logic flow diagram depicting these tasks, their interrelationships, and study inputs and outputs is shown on Figure 1.

This study was a logical extension of earlier studies such as, the Orbit-to-Orbit Shuttle Engine Design Study (OOS), the Space Tug Storable Engine Study and the Design Study of RL-10 Derivatives as well as the other studies listed on Figure 1. The data, analyses and results of these previous studies were used and updated to meet the OTV requirements wherever possible. This resulted in a cost effective study program by permitting the study funds to be concentrated upon the new major issues.

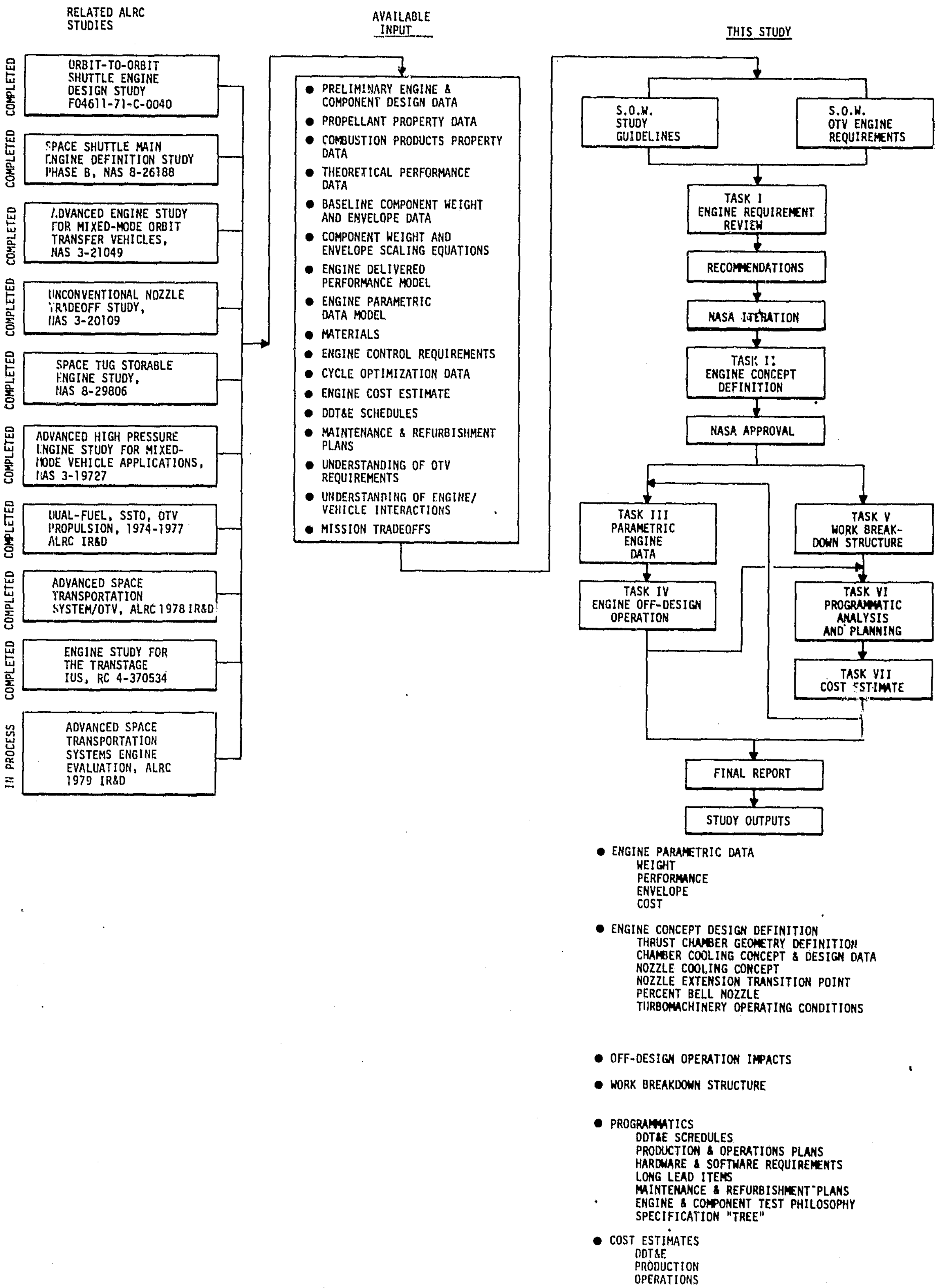


Figure 1. Study Flow Logic Diagram

Detailed engine design, programmatic and cost data for several O_2/H_2 engine cycles and concepts are available from the OOS study final report documentation, Reference 2. Similar data on the RL-10, Reference 3, was reviewed along with the technology data and information available on the Advanced Space Engine (ASE), including References 4 and 5. Other particularly noteworthy efforts which added experience, understanding and data to the study are: (1) ALRC's SSME Phase B Design, Reference 6, (2) Space Tug Storable Engine Study, Reference 7, and (3) the Advanced Engine Study for Mixed-Mode Orbit Transfer Vehicles, Reference 8. Parametric and programmatic data in Reference 9 was also used to support this study.

ALRC in-house efforts have supported the Advanced Space Transportation System definition efforts for the past five years. Emphasis in the past few years on the OTV lead to the formulation and computer modeling of various OTV engine candidates under these in-house efforts. These engine models are capable of generating parametric delivered performance, weight and envelope data. These models were used to support the study and generate the Task III, Parametric Engine Data. The models generate the performance data for O_2/H_2 engines for a range of engine mixture ratios from 5 to 10 using accepted simplified JANNAF methodology for preliminary design studies.

The engine requirements were reviewed in Task I, recommendations were made and iterated with NASA/MSFC. Based upon these requirements and a preliminary analysis of candidate engine cycles, an engine concept that was capable of meeting the requirements was defined in Task II. Parametric analyses were conducted for all candidate cycles in Task III over a thrust range of 10K to 30K lb thrust and detailed analytical studies were conducted on the engine concept selected in Task II. Tank-head and pumped idle mode operation were evaluated in Task V along with one time emergency operation at O/F = 10. A WBS was established in Task V and this WBS was used to conduct the programmatic analysis and cost estimates for the engine DDT&E, production and operations phases in Tasks VI and VII.

II. SUMMARY

A. STUDY OBJECTIVES AND SCOPE

The major objective of this Phase "A" engine study was to provide design and parametric data on the OTV engine for use by NASA and the OTV systems contractors. These data and the systems analyses will ultimately lead to the identification of the OTV engine requirements so that the conceptual design phase can be initiated. Specific study objectives were:

- Review the OTV engine requirements identified in the Statement of Work, make recommendations and iterate with NASA/MSFC.
- Conduct trade studies and system analyses necessary to define the engine concept(s) which meets the OTV engine requirements.
- Generate parametric OTV engine data and provide this data in suitable format for use by the OTV system contractors.
- Prepare a final report at the completion of the study which documents the technical and programmatic assessments of the OTV engine concepts studied. This final report is submitted in three volumes.

  Volume I - Executive Summary
  Volume II - Study Results
  Volume III - Study Cost Estimates

To accomplish the program objectives, a study program consisting of seven major technical tasks and a reporting task was conducted. These tasks are:

Task I - Engine Requirement Review
Task II - Engine Concept Definition
Task III - Parametric Engine Data
Task IV - Engine Off-Design Operation
Task V - Work Breakdown Structure
Task VI - Programmatic Analysis and Planning
Task VII - Cost Estimate

This final report volume is structured by the study tasks. The results and analyses for the first six tasks are reported in the subsequent sections of this volume and Task VII is reported in Volume III.

B. RESULTS AND CONCLUSIONS

The minimum engine performance and man-rating requirements were found to be major concept selection drivers. These requirements, along with the requirements for reusability and long service life, make the development of a new engine necessary.

The man-rating requirement is what makes the OTV different from the engine studies conducted in support of the OOS and Space Tug studies performed in the early 1970's.

High area ratio nozzles are required to meet the performance requirement. However, the area ratio and the delivered performance is constrained by the envelope that is available for the engine in the Orbiter cargo bay. To minimize engine length, bell nozzles with extendible/retractable sections are employed. The maximum engine length with the extendible nozzle in the stowed position was varied parametrically but specified as a nominal 60 inches. This is a 120 inch engine length with the extendible nozzle deployed. To achieve the high area ratios and performance in this length, the engine throat size must be small. This can be accomplished by either increasing the thrust chamber pressure or with lower thrust, multiple engines. The delivered engine performance is shown on Figure 2 for single and twin engine installations over the study thrust range of 10K to 30K lbF for single installations and 16K to 40K for dual installations. Staged combustion, expander and gas generator engine cycles were considered in the engine concept definition task. The figure shows that only the staged combustion cycle engine meets the minimum performance requirement in a single engine installation. Both the expander and staged combustion cycle engines meet the minimum performance requirement in a twin engine installation. The gas generator cycle engine concept fails to meet the performance requirements in either single or dual installations.

Reliability and safety analyses were undertaken in the study to determine the impact of the man-rating requirement upon the engine design. The man-rating requirement implies crew safety. For the OTV, crew safety is measured in terms of the probability of safely returning the crew to the orbiter. This probability is in turn related to the operational reliability. The reliability and safety, although inter-active, are different measures and impose different requirements. For example, the crew safety requirement can detract from the overall mission reliability by resulting in a higher incidence of mission aborts.

The results of the reliability and safety analyses showed that no single engine concept, even with redundant components, could satisfy the reliability and safety requirements. Therefore, multiple engine installations were evaluated, considering both crew losses and mission losses. Crew risk was minimized at two engines. For additional engines, the increase in

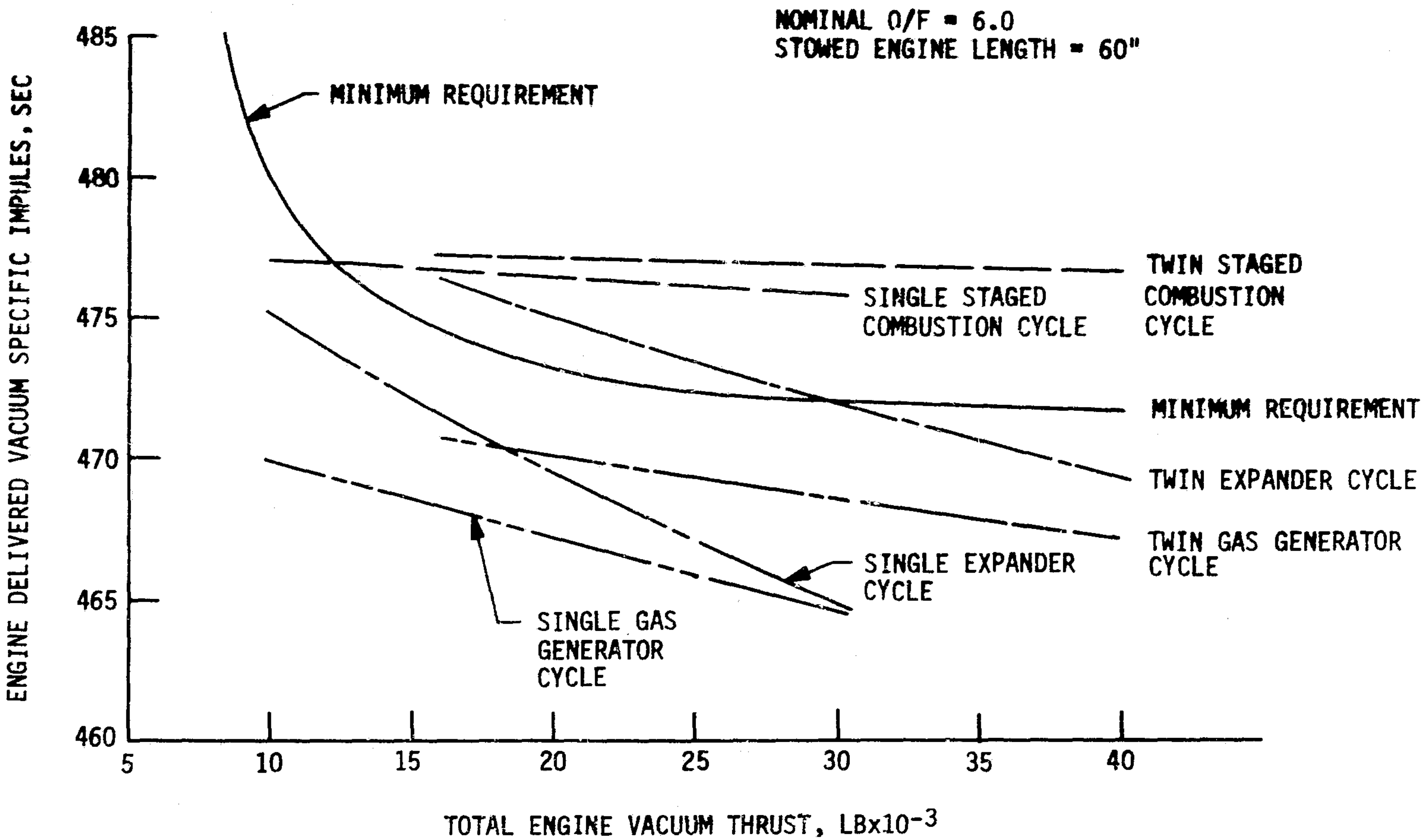


Figure 2. Engine Concept Performance Comparisons

catastrophic failures outweighed the reliability improvement. Mission risk was minimized with three engines. This assumed one engine-out capability. Assuming that only two engines are required for the first burn and the mission can be completed on one engine for the remaining burns, the two engine installation appeared to be the best choice from an overall cost, weight, payload and risk standpoint. The major conclusions of the reliability and safety analyses are:

- A minimum of two engines is mandatory.
- Series redundant main propellant valves are required.
- Redundant spark igniter is required.

Based upon the safety and reliability analyses, as well as the performance requirement analysis, twin engine installations were recommended as one of the results of the concept definition task. Therefore, weight and performance trade-offs were conducted for the various cycle candidates in dual engine installations to determine if any of the cycles were technically superior in terms of payload delivery capability. The engine weight data for the various cycle candidates that was used in conducting the trades is shown on Figure 3. The weight data is plotted as a function of the total engine thrust for two engine subassemblies. The figure shows that the higher pressure staged combustion cycle engine is heavier than the other two candidates. This occurs because of the additional components and higher pressure components. The engine length is fixed and hence, the envelope is always filled. Therefore, the nozzle weight at a given thrust level is approximately constant as a function of pressure.

Payload partials were derived using the data in NASA Technical Memorandum TMX-73394. These partials were used to conduct the trade-offs required in this study program. The payload partials are:

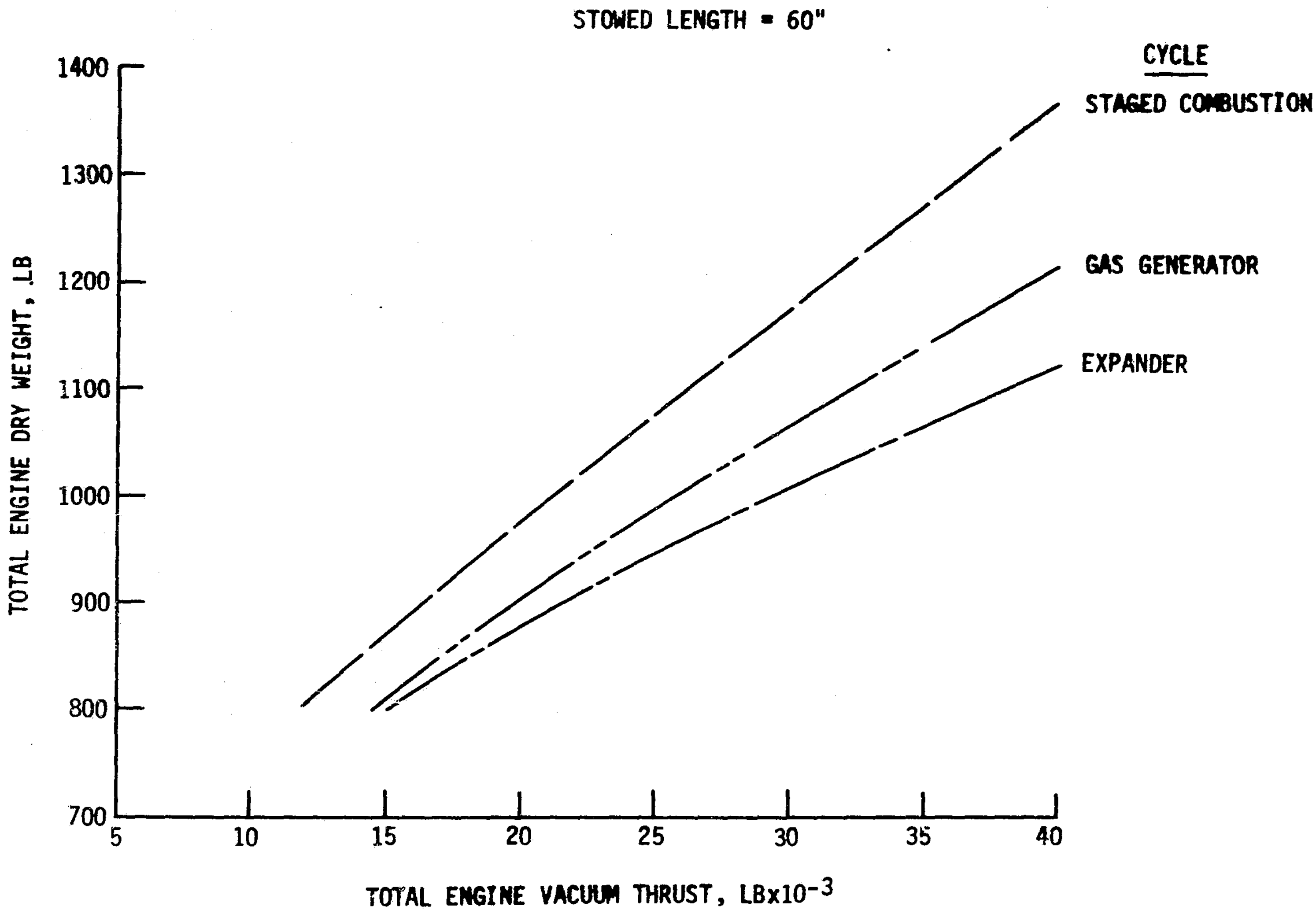


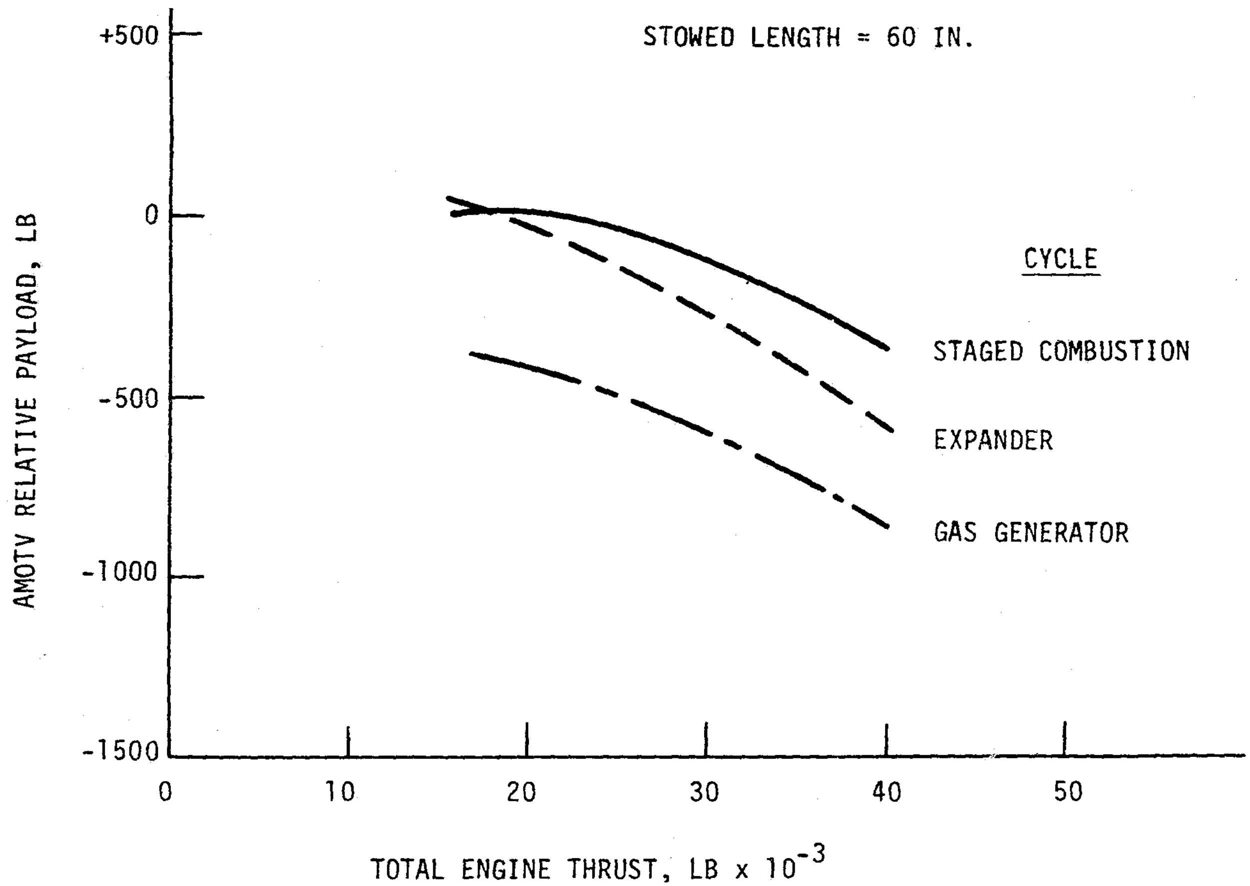
Figure 3. Twin Engine Installation Weight Comparisons

	AMOTV	APOTV
$\frac{\Delta W_{PL}}{\Delta I_s}$, lb/sec	73	60
$\frac{\Delta W_{PL}}{\Delta W_{ENG}}$, lb/lb	-1.1	-1.1

The results of the trade-off analyses using the AMOTV payload partials are shown on Figure 4. The payload changes were computed against the minimum specific impulse requirement and a single, staged combustion cycle, 20K engine baseline weight of 718 lb. The figure shows that the expander cycle and staged combustion cycle engines have approximately the same payload capability at the 20K lbF nominal total thrust level. The gas generator cycle engine results in relatively high payload penalties over the entire thrust range and is not competitive.

Because the staged combustion and expander cycle engine payload comparisons result in a technical stalemate, other factors had to be considered in the concept selection. These factors were development risk and life cycle cost. The expander cycle development risk is lower than either the staged combustion or gas generator cycles primarily because a critical combustion device is eliminated. In-house studies have also shown that this results in substantial cost savings. These cost results are supported by the data in Volume III.

Based upon the payload, safety, reliability, risk and cost evaluations, an expander cycle, twin engine installation was recommended and approved by NASA at the conclusion of Task II. The conclusions are summarized on Figure 5.



STAGED AND EXPANDER CYCLE PAYLOAD CAPABILITIES ABOUT EQUAL

Figure 4. Engine Cycle Payload Comparisons, Two Engine Installation

- MAN-RATING, PERFORMANCE, REUSABILITY AND SERVICE LIFE REQUIREMENTS MAKE DEVELOPMENT OF A NEW ENGINE NECESSARY
- CREW SAFETY DICTATES A MINIMUM OF TWO ENGINES
- MULTIPLE ENGINE VEHICLE RELATIVE PAYLOAD TRADES RESULT IN A STAGED COMBUSTION vs EXPANDER CYCLE DRAW
- GAS GENERATOR CYCLE ENGINE ELIMINATED BECAUSE OF PAYLOAD PENALTIES

IN-HOUSE STUDY RESULTS

- DEVELOPMENT RISK WITH EXPANDER CYCLE IS MUCH LESS BECAUSE A CRITICAL COMBUSTION DEVICE IS ELIMINATED
- POTENTIAL MAXIMUM LIFE CYCLE SAVINGS FOR EXPANDER CYCLE OF 480 MILLION DOLLARS

- RECOMMENDED ENGINE DESIGN APPROACH:

TWIN 10K LBF MAN-RATED EXPANDER CYCLE ENGINE

Figure 5. Engine Requirements and Concept Selection Study Conclusions

The advanced expander cycle engine was carried through the remaining study tasks, although parametric data was generated on the staged combustion and gas generator cycles. The parametric data is reported in Section V of this volume.

The baseline advanced cycle engine characteristics that evolved from this study are shown on Figure 6. The engine uses an extendible/retractable nozzle which extends from an area ratio of 297:1 to 792:1 at the exit. This nozzle extension is 60 inches long and is radiation cooled. It should also be noted that the length of the engine from the gimbal center to the fixed nozzle exit (ϵ = 297:1) is also 60 inches. This is to satisfy the Phase "A" engine requirement of a maximum engine length with the extendible nozzle in the stowed position of 60 inches (nominal). The maximum engine length with this nozzle deployed is 120 inches (nominal).

The advanced expander cycle engine schematic is shown on Figure 7. The description which follows is for the 10K baseline engine. A slotted copper chamber is cooled with 85% of the total hydrogen flow from an area ratio of 8:1 to the injector. The fixed portion of the nozzle from an area ratio of 8:1 to 297:1 is cooled with the remaining 15% of the hydrogen flow in a two-pass tube bundle. The nozzle and chamber coolants are combined to provide the warm (653°R) turbine drive gas. Six percent of the flow bypasses the turbines to provide margin and control. 69.6% of the hydrogen flow is used to drive the LH_2 TPA and 24.4% is used to drive the LOX TPA. The turbine exhaust and bypass flows enter the injector to be mixed and burned with the liquid oxygen.

- VACUUM THRUST = 10,000 LB
- VACUUM SPECIFIC IMPULSE = 475.1
- THRUST CHAMBER PRESSURE = 1300 PSIA
- MIXTURE RATIO = 6.0
- NOZZLE AREA RATIO = 792
- ENGINE LENGTH
  - EXTENDIBLE NOZZLE STOWED = 60"
  - EXTENDIBLE NOZZLE DEPLOYED = 120"
- NOZZLE EXIT DIAMETER = 61.5"
- ENGINE DRY WEIGHT = 437 LB

Figure 6. Baseline Advanced Expander Cycle Engine Characteristics

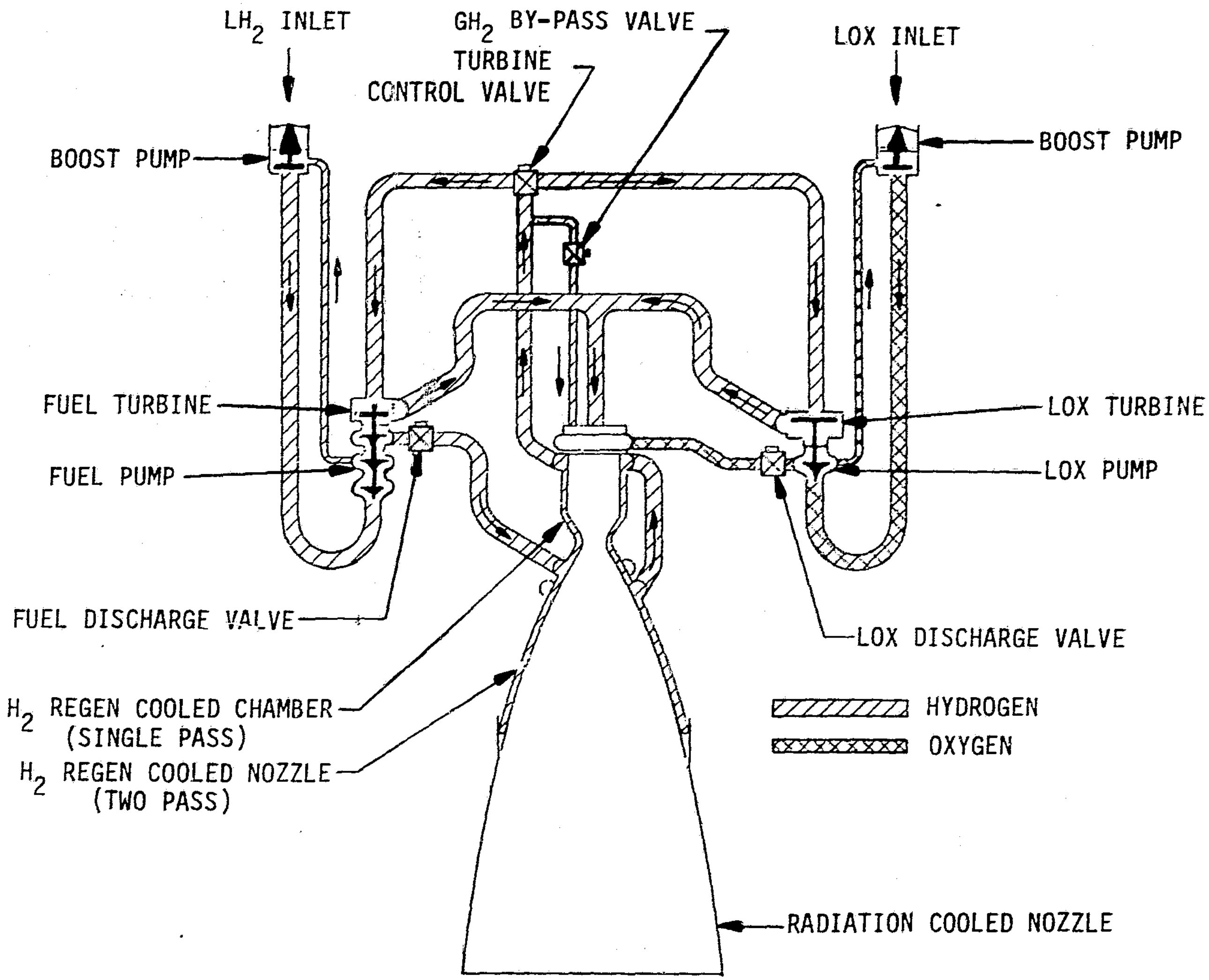


Figure 7. Advanced Expander Cycle Engine Schematic

III. TASK I - ENGINE REQUIREMENT REVIEW

A. OTV ENGINE REQUIREMENTS

The requirements for the OTV engine applicable to a vehicle of the type envisioned to be operational in 1987 have been derived from numerous NASA in-house and contracted vehicle and systems studies and are summarized as follows:

1. The engine will operate on liquid hydrogen and liquid oxygen propellants.

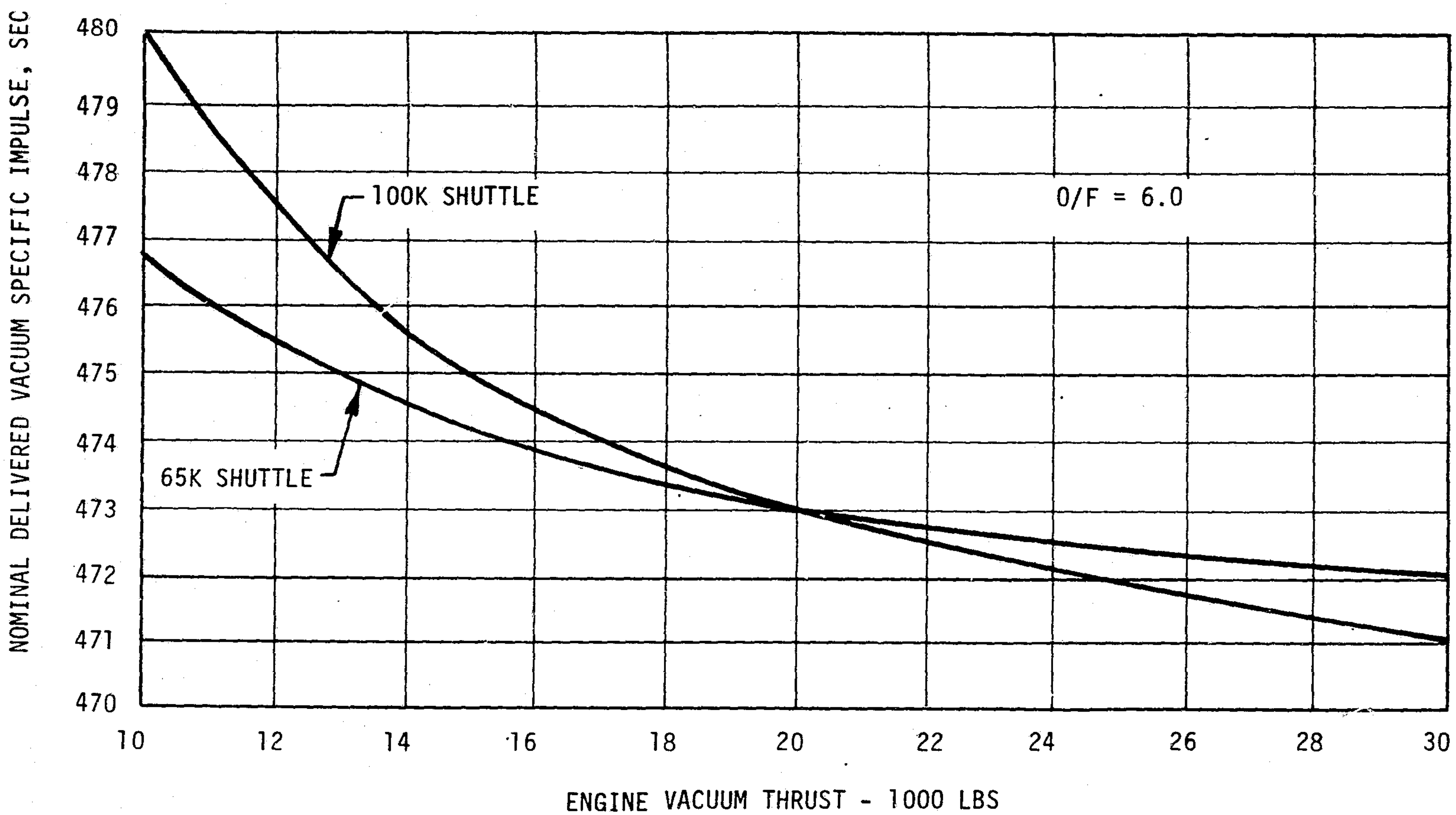
2. Engine design and materials technology are based on 1980 state-of-the-art, or start of Phase C/D contract.

3. The engine must be capable of accommodating programmed and/or commanded variations in mixture ratio over an operating range of 6:1 to 7:1 during a given mission. The effects on engine operation and lifetime must be predictable over the operating mixture ratio range.

4. The nominal specific impulse shall not be less than that specified in Figure 8. The higher of the two values shown must be used.

5. The engine chamber pressure is to be determined by the effects on total vehicle weight and stage performance.

6. The propellant inlet temperatures shall be 162.7°R for the oxygen boost pump and 37.8°R for the hydrogen boost pump. The boost pump inlet NPSH at full thrust shall be 2 ft for the oxygen pump and 15 ft for the hydrogen pump.



HIGH PERFORMANCE NECESSARY

Figure 8. OTV Engine Minimum Nominal Specific Impulse Requirement vs Engine Vacuum Thrust

7. The service free life of the engine cannot be less than 60 start/shutdown cycles or two hours accumulated run time, and the service life between overhauls cannot be less than 300 start/shutdown cycles or 10 hours accumulated run time. The engine shall have provisions for ease of access, minimum maintenance, and economical overhaul.

8. The engine when operating within the nominal prescribed range of thrust, mixture ratio, and propellant inlet conditions shall not incur during its service life chamber pressure oscillation, disturbances, or random spikes greater than $\pm$ 5 percent of the mean steady state chamber pressure. Deviations to be expected in emergency modes shall be predictable.

9. The engine weight is to be determined by the effects on stage weight and cost.

10. The engine nozzle is to be a contoured bell with an extendible/retractable section.

11. Engine gimbal requirements are +15 degrees and -6 degrees in the pitch plane and $\pm$ 6 degrees in the yaw plane.

12. The engine is to provide gaseous hydrogen and oxygen autogenous pressurization for the propellant tanks.

13. The engine is to be man-rated and capable of providing abort return of the vehicle to the Orbiter orbit.

In addition to the engine requirements, consideration was also given to the following general requirements which are critical to the overall OTV program.

1. Space Shuttle Payload requirements and constraints
2. Operational flexibility
3. Reusability
4. Reliability, quality and safety
5. Low-cost operations and minimum program cost
6. Performance and weight sensitivity
7. Development risk
8. Launch operations
9. Mission operations
10. Engineering development and test programs

Because the engine performance and man-rating requirements were found to be major concept selection and design drivers, they are discussed in this section. The man-rating requirement is what makes the OTV different from the engine studies conducted in support of the Space Tug studies performed in 1973. The service life requirement and structural guidelines also have a significant impact upon the engine design and are also discussed briefly herein.

B. PERFORMANCE REQUIREMENT

The minimum required nominal specific impulse specified by the SOW is shown on Figure 8. The higher of the two values shown was used. This figure assumes that the specific impulse will make up the gravity loss as the thrust to weight ratio decreases. This then keeps the payload constant. To achieve the high performance levels shown in the figure, high nozzle area ratios are required. Because the engine was also length constrained, this means high chamber pressures are required for a single engine installation. The engine length with the extendible nozzle retracted was varied as 50, 60 and 70 inches in the study. The 60 in. stowed length was the study nominal and much of the preliminary analyses were conducted for this value.

Figure 9 and Table I show the chamber pressures and nozzle area ratios necessary for a staged combustion cycle engine to meet the SOW specific impulse requirements. A single engine is assumed. It should be noted that not all operating points proved to be feasible and some performance penalty must be accepted at the 10K thrust level.

The expander cycle engine requires relatively long chamber lengths to heat the hydrogen to values sufficient to meet engine power balance requirements. This means that less engine length is available for the nozzle and higher chamber pressures would be required for single engine installations of expander cycle engines to meet the minimum performance requirements.

Figure 10 compares the expander and staged combustion cycle minimum chamber pressure and area ratio requirements for a single engine installation. The data for the expander cycle engine is also shown on Table II for comparison to that shown on Table I. Actually, the maximum operating chamber pressure level for an expander cycle is less than that of the staged combustion cycle. Therefore, the performance goals are more difficult to achieve with the expander cycle in a single engine installation. Again, many of the operating points shown on Table II for the expander cycle do not appear to be feasible.

The results of this performance requirement analysis, in part, lead to the consideration of multiple engines (2 or 3) to achieve the performance goals. By reducing the thrust per chamber, small throat sizes and

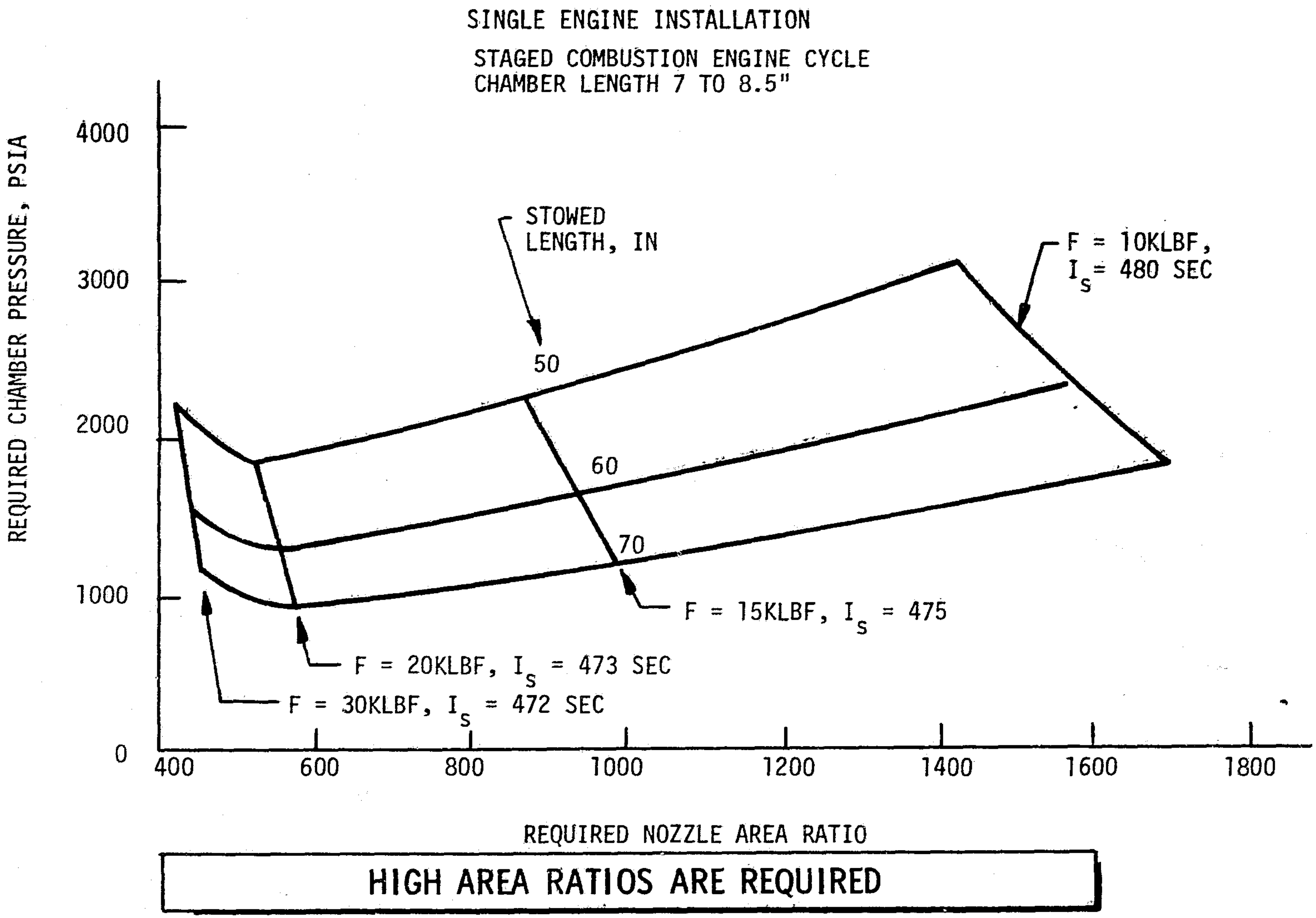


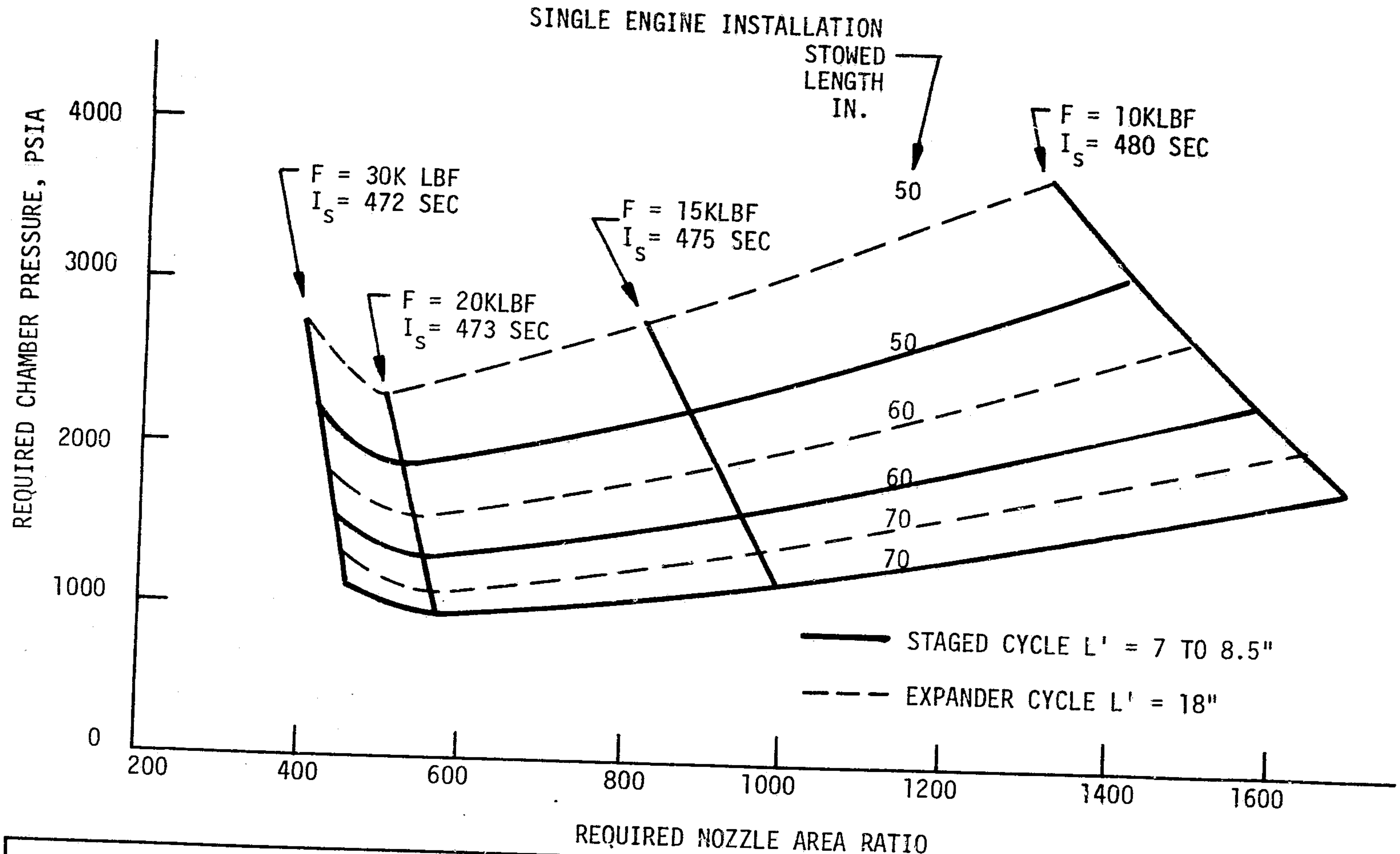
Figure 9. Sensitivity of Engine Design Point to Performance and Stowed Length Requirements

TABLE I

SENSITIVITY OF ENGINE DESIGN POINT TO
PERFORMANCE AND STOWED LENGTH REQUIREMENTS

Thrust, K lb	Minimum Required I_s, sec	Maximum Retracted Length, in.	Required Chamber Pressure, psia	Required Area Ratio
10	480	50	3120	1421
		60	2359	1572
		70	1850	1698
20	473	50	1847	522
		60	1281	548
		70	945	569
30	472	50	2245	415
		60	1550	439
		70	1124	454

Staged Combustion Cycle
Chamber Lengths: 7 to 8.5 in.



PERFORMANCE GOALS MORE DIFFICULT TO ACHIEVE WITH EXPANDER CYCLE

Figure 10. Effect of Chamber Length Requirement Upon Required Chamber Pressure and Area Ratio

TABLE II

EFFECT OF CHAMBER LENGTH REQUIREMENT UPON THE REQUIRED CHAMBER PRESSURE AND AREA RATIO

Thrust K lb	Minimum Required I_s, sec	Maximum Retracted Length, in.	Required Chamber Pressure, psia	Required Area Ratio
10	480	50	3699	1315
↓	↓	60	2745	1491
↓	↓	70	2117	1640
20	473	50	2285	499
↓	↓	60	1531	535
↓	↓	70	1084	556
30	472	50	2785	396
↓	↓	60	1852	429
↓	↓	70	1303	447

Expander Cycle
Chamber Length = 18 in.

hence, high area ratios are possible. The multiple engine analysis is reported in Section IV,B of this report as part of the concept definition task.

C. MAN-RATING REQUIREMENT

The man-rating requirement implies crew safety. For the OTV, crew safety can be measured in terms of the probability of safely returning the OTV crew to the orbiter. This probability is in turn related to operational reliability. The objectives of the reliability and safety analysis were to; (1) establish the special engine design requirements imposed by man-rating, (2) determine the desirable reliability and safety numerical requirements, and (3) establish the reliability and safety of each candidate engine concept to compare and determine if they meet the reliability and safety (R&S) requirements. The reliability and safety definitions are:

What is Mission Reliability?

Successful Insertion of Payload and Return of OTV to Shuttle Orbiter.

What is Crew Safety?

Safe Return of Crew to Orbiter Regardless of Payload Status.

Reliability and safety, although interactive, are different measures and impose different requirements. For example, the crew safety requirement can detract from the overall mission reliability by resulting in a higher incidence of mission aborts. The reliability requirement is driven by:

- Cost Effectiveness (Life Cycle Cost per Pound of Payload Delivered)

- Crew Safety (Safe Return of OTV Crew to Orbiter)

The crew safety requirement is driven by:

- Acceptable Risk to Life

To be effective, safety considerations must be incorporated into the initial system design concept. Man-rating has programmatic as well as, design implications. Some cost and schedule increases must be expected compared to a non-man-rated design. Man-rating considerations are shown on Figure 11.

In order to establish realistic reliability and safety requirements, it was necessary to first establish the influencing factors. Although there is inherent psychological difficulty in defining acceptable crew risk, somewhere there is a minimum acceptable threshold for crew safety and maximum acceptable levels for design and development and life cycle costs. An acceptable crew risk (ACR) of about 5×10^{-4} was assumed to be tolerable based upon historical precedence and the following logic.

<u>ACR Estimation</u>

Given: 200 manned mission program (APOTV)
If it is desired to have 90% confidence that no OTV crews are stranded in orbit, i.e., a 10% risk of losing one or more crews is acceptable, then

$$\text{Mission ACR} = 1 - (0.90)^{1/200} = 5.3 \times 10^{-4}$$

- MAN-RATING IMPACTS THE FOLLOWING AREAS

<u>DESIGN</u>
- SYSTEM CONFIGURATION
- OPERATIONAL CONTROL
- REDUNDANCY
- FAIL SAFE OPERATION
- BITE INSTRUMENTATION (FLIGHT READINESS)
- MDS INSTRUMENTATION (MALFUNCTION DETECTION)

<u>PROGRAM</u>
- COST
- WEIGHT
- TESTING
- SCHEDULE
- PROCEDURES
- QUALITY ASSURANCE

MAN-RATING INFLUENCES ALL ASPECTS OF PROGRAM

Figure 11. Man-Rating

C. E. Cornell, Institute of Aerospace Safety and Management, reported in Space/Aeronautics, October 1969, that "The chance an Apollo won't safely return its astronauts is (approximately) 10^{-3}". He also conjectured that, "the maximum allowable risk of a fatal accident for a manned space project might be 4×10^{-4}".

The propulsion system reliability requirement to achieve the above maximum crew risk is .999994 and was derived as follows:

- Assumed mission ACR is 5.3×10^{-4} deaths per mission.
- Corresponding engine reliability is:

$$R_{eng} = (((1\text{-ACR})^{1/M})^{1/V})^{1/B}$$

- where: M = average crew number = 4
  V = vehicle/engine allocation = 4 (assumes 1/4 of system failures are engine related)
  B = nominal engine burns per mission = 6

$$R_{eng} = (((1-5.3 \times 10^{-4})^{1/4})^{1/4})^{1/6} = 0.999994.$$

Based upon an evaluation of historical data and component failure rate projections, failure rate predictions were made for various cycle candidates. Engine failure (shutdown) was assumed to occur any time there is a malfunction in a critical component. The results of this analysis, shown on Figure 12, are that no single engine concept, even with redundant components, can satisfy the reliability and safety requirements. Therefore, multiple engine installations were evaluated considering both crew losses and mission losses.

Figure 13 summarizes the results of the reliability and safety analysis. The top bar represents the predicted mission losses (without crew losses) and the bottom bar the vehicle and crew losses. For this analysis, the crew is considered to be lost if there is no main engine left to return the OTV to LEO. Rescue vehicles might be used but were not factored into this analysis. It is doubtful that the use of rescue vehicles is as desirable

- FIRST-ORDER PREDICTION PERFORMED BASED ON PROJECTED 1980 COMPONENT FAILURE RATES.

	PREDICTED ENGINE FAILURE RATE*	
	NON-REDUNDANT COMPONENTS	REDUNDANT COMPONENTS
• STAGED COMBUSTION CYCLE	3.9	2.6
• GAS GENERATOR CYCLE	3.1	2.0
• EXPANDER CYCLE	2.7	1.8

*FAILURES PER THOUSAND ENGINE FIRINGS.

NO SINGLE ENGINE CONCEPT SATISFIES THE CREW SAFETY REQUIREMENT OF 6×10^{-6}

Figure 12. Predicted Engine Failure Rate

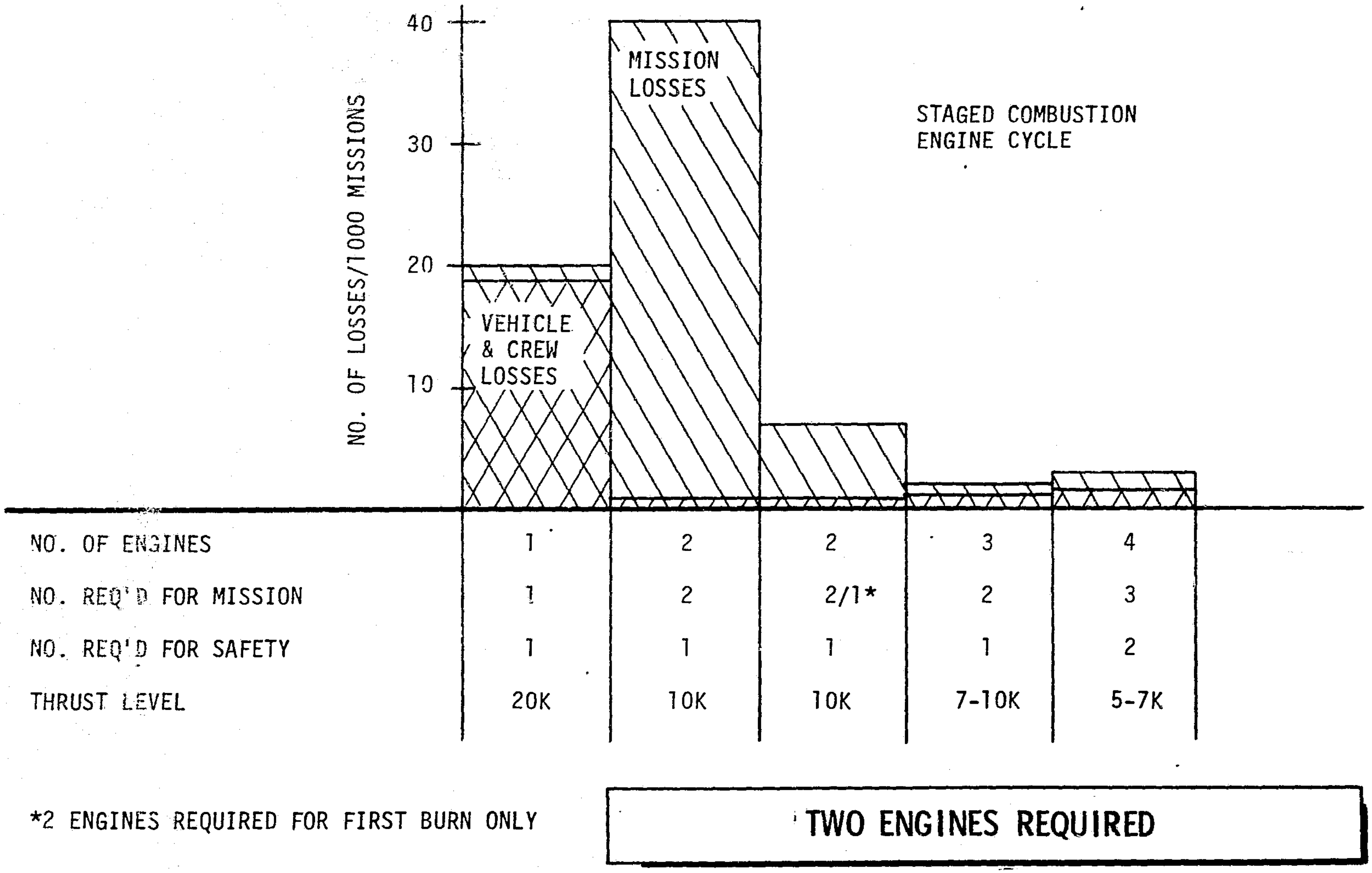


Figure 13. OTV - Mission and Vehicle Crew Risk

or cost effective as providing the safety (crew return capability) in the OTV itself. The single engine installation results in expected crew losses of 19 per 1000 missions which is very unacceptable. One additional failure per thousand results in only the loss of the mission and not the crew. This would occur when the engine would fail-to-start on the first burn while still in the vicinity of the orbiter.

The effect of multiple engine intallations was evaluated assuming that only the catastrophic failures (failures which release enough energy to do significant damage to the vehicle or adjacent engines) would result in vehicle/crew loss. For critical failures (resulting in engine shutdown), the engines were assumed to be independent. In other words, it was assumed that multiple engines would have separate power and propellant sources with cross-feed capability and have isolation components such as circuit breakers and isolation valves to permit operation of the remaining engine(s). The figure shows that crew risk is minimized at two engines. For additional engines, the increase in catastrophic failures outweighs the reliability improvement (considering that approximately 0.5 catastrophic failure occur per engine per thousand missions). Mission risk is minimized with three engines. This assumes one engine-out capability. The two engine installation results in a higher incidence of mission losses if both engines are assumed to be required to complete the mission on all burns because system reliability goes down even though engine reliability goes up. However, a more reasonable assumption is that the mission would be aborted only if an engine is lost during the first burn. Assuming that two engines are required for only the first burn and the mission can be completed on one engine for the remaining burns (called the 2/1 concept) the two engine installation appears to be the best choice from an overall cost, weight, payload and risk standpoint. During the apogee burn out of GEO to return to LEO and the LEO rendezvous burn, there is little choice but to complete the burn with whatever main engines are still operating. It is possible that the crew might elect not to complete the mission if an engine fails to start on the GEO insertion burn. The single vs multiple engine analysis and mission-abort or mission-completion options will be further examined and updated in the follow-on efforts to this contract.

The 2/1 engine concept was selected on the basis of the results shown on Figure 14. Specifically the propulsion system can be thoroughly checked-out prior to undocking and a go/no-go decision made. On the first burn, both engines must fire or the mission is aborted. On subsequent burns, (orbit insertion, de-orbit and rendezvous) only one engine is assumed to be required. Figure 14 shows that all candidate engine concepts satisfy the reliability and safety requirements in the 2/1 concept. The gas-generator and expander engine cycles appear to be better than the staged combustion engine cycle although there is no clear-cut winner.

The major conclusions derived from the reliability and safety analyses are:

- A minimum of two engines is mandatory.
- Series redundant main propellant valves are required.
- Redundant spark igniter is required.
- The igniter, gas-generator or preburner valves should be dual coil.

The redundancy requirements were factored into the engine study weight data and multiple engines were evaluated in the performance and weight tradeoff studies of Task II.

It should also be noted that, a two engine installation with series redundant main propellant valves was also selected for the OMS engine system. This selection was made after a rigorous evaluation that considered cost, weight, reliability and safety.

D. SERVICE LIFE REQUIREMENT

The original statement of work required that:

> "The structure shall be capable of withstanding four times the limit life where all components of structural behavior are magnified by 1.4 (e.g., mean stress or strain, alternating stress or strain)."

The pressure drop required for the chamber is very sensitive to the allowable strain resulting from the cycle life specification. Including

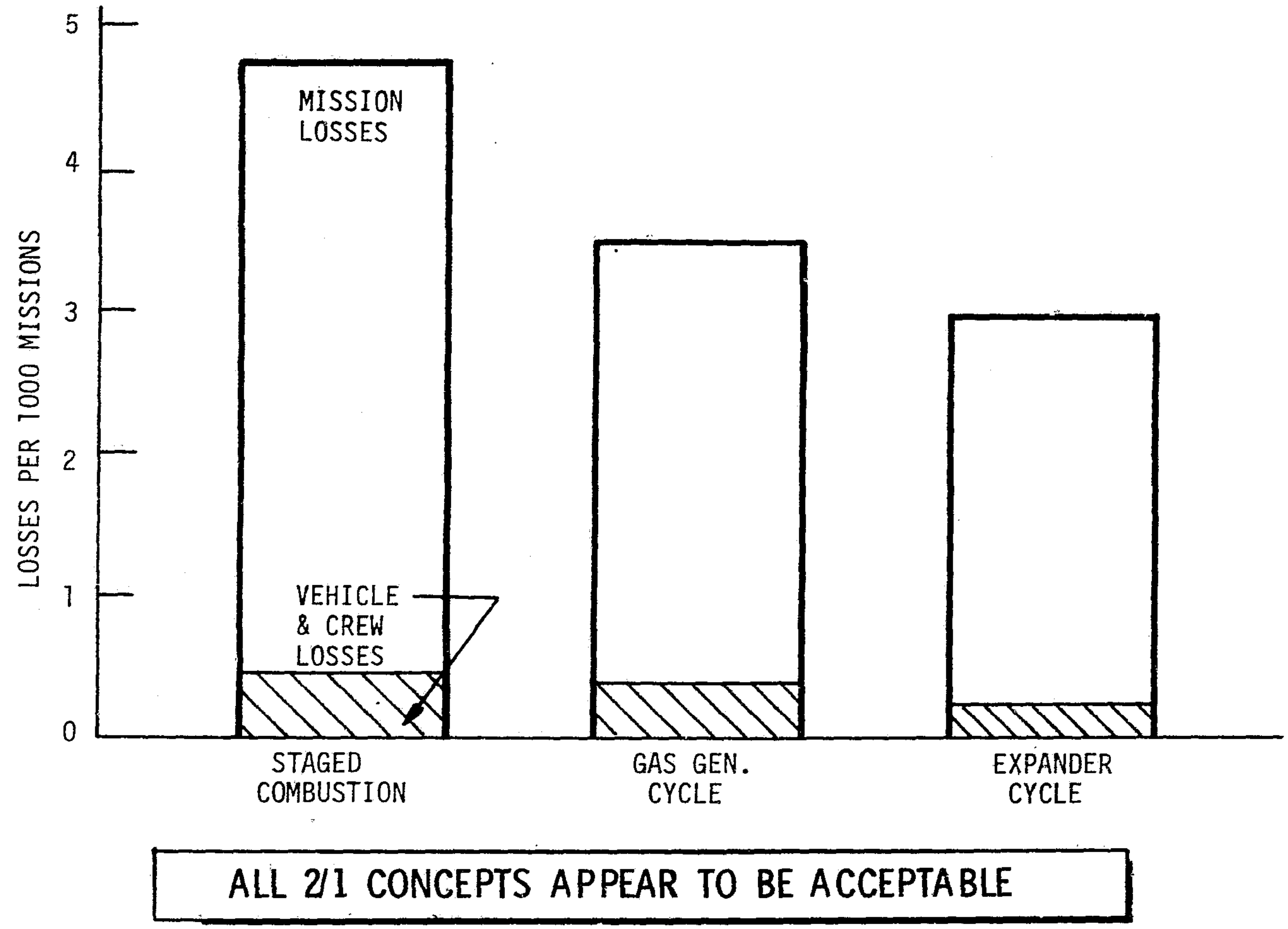


Figure 14. 2/1 Engine System Concept

the factor of safety on strain of 1.4, as originally required by the Statement of Work, tripled the pressure drop compared to a factor of safety of 1.0 as shown by Figure 15.

This data was discussed with the NASA/COR at the October 1978 Task I and Task II review and the 1.4 factor of safety on strain was dropped for further thermal study efforts in Task III. All further design analysis was conducted with a factor of four times the limit life. However, the cooling decisions made during the concept selection phase were assumed to remain valid. The requirement for the statement of work was changed to read:

> "The structure shall be capable of withstanding at least four times the limit life based on lower bound fatigue property data."

The figure also shows that a mixture ratio of 6.0 design point requires more pressure drop than at a mixture ratio of 7.0. This occurs because there is a significant reduction in the gas-side heat transfer coefficient with increasing mixture ratio caused by transport property changes.

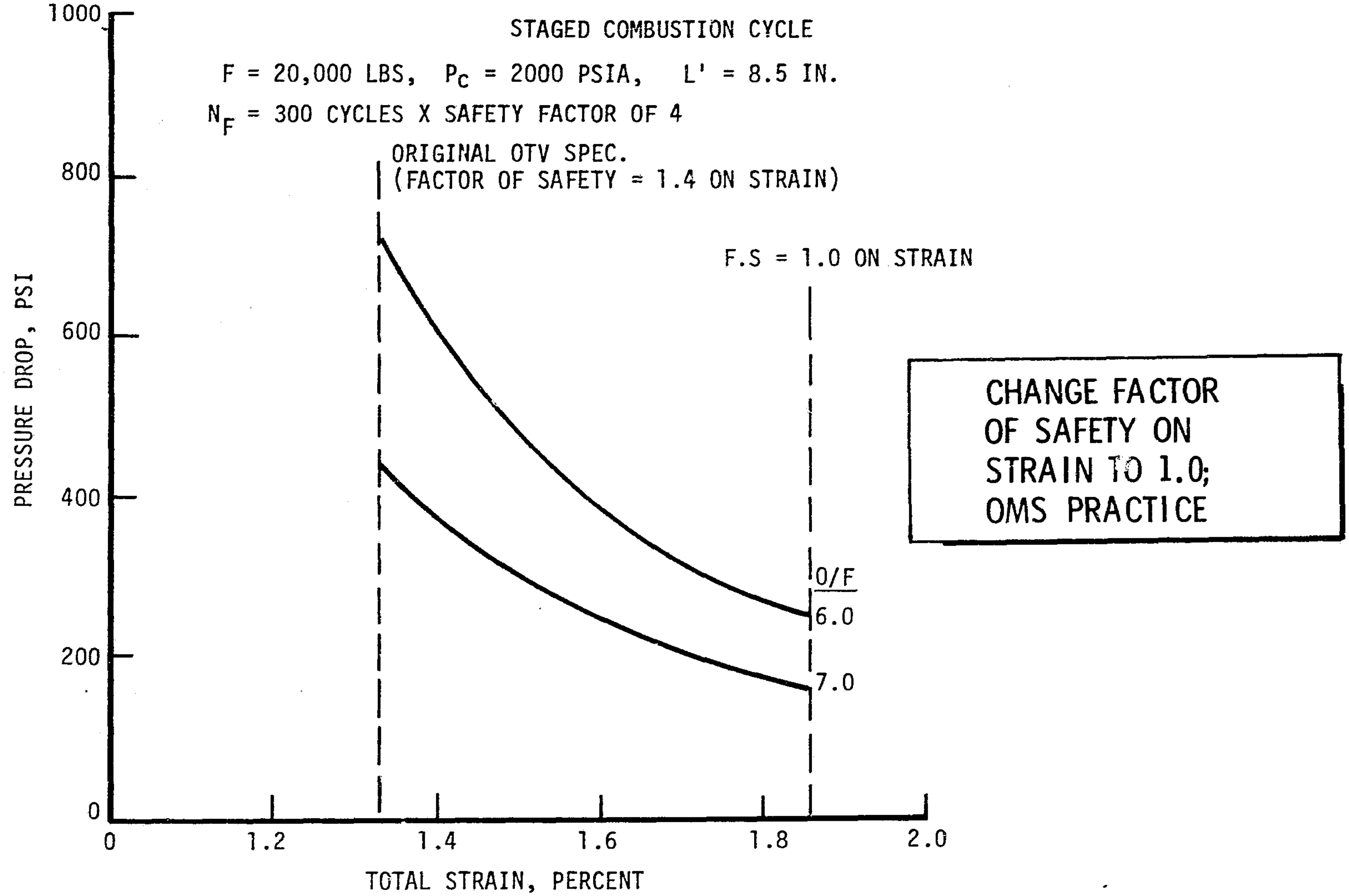


Figure 15. Effect of Allowable Strain on Combustion Chamber Pressure Drop

IV. TASK II - ENGINE CONCEPT DEFINITION

A. CONCEPTS CONSIDERED

The primary pump-fed engine cycle candidates for use with O_2/H_2 propellants are; (1) expander cycles (RL-10 type), (2) conventional turbine bleed cycles (i.e., gas-generator) (J-2 type), and staged combustion cycles (SSME and ASE types).

Both the expander cycle and the gas generator cycle are limited to moderately high chamber pressure operation. The expander cycle is limited by the amount of heat that can be put into the hydrogen. The gas generator cycle is limited because of the performance loss associated with the turbine drive flow. The upper limit on chamber pressure of the staged combustion cycle engine is set by the service life requirements. These effects were investigated and are presented in this section.

The staged combustion cycle evaluated in the concept definition phase is similar to the Advanced Space Engine (ASE). A simplified schematic is shown on Figure 16. It uses a single fuel-rich preburner to produce 1860°R turbine drive gases. Turbomachinery efficiencies used to perform power balances were obtained from documentation on the ASE components. Fuel and oxidizer pump efficiencies used, are 63.5% and 65.7%, respectively. Fuel and oxidizer pump turbine efficiencies were 82.8% and 63.7%, respectively. The engine combustion chamber is regeneratively cooled in a slotted copper chamber to an area ratio of about 8:1. A tube bundle nozzle is cooled in parallel with the chamber using about 22% of the total hydrogen flow. The nozzle extension is radiation cooled. This selection is an ALRC choice. The ASE extension is hydrogen dump cooled. This was not selected because of the performance loss associated with the small dump cooling flow. All three cycles are assumed to have radiation cooled nozzles which puts the performance comparisons on a common basis.

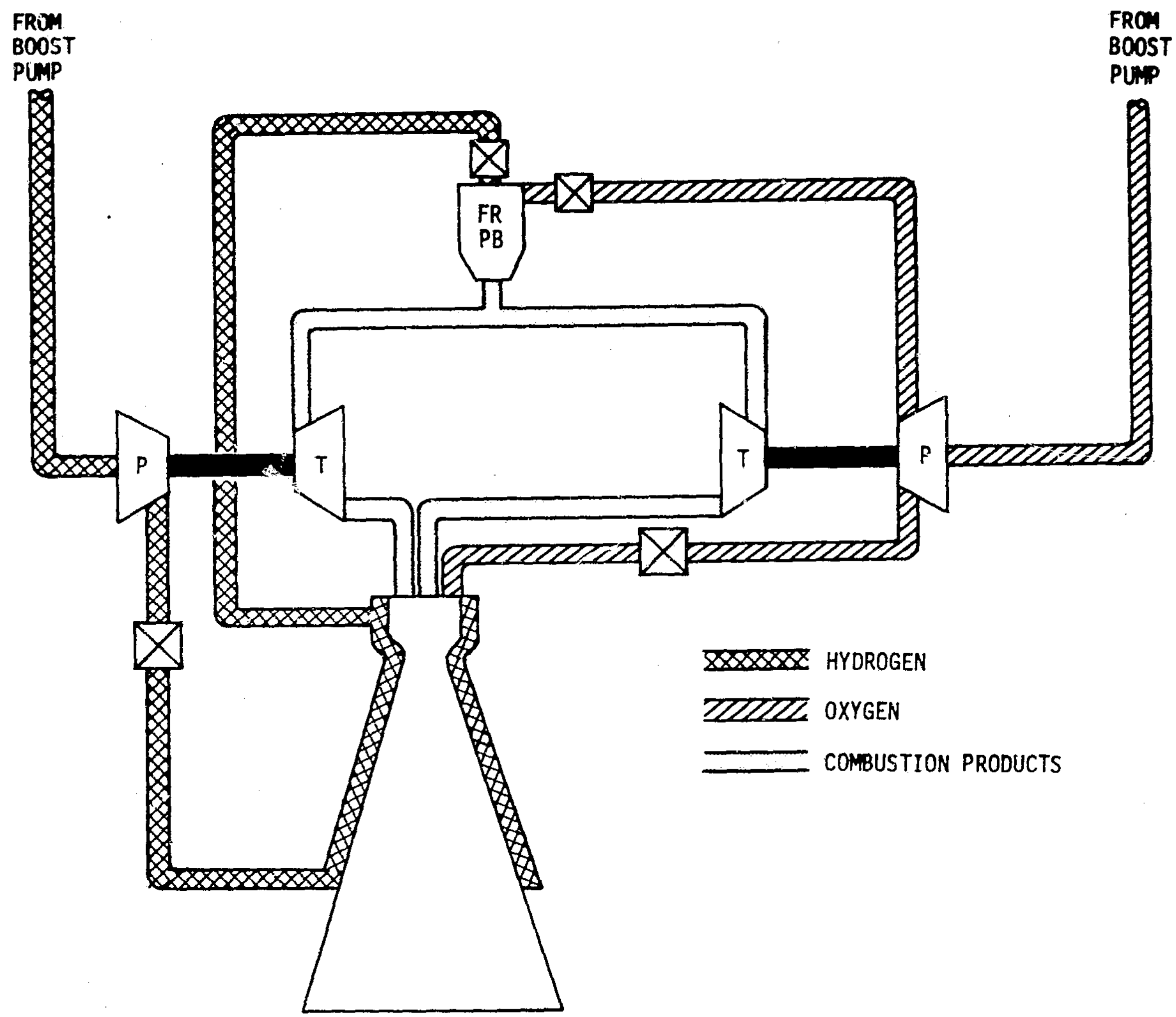


Figure 16. Single Fuel-Rich Staged Combustion Cycle

The expander cycle engine evaluated in the concept definition phase is a parallel turbine drive concept shown on Figure 17. Data from the OOS and RL-10 Derivative studies of expander cycle engines was used to support the analysis. A chamber length of 18 inches was selected after reviewing these analyses. A contraction ratio of 3.66 was selected on the basis of the ASE design. Turbomachinery efficiencies used to perform the preliminary power balances were estimated from the RL-10 Derivative documentation. Specifically, efficiencies are: fuel pump 60%, oxidizer pump 69%, fuel turbine 67%, and oxidizer turbine 74%. These efficiencies were evaluated, revised and power balances rerun in later Task III efforts. The revised power balance data does not differ much from these preliminary analyses. The engine combustion chamber is regeneratively cooled in a slotted copper chamber to an area ratio of approximately 8:1. A tube bundle nozzle is cooled in parallel to the chamber with 15% of the total hydrogen flow,which is based upon cooling evaluation results. A radiation cooled nozzle extension is used.

The gas generator cycle engine evaluated in the concept definition phase (shown on Figure 18) uses a fuel-rich gas generator which produces 1860°R turbine drive gas. The pumps are driven in parallel with 20:1 pressure ratio turbines. The turbine exhaust gases are dumped into the nozzle extension to be mixed and expanded over the remaining area ratio. These exhaust gases could be used as nozzle extension flange coolant. Turbomachinery efficiencies used to perform the power balances were obtained from studies of similar components for Contract NAS 3-21049, Advanced Engine Study for Mixed-Mode Orbit-Transfer Vehicles. They are: fuel pump 60%, oxidizer pump 63%, fuel and oxidizer turbines 60%. Coolant flow paths are similar to those described for the staged combustion and expander cycles.

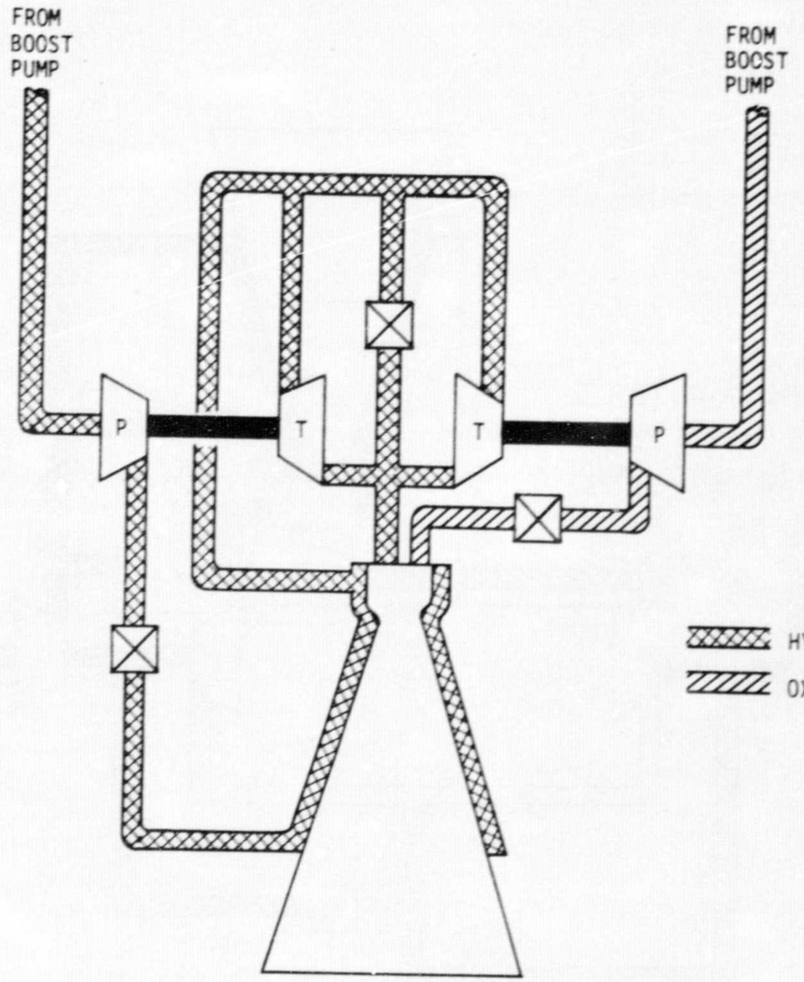


Figure 17. Expander Cycle

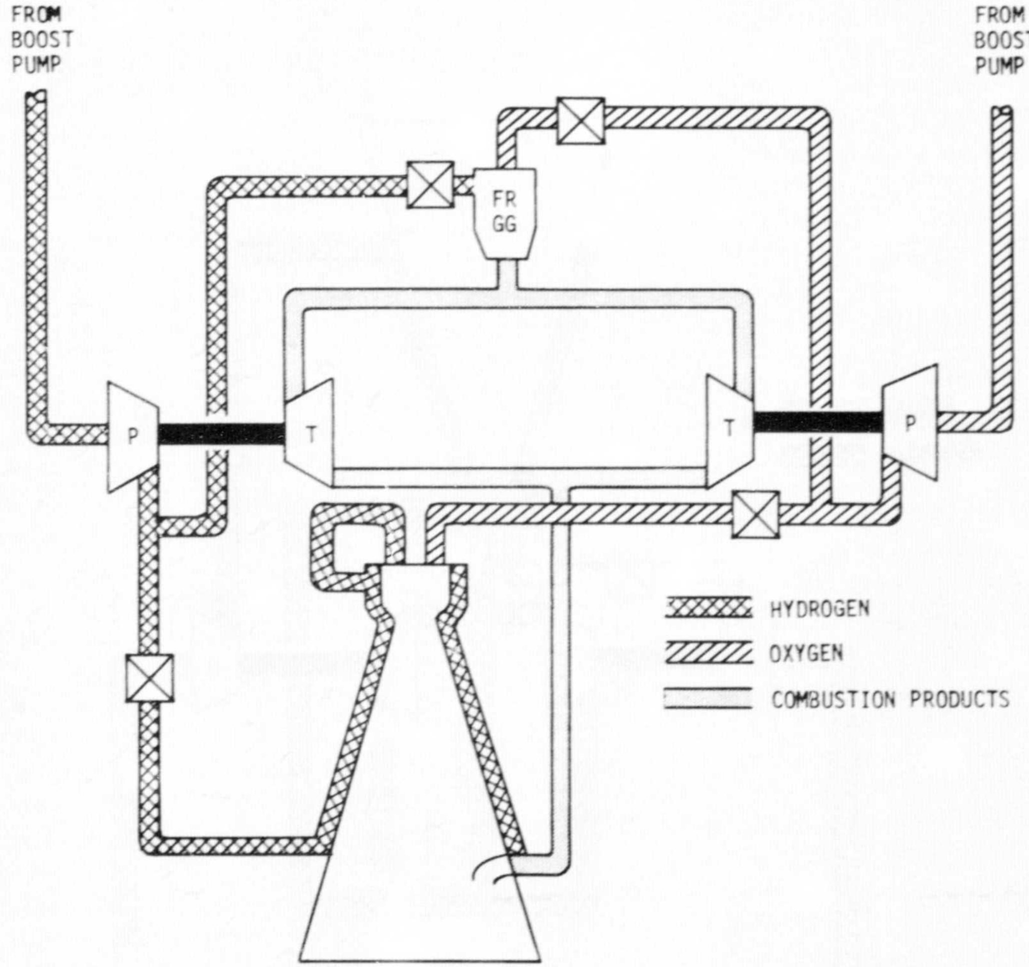


Figure 18. Gas Generator Cycle

B. CYCLE ANALYSIS

Engine system analysis was conducted on each cycle to establish the maximum operating chamber pressure for each as a function of thrust. These analyses consisted of service life, power balance and performance/weight trade-offs. The evaluations were supported by preliminary analyses to establish coolant jacket pressure drop and temperature rise as a function of chamber pressure, thrust and chamber length. As discussed in Section III,D, the use of a safety factor of 1.4 on strain created some invalid thermal results early in the study. Therefore, the data used was obtained through scaling of data available from ASE, RL-10 Derivative and OOS documentation as well as, the preliminary thermal analysis which showed the trends. This data is summarized below:

Thrust, lb	Cycle	Chamber Pressure, psia	Chamber Length, in.	Coolant Jacket Pressure Drop, psia	Turbine Inlet Temperature, °R
10,000	Staged	1500	8.0	320	1860
20,000	Staged	2000	8.0	340	1860
30,000	Staged	2300	8.0	375	1860
10,000	Expander	1300	18.0	305	815
20,000	Expander	1000	18.0	220	590
30,000	Expander	850	18.0	205	465
10,000	Gas Generator	1500	13.0	345	1860
20,000	Gas Generator	1500	13.0	293	1860
30,000	Gas Generator	1500	13.0	280	1860

The lower operating pressure of the expander cycle engine tends to reduce the pressure drop while the longer chamber length causes a pressure drop increase.

Analysis in Task III showed that both the pressure drop and temperature rise data for the expander cycle engine were too high however, the revised thermal data did not significantly change the cycle analysis power balance results (see Section V,E).

Additional power balance analysis guidelines were:

System Pressure Losses ($\Delta P/P$ upstream)

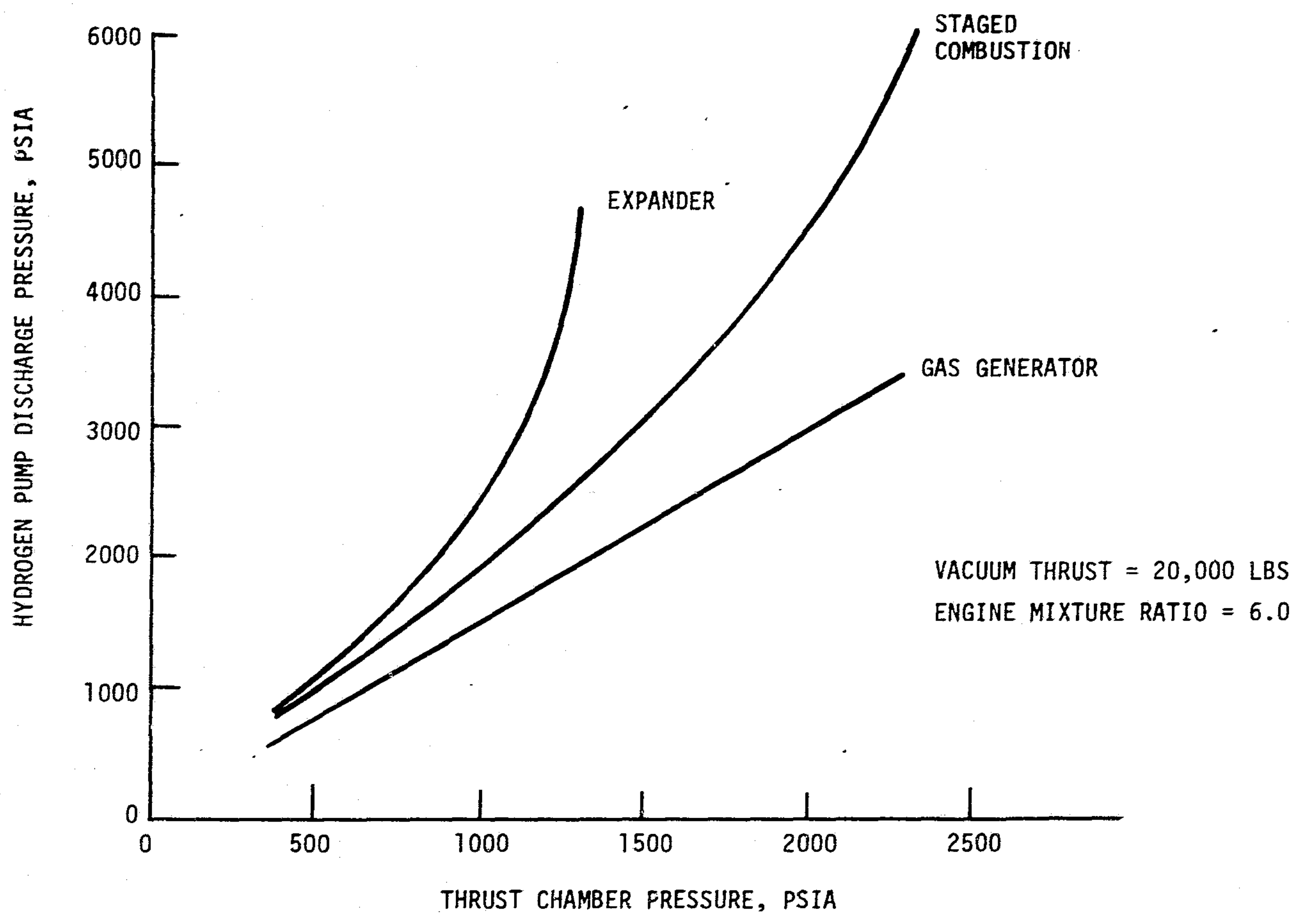
Injectors:

Liquid - 15% (minimum)
Gas - 8% (minimum)

Valves:

Shutoff - 1%
Liquid Control - 5% (minimum)
Gas Control - 10% (minimum)

The cycle power balance results are shown on Figures 19 and 20. Figure 19 shows that the expander cycle engine is power balance limited. The staged combustion cycle engine is ultimately power balance limited but not in the range of chamber pressures investigated in this study. At a given chamber pressure, the expander cycle requires higher hydrogen pump discharge pressures than the staged combustion cycle engine because of the



HIGH LH_2 PUMP DISCHARGE PRESSURES REQUIRED

Figure 19. Hydrogen Pump Discharge Pressure Requirements for Various Engine Cycles

relatively low turbine inlet temperature. The gas generator cycle is not power limited. These power balance results were checked against known design points and were found to be in close agreement. The data used as a check is shown by the following table:

Engine	Thrust, lb	Chamber Pressure, psia	Fuel Pump Discharge Pressure, psia	Oxidizer Pump Discharge Pressure, psia
ASE	20,000	2,000	4,560	4,320
Cat. IV RL-10	15,000	915	2,050	1,320

Figure 20 shows that the oxygen pump discharge pressure requirements do not govern the power balance. The gas generator and expander cycle engines have the same oxidizer pump discharge pressure requirements. The oxygen flow path for both of these cycles is from the pump, through the main oxidizer shutoff valve, and then into the thrust chamber injector.

The expander cycle power balance limit was evaluated as a function of thrust. The results of this analysis is shown on Figure 21. The expander cycle is harder to power balance at higher thrusts because the thrust chamber coolant outlet temperature (turbine inlet temperature) decreases with increasing thrust. This occurs because the chamber surface area increases with only the square root of thrust while the available coolant is directly proportional to thrust. In other words, the harder-to-cool engines are easier to power balance. The thrust chamber pressure points for this phase of the study were selected at 80% of the maximum attainable chamber pressure.

The practical upper limit on chamber pressure for the staged combustion cycle engine is governed by the service life requirement. Chamber life cycle and thermal analyses conducted for this study, and the OOS and ASE design studies have shown that there is a practical upper limit on

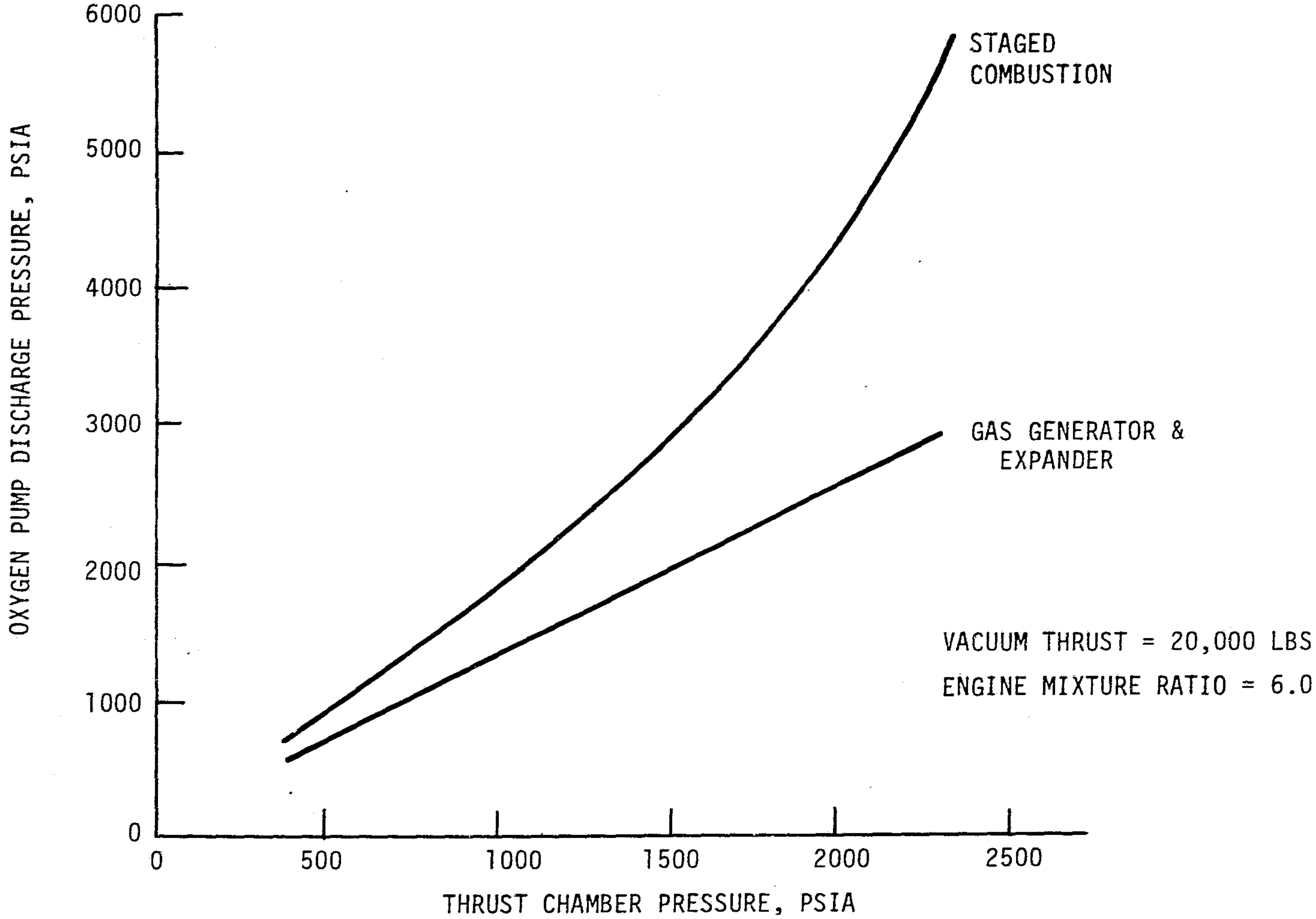
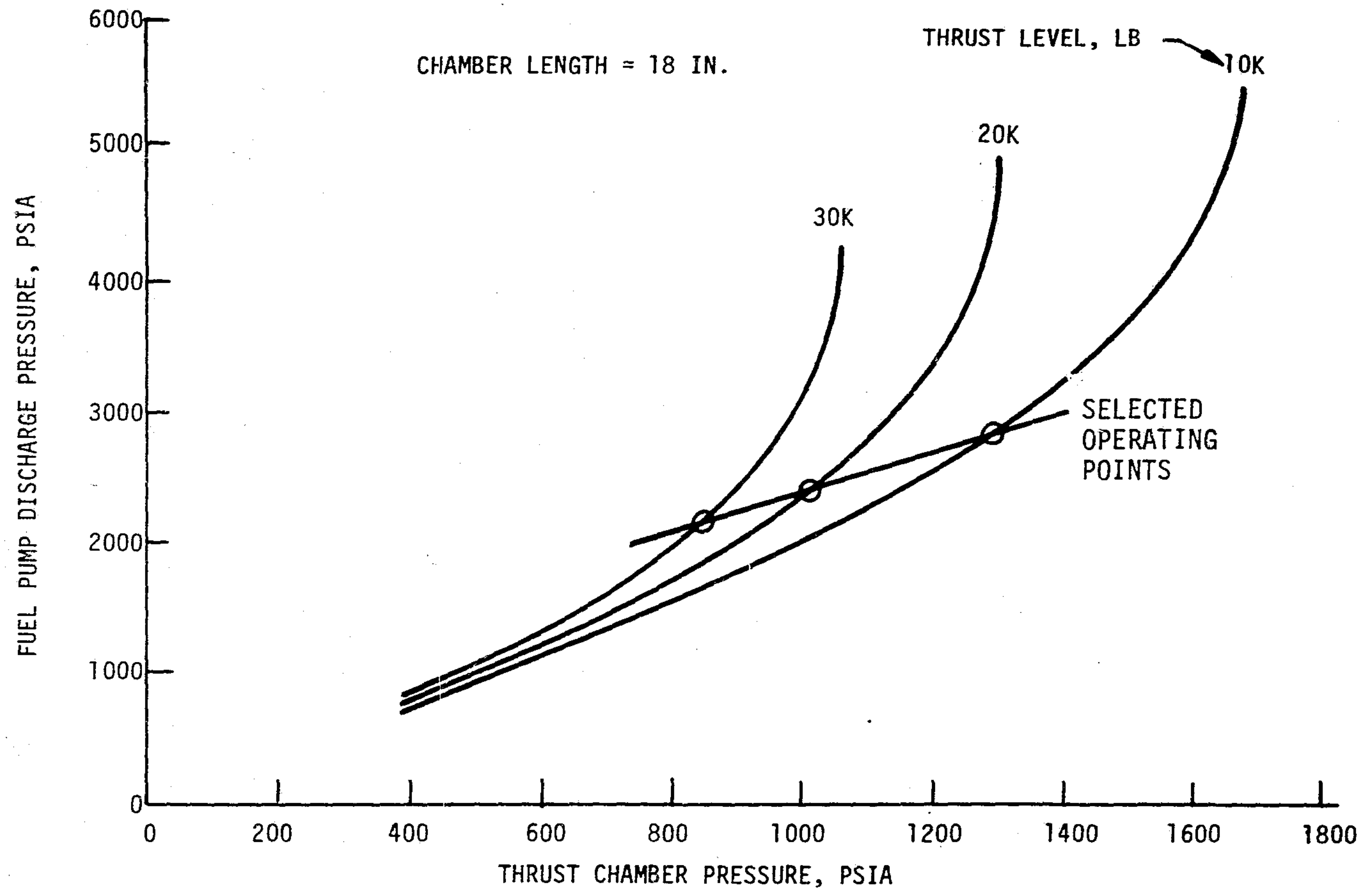


Figure 20. Oxygen Pump Discharge Pressure Requirements for Various Engine Cycles



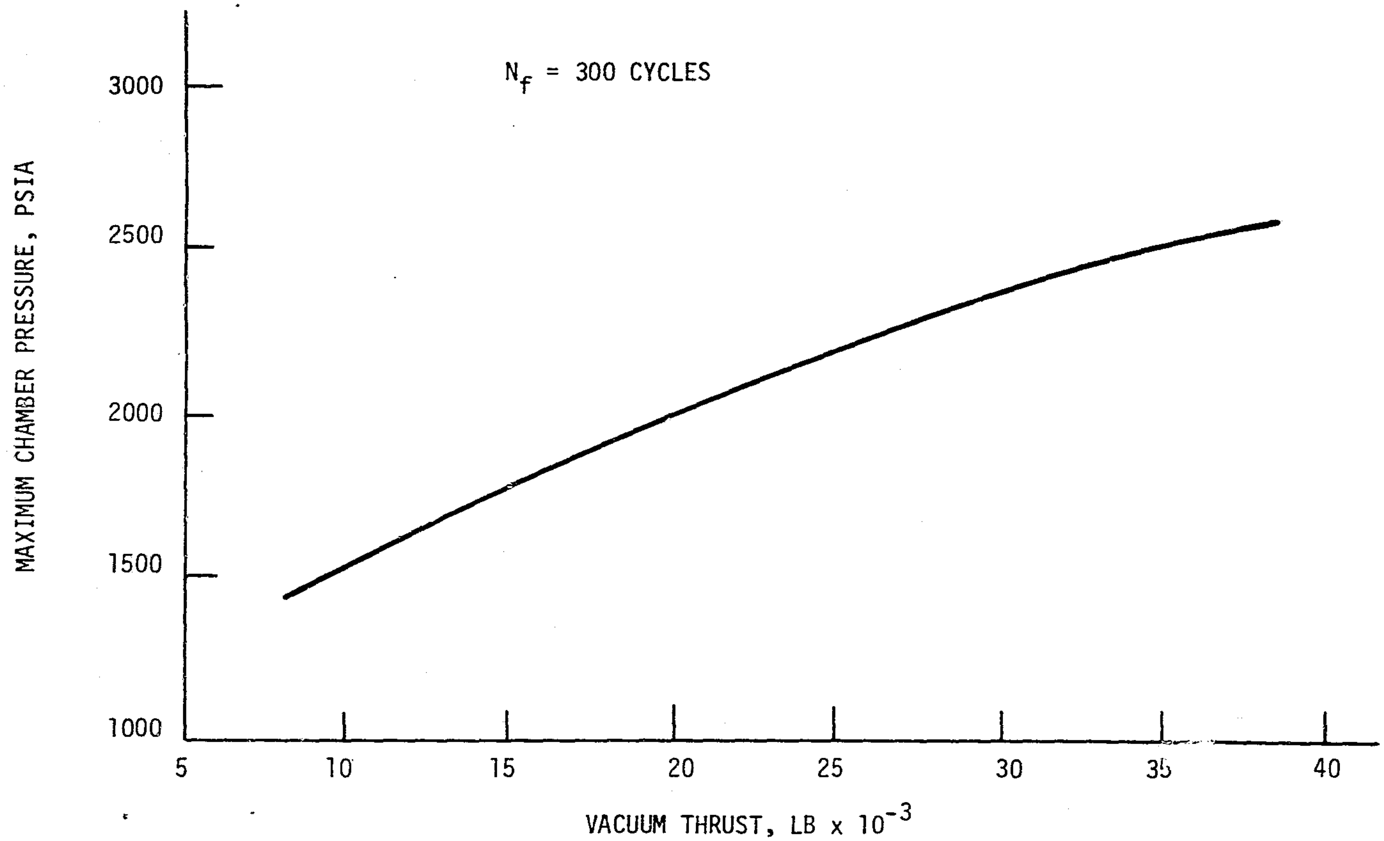
POWER BALANCE MORE DIFFICULT AT HIGH THRUST

Figure 21. Effect of Thrust Level Upon Expander Cycle Power Balance Limits

operating chamber pressure. Either pressure drops become too high because of high coolant velocities or the coolant bulk temperature rise limit reached. The chamber pressure limits imposed by the chamber life requirement are shown on Figure 22.

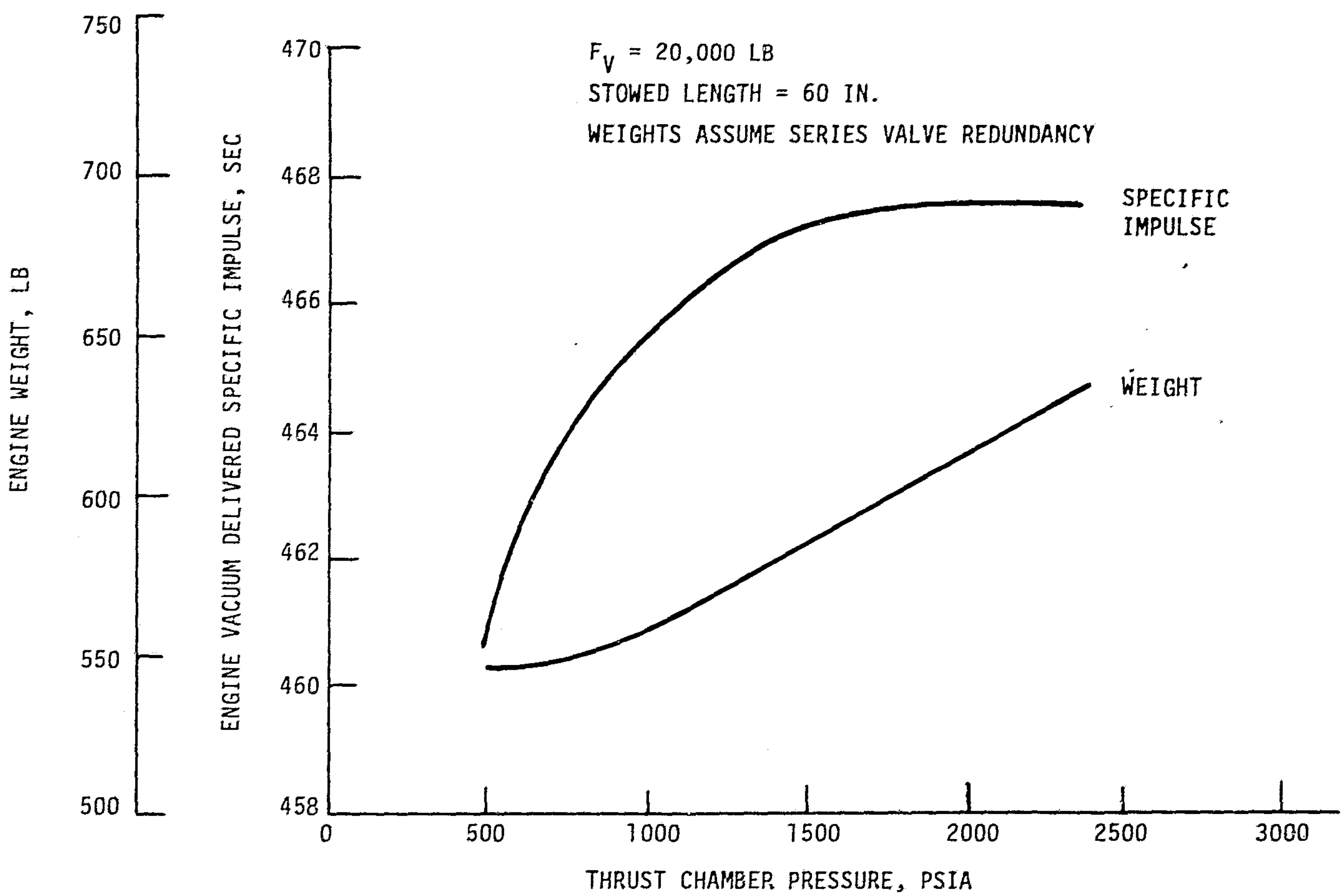
The gas generator cycle could operate at chamber pressures at least as high as the staged combustion cycle. However, because of the large performance loss due to the turbine drive flow at high pressures, it was not found desirable to do so. Figure 23 shows that the gas generator cycle engine specific impulse levels off with increasing chamber pressure. The performance loss associated with the turbine drive flow is almost directly proportional to chamber pressure. The increase in theoretical performance obtained at the higher nozzle area ratios resulting from chamber pressure increases does not make up for the turbine exhaust loss beyond 2000 psia. Engine weight increases with chamber pressure because turbopump and other pressure dependent component weights increase. Engine length is fixed. Therefore, the envelope is always filled and nozzle weight remains almost constant. Weight/I_S trades were performed using the AMOTV payload partials previously presented. This resulted in a maximum recommended chamber pressure of 1500 psia for the gas generator cycle as shown on Figure 24. This value was used in all remaining study efforts for this cycle.

The maximum operating pressures for the engine cycle candidates investigated in this concept definition phase are shown on Table III and Figure 25 as a function of thrust level. All cycles appear to be life cycle limited at a thrust level of approximately 8K lbF and lower.



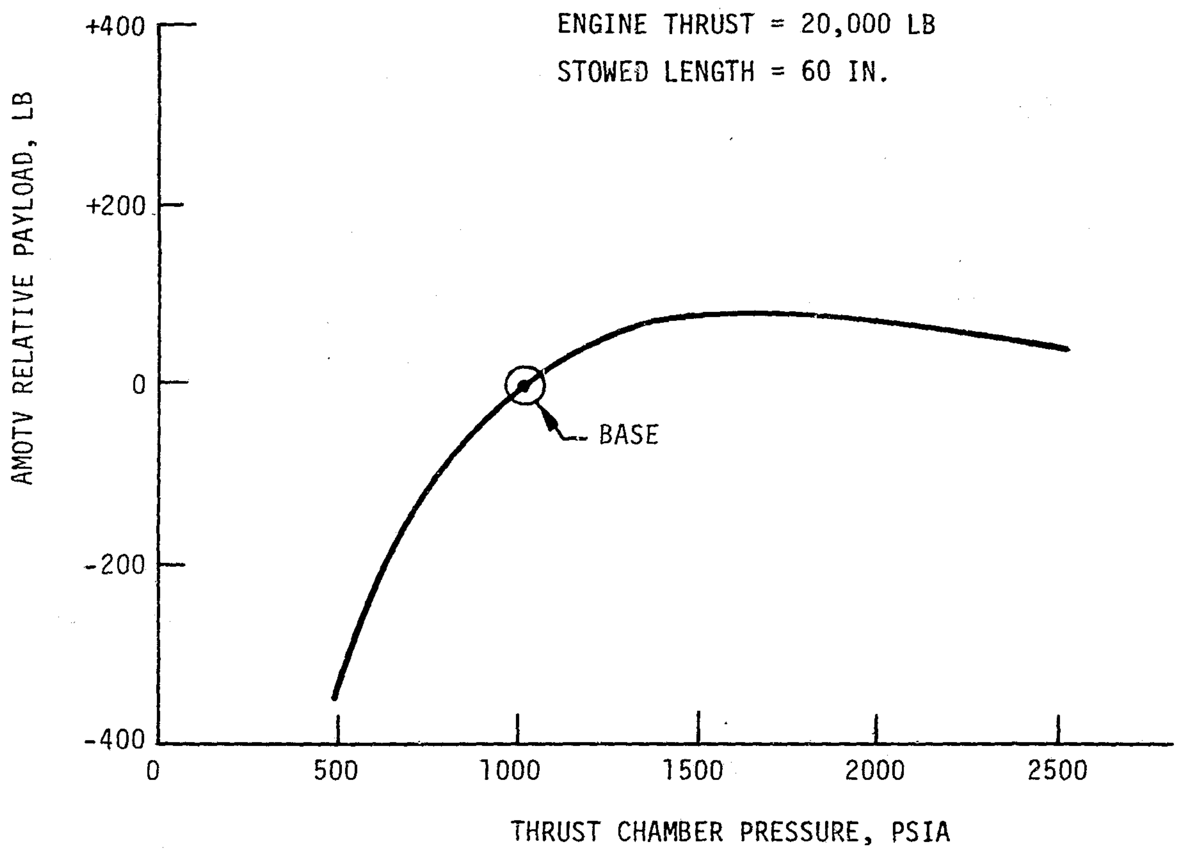
STAGED COMBUSTION CYCLE LIFE LIMITED

Figure 22. Predicted Maximum Chamber Pressure for Chamber Life Cycle Requirement vs Thrust



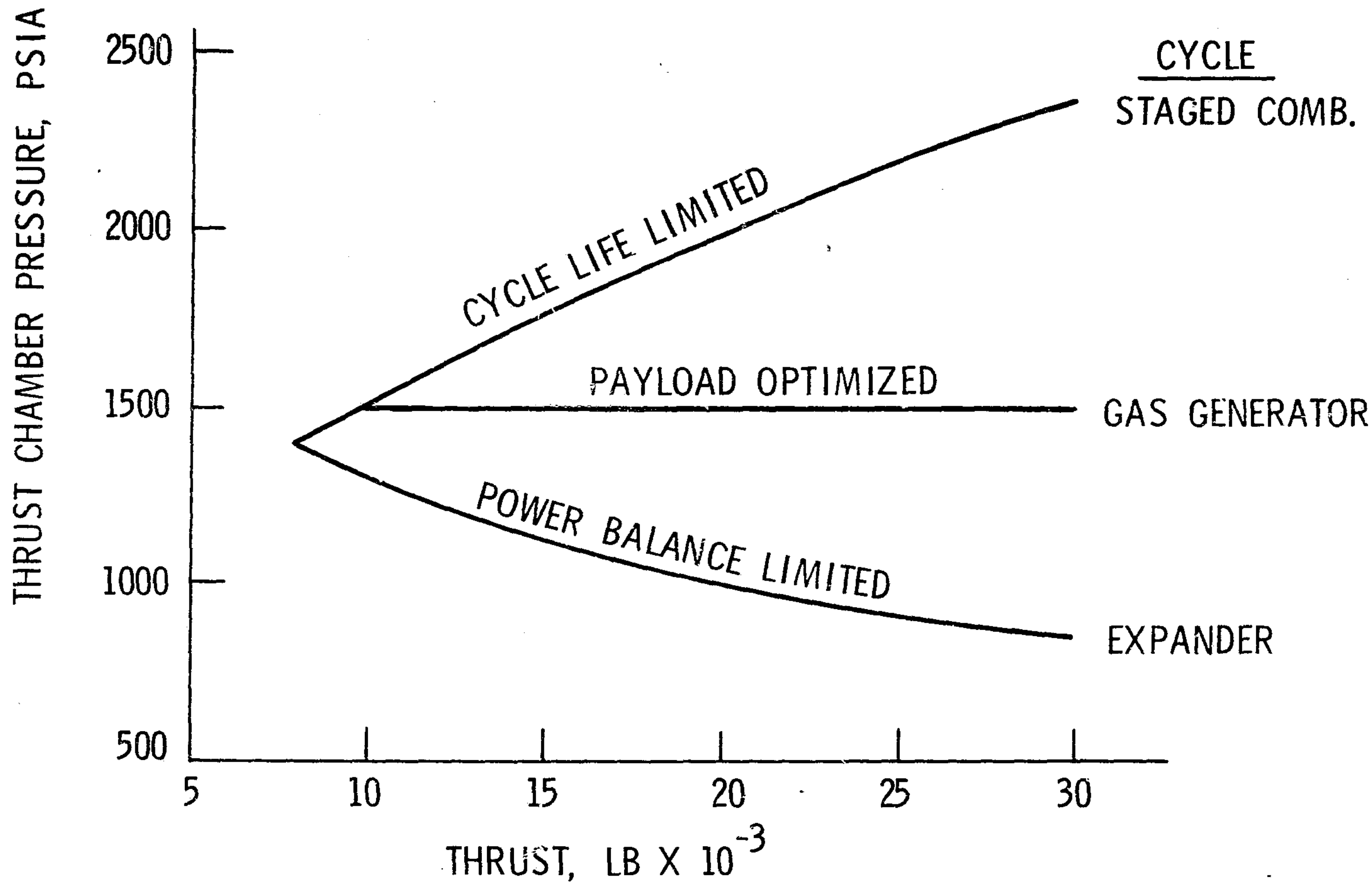
PERFORMANCE GAINS BEYOND 1500 PSIA ARE SMALL

Figure 23. Gas Generator Cycle Engine Performance and Weight Data



PAYLOAD MAXIMIZES AT 1500 PSIA

Figure 24. Gas Generator Cycle Chamber Pressure Optimization



ALL CONCEPTS CYCLE LIFE LIMITED BELOW 8K LBF

Figure 25. Maximum Engine Operating Pressures

TABLE III

TASK II MAXIMUM ENGINE OPERATING PRESSURES

Cycle	Vacuum Thrust, K lb	Thrust Chamber Pressure, psia	Limitation
STAGED COMBUSTION	8	1400	CHAMBER CYCLE LIFE
	10	1500	
	20	2000	
	30	2350	
EXPANDER	8	1400	CHAMBER CYCLE LIFE
	10	1300	POWER BALANCE
	20	1000	
	30	850	
GAS GENERATOR	8	1400	CHAMBER CYCLE LIFE
	10	1500	PERFORMANCE
	20	1500	
	30	1500	

C. CANDIDATE CYCLE PERFORMANCE, WEIGHT AND TRADEOFF ANALYSIS

The performance and weight data for the various cycle candidates were calculated in order to perform the system tradeoffs. The data was evaluated as a function of thrust, at the chamber pressures established by the cycle analysis and for a maximum engine length with the extendible nozzle retracted (stowed length) of 60 inches.

Engine delivered performance data were calculated for an energy release efficiency goal of 99.5% and minimum chamber length requirements were established accordingly. Chamber lengths of approximately 8 and 13 inches were established for the staged combustion and gas generator cycle engines, respectively. A chamber length of 18 inches was selected for the expander cycle engine for power balance reasons.

Simplified JANNAF performance prediction techniques (Reference 10) were used to determine the other performance losses. The boundary layer loss charts in the simplified procedures were adjusted to agree with the latest experimental data obtained at an area ratio of 400:1, a thrust level of 20,000 lb and a chamber pressure of 2000 psia (Reference 11). For these test conditions, the experimental data indicates that the old procedures predicted a boundary layer loss approximately 4 seconds too high.

Engine weight calculation requires careful consideration of not only those components included, but what is not included in the weight statement. For this study the components included in the total engine weight are:

Gimbal
Injector
Copper Chamber and Nozzle

Tube Bundle Nozzle
Radiation Cooled Nozzle Extension
Nozzle Deployment Mechanism
Main Valves and Actuators
LOX Boost Pump
LH_2 Boost Pump
LOX Main TPA
LH_2 Main TPA
Preburner/Gas Generator (where applicable)
Propellant Lines
Ignition System
Heat Exchanger
Engine Controller
Miscellaneous Valves
Miscellaneous (Electrical Harness, Instrumentation and Brackets)

Weights do not include the gimbal actuators and actuation system, pre-valves or a contingency which is normally included in the vehicle weight statement.

For each engine component in the weight statement, weight correlation equations have been developed and modeled as a function of the main design parameters such as flow rate, pressure, area ratio, thrust, etc. This method was used successfully in the parametric engine weight development on many previous design studies such as the UNTS (Contract NAS 3-20109), Hi P_c (Contract NAS 3-19727), OOS (Contract F04611-71-C-0040), Space Tug Storable Engine (Contract NAS 8-29806), and the parametric analyses conducted for the early Phase B Space Shuttle Main Engine Definition Study (Contract NAS 8-26188). Existing pump-fed engines such as, the Agena, RL-10 and Titan II second stage and study engines such as, the OOS, Storable Tug and RL-10 derivatives provided the original basis for component weight data in the desired thrust range.

The model was updated to meet the requirements of the Mixed-Mode Orbit Transfer Vehicle Study, Contract NAS 3-21049 and includes the ASE component weight data base. Where historical component weight data did not exist, weights were obtained from past related studies and preliminary design information.

The engine weight data used in the concept evaluation is shown on Table IV and Figure 26. The figure shows the weights with series valve redundancy and redundant igniters which was the reliability and safety analysis recommendations. For single engine installations, the engine must have quad redundant valves. This is based upon the Apollo SPS experience. These weights are also shown in the table. The data show that the higher pressure engines are heavier because the pressure dependent component weights increase while the nozzle weight remains almost constant. The available envelope is always totally used. In addition, the gas generator and staged combustion cycle engines require more components than an expander cycle engine.

The performance of the various engine cycle candidates in a single engine installation are compared to each other and the minimum requirement on Figure 27. As expected, the figure shows that the staged combustion cycle specific impulse is the highest. None of the cycles can achieve the minimum specific impulse requirement at 10K lbF thrust. However, the staged combustion and expander cycle performance values close in at low thrust.

Table V shows the pertinent single engine installation data. It should be noted that the engine weight data includes quad redundant main propellant valves and redundant igniters in all combustion devices. Trade-offs of engine performance and weight for each of the cycles were performed using the derived AMOTV payload partials. The change in payload with I_S was computed against the minimum I_S requirement and the change in payload with weight was computed against the 20K lbF thrust baseline staged combustion cycle weight. This data is shown on the table and Figure 28. The staged combustion

TABLE IV

TASK II ENGINE WEIGHT COMPARISONS

CYCLE	THRUST K LB	CHAMBER PRESSURE PSIA	ENGINE DRY WEIGHT, LB		
			NO REDUNDANCY	WITH SERIES VALVE REDUNDANCY(1)(2)	WITH QUAD. VALVE REDUNDANCY(1)(3)
STAGED COMBUSTION	8K	1400	404	442	469
	10K	1500	443	484	514
	20K	2000	619	674	718
	30K	2350	778	844	899
EXPANDER	8K	1400	365	397	424
	10K	1300	402	435	462
	20K	1000	516	551	580
	30K	850	623	660	691
GAS GENERATOR	8K	1400	375	410	434
	10K	1500	407	444	470
	20K	1500	542	585	617
	30K	1500	652	702	741

(1) ALSO INCLUDES REDUNDANT IGNITION SYSTEMS IN ALL COMBUSTION DEVICES.

(2) WEIGHTS FOR MULTIPLE ENGINE INSTALLATIONS.

(3) WEIGHTS FOR SINGLE ENGINE INSTALLATIONS.

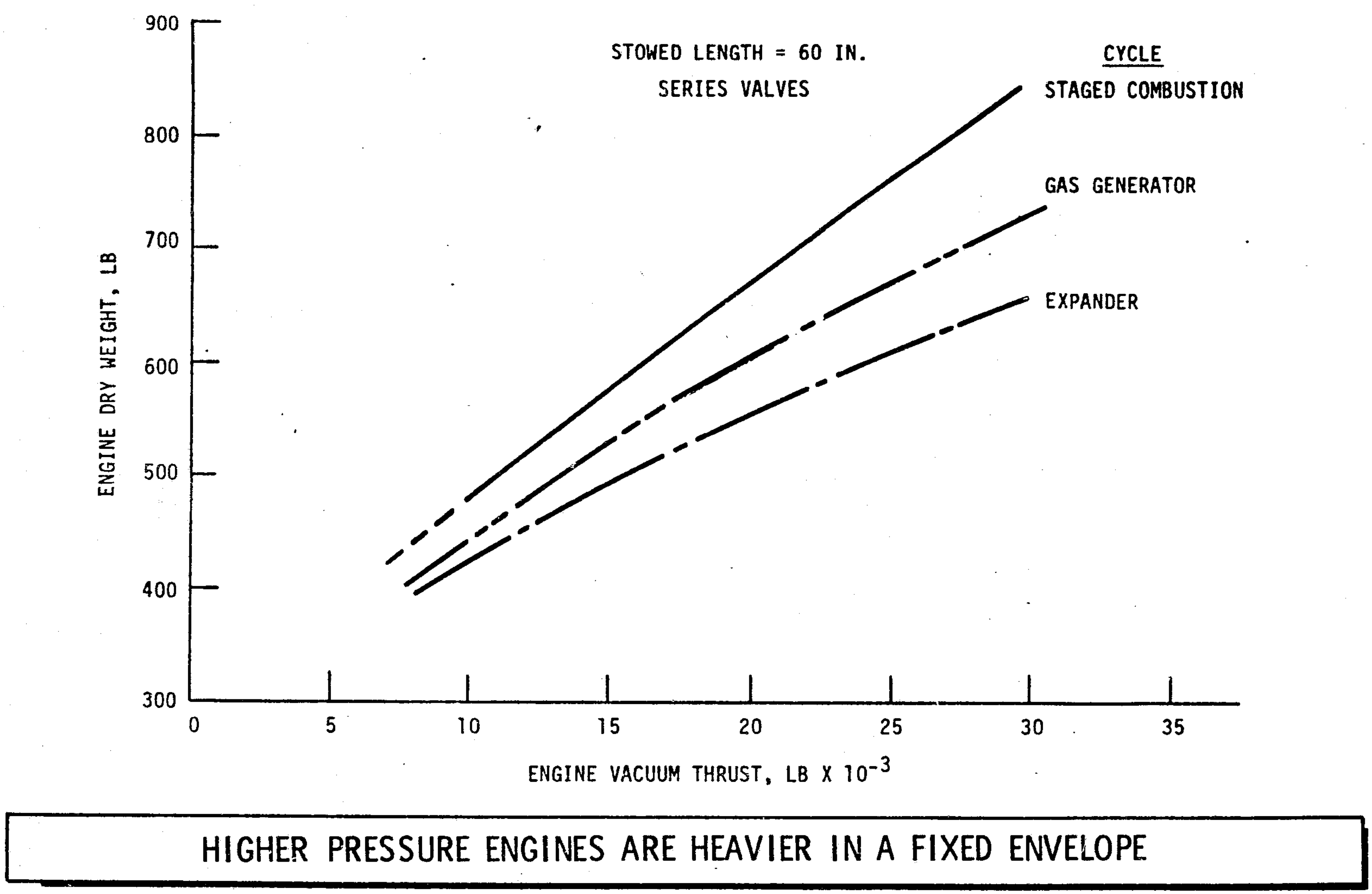
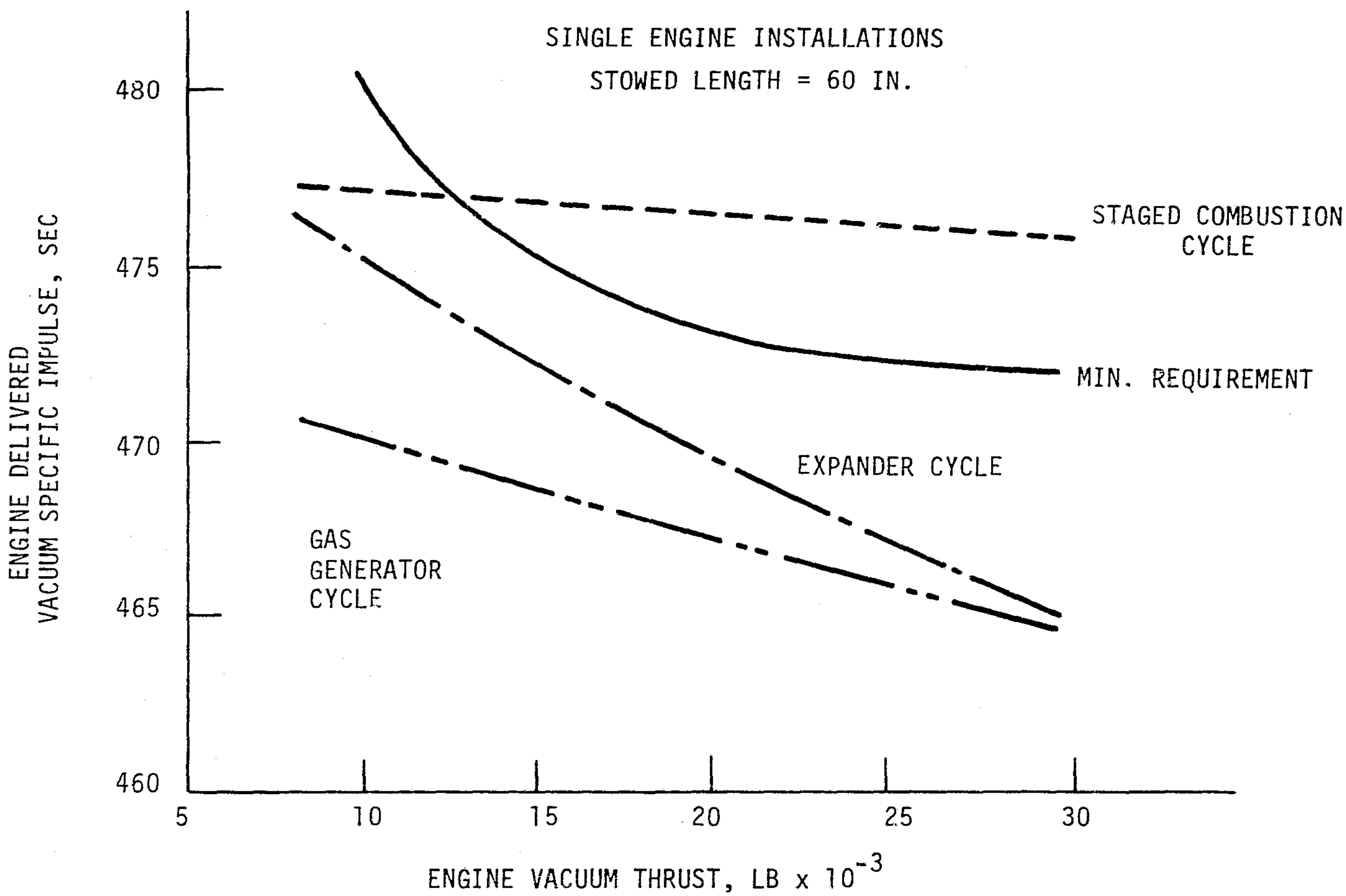


Figure 26. Engine Weight Comparisons



STAGED CYCLE ENGINE DELIVERS HIGHEST I_S

Figure 27. Engine Cycle Performance Comparisons at Maximum Operating Pressures

TABLE V

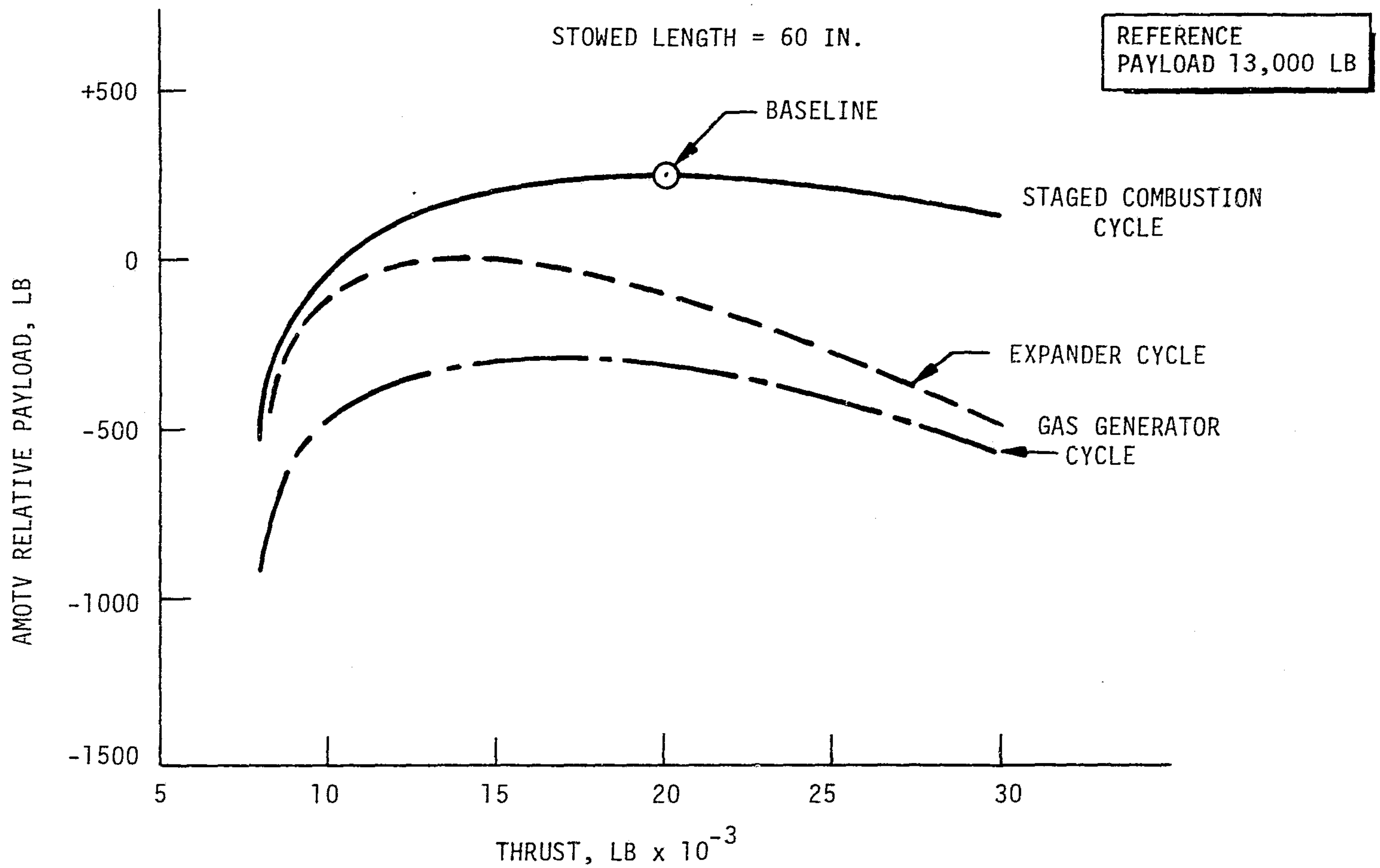
SINGLE ENGINE INSTALLATION DATA SUMMARY

(STOWED ENGINE LENGTH = 60 IN.)

CYCLE	THRUST, LB	MIN. REQ. I_S SEC	CHAMBER PRESSURE, PSIA	NOZZLE AREA RATIO	VACUUM DELIVERED SPECIFIC IMPULSE, SEC	ENGINE DRY WEIGHT,[2] LB	AMOTV RELATIVE PAYLOAD,[1] LB
STAGED COMB.	8K	488	1400	1034	477.2	469	-515
	10K	480	1500	938	477.0	514	+5
(BASELINE)	20K	473	2000	699	476.5	718	+256
	30K	472	2350	552	475.7	899	+71
EXPANDER	8K	488	1400	898	476.2	424	-538
	10K	480	1300	782	475.1	462	-76
	20K	473	1000	309	469.4	580	-111
	30K	472	850	186	464.9	691	-489
GAS GENERATOR	8K	488	1400	968	470.8	434	-943
	10K	480	1500	858	470.0	470	-457
	20K	473	1500	486	467.2	617	-312
	30K	472	1500	327	464.6	741	-566

[1] COMPUTED FOR MIN I_S REQUIREMENT AT EACH THRUST LEVEL
BASE ENGINE WEIGHT = 718 LB.

[2] ASSUMES QUAD. MAIN ENGINE VALVES



EXPANDER CYCLE IS COMPETITIVE AT LOW THRUST

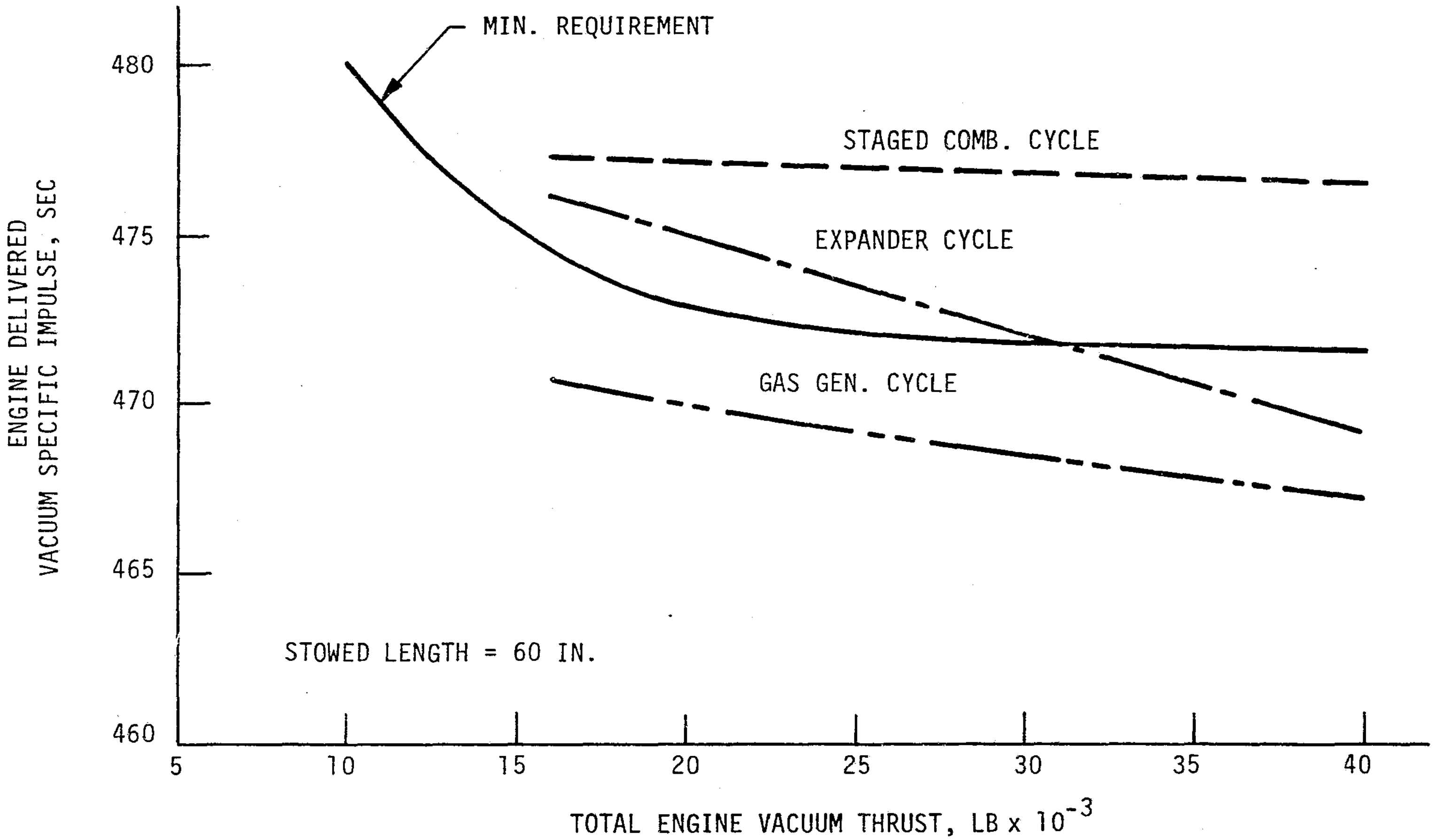
Figure 28. Engine Cycle Payload Comparisons, Single Engine Installation

and expander cycles have approximately the same payload capability at low thrust. The gas generator cycle results in significant payload losses.

The payload and performance data indicate that low thrust engines are attractive. The reliability and safety analysis concluded that a minimum of two engines are required. Therefore, multiple engine installations were investigated in this concept definition phase.

Engine performance for a twin engine installation is shown on Figure 29. Both the staged and expander cycle engines are capable of meeting the minimum specific impulse requirements in this configuration. Engine performance and weight data are shown on Table VI. The weights shown include series main propellant valve redundancy and redundant ignition systems in all combustion devices. Performance and weight trade-offs were performed using the payload partials for the AMOTV. Payload changes were computed against the minimum I_S requirement and a single engine installation baseline weight of 718 lb. Figure 30 shows that the staged combustion and expander cycle engines have approximately the same payload capability at a 20K lbF total thrust level. The gas generator cycle results in relatively high payload penalties over the entire thrust range.

Performance, weight and relative payload data are shown for an installation of three engines on Figure 31, Table VII and Figure 32. Although the staged combustion and expander cycle engines exceed the minimum specific impulse requirement, the weight penalty results in significant payload losses. The staged combustion and expander cycles have approximately equal payload capability. High payload penalties again result with a gas generator cycle. The relative payload capability was again computed against the minimum I_S requirement and a single engine staged combustion cycle baseline weight of 718 lb.



STAGED AND EXPANDER CYCLES MEET MIN. I_S REQUIREMENTS

Figure 29. Multiple Engine Installation Performance, Two Engines

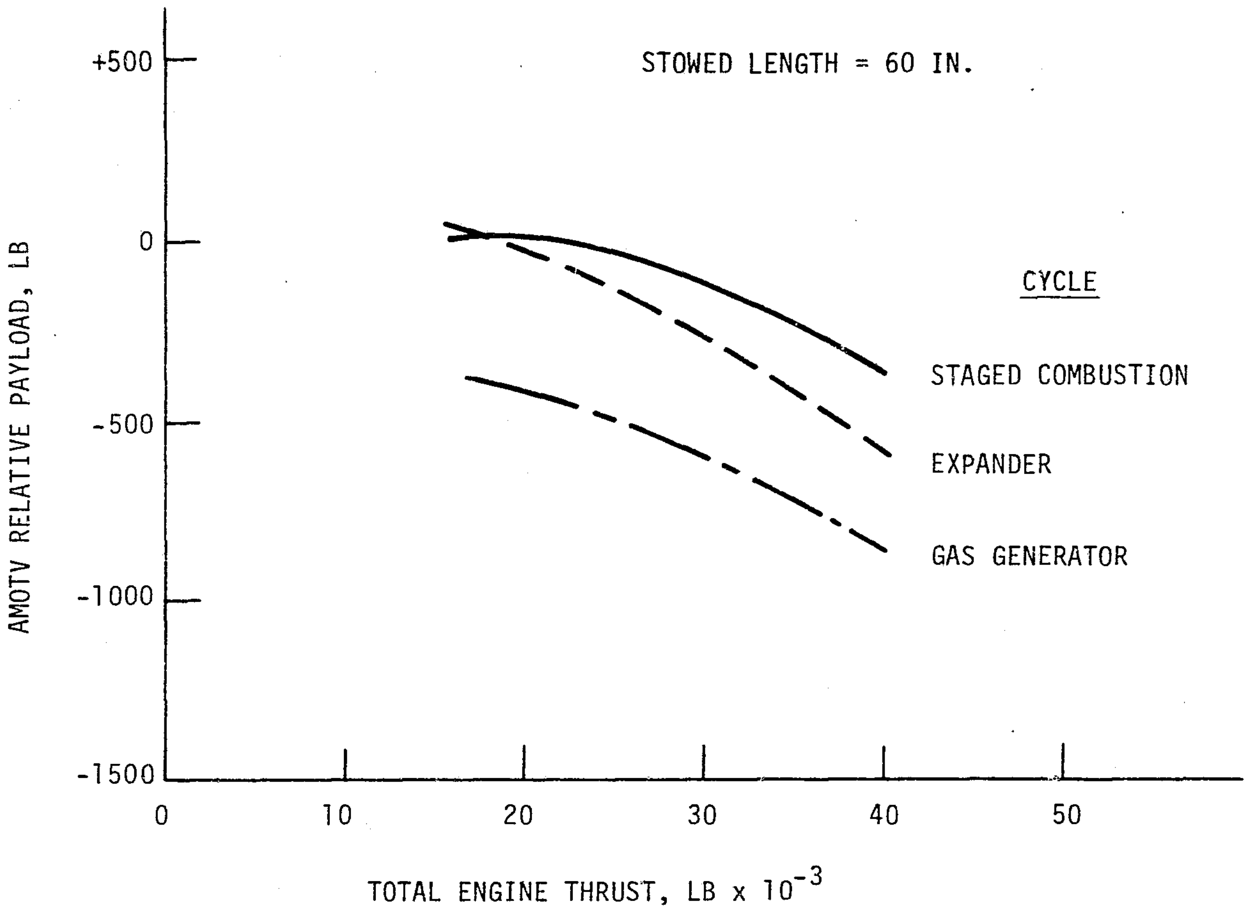
TABLE VI

TWIN ENGINE INSTALLATION DATA SUMMARY

(STOWED ENGINE LENGTH = 60 IN.)

CYCLE	TOTAL THRUST, KLB	MIN I_S REQ., SEC	NO. OF ENGINES	THRUST PER CHAMBER, KLB	VACUUM DELIVERED SPECIFIC IMPULSE, SEC	TOTAL ENGINE DRY(1) WEIGHT, LB	AMOTV RELATIVE PAYLOAD, LB
STAGED	16	474.5	2	8	477.2	893	+5
	20	473.0		10	477.0	979	+5
	24	472.4		12	477.0	1053	-33
	30	472.0		15	476.9	1170	-140
	40	471.7		20	476.5	1369	-366
EXPANDER	16	474.5	2	8	476.2	803	+31
	20	473.0		10	475.1	881	-26
	24	472.4		12	474.0	929	-115
	30	472.0		15	472.0	1004	-315
	40	471.7		20	469.4	1123	-613
GAS GENERATOR	16	474.5	2	8	470.8	829	-392
	20	473.0		10	470.0	899	-418
	24	472.4		12	469.4	969	-495
	30	472.0		15	468.5	1076	-649
	40	471.7		20	467.2	1191	-849

(1) Assumes series valve redundancy and redundant ignition systems.



STAGED AND EXPANDER CYCLE PAYLOAD CAPABILITIES ABOUT EQUAL

Figure 30. Engine Cycle Payload Comparisons, Two Engine Installation

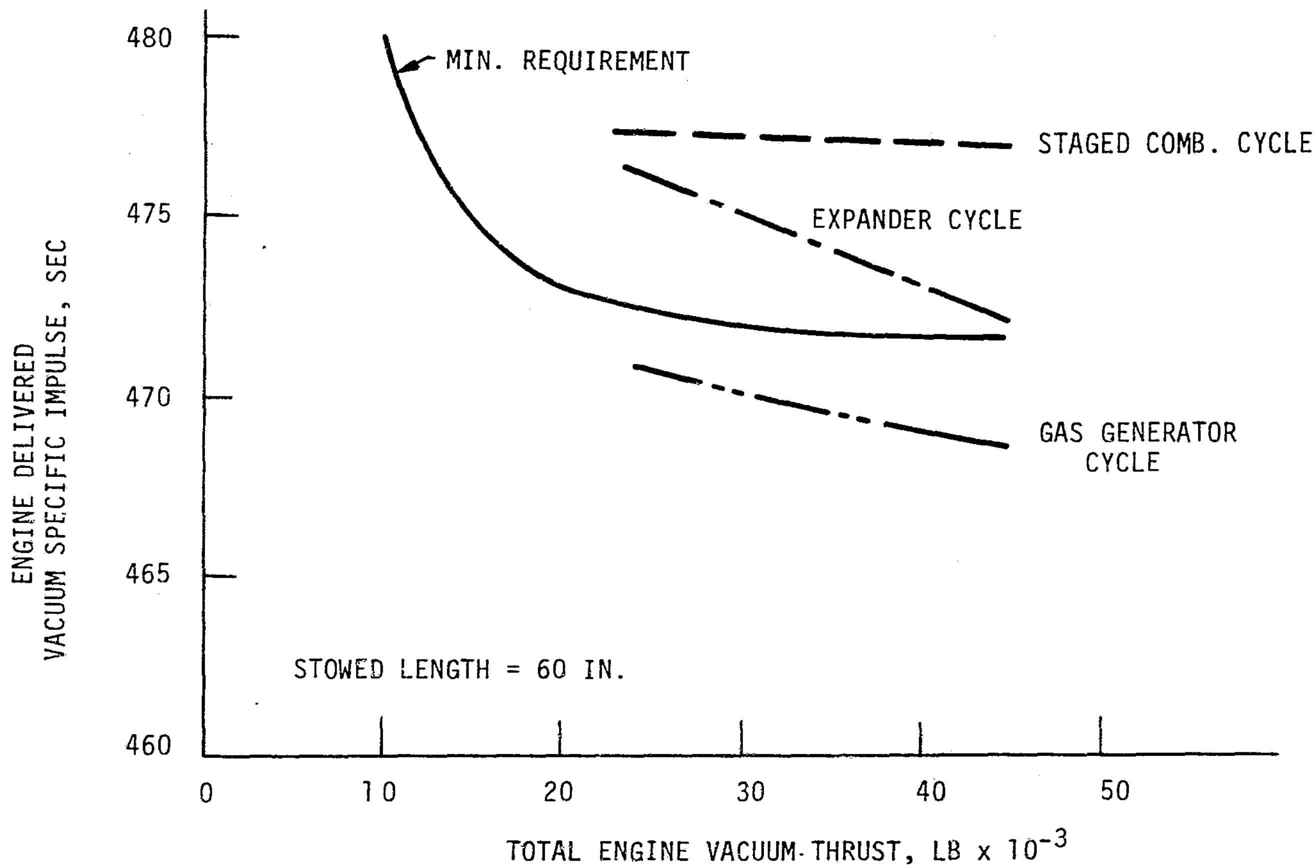


Figure 31. Multiple Engine Installation Performance, Three Engines

TABLE VII

THREE ENGINE INSTALLATION DATA SUMMARY

STOWED LENGTH = 60 IN.

CYCLE	TOTAL THRUST, KLB	MIN I_s REQ., SEC	NO. OF ENGINES	THRUST PER CHAMBER, KLB	VACUUM DELIVERED SPECIFIC IMPULSE, SEC	TOTAL DRY ENGINE WEIGHT, LB (1)	AMOTV RELATIVE PAYLOAD LB
STAGED COMB.	24	472.4	3	8	477.2	1339	-333
↓	30	472.0	↓	10	477.0	1468	-460
↓	36	471.8	↓	12	477.0	1579	-568
↓	45	471.6	↓	15	476.9	1755	-754
EXPANDER	24	472.4	3	8	476.2	1191	-243
↓	30	472.0	↓	10	475.1	1321	-437
↓	36	471.8	↓	12	474.0	1393	-582
↓	45	471.6	↓	15	472.0	1506	-838
GAS GENERATOR	24	472.4	3	8	470.8	1243	-694
↓	30	472.0	↓	10	470.0	1348	-839
↓	36	471.8	↓	12	469.4	1454	-985
↓	45	471.6	↓	15	468.5	1614	-1212

(1) Weights include series valve redundancy and redundant ignition systems.

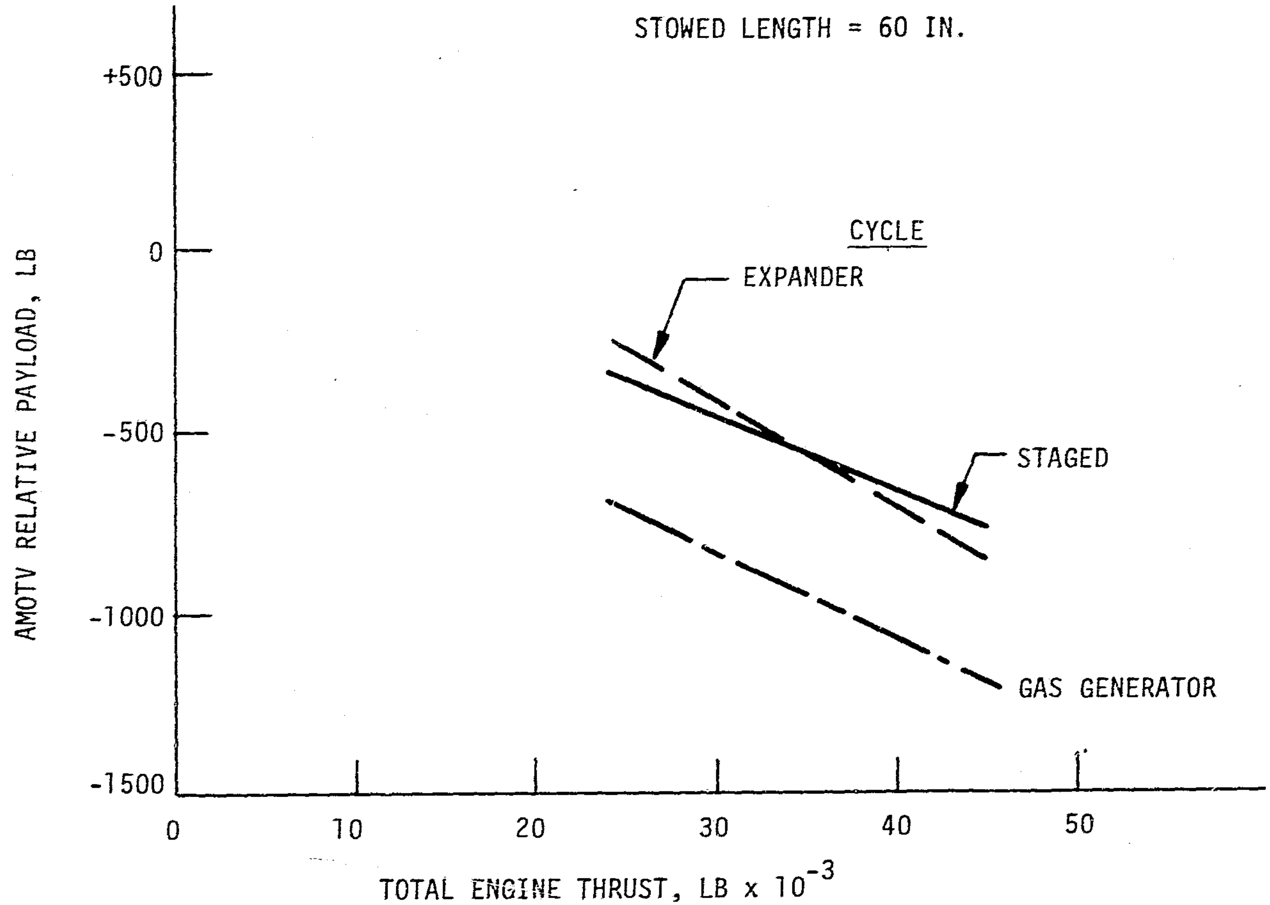


Figure 32. Engine Cycle Payload Comparisons, Three Engine Installation

The effect of the number of engines upon the relative payload capability of the AMOTV is summarized on Figure 33. Multiple engine installation payload capabilities of staged combustion and expander cycle engines are essentially a "draw". The payload capability of multiple gas generator cycle engines is approximately 500 lb less than the other two candidates.

In addition to the concept comparison studies, analyses were conducted to determine the sensitivity of the OTV performance to the engine requirements. The effect of the thrust and stowed length requirements upon the OTV relative payload capability is shown on Figure 34. The figure shows that 20,000 lb thrust is about optimum at 60 inches stowed lengths. The change is toward a lower total thrust level for smaller stowed length. The change in payload between 60 and 70 inches stowed length is not as large as between 50 and 60 inches because the rate of change in specific impulse with nozzle area ratio diminishes at the high stowed length.

The sensitivity of the vehicle performance to the engine operating chamber pressure was also evaluated. Figure 35 shows that small changes in operating pressures, particularly at low thrust, have a minor effect upon the vehicle payload capability. At high thrust (30K), reductions in chamber pressure for the expander cycle engine have a bigger affect than for a staged combustion cycle engine. This occurs because of the relatively low operating pressure and area ratio of the expander cycle engine at 30K lb thrust. At 10K lb thrust the change in payload capability is about the same for either cycle.

D. CONCEPT SELECTION

Based upon the engine requirements review, cycle performance comparisons and ALRC in-house studies which evaluated development risk and life cycle cost for each candidate, an engine concept was selected as a baseline

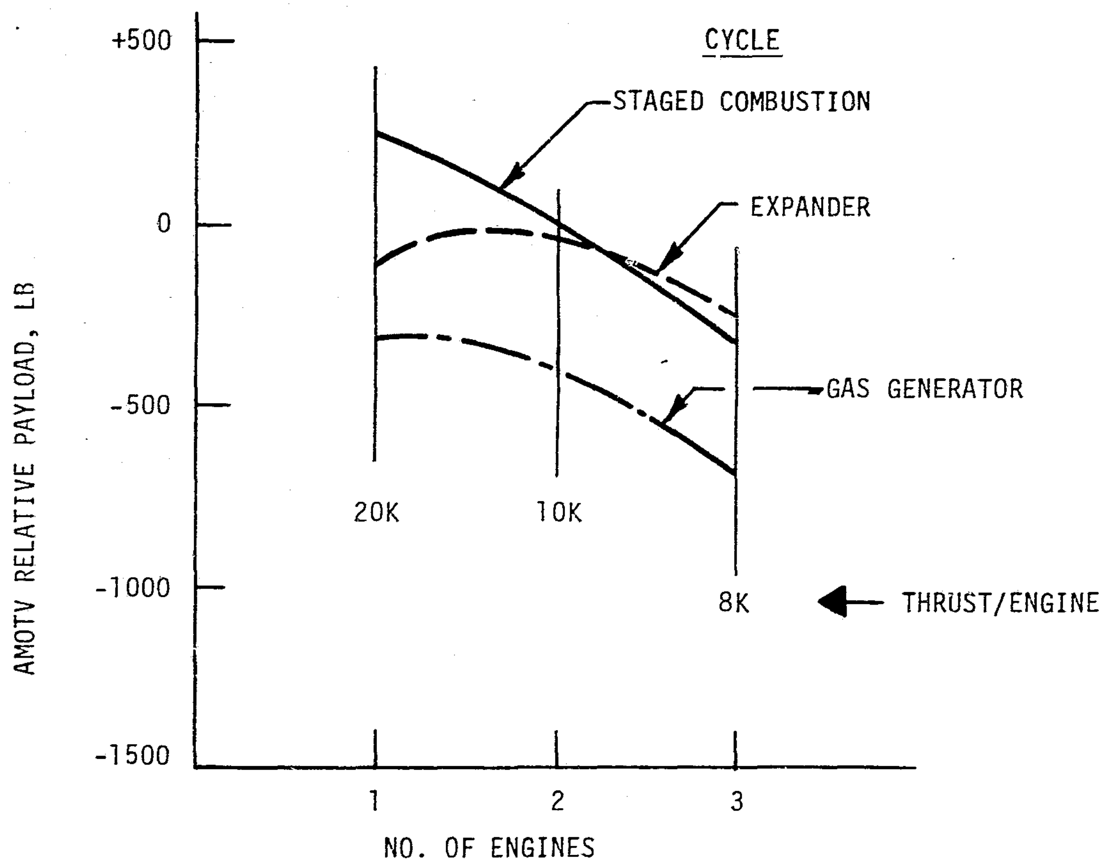


Figure 33. Effect of Number of Engines Upon Relative Payload Comparisons

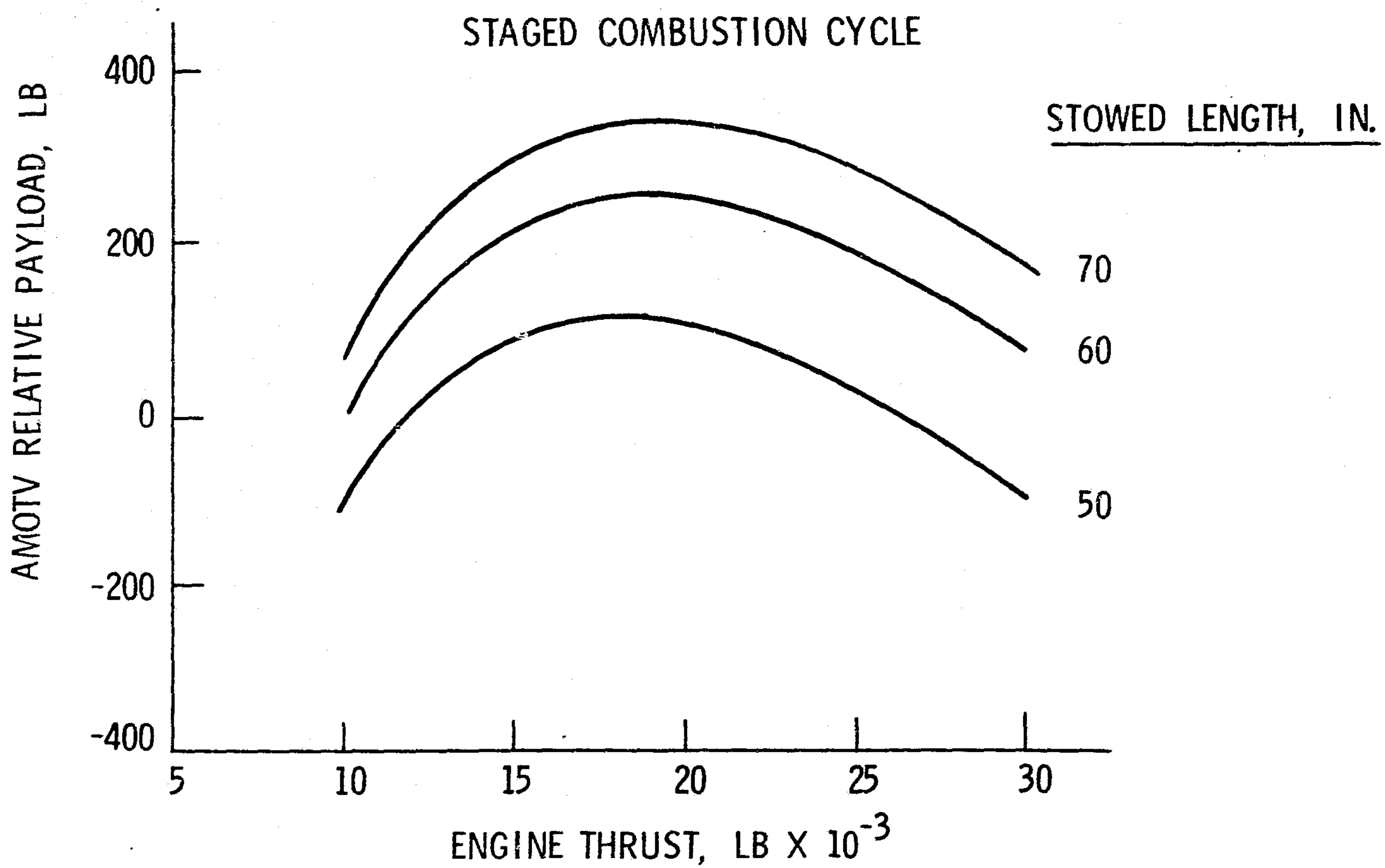


Figure 34. Effect of Thrust and Stowed Length Requirements Upon Relative Payload Capability

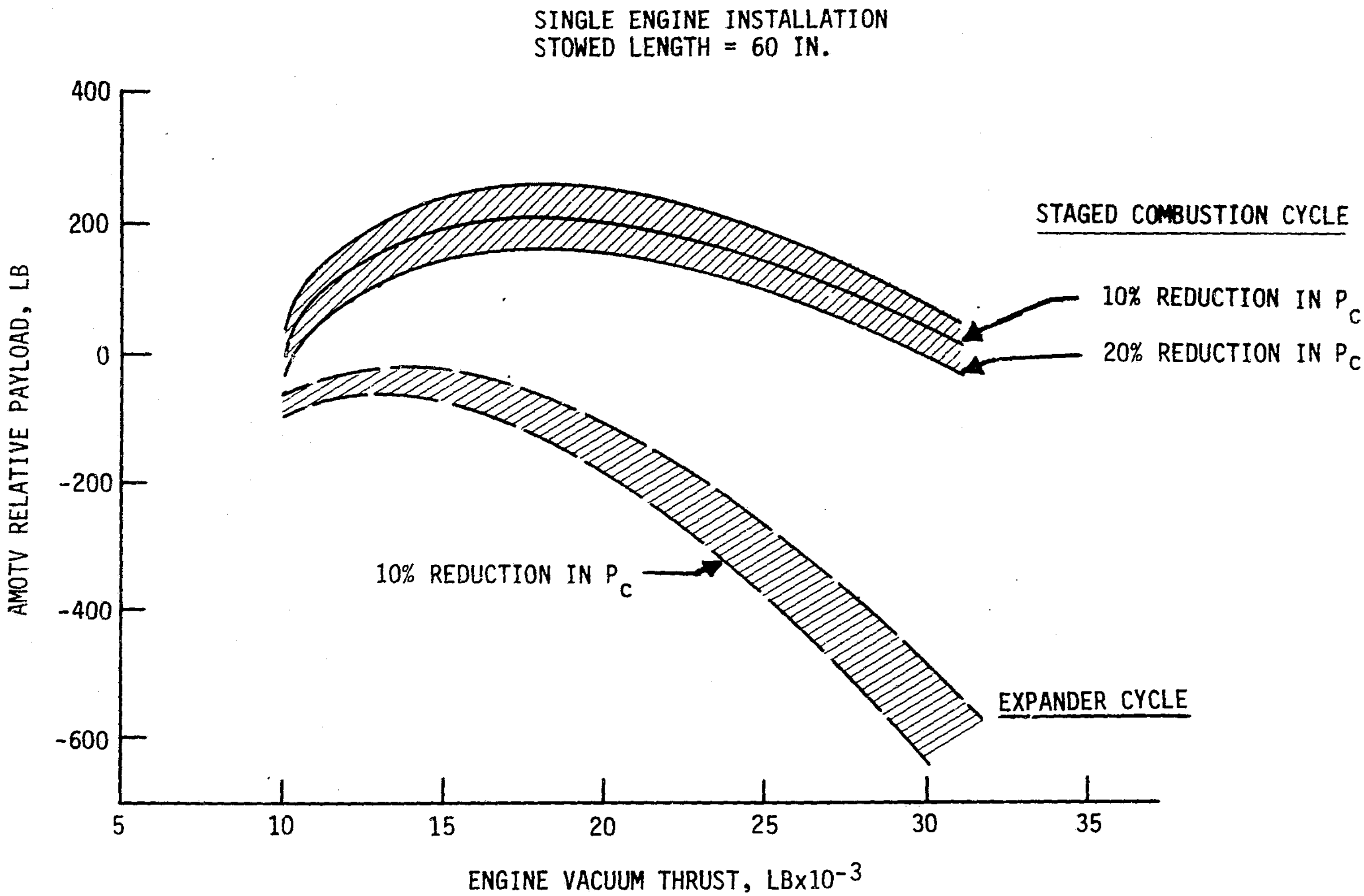


Figure 35. Effect of Operating Chamber Pressure on OTV Relative Payload Capability

for the remaining study efforts. The recommended engine configuration is twin 10K lb thrust expander cycle engines. This recommendation was approved by NASA for continuation of the study. The logic for this selection is summarized herein.

Figures 36 and 37 summarize the trade study and the requirements and concept selection study conclusions. Crew safety dictates a minimum of two engines. The payload trades show that 8K to 12K lb thrust engines provide the best multiple engine installation. In this thrust range, the expander cycle and staged combustion cycles have approximately the same payload capability, creating a technical stalemate. The gas generator cycle engine was eliminated because of its low performance and resulting payload penalties. Because a cycle selection could not be made purely on a technical (payload) basis, in-house studies were conducted which evaluated the development risk and life cycle costs of the engine cycles. These studies concluded that the expander cycle development risk is lower than the staged combustion and gas generator cycles primarily due to the elimination of a critical combustion device. The risk, reliability and safety analyses for staged combustion and expander cycles will be amplified in an extension to this study. The preliminary in-house costing study concluded that the expander cycle engine offers a large potential life cycle cost savings. This conclusion is supported by cost estimate results presented in Volume III of this final report. Further cost analyses and comparisons will be made in the Phase A, Extension I Study for this contract.

- A MULTIPLE ENGINE INSTALLATION OF EXPANDER OR STAGED CYCLE ENGINES HAVE APPROXIMATELY THE SAME PAYLOAD CAPABILITY
- 8K TO 12 K ENGINES PROVIDE ATTRACTIVE MULTIPLE ENGINE INSTALLATIONS
- A TOTAL ENGINE THRUST OF 20K LBF APPEARS TO BE ABOUT OPTIMUM ON A PAYLOAD BASIS
- GAS GENERATOR CYCLE ENGINES RESULT IN PAYLOAD PENALTIES IN SINGLE OR MULTIPLE ENGINE INSTALLATIONS
- TWO ENGINES ARE BETTER THAN THREE ON A PAYLOAD AND CREW SAFETY BASIS

Figure 36 Engine Cycle Payload/I_s/Weight Trade Conclusions

- MAN-RATING, PERFORMANCE, REUSABILITY AND SERVICE LIFE REQUIREMENTS MAKE DEVELOPMENT OF A NEW ENGINE NECESSARY
- CREW SAFETY DICTATES A MINIMUM OF TWO ENGINES
- MULTIPLE ENGINE VEHICLE RELATIVE PAYLOAD TRADES RESULT IN A STAGED COMBUSTION vs EXPANDER CYCLE DRAW
- GAS GENERATOR CYCLE ENGINE ELIMINATED BECAUSE OF PAYLOAD PENALTIES

IN-HOUSE STUDY RESULTS

- DEVELOPMENT RISK WITH EXPANDER CYCLE IS MUCH LESS BECAUSE A CRITICAL COMBUSTION DEVICE IS ELIMINATED
- POTENTIAL MAXIMUM LIFE CYCLE SAVINGS FOR EXPANDER CYCLE OF 480 MILLION DOLLARS

- RECOMMENDED ENGINE DESIGN APPROACH:

  TWIN 10K LBF MAN-RATED EXPANDER CYCLE ENGINE

Figure 37. Engine Requirements and Concept Selection Study Conclusions

V. TASK III - PARAMETRIC ENGINE DATA

The primary objective of this task was to provide parametric engine data for the three candidate engine cycle concepts. The parametric ranges were:

Thrust Level	- 10,000 to 30,000 lb
Maximum Stowed Length	- 50, 60 and 70 inches

Nominal mixture ratio was specified as 6.0, although off-design operation up to a mixture ratio of 7.0 was evaluated.

Supporting analyses were conducted to provide the data necessary to perform the parametric studies. These supporting studies included performance, structural, thermal, turbomachinery, controls, cycle and materials analyses. These analyses and their results are discussed in this section. Primary emphasis was placed upon the recommended expander cycle engine concept.

A. PERFORMANCE ANALYSIS

Engine delivered performance was calculated using the previously referenced simplified JANNAF performance prediction techniques and summarized on Figure 38. The purpose of this analysis was to verify that the energy release goal of 99.5% is attainable with a coaxial injector.

The expander cycle thrust chamber geometry requirements are more critical for the heat transfer and power balance requirements than for the combustion or performance requirements. The combustion process of the proposed OTV propellants (gaseous hydrogen and liquid oxygen) was divided and analyzed as two different combustion mechanisms. First, the mixing efficiency of the gas-liquid combination was based upon a cold flow correlation (Reference 12). This correlation yields the mixing efficiency for a coaxial

- SIMPLIFIED JANNAF PERFORMANCE PREDICTION PROCEDURES
  - THEORETICAL PERFORMANCE - ODE PROGRAM
  - ENERGY RELEASE EFFICIENCY - 99.5%
  - TWO DIMENSIONAL AERODYNAMIC NOZZLE EFFICIENCY - DIVERGENCE LOSS CHARTS
  - KINETIC EFFICIENCY - ODK PROGRAM
  - BOUNDARY LAYER - SIMPLIFIED CHARTS

    BOUNDARY LAYER LOSS DATA ADJUSTED FOR:

    I_S = 473 SECS AT:

    F_V = 20,000 LB

    P_C = 2000 PSIA

    ϵ = 400

Figure 38. Performance Prediction Methodology

type injector as a function of the momentum of the injected propellants. Second, the chamber length necessary to achieve 99.5+ % vaporization of the liquid oxygen was obtained from a simplified Priem analysis (Reference 13).

The OTV thrust chamber lengths necessary to achieve 99.5% energy release efficiency (includes mixing and vaporization) were evaluated over a range of chamber pressure, thrust, contraction ratio, mixture ratio and injector element density.

The injector element density has a strong influence on the chamber length as shown on Figure 39. The element density and physical size is constrained by manufacturing tolerances. A density of 6 per square inch was selected for this study on the basis of the ASE design experience.

Figure 40 shows the chamber length (L') requirement is not thrust dependent and an L' of 18 inches is more than adequate for the expander cycle to achieve the 99.5% ERE goal.

Chamber length requirements are a strong function of thrust chamber pressure as shown on Figure 41. The 18 in. length chamber chosen for the expander cycle is still more than adequate over the total chamber pressure range analyzed. For engines, like the staged combustion cycle, which are not dependent upon the heat input into the hydrogen for power balancing, shorter chamber lengths can be used.

Figures 42 and 43 show the effect of the chamber contraction ratio and design mixture ratio upon the chamber length requirement. Mixture ratio has little affect while the L' decreases as the contraction ratio is increased.

The performance analysis indicates that the ERE goal of 99.5% appears to be practical for the OTV engine.

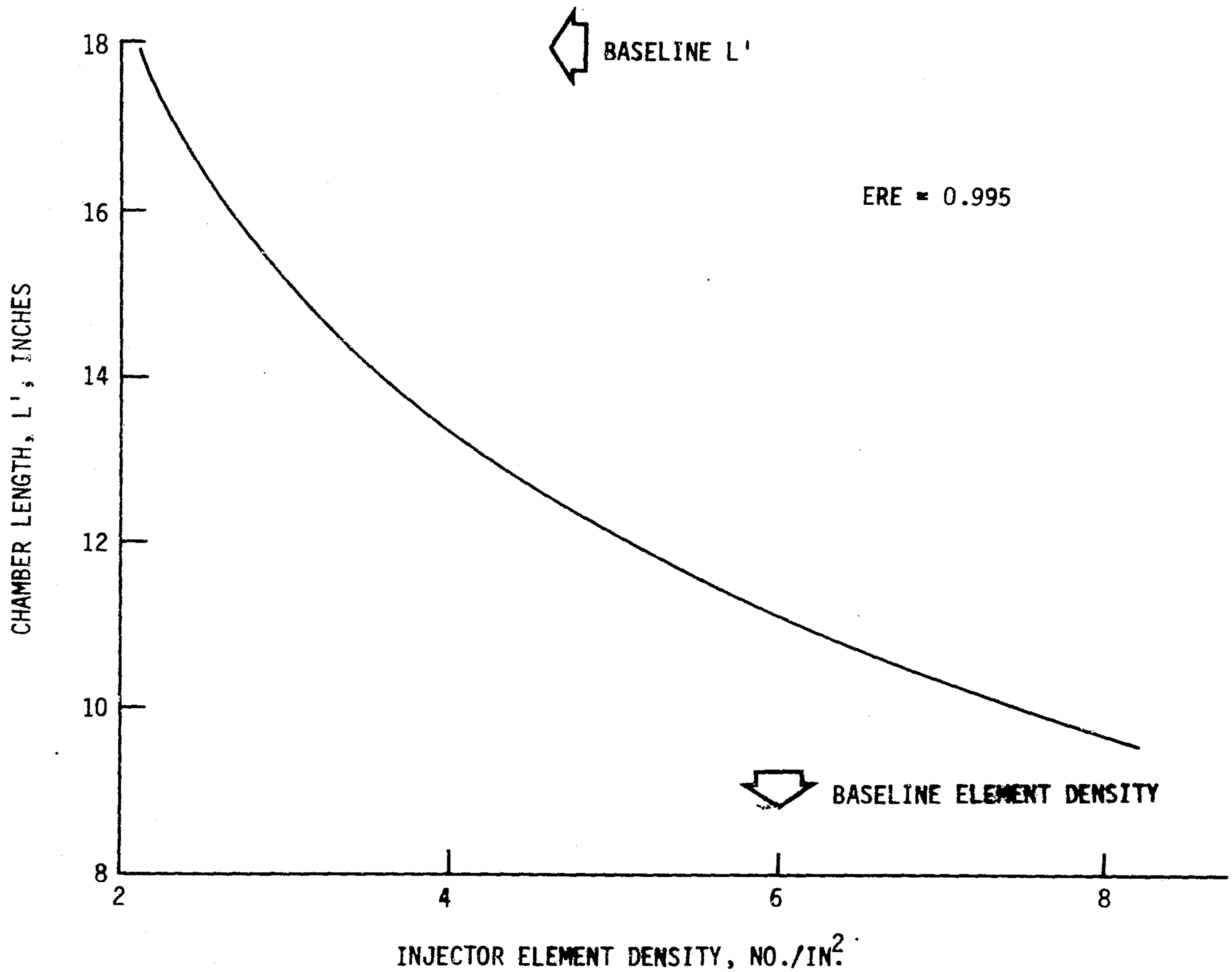


Figure 39. Chamber Length vs Injector Element Density

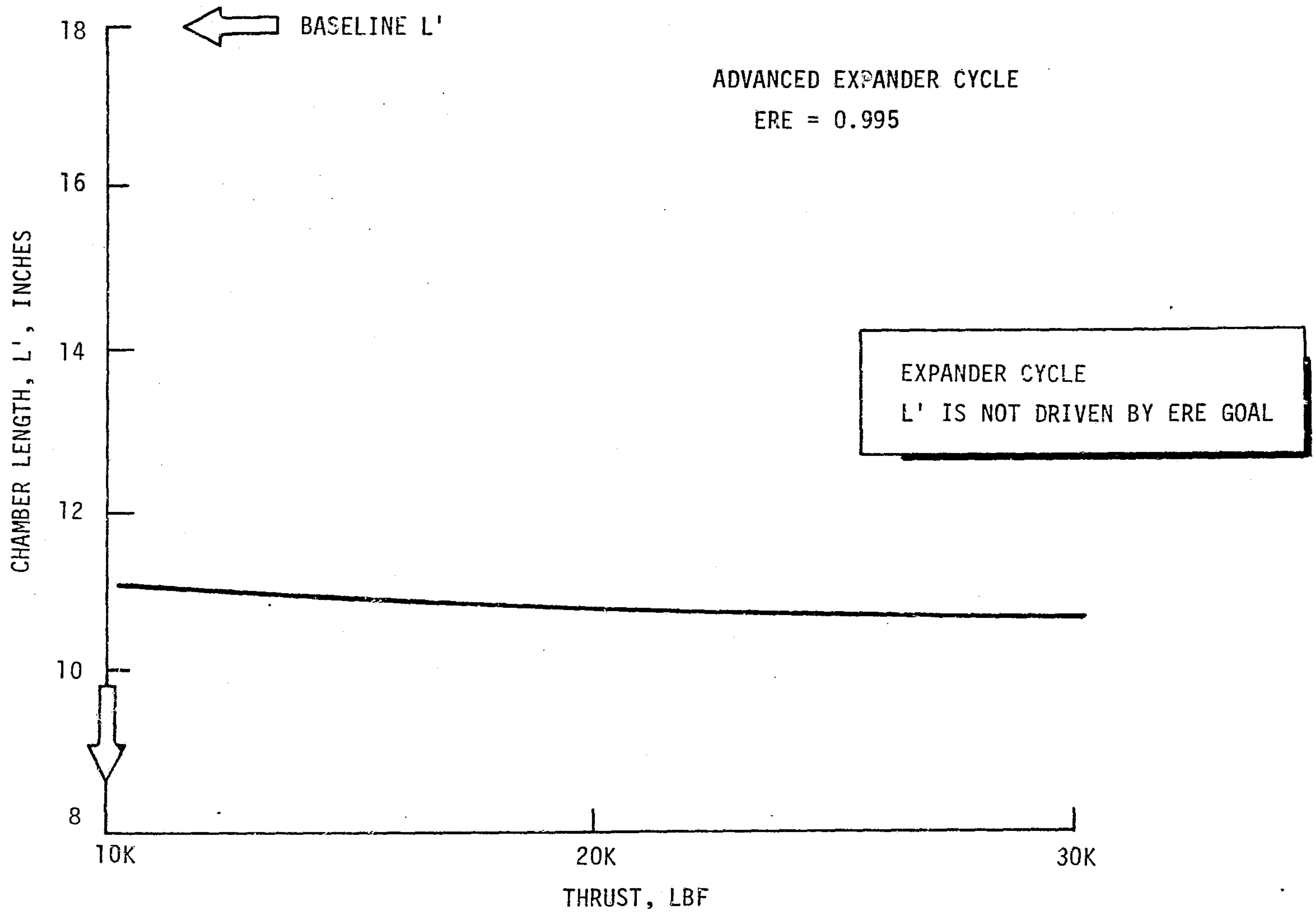


Figure 40. Minimum Chamber Length Requirements to Meet Performance Goal

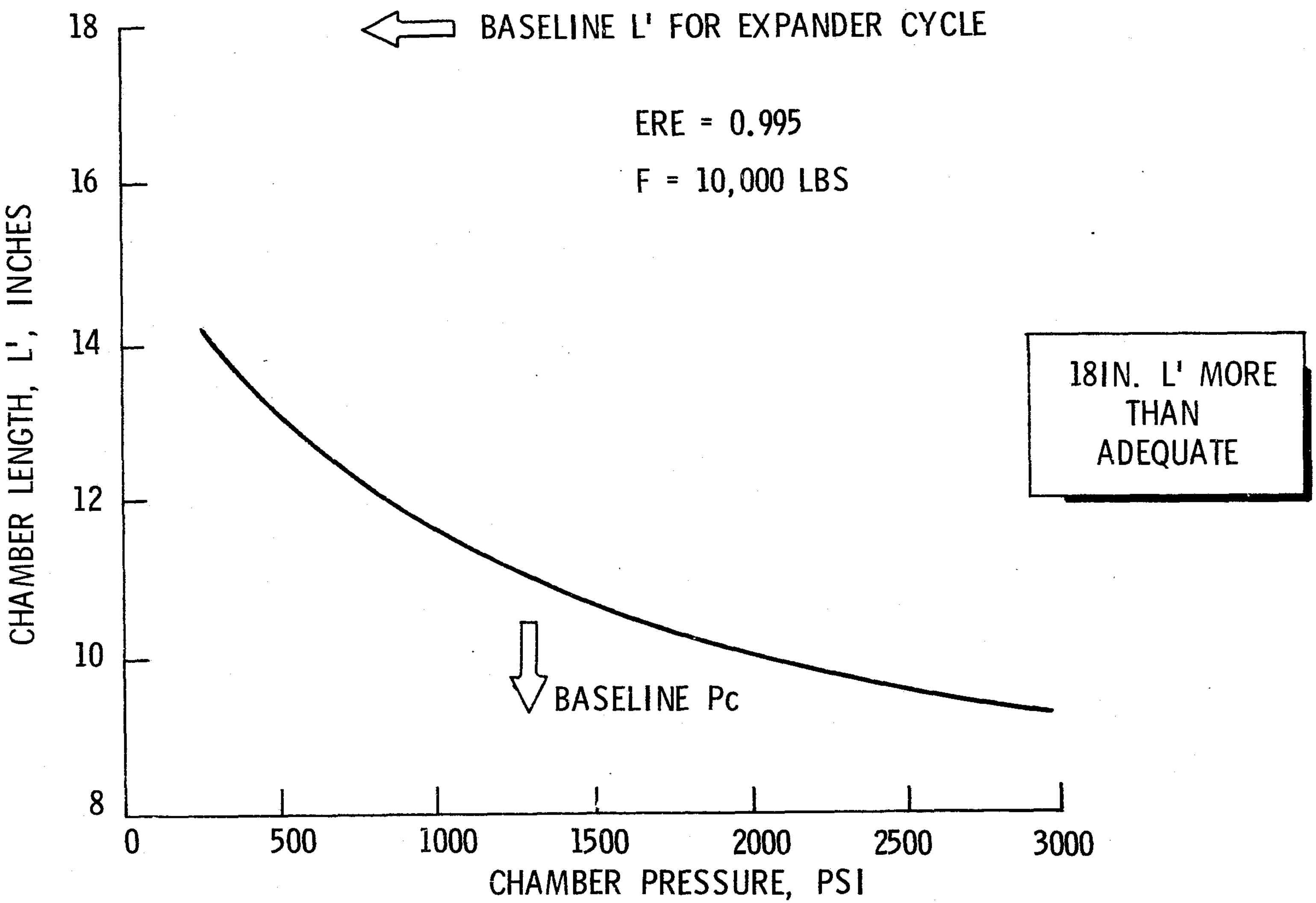


Figure 41. Effect of Chamber Pressure Upon Chamber Length Requirements

C-2

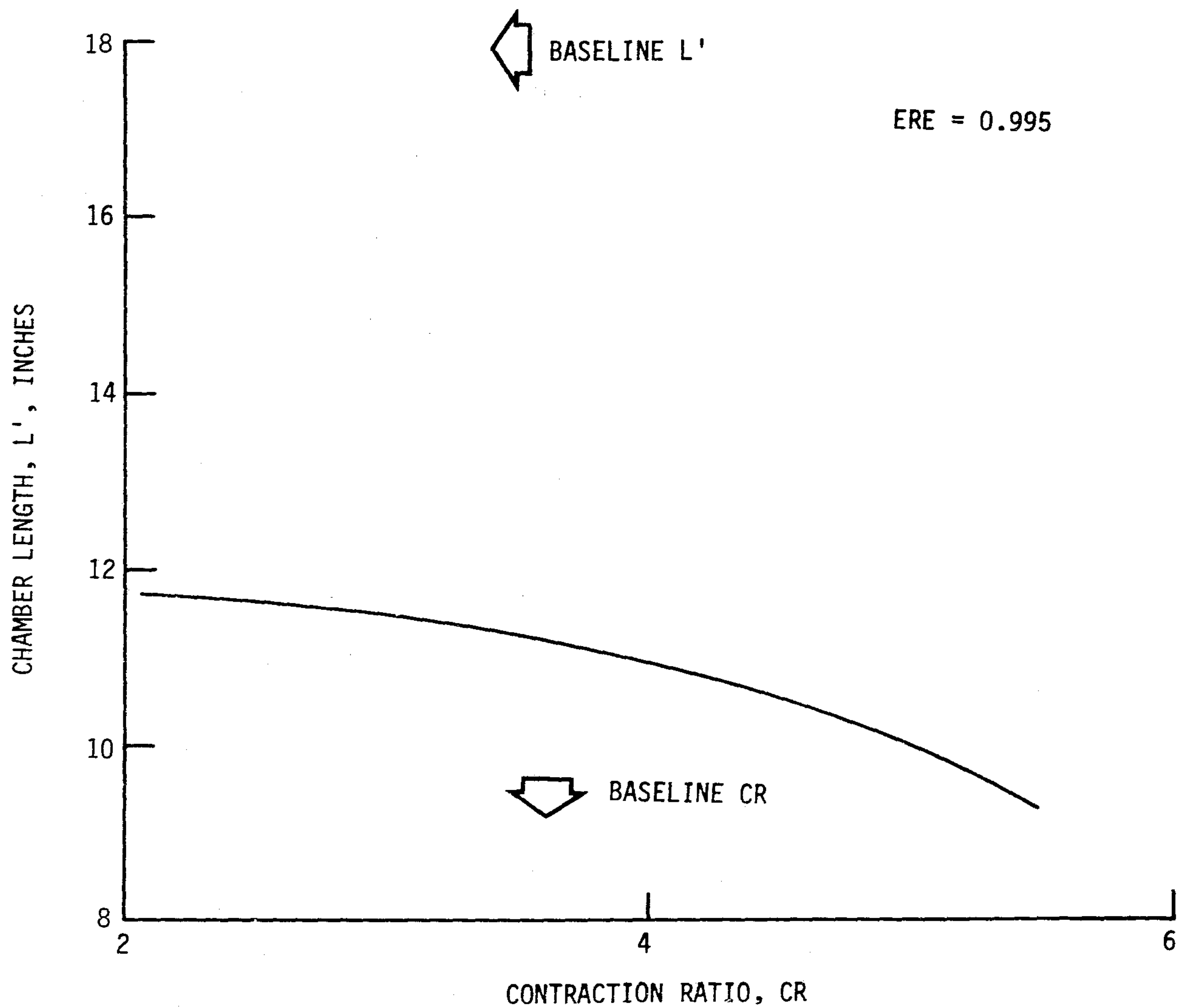


Figure 42. Chamber Length vs Contraction Ratio

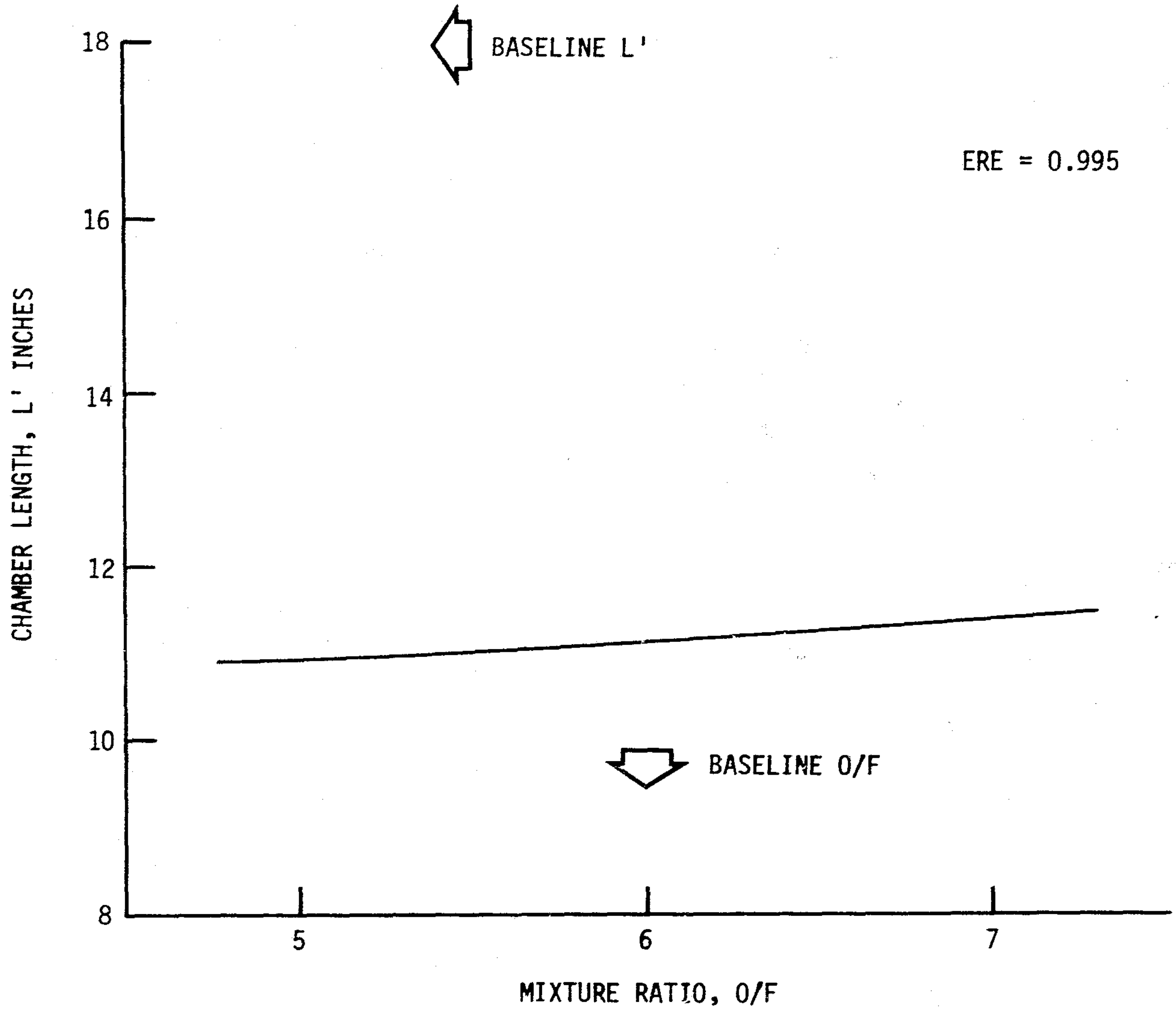


Figure 43. Chamber Length vs Mixture Ratio

B. STRUCTURAL ANALYSIS

Structural analyses were undertaken to determine the design constraints imposed by low cycle thermal fatigue and creep-rupture strength. These analyses were conducted in conjunction with the coolant heat transfer evaluation to establish the chamber temperature, pressure and coolant channel geometry limits created by the chamber service life requirements. For this analysis, the service life between overhauls is 300 cycles times a safety factor of 4 (1200 total cycles) or 10 hours accumulated run time.

The parametric low cycle fatigue analyses was conducted at 10K, 20K and 30K thrust levels at a mixture ratio of 6.0. Off-design mixture ratio operation was evaluated up to an O/F of 10 at the 10K thrust level.

The material used for the combustion chamber (non-tubular portion) is zirconium copper in a mill slotted configuration. The outer shell of the chamber is electroformed nickel with adequate thickness to remain elastic under the pressure loading and copper expansion forces. Although the electroformed nickel jacket thickness was not optimized for this analysis, the predicted thicknesses in the throat and barrel sections are 0.03 to 0.040 and 0.070 to 0.080 inches, respectively.

The low cycle fatigue is dependent upon the total strain range induced on the hot gas-side wall of the regen-cooled thrust chamber. The large number of chamber configurations and thermal loadings in the parametric studies precluded the use of finite element computer analysis at each point. A simplified strain prediction method was developed, based upon a strain concentration factor (K_ε), thermal expansion coefficient (α), and the temperature differential between the gas and backside walls (ΔT).

$$\Delta\varepsilon = K_\varepsilon \ \alpha \ \Delta T$$

A plane strain computer analysis was conducted at the throat, and at one point in the barrel section. Maximum stresses and strains were determined. This strain data was used to calculate K_ε in the above equation and checked against the design curve shown by Figure 44 which was established by computer solutions for many other designs. Lower gas-side wall temperatures exhibit lower K_ε values due to reduced plasticity and relief from outward deflection of the outer chamber shell. Higher gas-side temperatures exhibit higher K_ε values due to less outward deflection of the shell when the copper softens, and from uneven strain distributions when the copper liner moves further into the plastic range and pressure-induced strains become significant. The detailed analyses results fit the historical base well as shown on the figure. Therefore, the design curve was used to predict strain at other points in the chamber and for the parametric studies.

The life cycle analysis is an iterative process between the structural and thermal analysts. With the predicted strain, strain concentration factor and material properties, the maximum temperature differential between the hot gas-side wall and the cooler backside-wall was established. This temperature differential for the slotted zirconium copper chamber is shown on Figure 45 for a range of allowable strains. A strain of 1.86% was predicted for the initial analysis. This allowable temperature differential data was used by the thermal analyst to conduct the chamber and channel design studies.

The thermal analysis established channel designs and wall temperature profiles in the chambers. A typical strain vs cycle life curve for zirconium copper at 900°F is shown on Figure 46. Similar figures were constricted at other temperatures and used to check the design for the cycle life requirement at the predicted wall temperatures. For example, for a 10 hr hold time and a predicted maximum strain of 1.86%, the figure shows that 1200 thermal cycles are predicted. This meets the 300 cycle service life requirement when the safety factor of four is applied.

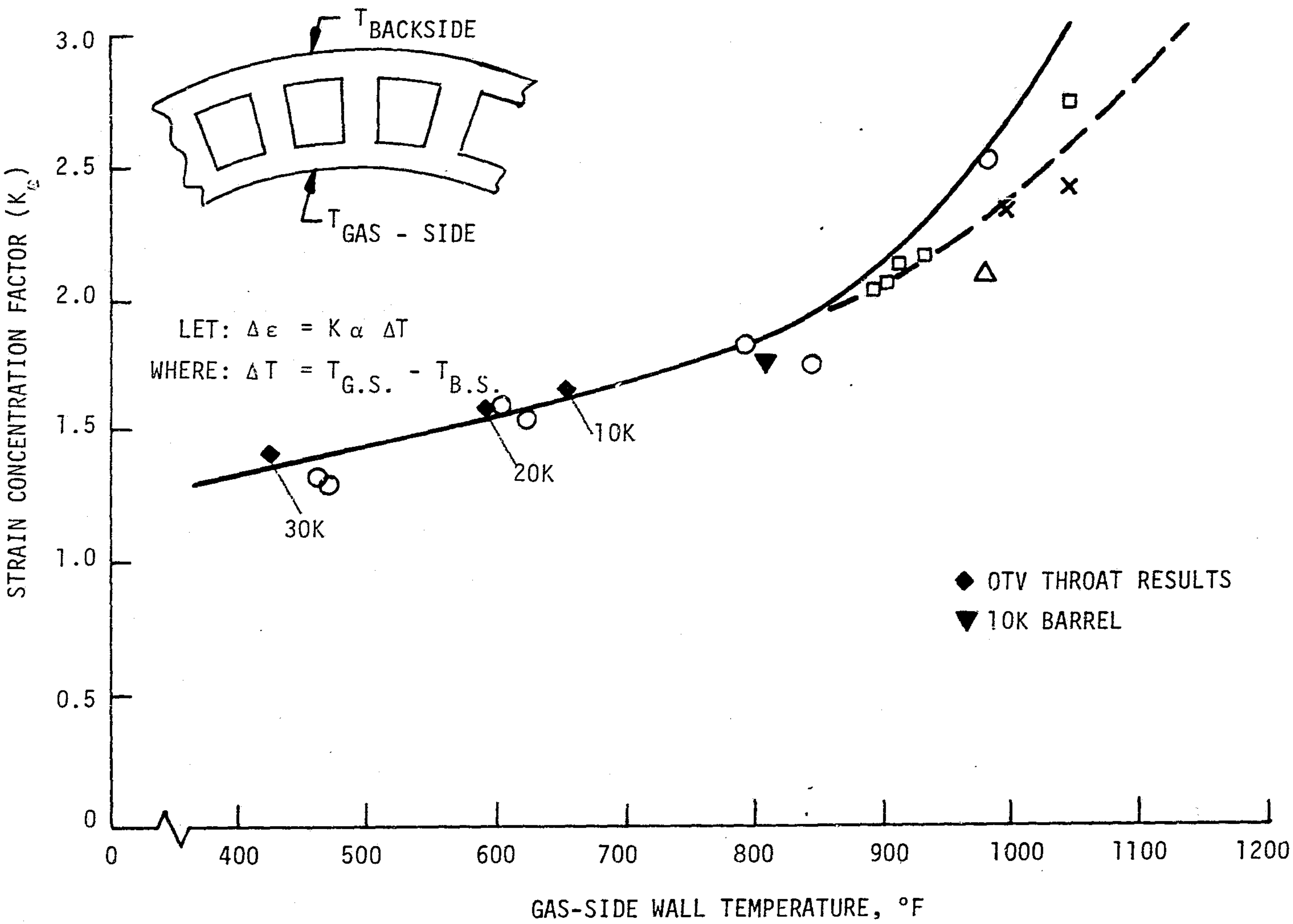


Figure 44. Predicted Strain Concentration Factor vs Gas - Side Wall Temperature

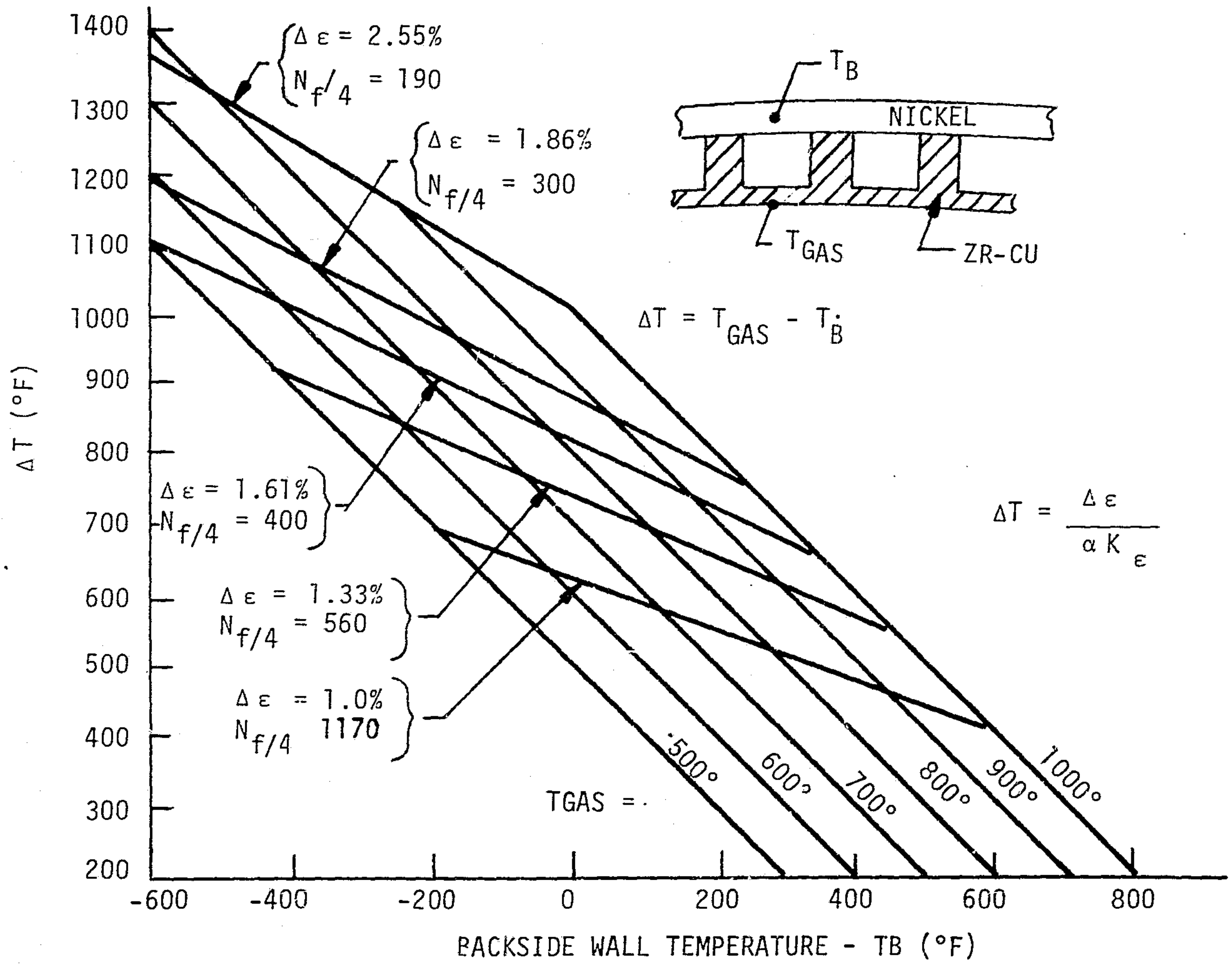


Figure 45. Allowable Temperature Differential

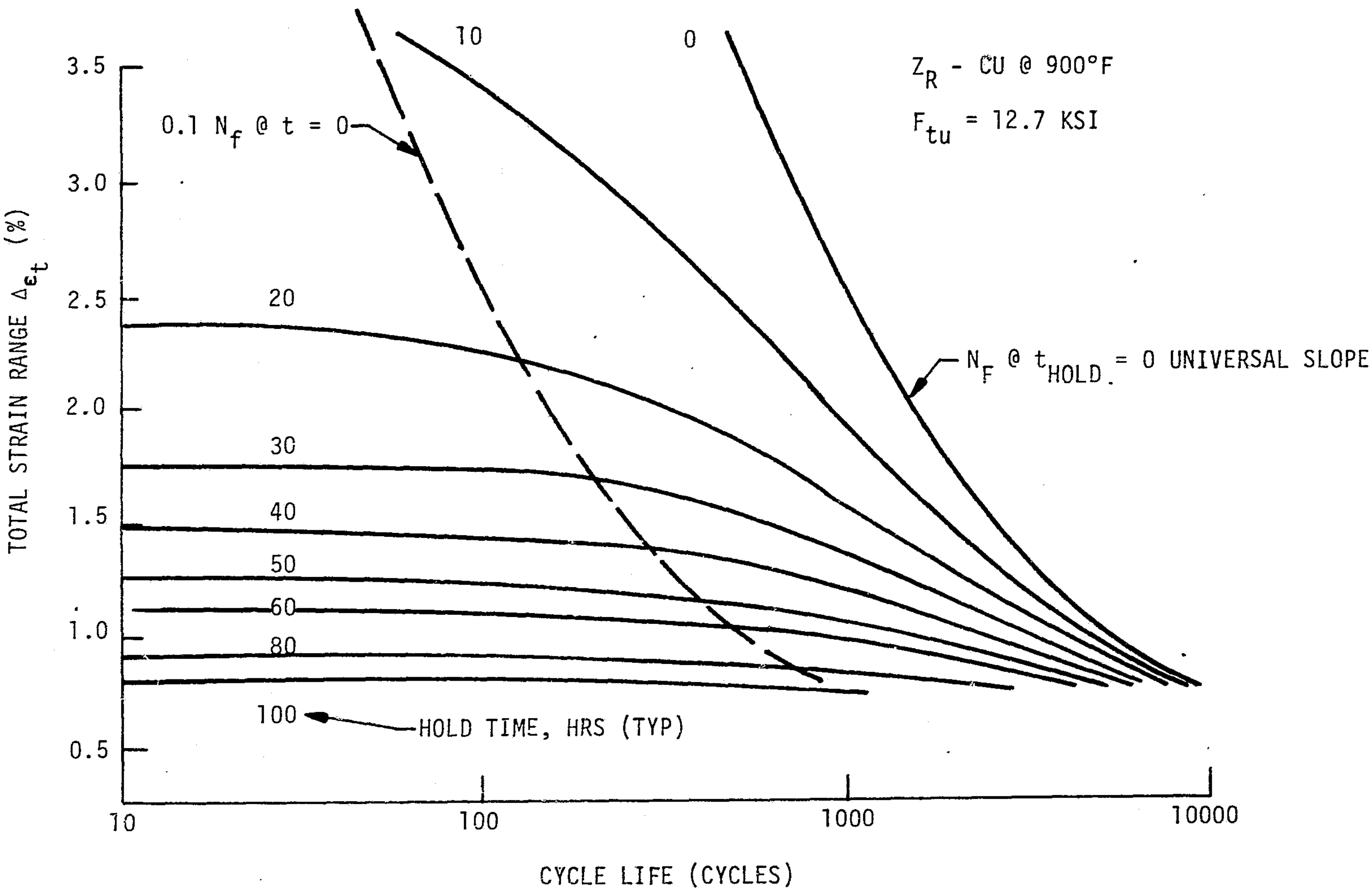


Figure 46. Total Strain Range vs Cycle Life

A copper chamber wall channel width to thickness ratio criteria was also established to conduct the studies. This data is shown as a function of the gas-side wall temperature and pressure differential across the wall on Figure 47.

Land widths were selected to provide maximum channel widths consistent with the wall strength criteria shown. A minimum practical land width was set as 0.040 in. The minimum channel width was also limited to 0.040 in. and a practical channel depth to width limit of 5:1 was imposed for the studies.

All designs established by the thermal analysis meet or exceed the 300 cycle life requirement using a hold time of 10 hours. The off-design study showed significantly lower strain levels in both the throat and barrel regions. Therefore, the off-design operation is less critical structurally than the baseline O/F operation. An emergency one-time operation at an O/F of 10 would not alter the chamber cycle life.

A two-pass A-286 tube bundle was selected for the fixed portion of the nozzle. This tube bundle extends from an area ratio of 8:1 to the point at which a radiation cooled nozzle is attached. The tube wall strength and cycle life criteria developed for use in this study are shown on Figures 48 and 49, respectively. Figure 48 shows the required tube radius to wall thickness ratio as a function of the average hot wall temperature and the coolant pressure. Figure 49 shows the maximum temperature differential that can be taken between the gas-side tube wall and the backside tube wall to meet the service life requirement. The applied strain line without the 1.4 safety factor was used in the Task III thermal analysis.

A radiation cooled nozzle was selected as a baseline. The nozzle material is FS-85 Columbium with an R-512E Silicide coating. This material

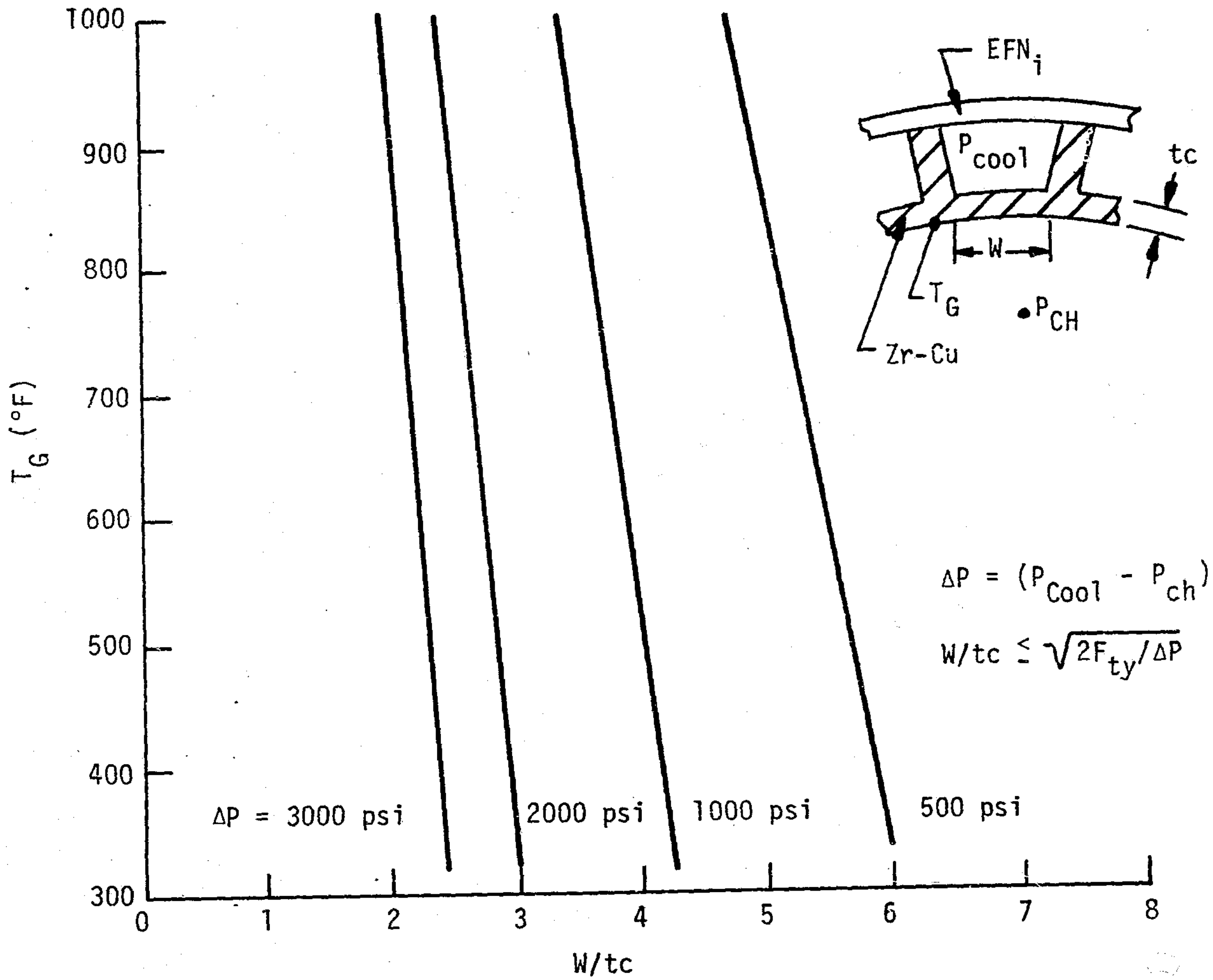


Figure 47. Zr-Zu Chamber Wall Strength Criteria

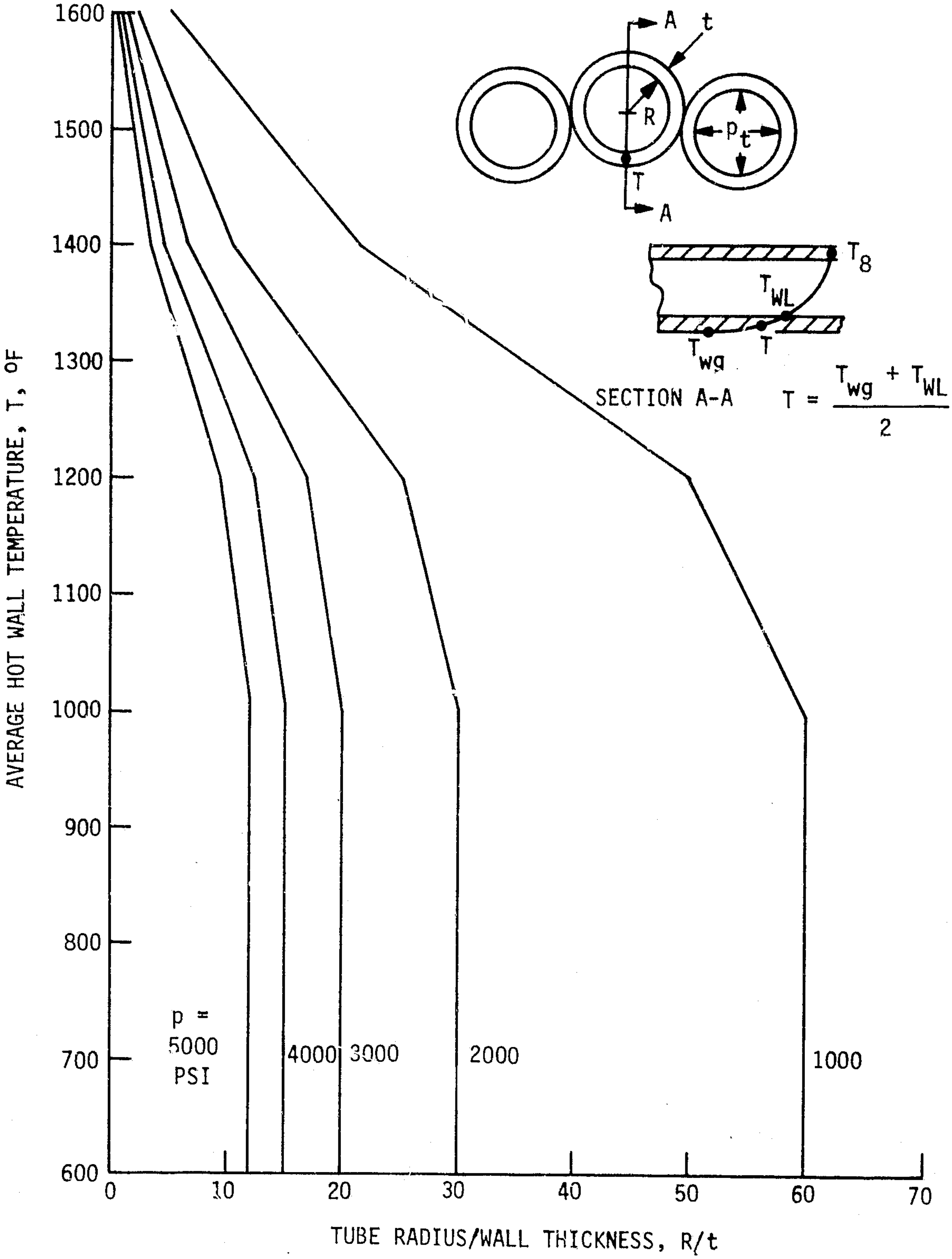


Figure 48. A-286 Tube Wall Strength Criteria

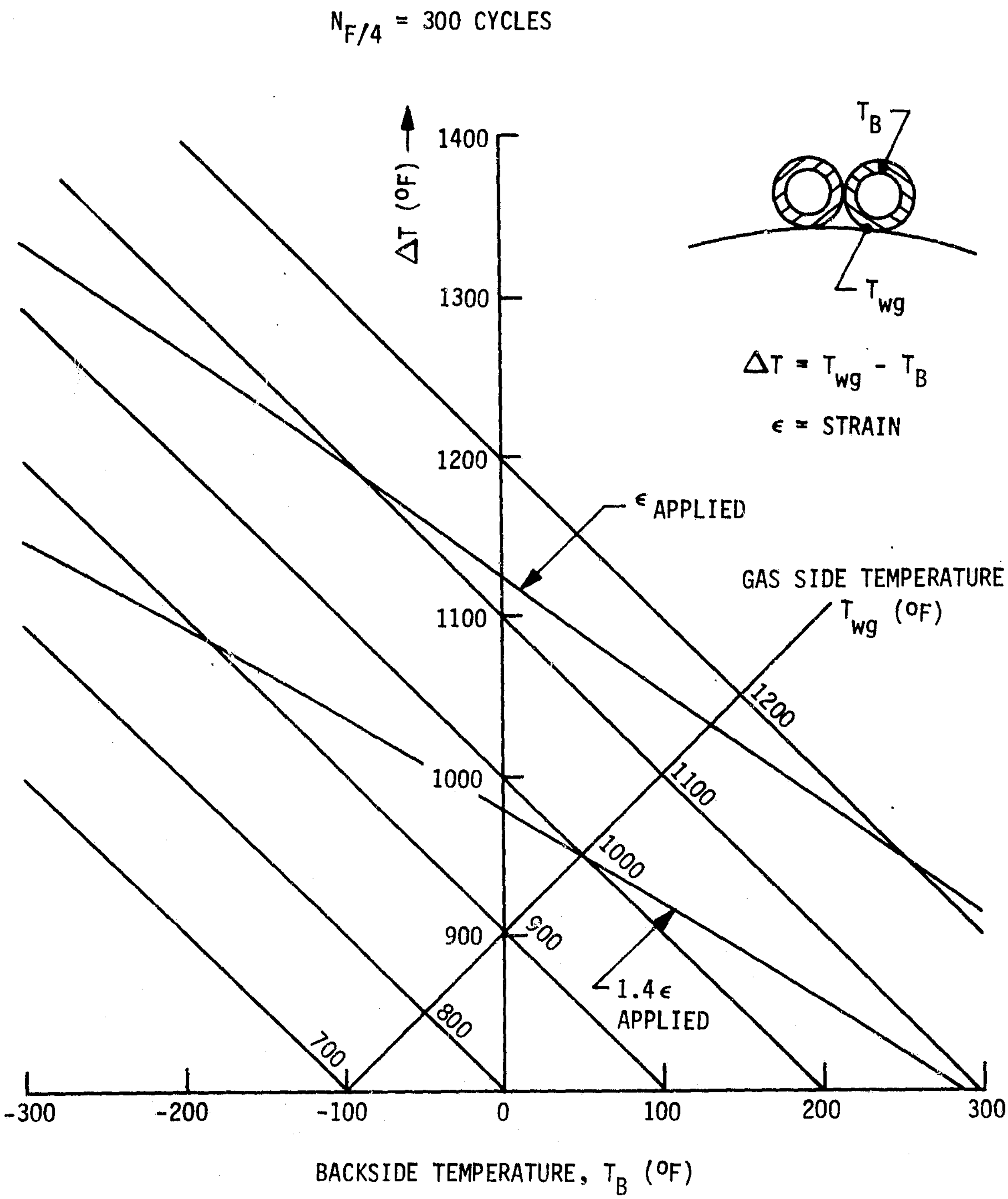


Figure 49. A-286 Tube Wall Temperature Criteria

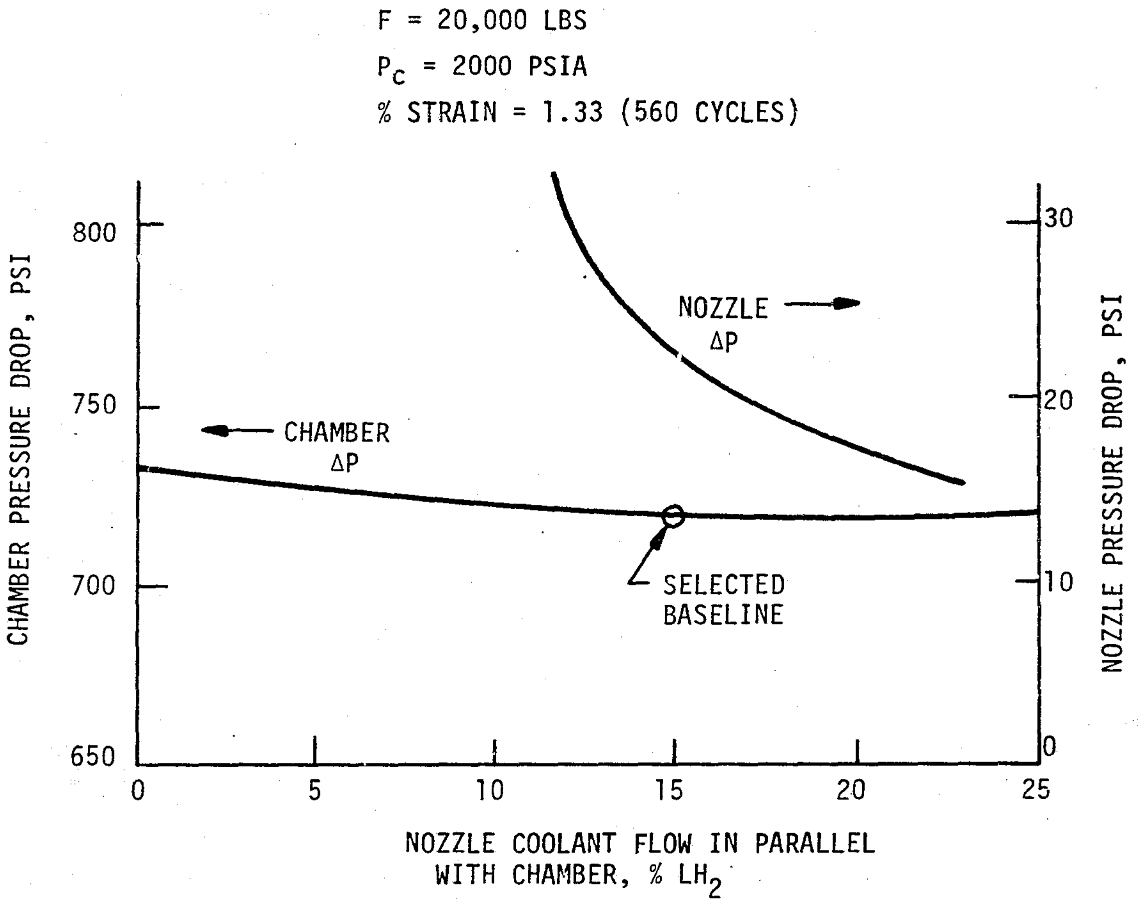
and coating is the same as that being used for the OMS engine nozzle extension. The OMS-E is designed for a cycle life requirement of 1000 cycles times a safety factor of 4 (4000 cycles). Therefore, a radiation cooled nozzle should be able to satisfy the OTV service life requirements.

C. THERMAL ANALYSIS

Parametric thermal analyses were conducted to support the design and power balance analysis required in this study. These parametric analyses considered variations in thrust level, stowed engine length, chamber length, contraction ratio, cycle life requirement, off-design mixture ratio operation, and the flow split between the combustion chamber and the fixed nozzle.

The zirconium copper chamber evaluated in this study is regeneratively cooled in a single pass with the coolant flowing from an area ratio of 8:1 to the injector head end. Flowing the hydrogen coolant from the injector to the aft end results in higher coolant Mach numbers and pressure drops then the counterflow arrangement chosen.

The coolant flow split between the regeneratively cooled nozzle and the combustion chamber was investigated during the concept definition phase. The results of this study were assumed to be valid for the conduct of the remaining study efforts. The study showed that reducing the chamber coolant flow rate results in a small decrease in the required chamber pressure drop. This occurs because the hydrogen heat transfer coefficient increases with increasing bulk temperature. Figure 50 shows that cooling the nozzle with 15% of the flow, that is in parallel with the chamber coolant, results in approximately 14 psi less chamber pressure drop. Based upon these results, cooling the nozzle in parallel with 15% of the total hydrogen flow was selected in order to minimize the system pressure drop.



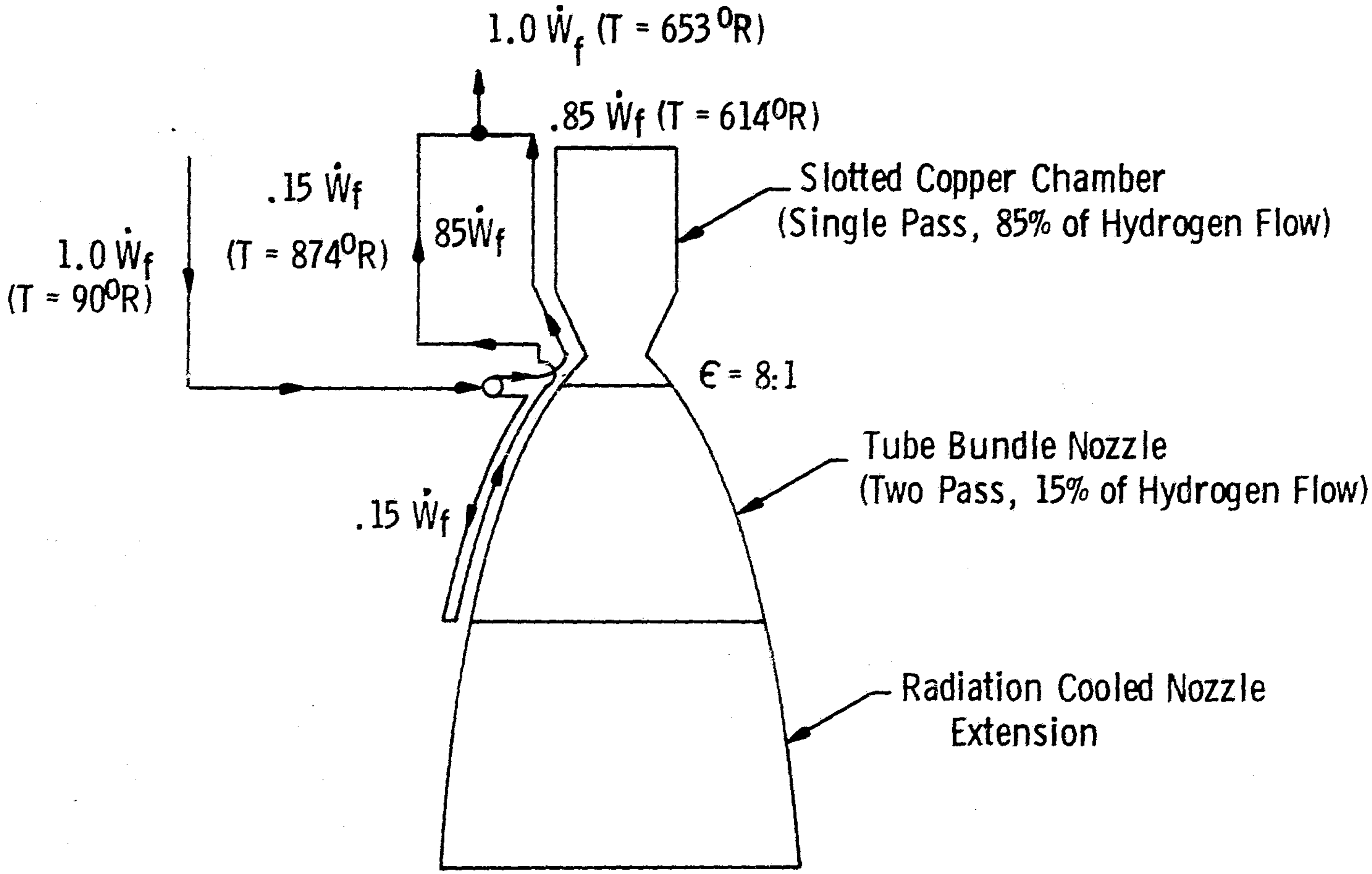
COOL NOZZLE IN PARALLEL WITH CHAMBER

Figure 50. Coolant Flow Split Optimization

The selected coolant flow schematic used in the Task III evaluations is shown on Figure 51. The zirconium copper chamber is cooled with 85% of the total hydrogen flowing from an area ratio of 8:1 to the injector. A two-pass A-286 tube bundle was designed to cool the nozzle with 15% of the total hydrogen flow from an area ratio of 8:1 to the end of the fixed nozzle. The fixed nozzle length is determined to meet the minimum stowed length requirements. The extendible nozzle is radiation cooled and is made of FS-85 columbium with an R-512E silicide coating. Analyses in support of the OMS-E have shown that this material will meet the service life requirements. A dump cooled nozzle extension was also investigated in the concept definition phase but eliminated because of complexity and performance loss considerations. A two pass regeneratively cooled nozzle extension was also eliminated for complexity reasons.

The minimum attachment area ratio for a radiation-cooled nozzle extension was established as a function of thrust chamber pressure and thrust. The results are shown on Figure 52. An attachment point temperature of 2450°F was assumed which is considered to be slightly conservative. The OMS-E nozzle extension was designed to meet a 4000 cycle life (1000 cycles x 4) requirement for the same wall temperature criteria.

Because the desired retraction point to minimize stowed length may not be consistent with the radiation-cooled attachment area ratio criteria, these effects were investigated. Figure 53 shows the area ratio at which the nozzle must be split to stay within a stowed engine length of 50 in. Long chamber lengths result in short fixed nozzle lengths and smaller area ratios at the split point. The dashed lines on the figure represent the minimum area ratios at which a radiation cooled nozzle can be attached. The figure shows that at 20 and 30K lb thrust, an 18 in. long chamber, which was preliminarily selected, results in a nozzle transition



Note: Temperatures are shown for 10KLBF baseline engine.

Figure 51. Advanced Expander Cycle Engine Coolant Flow Schematic

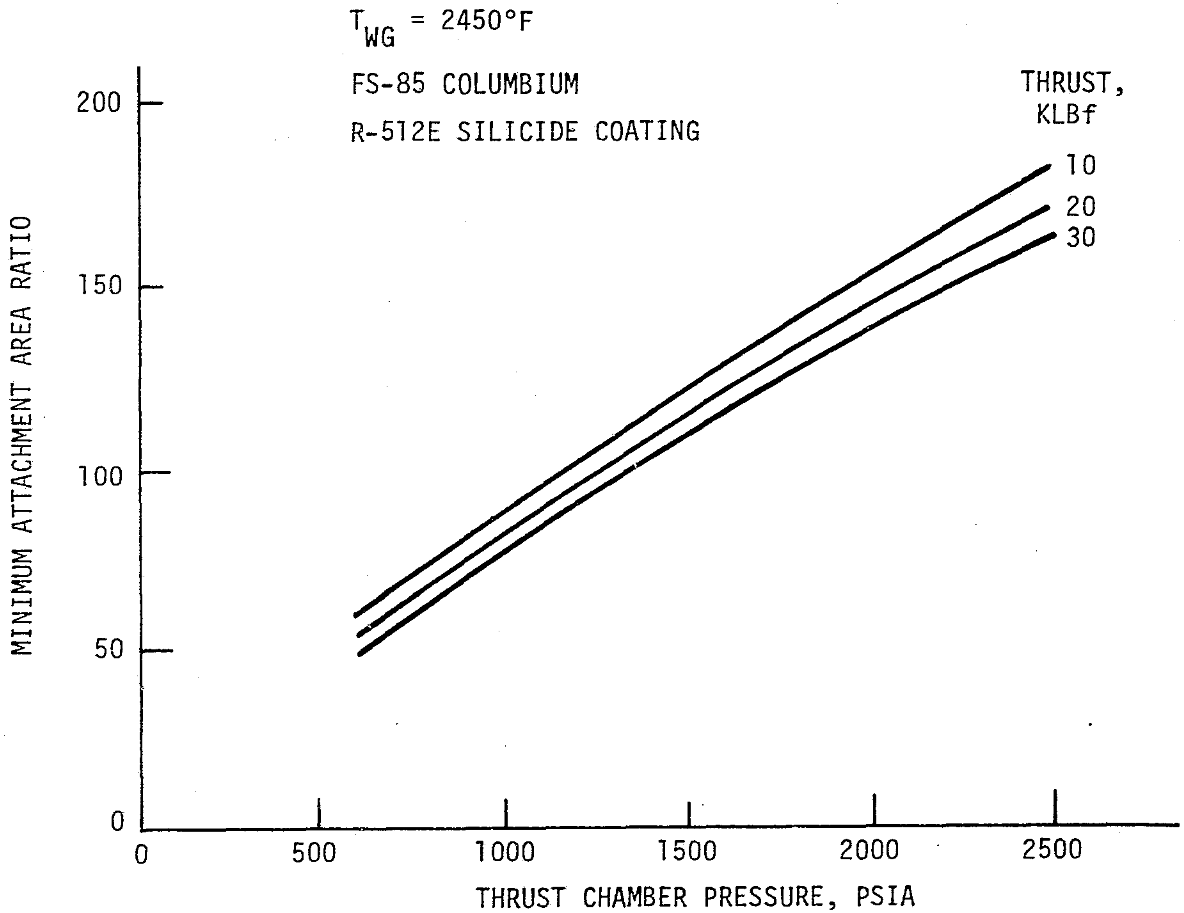
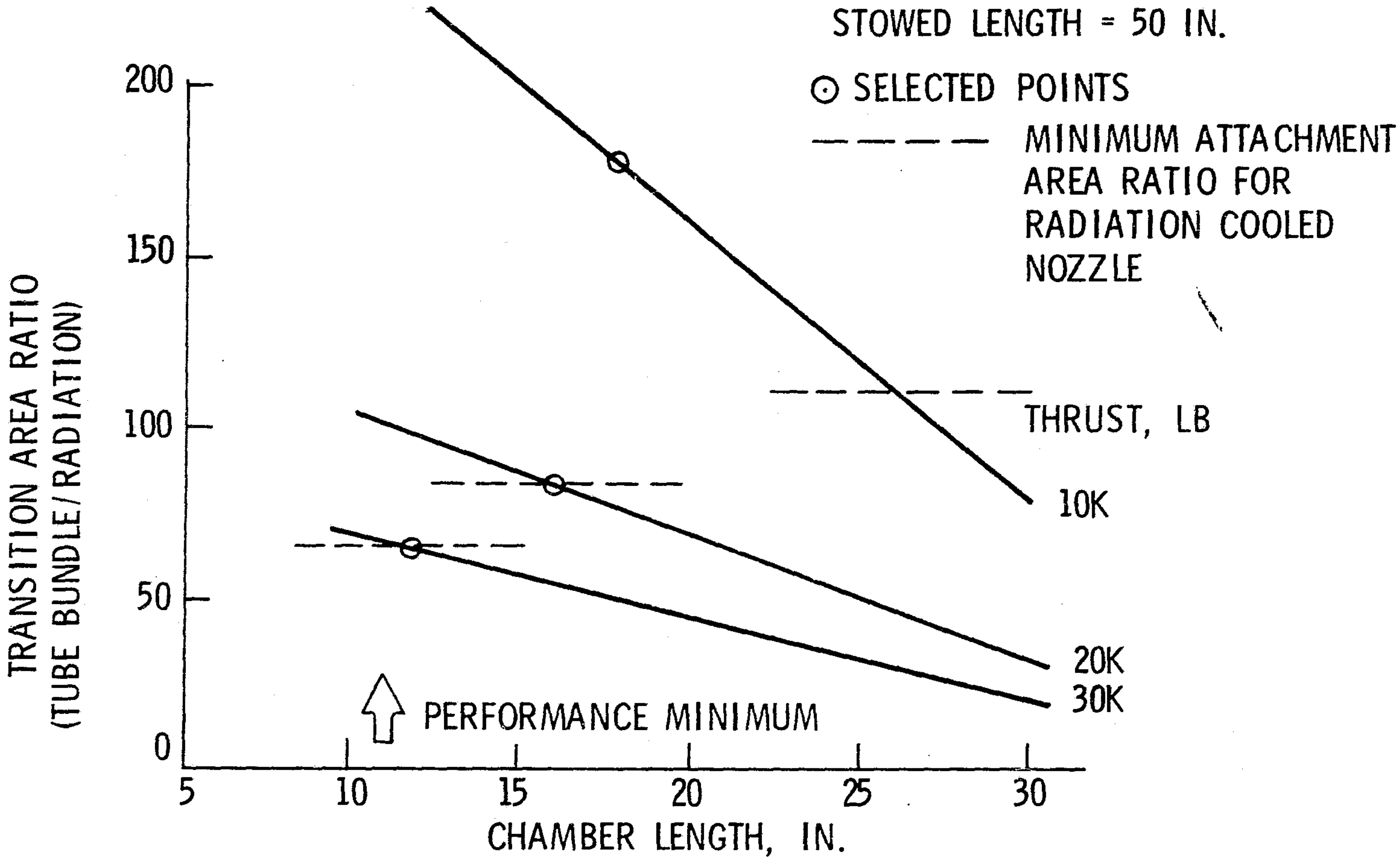


Figure 52. Radiation Cooled Nozzle Attach Area Ratio



50" STOWED LENGTH AFFECTS CHAMBER LENGTH AT HIGH THRUST

Figure 53. Required Nozzle Transition Area Ratio to Minimize Stowed Length

area ratio that is less than that feasible for the radiation extension. Therefore, the chamber length was shortened to accommodate the radiation cooled extendible nozzle criteria rather than assuming a partial tube bundle in the extension. The figure also shows that at 10K lb thrust, the fixed nozzle could be partially radiation cooled when an 18 in. long chamber is used. However, for this parametric study the tube bundle was assumed to extend to the area ratio at which the extendible nozzle is attached (i.e., 177:1 at 10K lbF). For engine stowed lengths of 60 and 70 in., an 18 in. long chamber can be used at all thrust levels without violating the radiation cooled nozzle attachment area ratio criteria.

A chamber L' of 18 inches and a contraction ratio of 3.66 were baselined during the concept definition phase. These values were selected initially from the results of studies conducted in the past by Pratt & Whitley for their RL-10 Derivative studies, Rocketdyne for the Advanced Space Engine and also by ALRC for the Orbit-to-Orbit Shuttle study. Preliminary analyses were undertaken to substantiate these selections at 20,000 lb thrust.

Performance analyses show that a minimum chamber length of about 12 inches is required to meet the Phase "A" ERE goal of 99.5%. However, longer chamber result in higher hydrogen outlet temperatures as shown by Table VIII. This increases the turbine inlet temperature, lowers the turbine pressure ratio and increases the thrust chamber pressure. For an engine with a fixed envelope (length), chamber pressure increases result in higher area ratios and hence, performance (I_S). Conversely, longer chambers reduce the length of the nozzle that can be fit in the fixed length constraint, thereby reducing the area ratio and performance. Longer chambers also result in heavier engine weights. Tradeoffs were performed using the AMOTV payload partials and the results are shown on the top two plots of Figure 54. The I_S and engine data used in the trades is shown on Table IX.

TABLE VIII

INITIAL CHAMBER GEOMETRY SURVEY

F = 20,000 lb

Contraction Ratio	Chamber Length, in.	Chamber Pressure, psia	Chamber Coolant Bulk Temperature Rise, °R	Chamber Coolant Pressure Drop, psi	Total Coolant Temperature, °R
3.66	12	600	211	28	355
3.66	12	1000	240	50	384
3.66	12	1400	264	127	408
3.66	18	600	282	32	406
3.66	18	1000	328	57	452
3.66	18	1400	364	148	488
3.66	24	600	357	35	455
3.66	24	1000	422	65	520
3.66	24	1400	475	176	573
3.66	30	600	435	39	507
3.66	30	1000	520	77	592
3.66	30	1400	588	217	660
2.0	18	600	311	84	431
2.0	18	1000	370	204	488
2.0	18	1400	418	865	534
5.0	18	600	264	29	391
5.0	18	1000	304	46	432
5.0	18	1400	336	103	464

Note: Nozzle coolant jacket in parallel with the chamber.

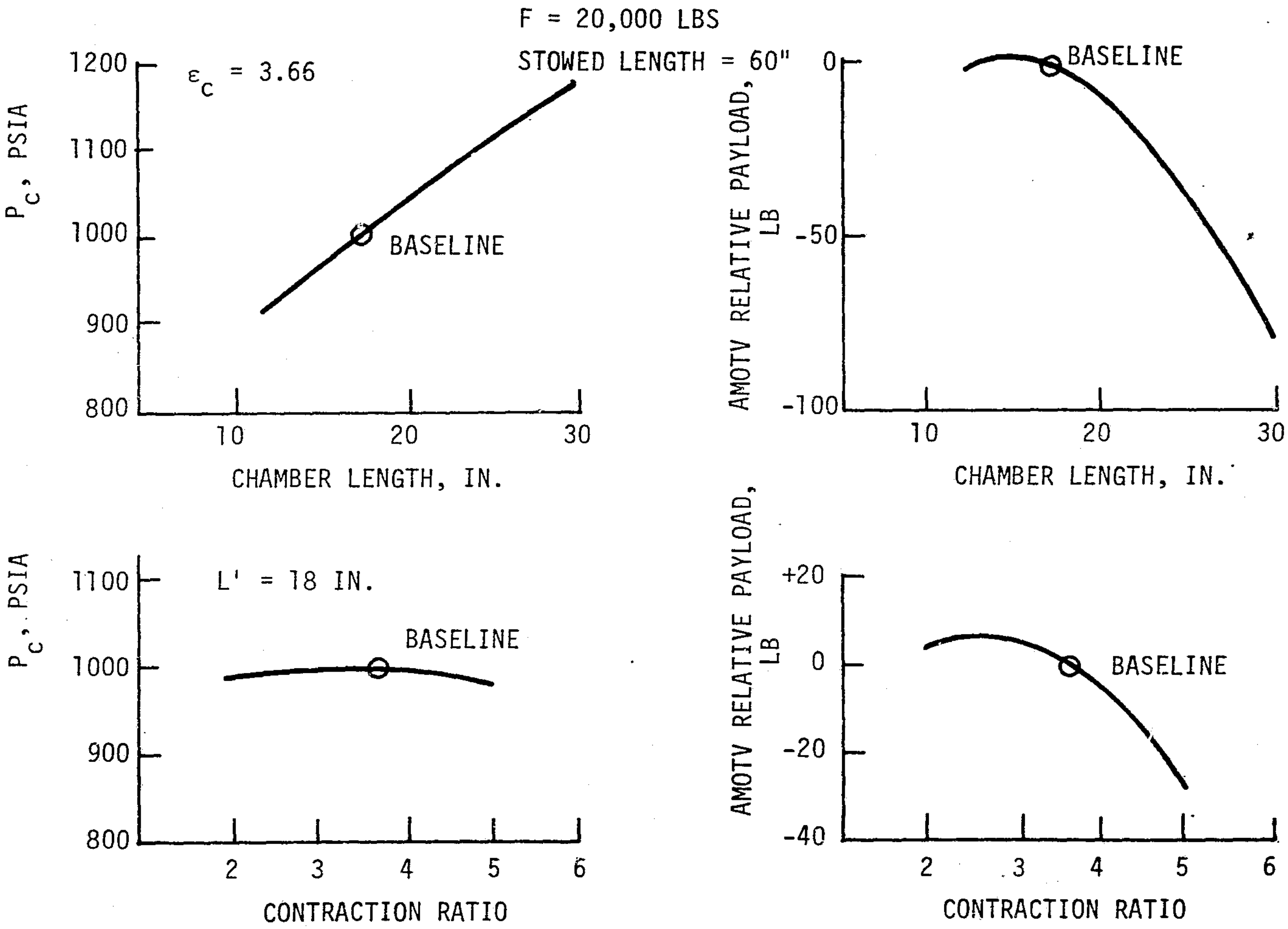


Figure 54. Preliminary Chamber Length and Contraction Ratio Study

TABLE IX

EFFECT OF CHAMBER LENGTH AND CONTRACTION RATIO

F = 20,000 lb

Chamber Length, in.	Contraction Ratio	Chamber Pressure, psia	Nozzle Area Ratio	Engine Vacuum Specific Impulse, sec	Engine Weight, lb	AMOTV Relative Payload, lb
12	3.66	910	377	469.13	562.2	-3
18	3.66	1007	371	469.24	567.1	0
24	3.66	1095	359	468.89	573.6	-33
30	3.66	1176	341	468.37	581.6	-80
18	2.0	991	367	469.08	552.7	+4
18	5.0	980	364	468.96	575.9	-30

Reducing the chamber contraction ratio much beyond 3.0 results in a small chamber pressure decrease. This occurs because the coolant jacket pressure drop goes up significantly and was not made up by the gain in the coolant outlet temperature (Table VIII) when the cycle power balance was evaluated. Increasing the contraction ratio resulted in lower coolant outlet temperatures, but not much lower pressure drops. Chamber contraction ratio increases also result in heavier chambers. The results of these trades are shown on the bottom two plots of Figure 54.

The data show that a chamber L' of 12 to 20 inches and a chamber contraction ratio of 2 to 4 are acceptable at 20,000 lb thrust. (Original Phase A baseline.) Similar analyses are planned over a thrust range of 10K to 20K lbF in the Phase A, Extension I efforts to optimize the expander cycle engine.

Based upon the results of this study, a nominal chamber length of 18 inches (where feasible) and a contraction ratio of 3.66 were baselined.

Thermal analyses were conducted with the selected chamber length and contraction ratio to support the power balance efforts and parametric studies. The results are summarized on Figure 55. The figure shows that the turbine inlet temperature increases with decreasing thrust level and increasing chamber pressure. The chamber coolant jacket pressure drop is also the highest at 10K lb thrust. As discussed previously, a channel depth to width ratio of 5:1 was imposed in the study. This resulted in overcooling the 30K thrust level which is reflected by slightly increased (but not very significant) pressure drops relative to the 20K designs. This problem might be alleviated by the use of wider channels in the throat region or by reducing the chamber coolant flow. If a 30K single engine thrust level is of interest, further study of the channel geometry and coolant flow split would be required.

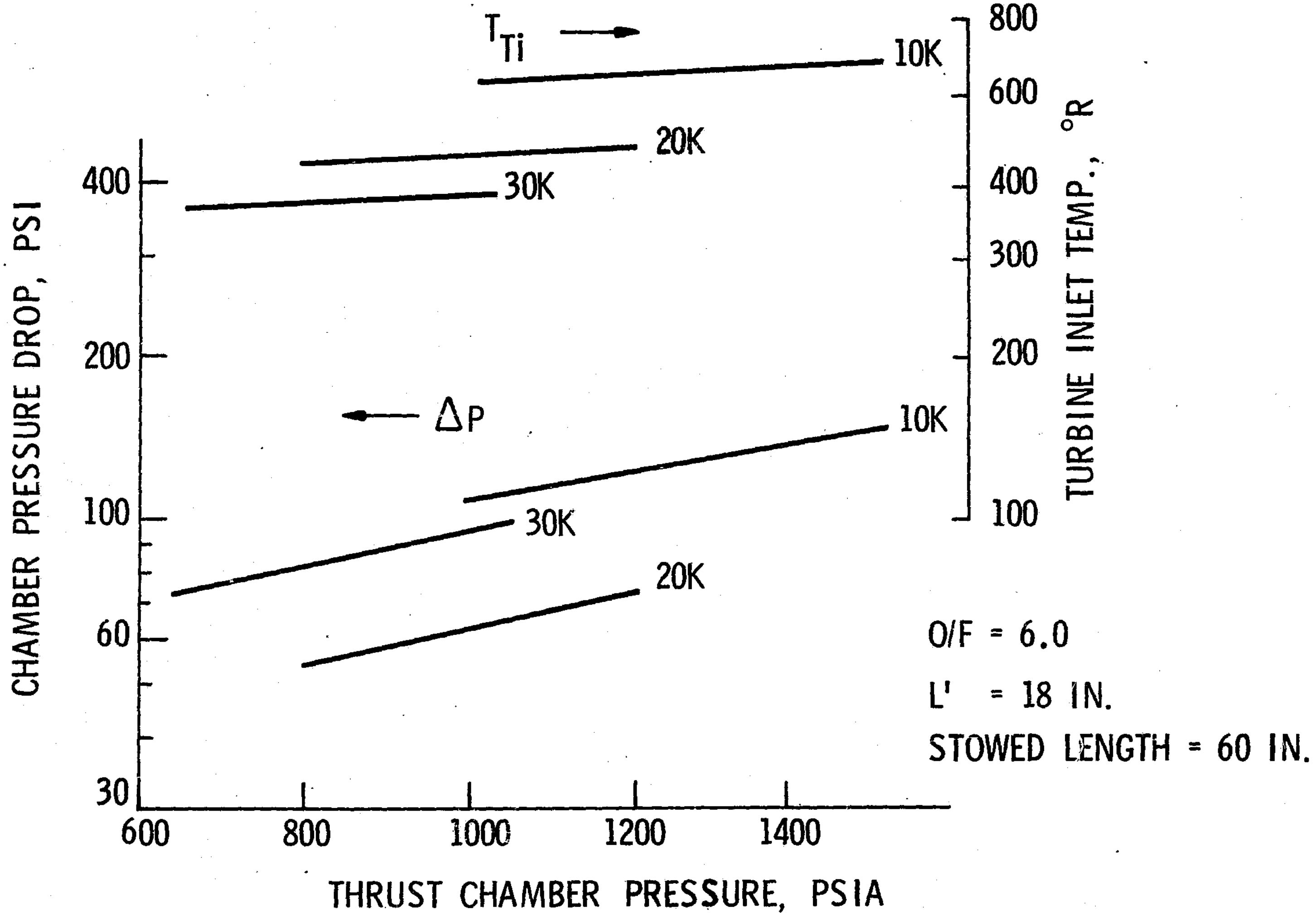


Figure 55. Expander Cycle Parametric Thermal Analysis

The chambers evaluated in this study are one-pass channel designs cooled with 85% of the total hydrogen flow. Cycle life, creep and wall strength criteria are as defined in Section V,B. Figure 56 shows the channel layout used. Throat channel and land widths were selected as 0.040 in. This is the minimum dimension considered to be practical for fabrication, thereby establishing the number of channels. Cylindrical section channel widths were varied up to the maximum allowed by the wall strength criteria in order to find the width which minimizes the pressure drop. The results obtained were as follows:

Thrust, lbF	No. of Channels	Maximum Barrel Channel Width, in.	Optimum Barrel Channel Width, in.
10K	89	0.067	0.056-0.061
20K	142	0.077	0.077
30K	188	0.082	0.082

Barrel land widths were continued into the first part of the convergent section, i.e., assuming straddle-mill machining. A constant channel width of 0.040 in. extended from the intersection with the straddle mill region in the convergent section through the throat to area ratio 1.58, where the land width reaches 0.060 in. This land width was retained for the continuous lands throughout the remainder of the nozzle section. As the channel width increases in the nozzle, it becomes necessary to increase the wall thickness or split the channel in order to satisfy the wall strength criteria. The selected design does both. A nominal wall thickness of 0.030 in. was used from the injector to area ratio 2.64, with the wall thickness increasing to 0.045 in. at area ratio 3.77. Rather than increase the wall thickness further, the channel was split at this point. The double channel region has two sections axially: a section of constant channel width of 0.047 in. extending to area ratio 6.04, followed by a section of constant land width

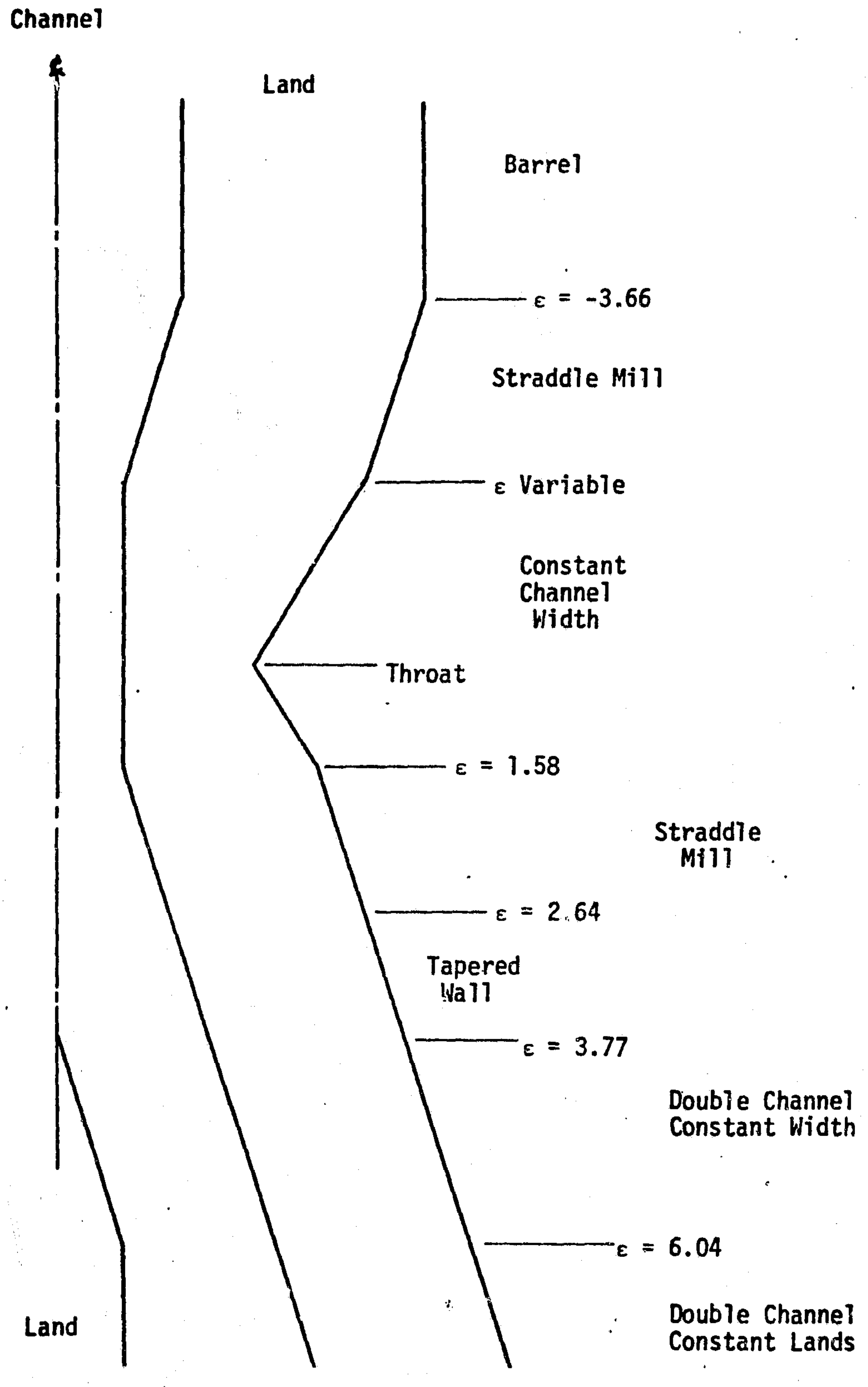


Figure 56. Channel Layout

of 0.040 in. Channel widths increase to 0.061 in. at the end of the latter section. A wall thickness tolerance of $\pm$ 0.005 in. was assumed, with the minimum thickness used in the strength criteria evaluation and the maximum used for wall temperature calculation.

The two-pass A-286 tube bundles are cooled with 15% of the hydrogen flow. Wall strength and cycle life criteria are as defined in Section V,B. The coolant inlet is limiting from a cycle life standpoint. The wall temperature at this point is limited to 1015°F. All designs were based on round tubes with a linearly tapered wall thickness which varies from 0.007 in. at the forward end to 0.010 in. at the aft end. For each design point, the number of tubes was varied in order to meet the wall temperature limit established by the cycle life criteria.

The thermal analysis data results are summarized for the ultimately selected Task III operating chamber pressures on Table X. Stowed engine lengths of 50, 60 and 70 inches were evaluated to support the parametric data evaluations. The stowed length affects the nozzle coolant heat pickup and in some cases, the chamber heat pickup because shorter chamber lengths must be used, as previously discussed in this section. These shorter chamber lengths result in much lower turbine inlet temperatures and affect the power balance and parametric data at these points.

Table XI shows the tube bundle design data with the number of tubes and the corresponding tube bundle pressure drops. Since the nozzle pressure drops are small relative to the chamber values, the nozzle circuit is orificed to provide the desired flow split.

The expander cycle turbine inlet temperatures resulting from the thermal analysis and used in the Task III studies are plotted as a function of thrust on Figure 57. Low thrust levels result in significantly higher turbine inlet temperatures.

TABLE X

ADVANCED EXPANDER CYCLE ENGINE THERMAL ANALYSIS DATA SUMMARY
Nominal O/F = 6.0

Thrust,	Chamber Pressure psia	Stowed Length, in.	Chamber Length, in.	Chamber ΔP, psia	Chamber ΔT, deg.	Nozzle ΔT, deg.	Coolant Outlet Temp., °R
10,000	1300	50	18	131	524	661	635
	↓	60	18	↓	↓	784	653
	↓	70	18	↓	↓	896	670
20,000	1100	50	16	73	310	460	422 *
	↓	60	18	76	340	563	463
	↓	70	18	76	340	659	477
30,000	950	50	12	87	205	378	321 *
	↓	60	18	91	269	423	382
	↓	70	18	91	269	525	397

NOTES: COOLANT INLET TEMP. = 90°R
CHAMBER COOLANT = 85% OF TOTAL H_2 FLOW
NOZZLE COOLANT = 15% OF TOTAL H_2 FLOW

*L' is constrained by minimum radiation cooled nozzle attach area ratio criteria.

TABLE XI

NOZZLE TUBE BUNDLE DESIGNS

Thrust 10^3 lbF	Stowed Length, in.	Cooled Area Ratio	No. of Tubes	Tube ΔP, psi
10	50	177	310	19
10	60	297	298	12
10	70	411	298	14
20	50	72	372	9
20	60	111	362	9
20	70	173	344	8
30	50	48	402	7
30	60	64	388	6
30	70	98	376	7

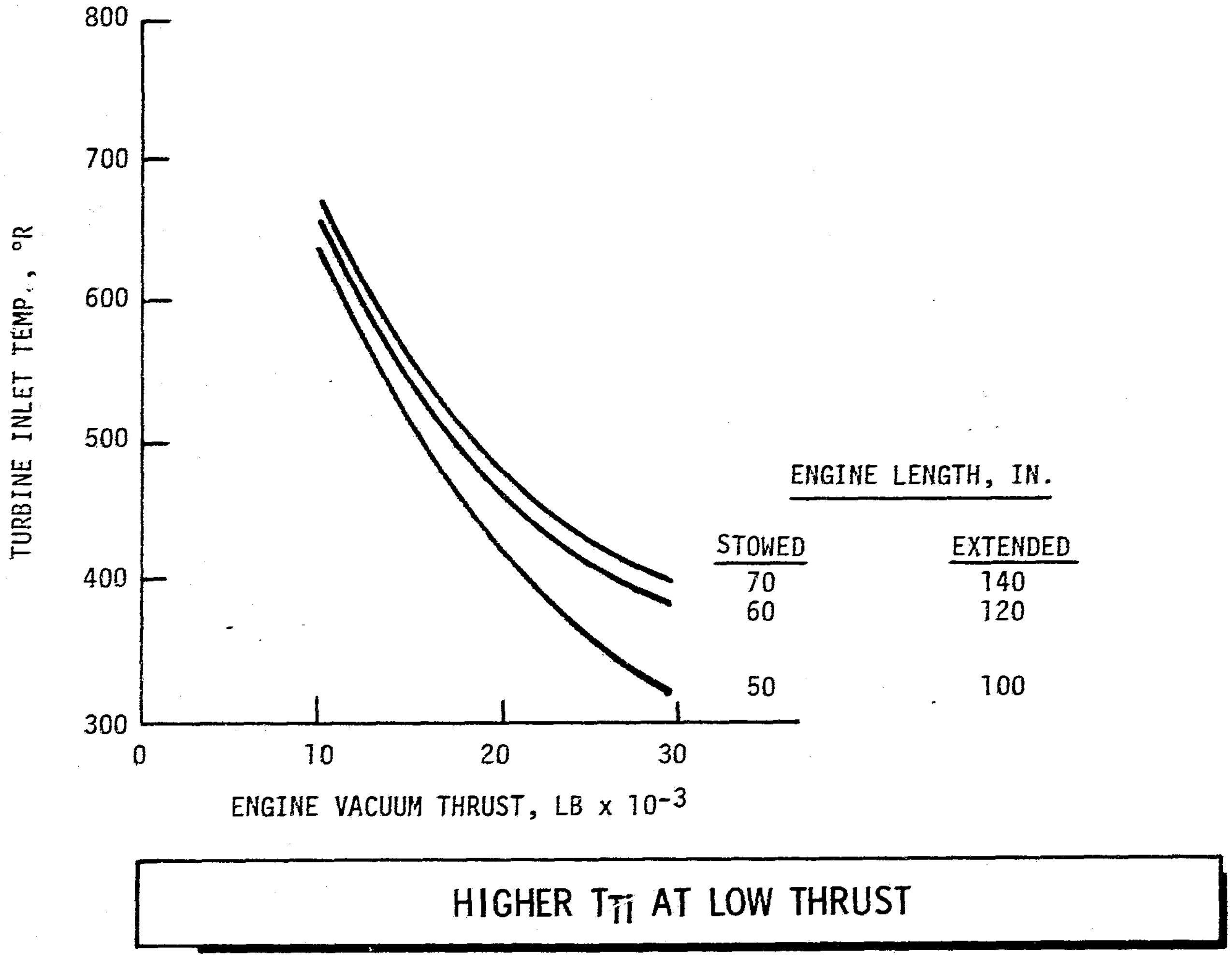


Figure 57. Advanced Expander Cycle Engine Turbine Inlet Temperature

The engine is required to operate over a mixture ratio range of 6.0 to 7.0. The thermal analysis results for these operating conditions are summarized on Table XII. The thermal and low cycle fatigue analysis showed that the engine could be designed at a nominal O/F of 6.0 and operate without cycle life degradation at a mixture ratio of 7.0. A mixture ratio of 7.0 requires less pressure drop than 6.0 even though the coolant flow is reduced and the stagnation temperature is increased. This result occurs because of a reduction in the gas-side heat transfer coefficient.

The engine is also required to operate in a one-time emergency mode at a mixture ratio of 10. The thermal analysis data for the baseline thrust level is shown on Table XIII. Low cycle thermal fatigue analysis also showed that this off-design O/F operation has a negligible affect upon the chamber service life.

The effect of the required cycle life upon the baseline chamber pressure drop was evaluated. The results are shown on Figure 58. As shown by the figure, significant increases in chamber coolant pressure drop occurs as the cycle life requirement is increased. An increased cycle life requirement would result in reduced operating chamber pressure and attendant performance losses. The data is summarized below:

Number of Thermal Cycles ($N_F/4$)	% Strain	Chamber ΔP, psi	Chamber Pressure, psia
300	1.86	131	1300
400	1.61	192	1265
560	1.33	333	1210

TABLE XII

DESIGN AND OFF-DESIGN OPERATION THERMAL ANALYSIS SUMMARY
(Expander Cycle)

Thrust, KlbF	Mixture Ratio	Stowed Length, inches	Chamber Pressure, psia	Chamber Pressure Drop, psia	Turbine Inlet Temperature, °R
10	6.0	50	1300	131	635
10	6.0	60	1300	131	653
10	6.0	70	1300	131	670
10	6.5	50	1278	120	650
10	6.5	60	1278	120	668
10	6.5	70	1278	120	685
10	7.0	50	1261	108	662
10	7.0	60	1261	108	680
10	7.0	70	1261	108	697
20	6.0	50	1100	73	422
20	6.0	60	1100	76	463
20	6.0	70	1100	76	477
20	6.5	50	1083	67	433
20	6.5	60	1083	70	474
20	6.5	70	1083	70	488
20	7.0	50	1070	60	441
20	7.0	60	1070	63	482
20	7.0	70	1070	63	496
30	6.0	50	950	87	321
30	6.0	60	950	91	282
30	6.0	70	950	91	397
30	6.5	50	937	80	330
30	6.5	60	937	83	391
30	6.5	70	937	83	406
30	7.0	50	927	72	337
30	7.0	60	927	75	398
30	7.0	70	927	75	413

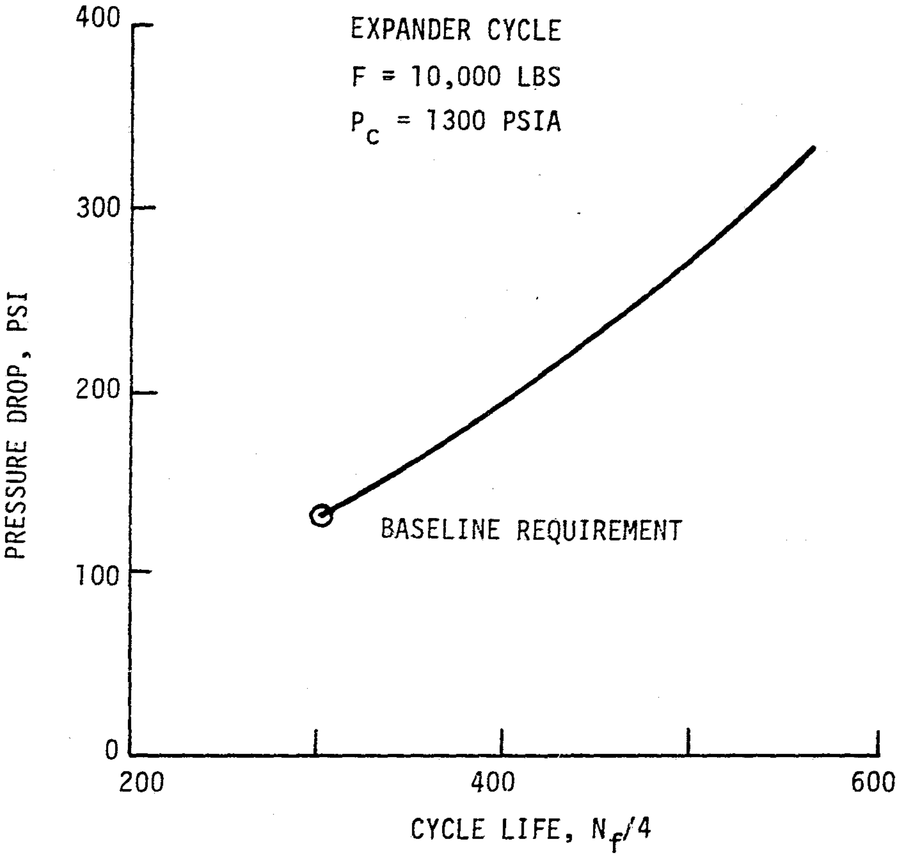
TABLE XIII

EMERGENCY ONE-TIME OPERATION THERMAL ANALYSIS DATA SUMMARY

O/F = 10.0

Thrust = 10,000 lb

Stowed Length, inches	Chamber Pressure, psia	Chamber Pressure Drop, psia	Turbine Inlet Temperature, °R
50	1260	90	838
60	1260	90	856
70	1260	90	873



PRESSURE DROP INCREASES ARE SIGNIFICANT

Figure 58. Effect of Required Cycle Life on Baseline Chamber Pressure Drop

D. TURBOMACHINERY ANALYSIS

The primary objective of the turbomachinery analysis was to determine the efficiencies of the oxygen and hydrogen pumps and turbines as a function of thrust level at both design and off-design O/F operation. This data was required for use in the Task III power balance analysis. The efficiencies were established through analysis, literature reviews and comparisons of design predictions to the efficiencies of existing turbopumps. The primary data sources used were the previously cited References 1 through 5, NASA SP-8109 (Reference 14) and the ASE turbopump documentation (References 15 and 16).

The following limiting parameters and guidelines were used in conducting this analysis:

- Suction Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$

Fuel Boost	40,000
Oxidizer Boost	30,000
Main	20,000

- Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$

Minimum	600

- Inducer Inlet Velocity

Fuel Boost	$\sqrt{\frac{2g\ NPSH}{1.3}}$
Oxidizer Boost	$\sqrt{\frac{2g\ NPSH}{2.3}}$

- Rolling Contact Bearings

Fuel	2×10^6 DN maximum
Oxidizer	1.5×10^6 DN maximum
Bearing Bore	20 MM minimum
Life B_{10}	100 hours

The main hydrogen pump operating specifications are shown on Table XIV as a function of thrust level. Three stage hydrogen pumps were selected at the 10K and 20K lbF thrust levels in order to achieve reasonable specific speeds (N_s) and efficiencies. Studies have shown that pump efficiencies drop off rapidly at N_s values less than 600. This is shown by the two versus three stage efficiency predictions at the 20K lbF thrust level. High pump efficiencies are obtained with two stages at the 30K lbF thrust and were selected for the study.

Specific Speed (N_s)/impeller tip diameter trades were considered in selecting the operating points shown on Table XIV. At the 10K thrust level, the hydrogen turbopump is bearing DN limited. The high speed is required to maintain a reasonable N_s value even though this results in a small impeller diameter which is detrimental to efficiency. At the 20 and 30K thrust levels, it is possible to slow the pumps down slightly from the bearing DN limit. This reduces the N_s somewhat but increases the impeller size. The pump discharge pressure decreases with increasing thrust because the expander cycle engine chamber pressure decreases with increasing thrust. The main pump inlet pressures shown on the table are provided by boost pumps which were assumed to be hydraulically driven. The main hydrogen pump horsepower is increased by 3% to provide the boost pump drive flow.

TABLE XIV

HYDROGEN PUMP OPERATING CHARACTERISTICS SUMMARY
(Expander Cycle)

Engine Thrust Level, lb	10,000	20,000	20,000	30,000
Inlet Pressure, psia	51.0	51.0	51.0	51.0
NPSH, ft	1,080	1,080	1,080	1,080
Volumetric Flow, gpm	307.1	620.2	620.2	939.5
Suction Specific Speed, (rpm) $(gpm)^{1/2}/(ft)^{3/4}$	9,300	11,900	11,900	13,015
Speed, rpm	100,000	90,000	90,000	80,000
Discharge Pressure, psia	3,130	2,605	2,605	2,230
Number of Stages	3	3	2	2
Specific Speed, (rpm) $(gpm)^{1/2}/(ft)^{3/4}$	706	1,039	767	945
Impeller Tip Diameter, in.	3.33	3.55	5.93	4.47
Efficiency, %	64.7	67.5	63.4	68.2

The main oxygen pump operating specifications are shown on Table XV as a function of thrust level. A single stage oxidizer pump can be used in all cases. The main oxidizer pumps operate at the maximum suction specific speed limit of 20,000. The rotating speed of the pumps was chosen considering the boost pump pressure rise requirements and the main pump impeller sizes. Operating the oxygen pumps at a speed of 75,000 RPM would result in high boost pump head rise requirements and extremely small main pump imepllers. The oxygen boost pumps were again assumed to be hydraulically driven. The main oxygen pump horsepower is increased by 5% to provide the boost pump drive flow.

A parallel flow turbine drive concept was selected for the main pumps of the baseline engine. This flow path was selected because there is less interaction between components and may offer development advantages. The expander cycle engine turbine operating specifications are shown on Tables XVI and XVII. The hydrogen TPA turbines are single stage, partial admission turbines. The oxygen pumps use single stage Terry turbines. This is a low flow, low efficiency machine with a high spouting velocity, as noted by the low U/C. The LOX TPA horsepower is low and therefore, this low turbine efficiency does not have as big an impact on the power balance as the hydrogen TPA efficiencies have. The LOX turbine efficiency, may be improved by a series, rather than the parallel flow turbine arrangement. However, the oxidizer TPA turbine inlet temperature is reduced in the series turbine case. The merits of parallel vs series turbines will be investigated further in a Phase A extension to this contract.

The main TPA efficiency data established to support the Task III expander cycle power balance analyses is shown on Figure 59 as a function of thrust level. The efficiencies improve as the turbopumps get larger. The fuel pump efficiency levels off slightly because of the change from three stage to two stage pumps between 20K and 30K lb thrust. The turbopump performance at off-design O/F operation was also evaluated and the results are shown on Table XVIII.

TABLE XV

OXYGEN PUMP OPERATING CHARACTERISTICS SUMMARY
(Expander Cycle)

Engine Thrust Level, lb	10,000	20,000	30,000
Inlet Pressure, psia	46.6	46.6	46.6
NPSH, ft	64.1	64.1	64.1
Volumetric Flow, gpm	114.0	230.5	349.2
Suction Specific Speed, (rpm) $(gpm)^{1/2}/(ft)^{3/4}$	20,000	20,000	20,000
Speed, rpm	42,440	29,840	24,245
Discharge Pressure, psia	1,585	1,350	1,170
Number of Stages	1	1	1
Specific Speed, (rpm) $(gpm)^{1/2}/(ft)^{3/4}$	1,085	1,229	1,374
Impeller Tip Diameter, in.	2.50	3.38	3.91
Efficiency, %	62.0	66.2	70.4

TABLE XVI

HYDROGEN TPA TURBINE OPERATING CHARACTERISTICS SUMMARY
(Expander Cycle)

Engine Thrust Level, lb	10,000	20,000	30,000
Turbine Type	Partial Admission	Partial Admission	Partial Admission
% Admission	13	21	32
Inlet Temperature, °R	653	463	382
Inlet Pressure, psia	2,850	2,390	2,000
Gas Flow Rate, lb/sec	2.095	4.18	6.421
Shaft Horsepower	878	1,410	1,803
Gas Properties			
C_p, Btu/lb-°R	3.53	3.774	3.875
Ratio of Specific Heats, γ	1.405	1.403	1.398
Pressure Ratio	2.02	2.00	1.94
U/C	0.34	0.46	0.44
Blade Height, in.	0.30	0.30	0.30
Mean Blade Diameter, in.	3.55	4.68	4.55
Efficiency, %	70.0	75.6	78.0

TABLE XVII

OXYGEN TPA TURBINE OPERATING CHARACTERISTICS SUMMARY

(Expander Cycle)

Engine Thrust Level, lb	10,000	20,000	30,000
Turbine Type	Terry	Terry	Terry
Inlet Temperature, °R	653	463	382
Inlet Pressure, psia	2,850	2,390	2,000
Gas Flow Rate, lb/sec	0.735	1.535	2.236
Shaft Horsepower	173	278	341
Gas Properties			
C_p, Btu/lb-°R	3.53	3.774	3.875
Ratio of Specific Heats, γ	1.405	1.403	1.398
Pressure Ratio	2.02	2.00	1.94
U/C	0.172	0.186	0.192
Wheel Diameter, in.	4.38	5.87	6.72
Efficiency, %	39.4	40.6	42.4

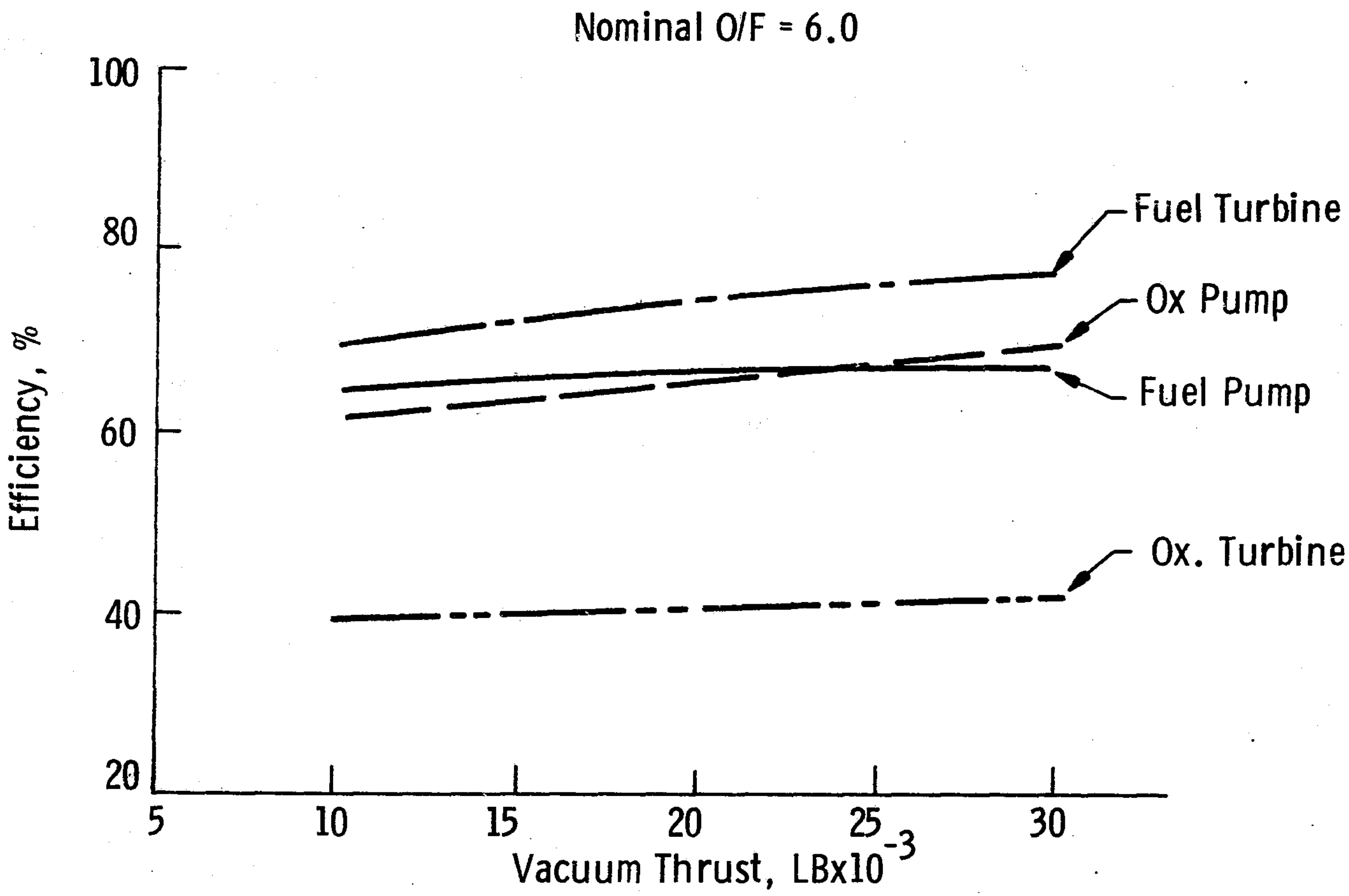


Figure 59. Advanced Expander Cycle Engine Turbomachinery Efficiencies

TABLE XVIII

TURBOPUMP PERFORMANCE AT OFF-DESIGN O/F

Thrust, lb	O/F	Pump Efficiencies, %		Turbine Efficiencies, %	
		Fuel	Oxidizer	Fuel	Oxidizer
10	6.0	64.7	62.0	70.0	39.4
10	7.0	63.0	62.0	69.5	39.4
10	10.0	60.4	59.5	66.0	39.3
20	6.0	67.5	66.2	75.6	40.6
20	7.0	66.8	66.2	74.5	40.6
30	6.0	68.2	70.4	78.0	42.4
30	7.0	67.5	70.4	77.9	42.4

E. CONTROLS ANALYSIS

The objective of the controls component preliminary design effort was to select the best valve concepts in terms of weight, envelope and ΔP to meet system operating requirements.

Since the system operating pressures are relatively high, and the engine overall size is small compared to the thrust level developed, the ideal valves for this system would have high flow efficiency and low required actuation forces to minimize valve envelope and weight.

For purposes of this study, it was assumed that the fuel and oxidizer discharge shutoff valves would be on-off type valves and that the turbine control and bypass valve would be the primary controlling elements in the system. Further study will be necessary to provide for requirements such as engine cooldown, tank pressurization, igniters, and instrumentation.

Estimated weights for the components were based on historical data, parametric curves and empirical equations developed from other engine programs.

A summary listing the typical valve configurations that could be used in this application along with pertinent information and estimated weights is presented on Table XIX. The data shown on the table are based upon preliminary power balance calculations and do not reflect the Task III update.

1. Fuel and Oxidizer Discharge Shutoff Valves

The primary function of these valves is to initiate propellant flow during the engine start sequence and shutoff the flow from the pumps to the system during engine shutdown.

TABLE XIX

VALVE SELECTION SUMMARY

Component	Type	Oper. Pressure (psia)	Oper. Temperature (°R)	Flow Rate lb/sec	Flow Port Size (in.-Diameter)	Pressure Drop and Operating Condition (psid)	Method of Actuation	Estimated (wt (lb)
Main LH_2 Shutoff Valve	Ball (on-off)	3160	90	3.01	0.750	5.3	Electric Motor	5.6
Main LO_2 Shutoff Valve	Ball (on-off)	1569	190	18.04	0.750	12.1	Electric Motor	5.6
Turbine Control and Bypass Valve Ports:								
LH_2 Turbine	Poppet (modulating)	2858	665	2.47	1.50	18.1	Electric Motor	10.4
LO_2 Turbine	Poppet (modulating)	2858	665	0.36	0.625	16	Electric Motor	10.4
GH_2 Bypass	Poppet (modulating)	2858	665	-	0.30	-	Electric Motor	

Note: Redundancy not included in valve weight. For series valves main shutoff valve weights should be doubled.

To minimize flow resistance and actuation forces, either a coaxial valve with a sleeve or balanced poppet closure element or a relatively small ball valve in a venturi section could be used. These valve concepts can be oriented in line with the flow direction and designed to minimize envelope size. Curves showing the relationship of valve flow port size to ΔP at nominal flow conditions are presented on Figure 60 for the fuel valve and Figure 61 for the oxidizer valve. As shown on these curves, the size advantage obtained using the ball valve concept is evident.

Final concept selection will depend on cycle life and leakage requirements. In general, ball valve shutoff seal cycle life is less and shutoff seal leakage is more than that provided by a poppet type coaxial valve unless an eccentric ball or seal liftoff device is utilized. As an example, Aerojet has recently designed and manufactured a series redundant bipropellant ball valve assembly for the space shuttle OME that utilizes the eccentric ball concept. This valve has successfully passed life cycle tests of 5500 cycles with ball seal leakage rates less than 20 SCCH using GHe.

2. GH_2 Bypass and Turbine Control Valves

The function of these valves is to provide proportional flow control to the fuel and oxidizer pump turbines, to assure proper mixture ratio, and to provide control of total GH_2 flow to maintain the required thrust level.

Based on the location of these valves in the engine system, a simple compact design could be in the form of a 'T' configuration with the inlet port connected to a double poppet, or balanced sleeve valve, with two outlet ports. One outlet port directs GH_2 flow to the fuel pump turbine and the other directs flow to the LOX pump turbine (see the control valve sketch on Figure 62). To provide turbine flow balance and mixture ratio control,

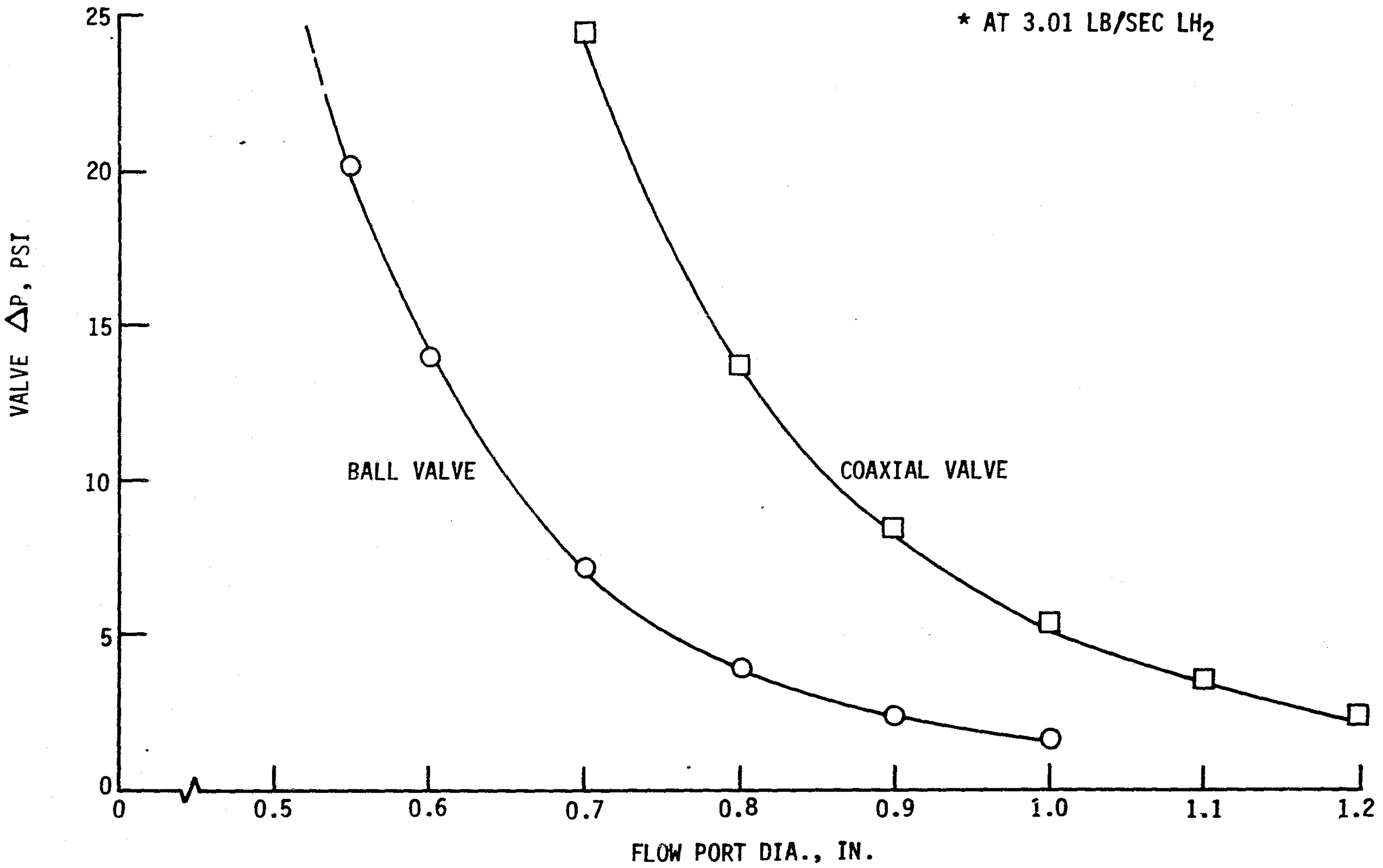


Figure 60. Fuel Discharge Valve ΔP vs Flow Port Diameter

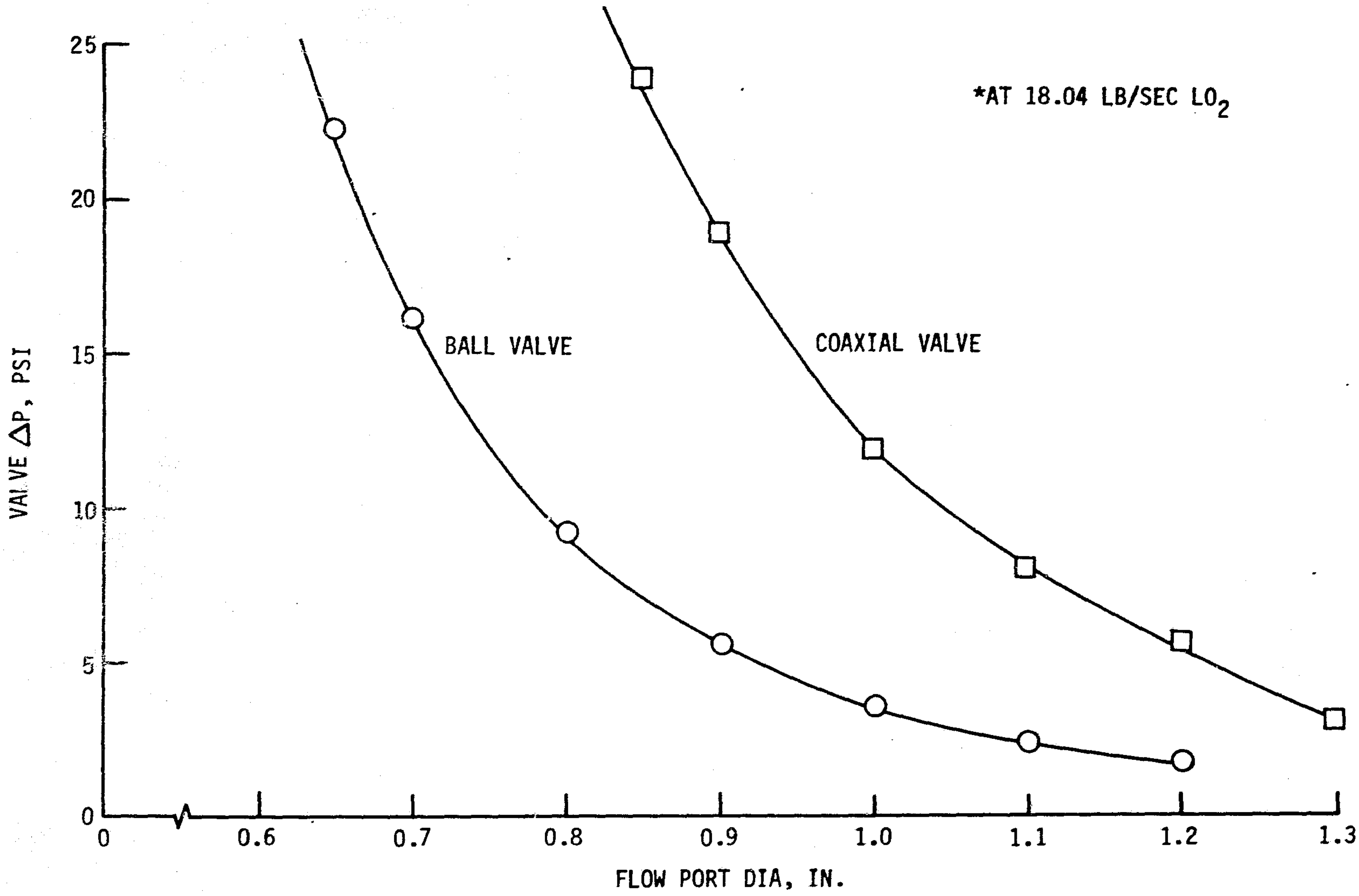


Figure 61. Oxygen Discharge Valve ΔP vs Flow Port Diameter

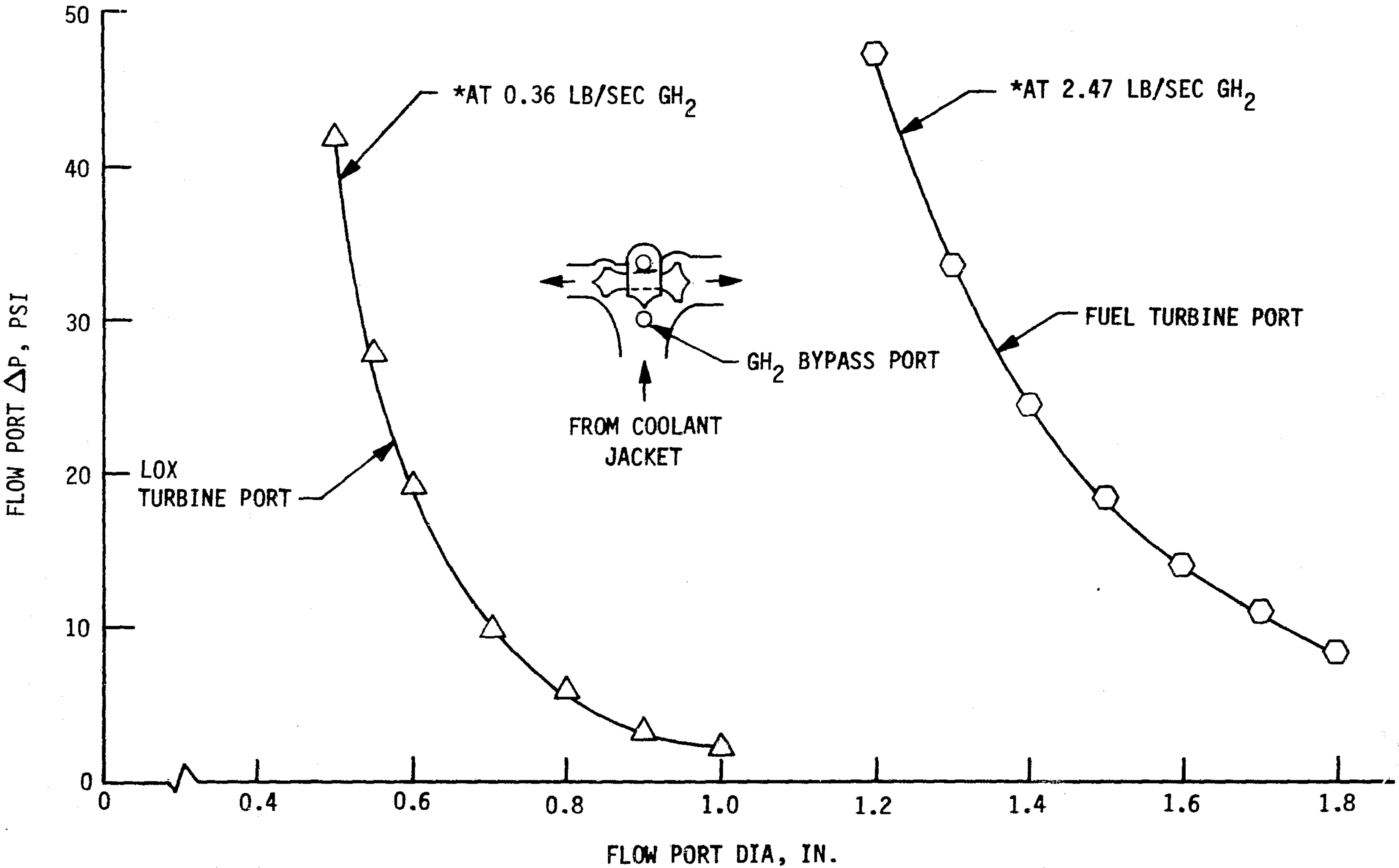


Figure 62. Turbine Control Valve ΔP vs Flow Port Diameter

translation of the poppet or sleeve would simultaneously increase the flow area passage from the turbine control valve inlet to one of the outlet ports and decrease flow area to the other port. Desired flow characteristics can be obtained by proper shaping of the selected valving element.

The GH_2 bypass valve could be attached to the turbine control valve body and physically located between the two outlet ports and adjacent to the inlet port. In conjunction with operation of the turbine control valve, the poppet type bypass valve would be modulated to control total flow to the pump turbines.

Based on nominal GH_2 flow conditions, the turbine control valve ΔP versus flow port diameter is shown on Figure 62. Nominal flow diameter of the GH_2 bypass valve is expected to be approximately 0.25 to 0.30 in. diameter.

3. Valve Actuation

Since accurate position control is required for the GH_2 bypass and turbine control valve, the use of electric motor actuators is preferred over pneumatic, propellant pressurized or hydraulic oil systems. Pneumatic systems pose a control problem in terms of gas compressibility and cryogenic collapse factors. The same problem occurs when using propellant pressure actuated systems unless the propellant used for actuation can be maintained in a liquid state by continuous bleed techniques. When using a hydraulic oil system, weight and envelope become a problem due to thermal barrier requirements to prevent excessive chilldown of the hydraulic oil.

Electric motor actuators also appear to be the best actuation method for the fuel and oxidizer discharge valves. This would provide component commonality, in terms of using the same basic motor design, and eliminate problems associated with storing, controlling and preventing leakage through the shutoff, relief, and pilot valves, regulators, lines and actuator necessary for a pneumatic actuation system, particularly under long term storage and cryogenic operating conditions. When the engine system and valve configurations have been further defined, additional trade studies will be conducted to assure the optimum method of valve actuation.

F. CYCLE ANALYSIS

The objectives of this subtask were to establish operating chamber pressures as a function of thrust for input into the parametric data analysis and to establish cycle sensitivities to operating conditions.

The thermal and turbomachinery analysis results were used to re-evaluate the expander cycle engine power balance. This new data resulted in slightly different results than the Task II Concept Definition Analysis as shown on Figure 63. However, the trends are the same and the results do not change the concept definition task conclusions. Slightly higher chamber pressures at 20K and 30K thrust levels are achieved for a given fuel pump discharge pressure than shown in Task II.

A turbine bypass flow rate of 6% was selected to provide engine power balance margin. This value was selected by evaluating the system power balance data, and reviewing OOS and RL-10 Derivative recommendations.

The operating thrust chamber pressures were selected at a point where the rate of change in the hydrogen pump discharge pressure with chamber pressure is small. The selected pressures have the same sensitivity at all thrust levels as shown on Figure 64. The chamber pressures used to conduct the Task III parametrics are compared to the Task II results:

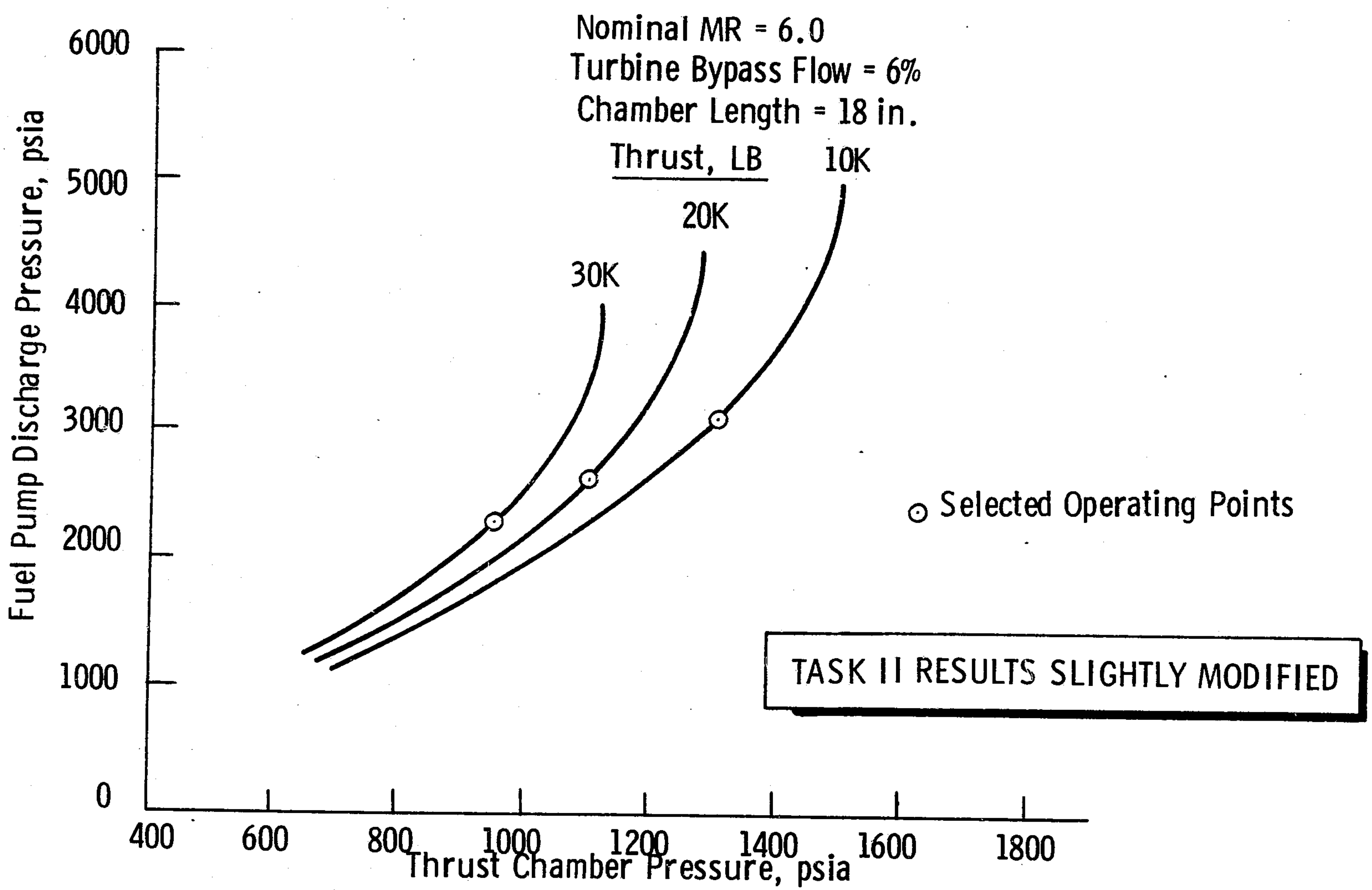


Figure 63. Task III Expander Cycle Power Balances

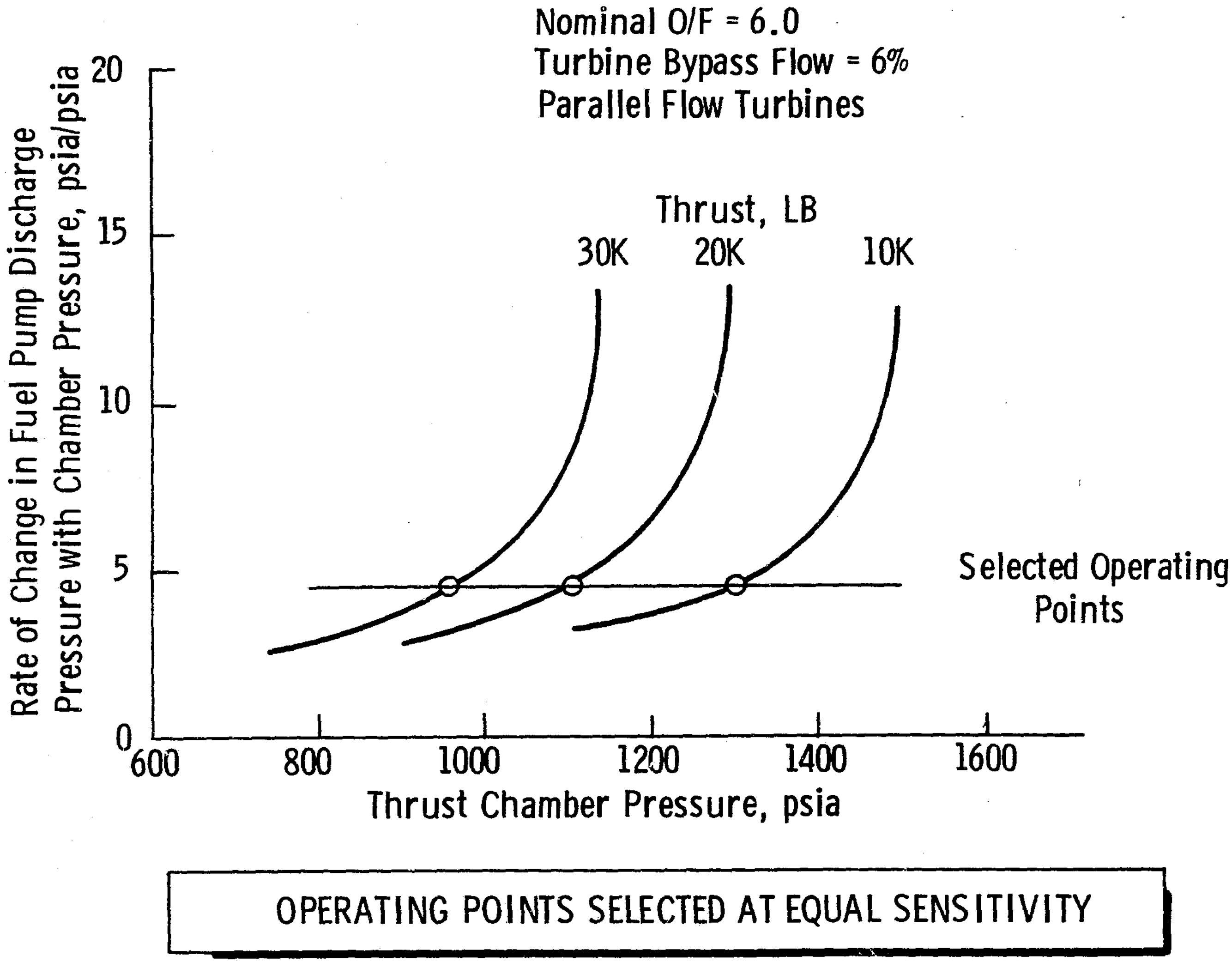


Figure 64. Advanced Expander Cycle Engine Power Balance Sensitivity

Chamber Pressure, psia	
Task II Results	Task III Results
1300	1300
1000	1100
850	950

Further sensitivity analyses were conducted for the expander cycle by calculating the delivered performance as a function of thrust and thrust chamber pressure. Figure 65 shows that the sensitivity is the lowest at a thrust level of 10K lbF. This is true because the area ratios at 10K are the highest and the change in ODE specific impulse at these high area ratios (i.e., $\sim$850:1) is small. The chamber pressure of the 10K engine could be reduced to about 950 psia and still meet the minimum specific impulse requirement of 473 sec in a twin engine installation.

The weight of the expander engines were also established as a function of thrust chamber pressure and thrust. Figure 66 shows that the weight changes with chamber pressure are small, although the higher pressure engines are heavier. Larger weight differences are not reflected because of the narrow chamber pressure band investigated. Using these results and those of the previous figure, along with the AMOTV payload partials, the sensitivity of a twin 10K engine installation to pressure is shown below:

Chamber Pressure, psia	Specific Impulse, sec	Engine Weight, lb	AMOTV Relative Payload, lb
1300	475.1	437	0
1100	474.0	430	-65
950	473.0	427	-131

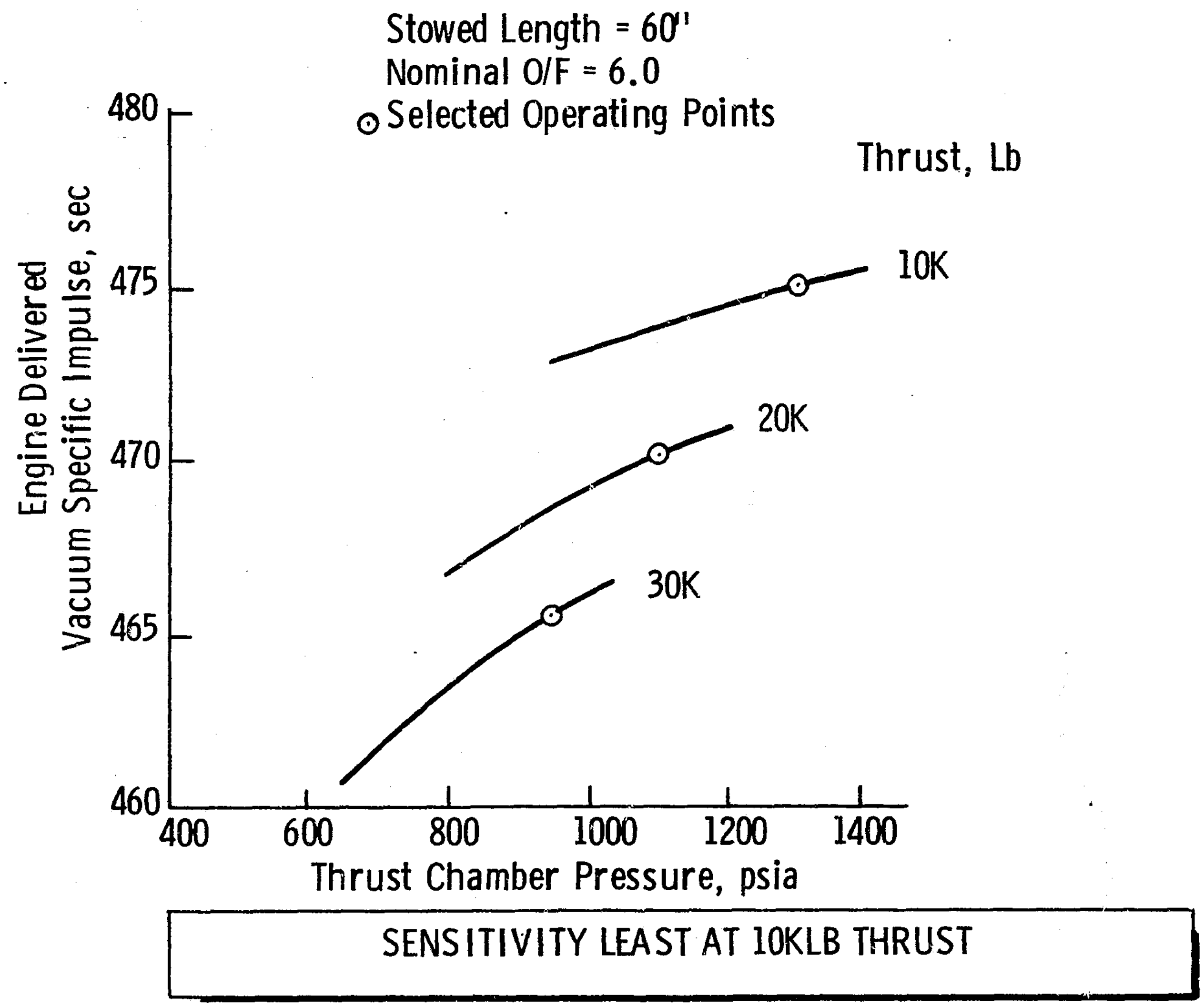
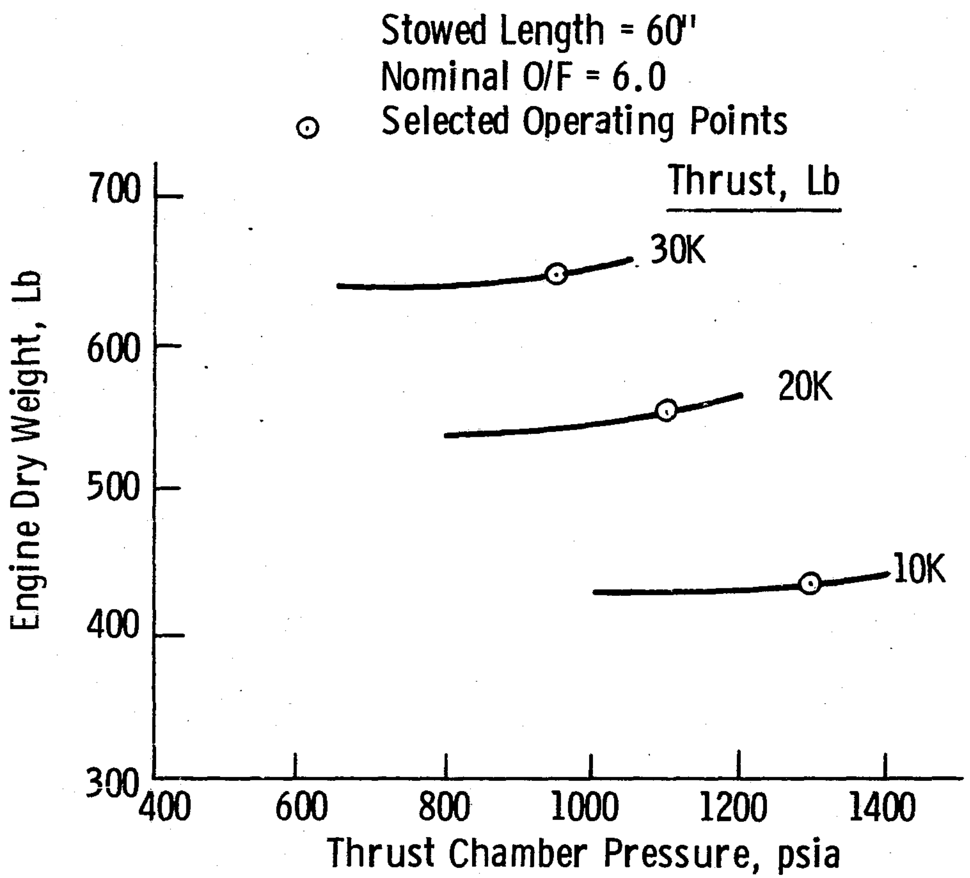


Figure 65. Expander Cycle Performance Sensitivity to Operating Chamber Pressure



WEIGHT CHANGES WITH P_C ARE SMALL

Figure 66. Expander Cycle Engine Weight Sensitivity to Operating Chamber Pressure

Preliminary engine operating specifications for an advanced expander cycle engine are shown on Tables XX, XXI and XXII at thrust levels of 10K, 20K and 30K lbF, respectively. These tables are the result of the detailed analyses performed for this task. The pressure budgets for these engines are shown on Tables XXIII, XXIV and XXV at 10K, 20K and 30K, respectively.

The recommended baseline advanced expander cycle engine characteristics that evolved from this study are shown on Figure 67. The engine uses an extendible/retractable nozzle which extends from 297:1 to 792:1 at the exit. This nozzle extension is 60 inches long and is radiation cooled. It should also be noted that the length of the engine from the gimbal center to the fixed nozzle exit (ε = 297:1) is also 60 inches. This is to satisfy the Phase "A" engine requirement of a maximum engine length with the extendible nozzle in the stowed position of 60 inches (nominal). The maximum engine length with this nozzle deployed is 120 inches (nominal).

Performance was calculated using the modified simplified JANNAF performance procedures and is summarized below:

ODE Specific Impulse, sec	491.3
Boundary Layer Loss, sec	-7.9
Kinetics Loss, sec	-3.1
Divergence Loss, sec	-2.7
Energy Release Loss, sec	-2.5
Delivered Performance, sec	475.1

The expander cycle engine schematic is shown on Figure 68. The description which follows is for the 10K baseline engine. The slotted copper chamber is cooled with 85% of the total hydrogen flow from an area ratio of

TABLE XX

PRELIMINARY ADVANCED EXPANDER CYCLE
ENGINE OPERATING SPECIFICATION

Rated Vacuum Thrust = 10,000 lb

Stowed Length = 60 in.

Engine	
Vacuum Thrust, lb	10,000
Vacuum Specific Impulse, sec.	475.1
Total Flow Rate, lb/sec	21.05
Mixture Ratio	6.0
Oxygen Flow Rate, lb/sec	18.04
Hydrogen Flow Rate, lb/sec	3.01
Thrust Chamber	
Vacuum Thrust, lb	10,000
Vacuum Specific Impulse, sec	475.1
Chamber Pressure, psia	1300
Nozzle Area Ratio	792
Mixture Ratio	6.0
Throat Diameter, in.	2.184
Chamber Diameter, in.	4.18
Chamber Length, in.	18.0
Chamber Contraction Ratio	3.66
Nozzle Exit Diameter, in.	61.5
Percent Bell Nozzle Length	86.5
Nozzle Length, in.	95.6
Combustion Chamber Coolant Flow Rate, lb/sec	2.56
Slotted Copper Chamber Area Ratio	8.0
Chamber Pressure Drop, psia	131
Coolant Inlet Temperature, °R	90
Chamber Coolant Temperature Rise, °R	524
Fixed Tube Bundle Nozzle Flow Rate, lb/sec	0.45
Tube Bundle Nozzle Area Ratio	297
Tube Bundle Coolant Pressure Drop, psia	12
Tube Bundle Coolant Temperature Rise, °R	784

TABLE XX (cont.)

F = 10,000 lb

Main Pumps	LOX	LH_2
Inlet Temperature, °R	162.7	37.8
Inlet Pressure, psia	46.6	51.0
Inlet Density, lb/ft^3	71.0	4.40
Vapor Pressure, psia	15.0	18.0
Net Positive Suction Pressure, psia	31.6	33.0
Net Positive Suction Head, ft	64.1	1080
Total Outlet Flow Rate, lb/sec	18.04	3.01
Volumetric Flow Rate, gpm	114.0	307.1
Suction Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	20000	9302
Speed, rpm	42435	100000
Discharge Pressure, psia	1585	3130
Head Rise, ft	3120	100767
Number of Stages	1	3
Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	1085	706
Head Coefficient	0.465	0.51
Impeller Tip Speed, ft/sec	465	1456
Impeller Tip Diameter, in.	2.50	3.33
Efficiency, %	62	64.7

Main Pump Turbines	LOX TPA	LH_2 TPA
Inlet Pressure, psia	2854	2854
Inlet Temperature, °R	653	653
Flow Rate, lb/sec	0.735	2.095
Gas Properties		
Cp, Specific Heat at Constant Pressure, BTU/lb-°R	3.53	3.53
γ, Ratio of Specific Heats	1.405	1.405
Shaft Horsepower	173.3(1)	877.9(2)
Pressure Ratio (Total to Static)	2.02	2.02
Static Exit Pressure, psia	1413	1413
Static Exit Temperature, °R	533.1	533.1
Efficiency, %	39.4	70.0
Turbine Bypass Flow Rate, lb/sec	0.18(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.

TABLE XXI

PRELIMINARY ADVANCED EXPANDER CYCLE ENGINE OPERATING SPECIFICATION

Rated Vacuum Thrust = 20,000 lb

Stowed Length = 60 in.

Engine	
Vacuum Thrust, lb	20000
Vacuum Specific Impulse, sec.	470.1
Total Flow Rate, lb/sec	42.54
Mixture Ratio	6.0
Oxygen Flow Rate, lb/sec	36.46
Hydrogen Flow Rate, lb/sec	6.08
Thrust Chamber	
Vacuum Thrust, lb	20000
Vacuum Specific Impulse, sec	470.1
Chamber Pressure, psia	1100
Nozzle Area Ratio	322
Mixture Ratio	6.0
Throat Diameter, in.	3.374
Chamber Diameter, in.	6.45
Chamber Length, in.	18.0
Chamber Contraction Ratio	3.66
Nozzle Exit Diameter, in.	60.52
Percent Bell Nozzle Length	87.2
Nozzle Length, in.	93.0
Combustion Chamber Coolant Flow Rate, lb/sec	5.17
Slotted Copper Chamber Area Ratio	8.0
Chamber Pressure Drop, psia	76
Coolant Inlet Temperature, °R	90
Chamber Coolant Temperature Rise, °R	340
Fixed Tube Bundle Nozzle Flow Rate, lb/sec	0.91
Tube Bundle Nozzle Area Ratio	122
Tube Bundle Coolant Pressure Drop, psia	9
Tube Bundle Coolant Temperature Rise, °R	563

TABLE XXI (cont.)

F = 20,000 lb

Main Pumps	LOX	LH_2
Inlet Temperature, °R	162.7	37.8
Inlet Pressure, psia	46.6	51.0
Inlet Density, lb/ft^3	71.0	4.40
Vapor Pressure, psia	15.0	18.0
Net Positive Suction Pressure, psia	31.6	33.0
Net Positive Suction Head, ft.	64.1	1080
Total Outlet Flow Rate, lb/sec	36.46	6.08
Volumetric Flow Rate, gpm	230.5	620.2
Suction Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	20000	11900
Speed, rpm	29840	90000
Discharge Pressure, psia	1350	2605
Head Rise, ft	2644	83585
Number of Stages	1	3
Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	1229	1039
Head Coefficient	0.44	0.46
Impeller Tip Speed, ft/sec	440	1396
Impeller Tip Diameter, in.	3.38	3.55
Efficiency, %	66.2	67.5

Main Pump Turbines	LOX TPA	LH_2 TPA
Inlet Pressure, psia	2392	2392
Inlet Temperature, °R	463	463
Flow Rate, lb/sec	1.535	4.18
Gas Properties		
Cp, Specific Heat at Constant Pressure, BTU/lb-°R	3.774	3.774
γ, Ratio of Specific Heats	1.403	1.403
Shaft Horsepower	278(1)	1410(2)
Pressure Ratio (Total to Static)	2.00	2.00
Static Exit Pressure, psia	1196	1196
Static Exit Temperature, °R	379.4	379.4
Efficiency, %	40.6	75.6
Turbine Bypass Flow Rate, lb/sec	0.365(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.

TABLE XXII

PRELIMINARY ADVANCED EXPANDER CYCLE
ENGINE OPERATING SPECIFICATION

Rated Vacuum Thrust = 30,000 lb

Stowed Length = 60 in.

Engine	
Vacuum Thrust, lb	30000
Vacuum Specific Impulse, sec.	465.5
Total Flow Rate, lb/sec	64.45
Mixtur Ratio	6.0
Oxygen Flow Rate, lb/sec	55.24
Hydrogen Flow Rate, lb/sec	9.21
Thrust Chamber	
Vacuum Thrust, lb	30000
Vacuum Specific Impulse, sec	465.5
Chamber Pressure, psia	950
Nozzle Area Ratio	180
Mixture Ratio	6.0
Throat Diameter, in.	4.464
Chamber Diameter, in.	8.54
Chamber Length, in.	18.0
Chamber Contraction Ratio	3.66
Nozzle Exit Diameter, in.	59.89
Percent Bell Nozzle Length	88.0
Nozzle Length, in.	90.98
Combustion Chamber Coolant Flow Rate, lb/sec	7.83
Slotted Copper Chamber Area Ratio	8.0
Chamber Pressure Drop, psia	91.0
Coolant Inlet Temperature, °R	90
Chamber Coolant Temperature Rise, °R	269
Fixed Tube Bundle Nozzle Flow Rate, lb/sec	1.38
Tube Bundle Nozzle Area Ratio	73
Tube Bundle Coolant Pressure Drop, psia	7
Tube Bundle Coolant Temperature Rise, °R	423

TABLE XXII (cont.)

F = 30,000 lb

Main Pumps	LOX	LH_2
Inlet Temperature, °R	162.7	37.8
Inlet Pressure, psia	46.6	51.0
Inlet Density, lb/ft^3	71.0	4.40
Vapor Pressure, psia	15.0	18.0
Net Positive Suction Pressure, psia	31.6	33.0
Net Positive Suction Head, ft.	64.1	1080
Total Outlet Flow Rate, lb/sec	55.24	9.21
Volumetric Flow Rate, gpm	349.2	939.5
Suction Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	20000	13015
Speed, rpm	24245	80000
Discharge Pressure, psia	1170	2230
Head Rise, ft	2278	71312
Number of Stages	1	2
Specific Speed, $(rpm)(gpm)^{1/2}/(ft)^{3/4}$	1374	945
Head Coefficient	0.427	0.47
Impeller Tip Speed, ft/sec	414	1562
Impeller Tip Diameter, in.	3.91	4.47
Efficiency, %	70.4	68.2

Main Pump Turbines	LOX TPA	LH_2 TPA
Inlet Pressure, psia	2004	2004
Inlet Temperature, °R	382	382
Flow Rate, lb/sec	2.236	6.421
Gas Properties		
Cp, Specific Heat at Constant Pressure, BTU/lb-°R	3.875	3.875
γ, Ratio of Specific Heats	1.398	1.398
Shaft Horsepower	341.2 (1)	1803 (2)
Pressure Ratio (Total to Static)	1.94	1.94
Static Exit Pressure, psia	1033	1033
Static Exit Temperature, °R	316.3	316.3
Efficiency, %	42.4	78.0
Turbine Bypass Flow Rate, lb/sec	0.553 (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.

TABLE XXIII

ADVANCED EXPANDER CYCLE ENGINE PRESSURE SCHEDULE

F = 10,000 lb

Pressure, psia	LOX	LH_2
Boost Pump Inlet	16	18.5
Boost Pump Discharge	56.6	61.0
Main Pump Inlet	46.6	51.0
Main Pump Discharge	1585	3130
ΔP Line	10	10
Main Shutoff Valve Inlet	1575	3120
ΔP Shutoff Valve	35	40
Shutoff Valve Outlet	1540	3080
ΔP Line	10	10
Coolant Jacket Inlet	-	3070
ΔP Chamber Coolant Jacket	-	131
Coolant Jacket Outlet	-	2939
ΔP Line	-	10
Turbine Valve Inlet	-	2929
ΔP Turbine Valve	-	65
Turbine Valve Outlet	-	2864
ΔP Line	-	10
Turbine Inlet	-	2854
ΔP Turbine	-	1441
Turbine Exit (static)	-	1413
Injector Inlet (total)	1530	1413
ΔP Injector	230	113
Chamber Pressure	1300	1300

TABLE XXIV

ADVANCED EXPANDER CYCLE ENGINE
PRESSURE SCHEDULE

F = 20,000 lb

Pressure, psia	LOX	LH_2
Boost Pump Inlet	16	18.5
Boost Pump Discharge	56.6	61.0
Main Pump Inlet	46.6	51.0
Main Pump Discharge	1350	2605
ΔP Line	10	10
Main Shutoff Valve Inlet	1340	2595
ΔP Shutoff Valve	35	40
Shutoff Valve Outlet	1305	2555
ΔP Line	10	10
Coolant Jacket Inlet	-	2545
ΔP Chamber Coolant Jacket	-	76
Coolant Jacket Outlet	-	2469
ΔP Line	-	10
Turbine Valve Inlet	-	2459
ΔP Turbine Valve	-	57
Turbine Valve Outlet	-	2402
ΔP Line	-	10
Turbine Inlet	-	2392
ΔP Turbine	-	1196
Turbine Exit (static)	-	1196
Injector Inlet (total)	1295	1196
ΔP Injector	195	96
Chamber Pressure	1100	1100

TABLE XXV

ADVANCED EXPANDER CYCLE ENGINE
PRESSURE SCHEDULE

F = 30,000 lb

Pressure, psia	LOX	LH_2
Boost Pump Inlet	16	18.5
Boost Pump Discharge	56.6	61.0
Main Pump Inlet	46.6	51.0
Main Pump Discharge	1170	2230
ΔP Line	10	10
Main Shutoff Valve Inlet	1160	2220
ΔP Shutoff Valve	32	40
Shutoff Valve Outlet	1128	2180
ΔP Line	10	10
Coolant Jacket Inlet	-	2170
ΔP Chamber Coolant Jacket	-	91
Coolant Jacket Outlet	-	2079
ΔP Line	-	10
Turbine Valve Inlet	-	2069
ΔP Turbine Valve	-	55
Turbine Valve Outlet	-	2014
ΔP Line	-	10
Turbine Inlet	-	2004
ΔP Turbine	-	971
Turbine Exit (static)	-	1033
Injector Inlet (total)	1118	1033
ΔP Injector	168	83
Chamber Pressure	950	950

- VACUUM THRUST = 10,000 LB
- VACUUM SPECIFIC IMPULSE = 475.1
- THRUST CHAMBER PRESSURE = 1300 PSIA
- MIXTURE RATIO = 6.0
- NOZZLE AREA RATIO = 792
- ENGINE LENGTH
    - EXTENDIBLE NOZZLE STOWED = 60"
    - EXTENDIBLE NOZZLE DEPLOYED = 120"
- NOZZLE EXIT DIAMETER = 61.5"
- ENGINE DRY WEIGHT = 437 LB

Figure 67. Recommended Baseline Advanced Expander Cycle Engine for Twin Engine Installation

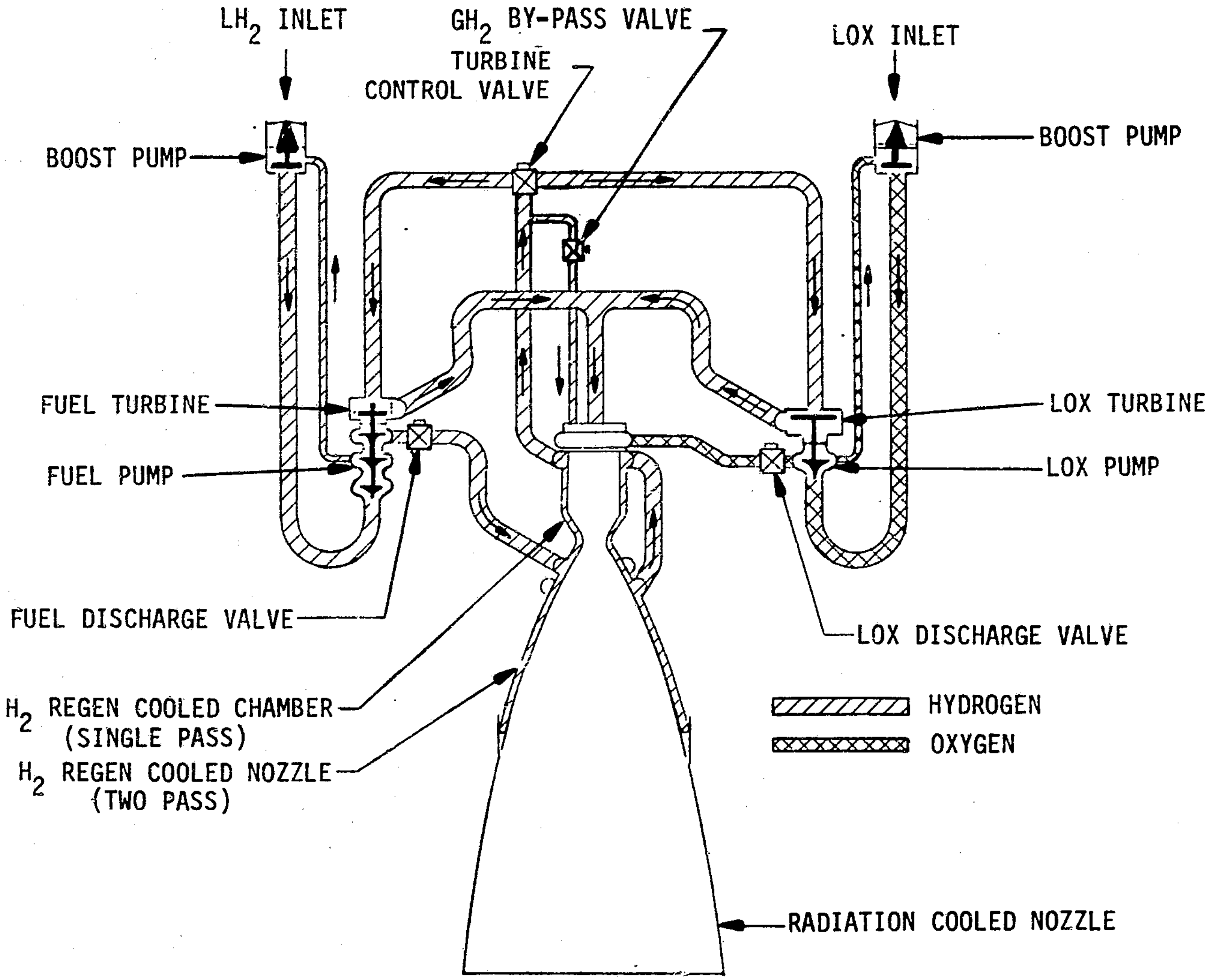


Figure 68. Task III Advanced Expander Cycle Engine Schematic

8:1 to the injector. The fixed portion of the nozzle from an area ratio of 8:1 to 297:1 is cooled with the remaining 15% of the hydrogen flow in a two-pass tube bundle. The nozzle and chamber coolants are combined to provide the warm (653°R) turbine drive gas. Six percent of the flow bypasses the turbines to provide margin and control. 69.6% of the hydrogen flow is used to drive the LH_2 TPA and 24.4% is used to drive the LOX TPA. The turbine exhaust and bypass flows enter the injector to be mixed and burned with the liquid oxygen.

G. MATERIALS SELECTIONS

The primary materials candidates for the major engine components are listed on Table XXVI. Where two or more material candidates are listed, the first is currently the preferred choice. The materials were selected to achieve lightweight with consideration of the design and long life requirements and the environmental and propellant compatibility aspects. They also meet the requirement to be within 1980 technology.

H. PARAMETRIC DATA

The objective of this subtask was to provide engine performance, weight and envelope parametric data for each of the candidate engine concepts. Parametric cost data was also established and is reported in Volume III of this final report. The parametric ranges considered in this study are shown on Table XXVII. The nozzle area ratio is dependent upon the available engine length and the operating chamber pressures selected for each of the concepts. The maximum engine length with the extendible nozzle in the retracted position was varied parametrically as shown on the table. The chamber pressures were selected on the basis of the power balance, cycle life or payload trade-off study results. The operating chamber pressures selected to conduct the parametric studies for each candidate engine concept are shown on Table XXVIII. Because the concept definition task results

TABLE XXVI

MATERIALS SELECTION

Component	Material Candidates
1. Low Speed TPA's	
a. Shaft	Inconel 718 A-286
b. Impeller and Turbine	7075 T-73 aluminum 2219 T-87 aluminum Inconel 718 5Al-2.5 Sn-ELI titanium (H_2 only)
c. Housing	A-356 T-6 aluminum casting 2219 T-87 aluminum
d. Fasteners	A-286
e. Bearings	CRES 440C Cobalt alloys, powder metallurgy
2. High Speed TPA's	
a. LOX TPA	
(1) Shaft	A-286 Inconel 718
(2) Impeller	7075 T-73 aluminum Inconel 718
(3) Pump Housing	Inconel 718 Armco Nitronic 50 (22-13-5)
(4) Turbine Housing	Armco Nitronic 50 (22-13-5)
(5) Turbine	A-286
(6) Fasteners	A-286
(7) Bearings	CRES 440C Cobalt Alloys, powder metallurgy
b. LH_2 TPA	
(1) Shaft	A-286
(2) Impeller	5Al-2.5 Sn-ELI titanium alloy
(3) Pump Housing	5Al-2.5 Sn-ELI titanium
(4) Turbine Housing	Armco Nitronic 50 (22-13-5)
(5) Turbine	A-286
(6) Fasteners	A-286
(7) Bearings	CRES 440C Cobalt alloys, powder metallurgy

TABLE XXVI (cont.)

Component	Material Candidates
3. Thrust Chamber Injector	CRES 304L CRES 347 Inconel 718
4. Combustion Chamber, Regenerative	Zirconium Copper Electroformed Nickel
5. Nozzle, Tubular	A-286 Armco Nitronic 50 (22-13-5) CRES 347
6. Nozzle, Radiation Cooled	FS-85 Columbium C-103 Columbium R-512E Coating Haynes 188
7. Valves	
a. Valve Bodies	A-286 CRES or Armco Nitronic 50 (22-13-5)
b. Shafts	A-286 CRES
c. Shutoff Seals	Phosphor bronze seal on CRES 347 (Electrolized) seat Gold plated CRES 347 seal on electrolized 347 CRES seat Filled teflons; Polyimides
d. Dynamic Shaft Seals	15% graphite filled Teflon (Delta seal)
e. Poppets	A-286
f. Guide Bushings	Teflon
g. Valve Springs	CRES 301 or CRES 302
h. Electric Motor Housings	356-T6 Aluminum Alloy

TABLE XXVII

PHASE A OTV PARAMETRIC RANGES

Thrust Level: 10,000 to 30,000 lb

Maximum Retracted Length: 50, 60 and 70 inches

Nozzle Area Ratio: TBD

Nominal Thrust = 20,000 lb

Nominal Retracted Length = 60 in.

Nominal Extended Length = 120 in.

Thrust Chamber Pressure: TBD

Nominal O/F = 6.0

Off-Design O/F = 6.5 and 7.0

TABLE XXVIII

TASK III CANDIDATE CYCLE OPERATING PRESSURES

Cycle	Vacuum Thrust, Klb	Thrust Chamber Pressure, psia	Criteria
Expander	10	1300	Task III Power Balance
Expander	20	1100	Task III Power Balance
Expander	30	950	Task III Power Balance
Staged Combustion	10	1500	Cycle Life*
Staged Combustion	20	2000	Cycle Life*
Staged Combustion	30	2300	Cycle Life*
Gas Generator	10	1500	Payload Optimized*
Gas Generator	20	1500	Payload Optimized*
Gas Generator	30	1500	Payload Optimized*

*Same as Task II.

showed that the gas-generator cycle engines low performance would preclude its use, data on this cycle was generated only for a nominal 60 inch stowed length.

The engine delivered performance data presented herein was calculated using the simplified JANNAF performance procedures, with the boundary layer loss modification previously discussed in Sections IV,C and V,A.

The engine weight data includes series redundant main propellant valves and redundant igniters in all combustors per the recommendations of the reliability and safety analysis. The components included in the total engine weight are:

Gimbal
Injector
Copper Chamber and Nozzle
Tube Bundle Nozzle
Radiation Cooled Nozzle Extension
Nozzle Deployment Mechanism
Main Valves and Actuators
LOX Boost Pump
LH_2 Boost Pump
LOX Main TPA
LH_2 Main TPA
Preburner/Gas Generator (where applicable)
Propellant Lines
Ignition System
Heat Exchanger
Engine Controller
Miscellaneous Valves
Miscellaneous (electrical harness, instrumentation and brackets)

Weights do not include the gimbal actuators and actuation system, pre-valves or a contingency which is normally included in the vehicle weight statement.

Weights are based upon scaling of historical data on similar components. Where component weight data did not exist, weights were obtained from past related studies and preliminary design information.

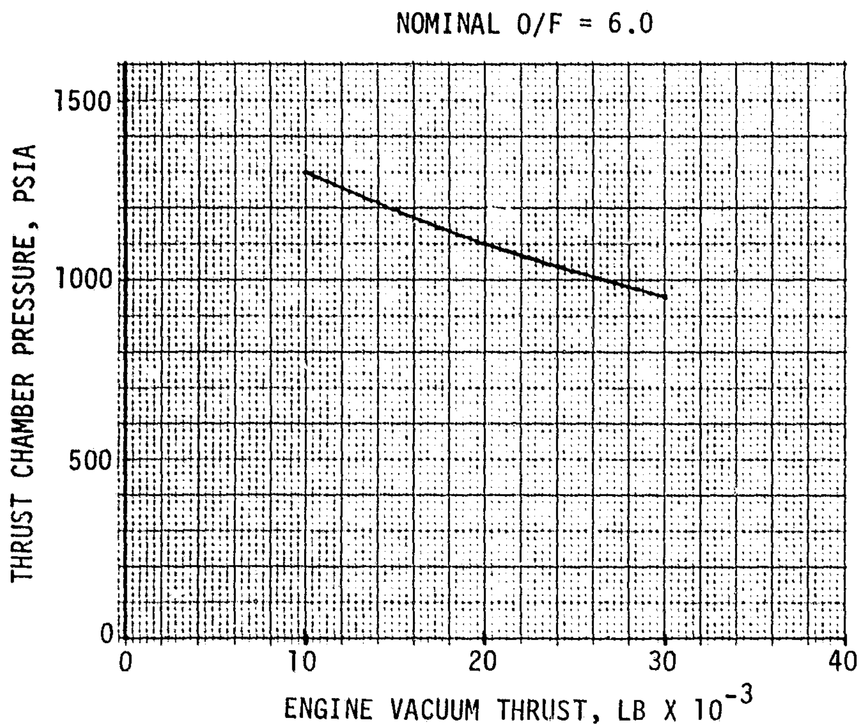
In all cases, the engine data is shown as a function of the single engine subassembly thrust level. It should be noted that the reliability and safety analysis recommended a minimum of two engines. Multiple engine installations should be considered in using the parametric data presented. For example, if a total engine thrust of 20,000 lb is desired, two 10K engines would meet this requirement and the weight and diameter data presented at 10K should be doubled.

1. Advanced Expander Cycle Engine

The advanced expander cycle engine analyzed in this task was depicted schematically on Figure 68.

The expander cycle engine thrust chamber pressure and nozzle area ratio variations with rated vacuum thrust, at nominal mixture ratio, are shown on Figure 69. As expected, the figure shows that low thrust, long length engines achieve the highest area ratios. The area ratios at 30K are reduced because of both the engine thrust size and the lower operating chamber pressure.

The variation of percent bell nozzle length and engine diameter (nozzle exit) with rated vacuum thrust is shown on Figure 70. The percent bell length at 10K and 70 inches stowed length is significantly larger



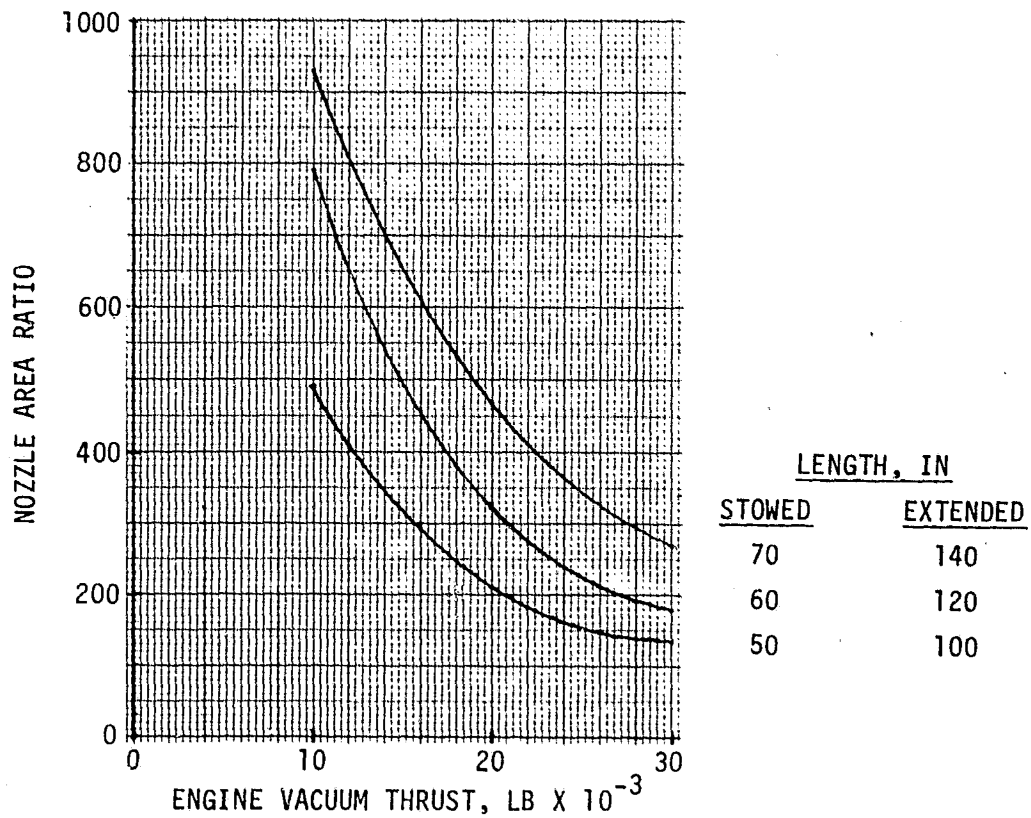
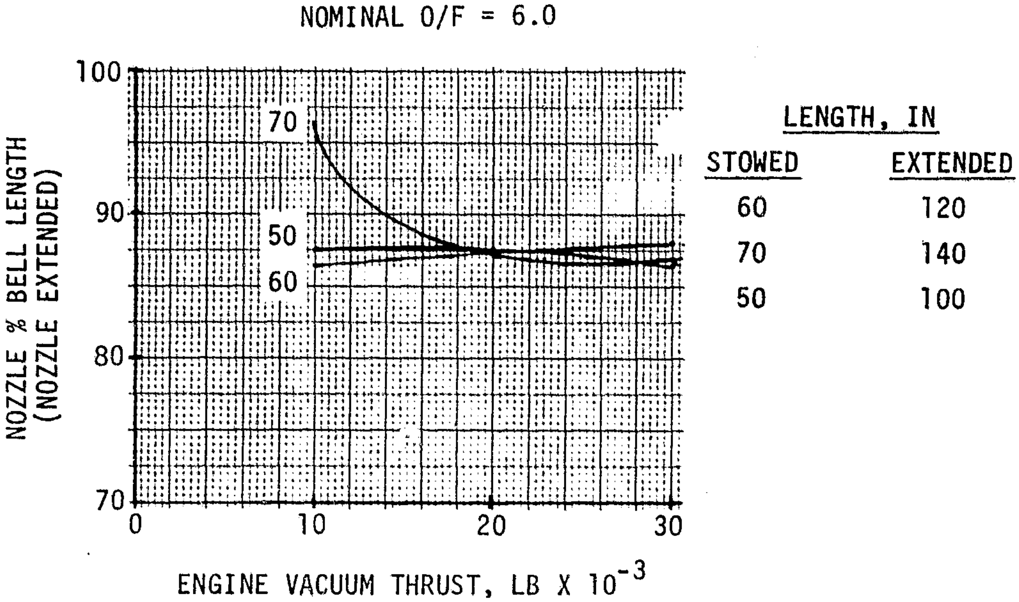


Figure 69. Advanced Expander Cycle Engine Thrust Chamber Pressure and Nozzle Area Ratio Variations with Rated Thrust



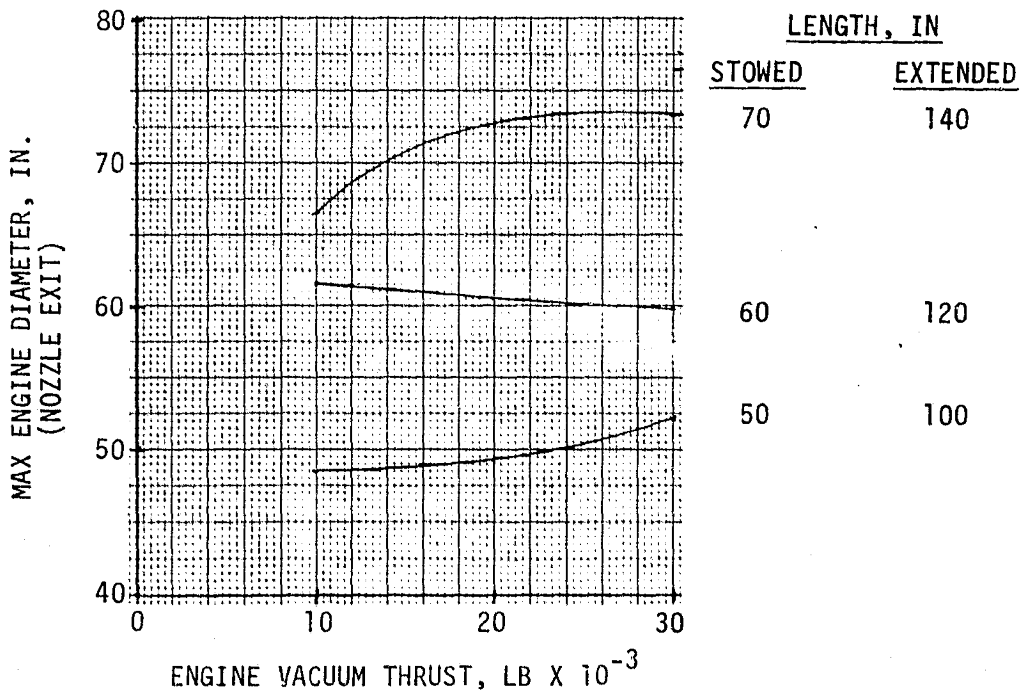


Figure 70. Advanced Expander Cycle Engine % Bell Length and Diameter Variations with Rated Thrust

than the 50 or 60 inch values because the performance/length/area ratio trades showed that the performance gains from increasing the bell length were greater than those obtained from increasing the area ratio. The diameters which evolved from the area ratio/% length trades are shown on the bottom of Figure 70.

The advanced expander cycle delivered performance and weight parametrics are shown on Figure 71. At 10K lb thrust and 60 inches stowed length, the delivered performance is approximately 96.7% of the theoretical one dimensional specific impulse value.

The expander cycle engine chamber length affects the available length for the nozzle and hence, the performance. The chamber lengths used in these evaluations are shown on Figure 72. At 50 inches stowed length, the chamber length is reduced with increasing thrust to accommodate a radiation cooled nozzle and minimize stowed length. This was discussed in Section V,C.

The turbine inlet temperatures are shown as a function of rated vacuum thrust on Figure 73. The turbine inlet temperatures at 20K and 30K lb thrust are much lower at 50 inches stowed length than at 60 or 70 inches because of the chamber length reduction. This, in turn, results in high fuel pump discharge pressure requirements as shown on Figure 74.

The advanced expander cycle engine performance and critical operating parameters such as, chamber pressure, turbine inlet temperature and pump discharge pressures are shown as a function rated thrust and stowed length over a mixture ratio range from 6.0:1 to 7.0:1 on Figures 75, 76, 77, 78, 79 and 80. Figure 75 shows that the peak performance shifts to higher mixture ratios as the thrust level is reduced. This occurs because the peak ODE performance is obtained at higher mixture ratios as the nozzle area ratio is increased.

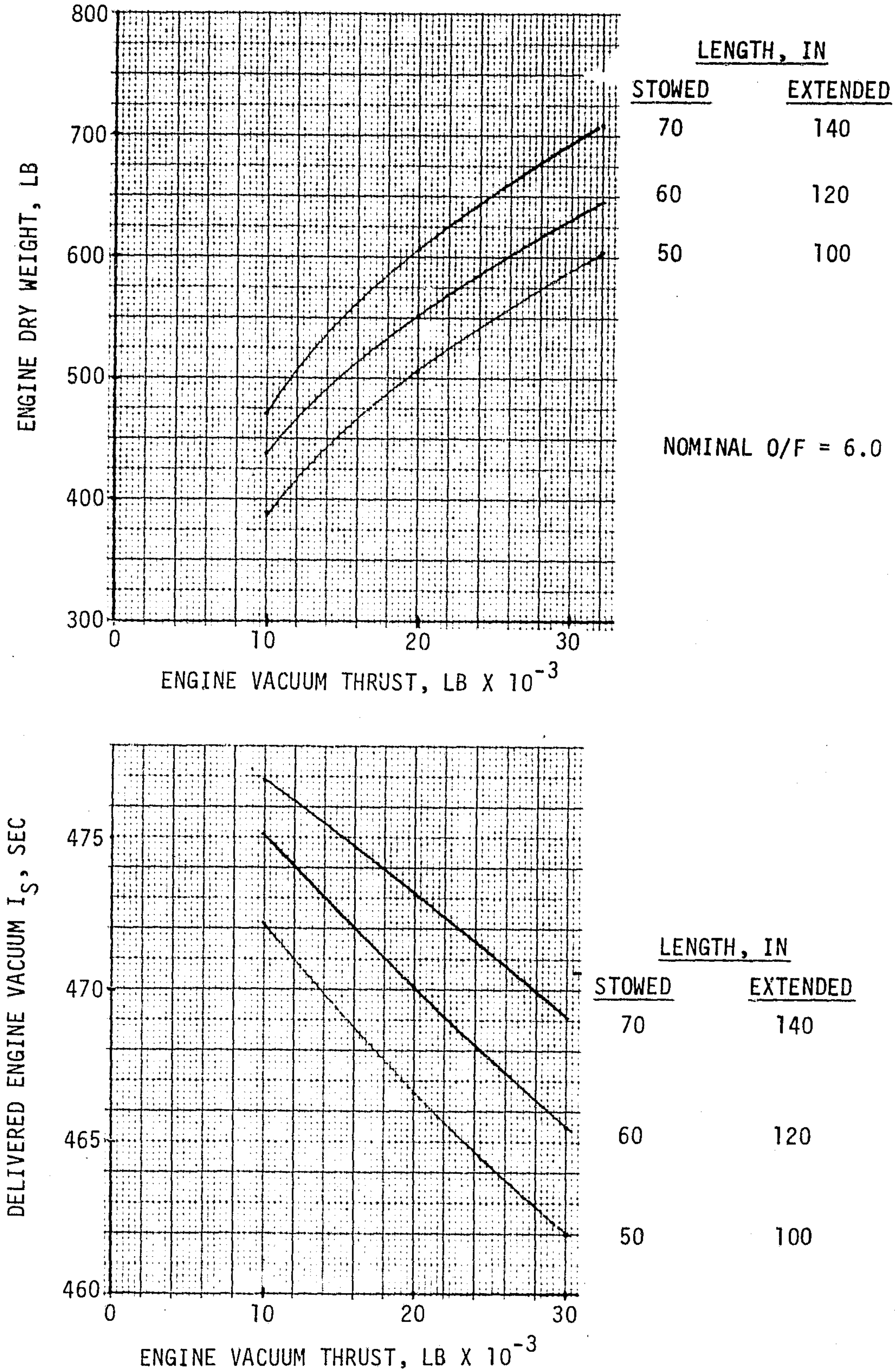
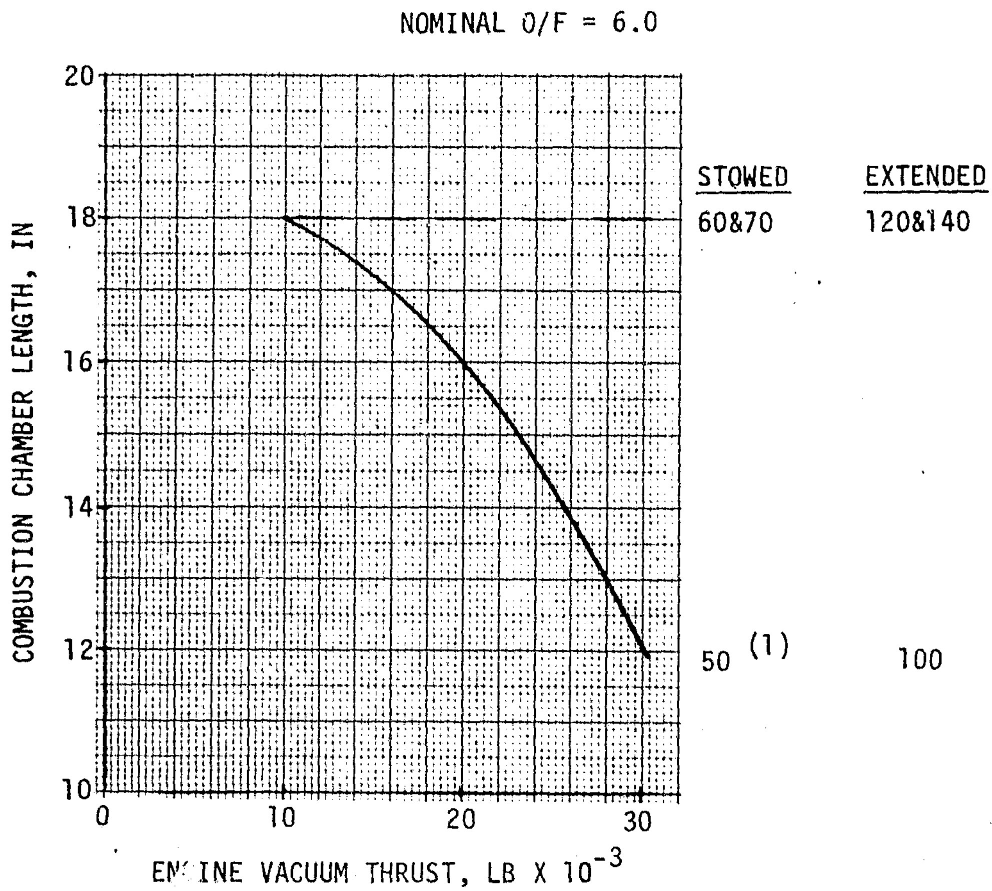


Figure 71. Advanced Expander Cycle Engine Weight and Performance Variations with Rated Thrust



(1) NOTE: CHAMBER LENGTH IS REDUCED TO MINIMIZE STOWED LENGTH

Figure 72. Advanced Expander Cycle Chamber Length

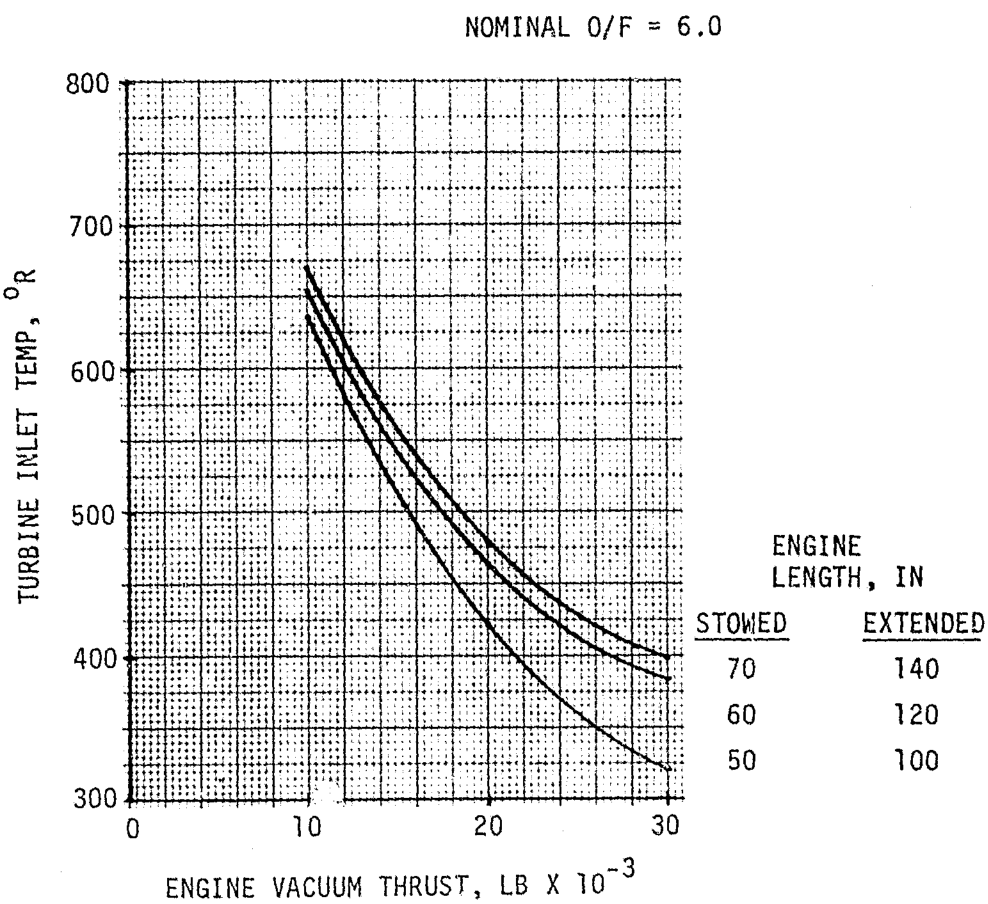
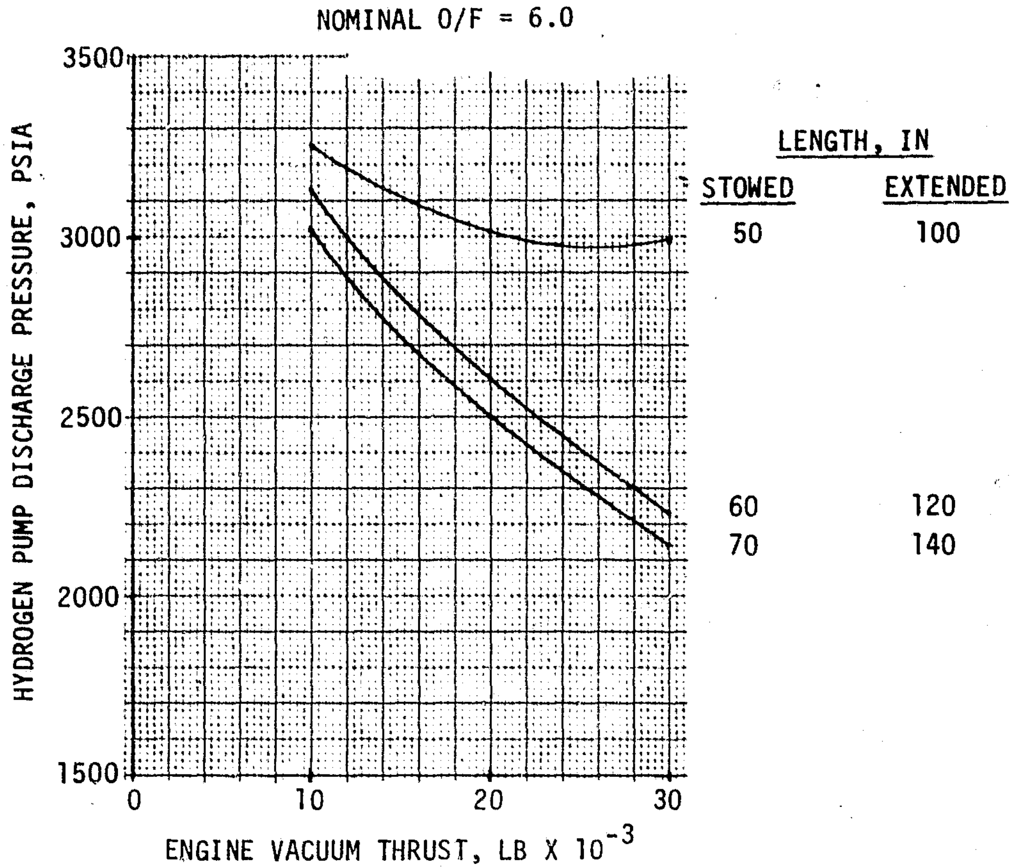


Figure 73. Advanced Expander Cycle Engine Turbine Inlet Temperature Variations with Rated Thrust



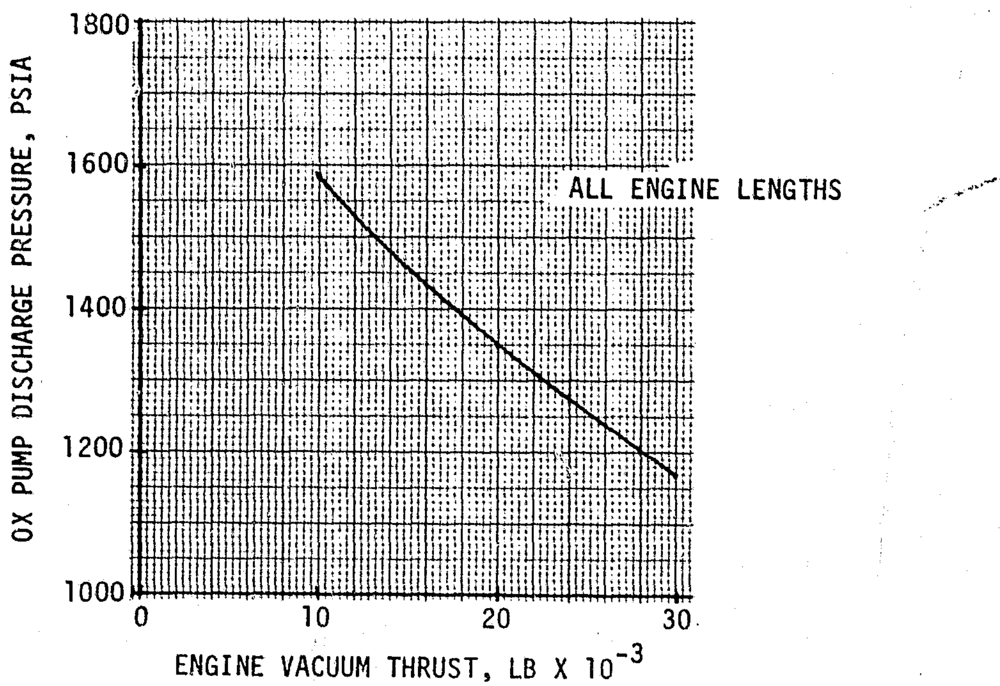


Figure 74. Advanced Expander Cycle Engine Pump Discharge Pressure Requirements as a Function of Rated Thrust

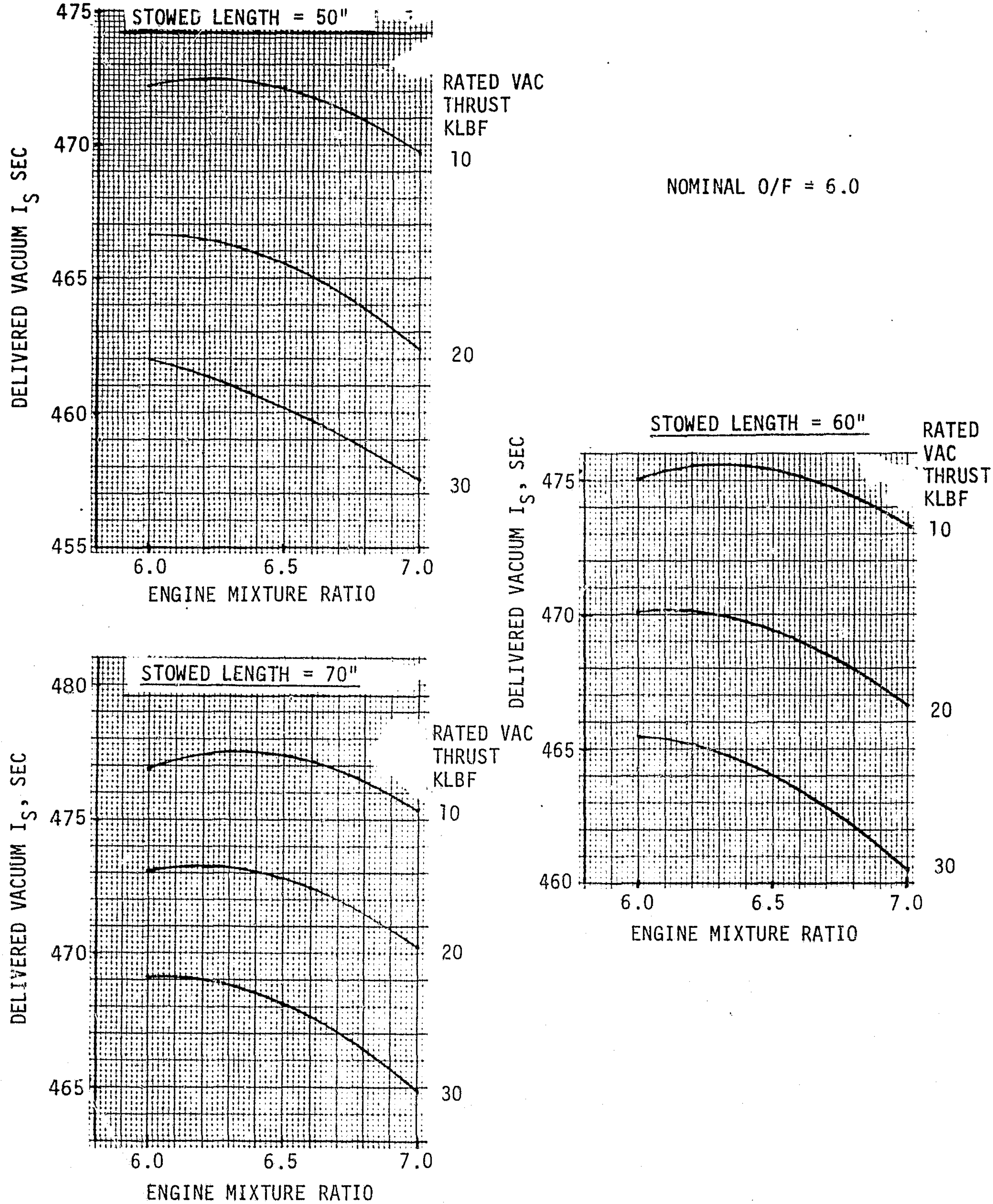


Figure 75. Advanced Expander Cycle Engine Performance at Design and Off-Design O/F

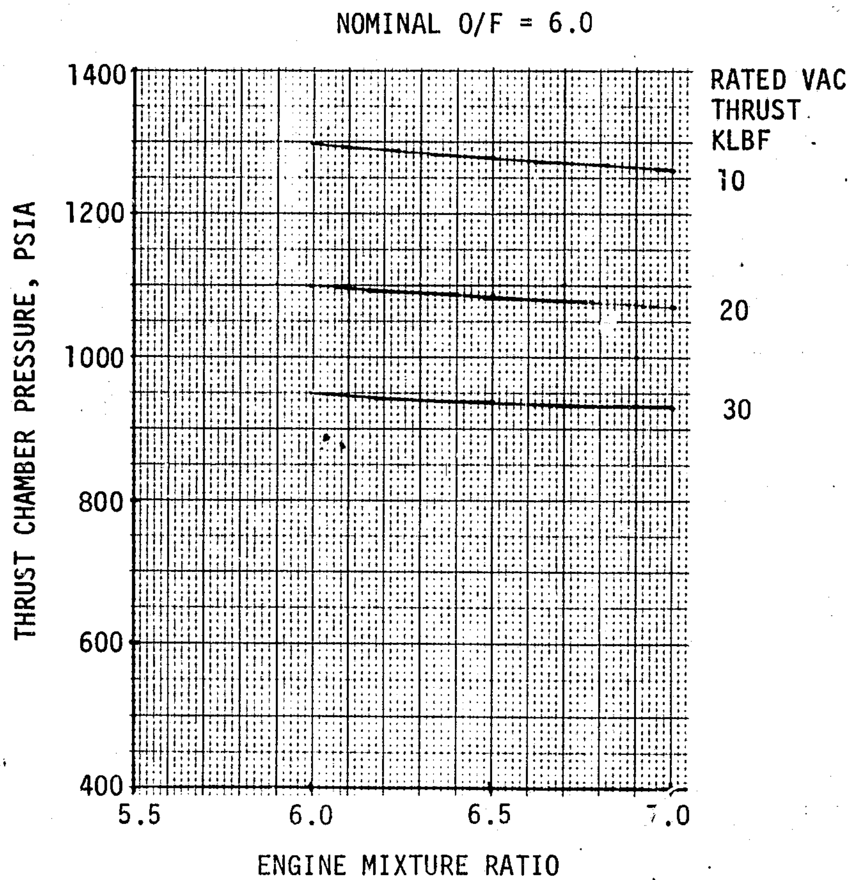


Figure 76. Advanced Expander Cycle Engine Thrust Chamber Pressure at Design and Off-Design O/F

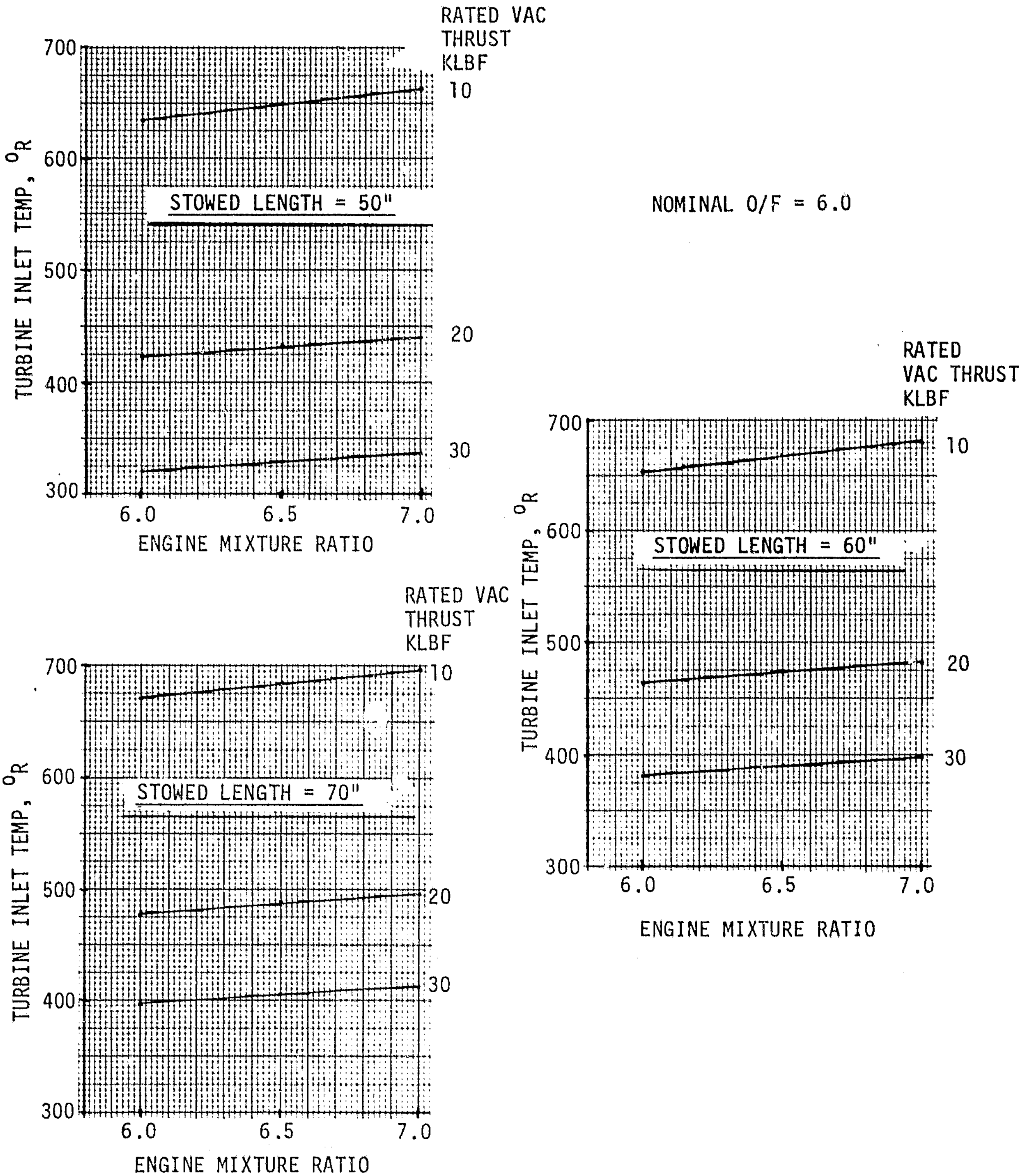


Figure 77. Advanced Expander Cycle Engine Turbine Inlet Temperature at Design and Off-Design O/F

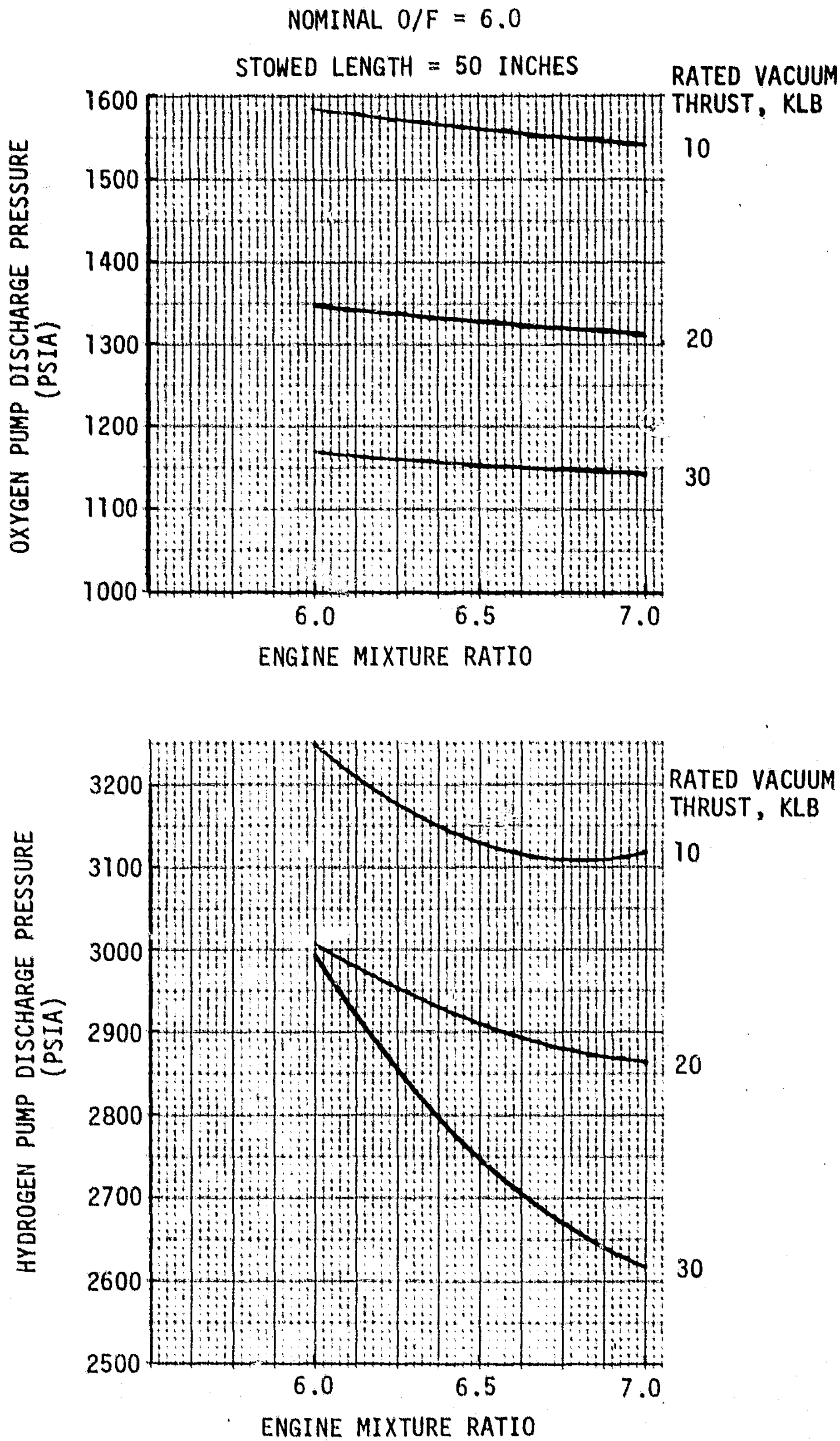


Figure 78. Advanced Expander Cycle Engine Pump Discharge Pressure Requirements at Design and Off-Design O/F

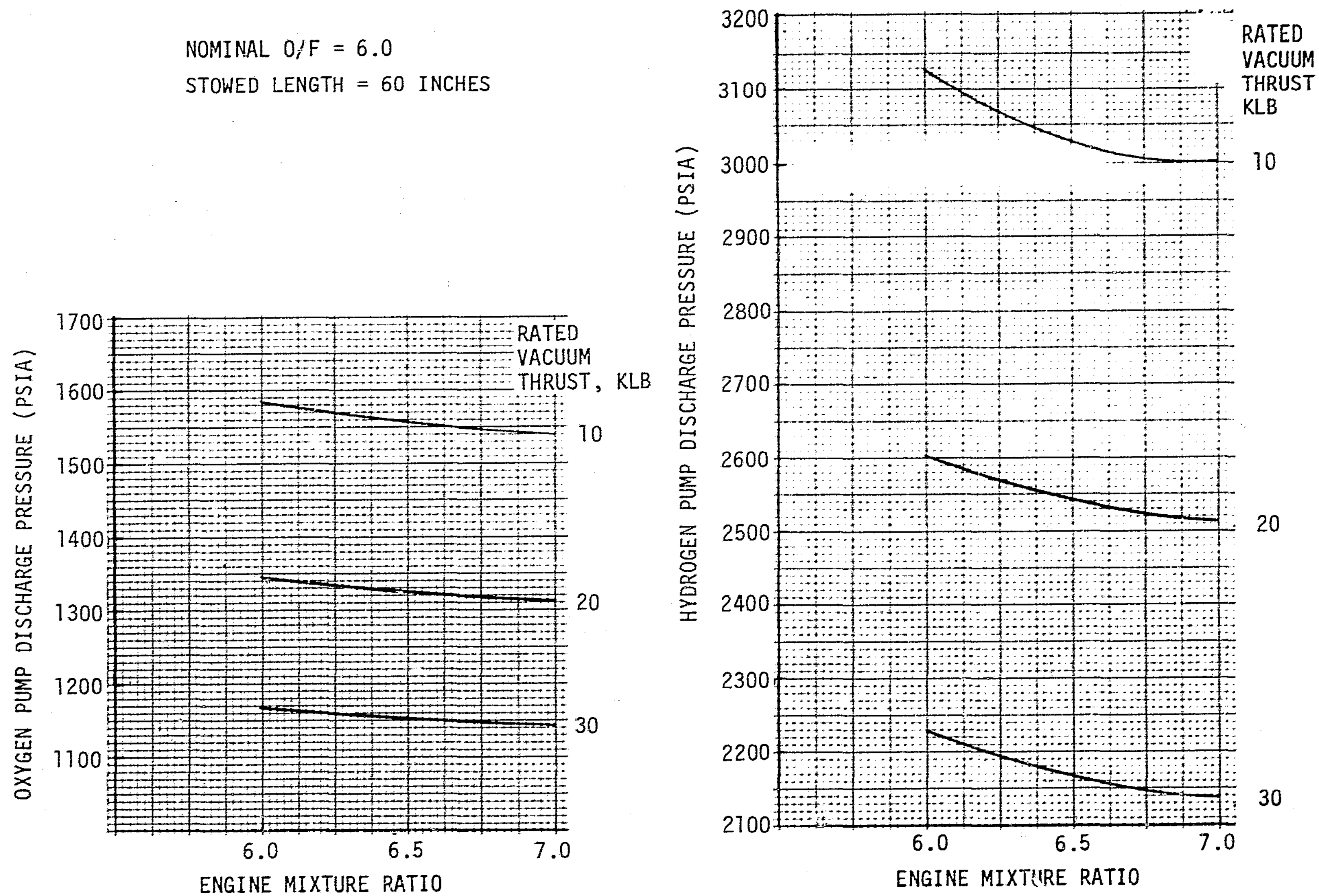


Figure 79. Advanced Expander Cycle Engine Pump Discharge Pressure Requirements at Design and Off-Design O/F

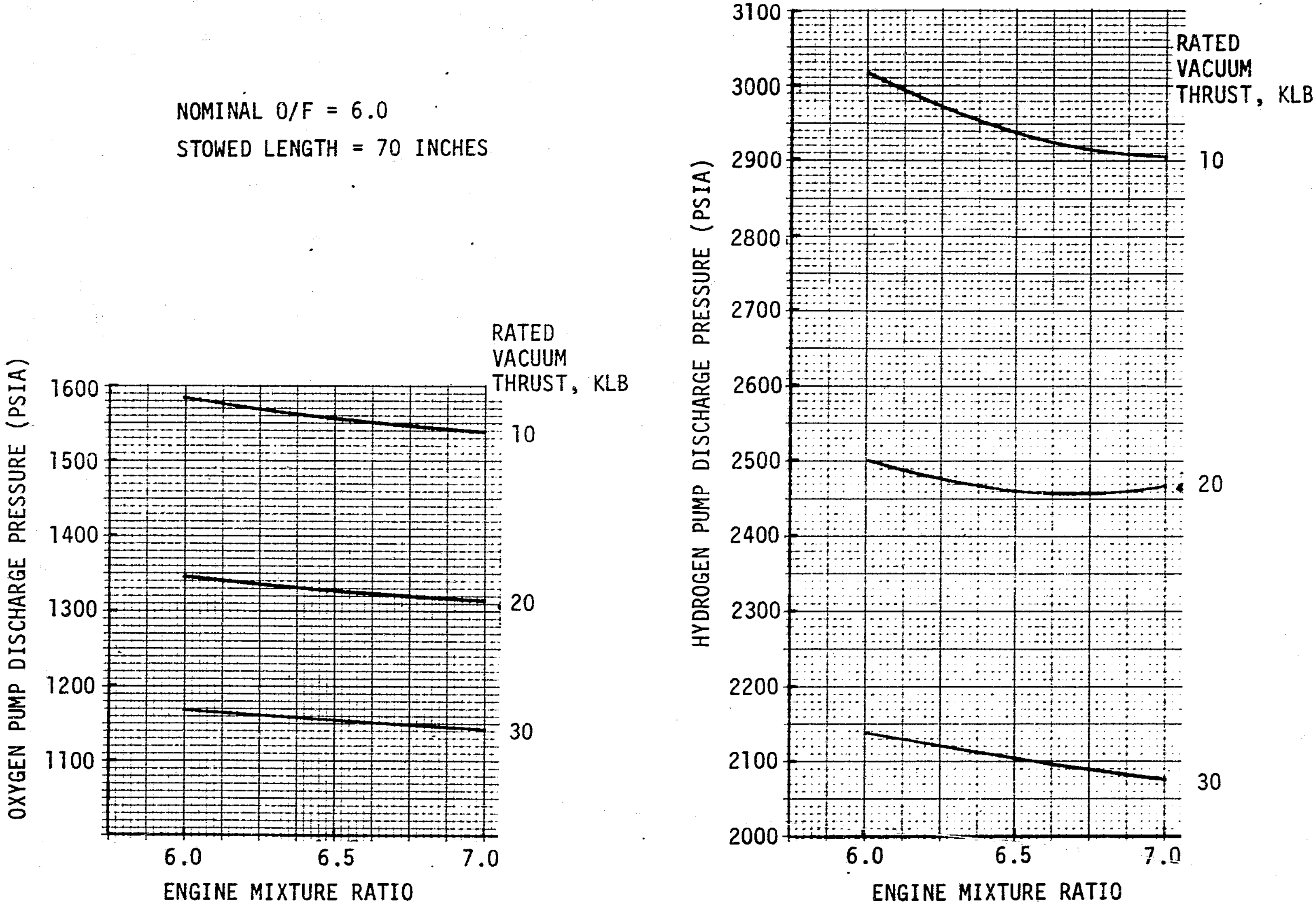


Figure 80. Advanced Expander Cycle Engine Pump Discharge Pressure Requirements at Design and Off-Design O/F

The engine boost pump inlet diameters and the change in engine weight with the net positive suction head requirement are shown on Figures 81 and 82, respectively. These figures are also applicable to the staged combustion and gas generator cycle engines. The boost pump inlet diameter variation was calculated using the inlet velocity guidelines presented in Section V,D. The change in engine weight with NPSH represents the changes in the individual boost pump weights. The rate of change in weight with NPSH is very large at NPSH values below the nominals. A requirement for lower NPSH values would require significant boost pump design definition effort.

2. <u>Staged Combustion Cycle</u>

The staged combustion cycle analyzed in this task is the same as that described in Section IV,A and shown on Figure 16.

The operating chamber pressure and nozzle area ratios established as a function of rated thrust for the staged combustion cycle are shown on Figure 83. The staged cycle area ratios are higher than the expander cycles because of the higher operating chamber pressures.

The variation of the percent bell nozzle length and the nozzle exit diameter with rated vacuum thrust and engine length are shown on Figure 84. The nozzle exit diameters are slightly larger than the expander cycle engines because of both the increase in area ratio and attendant increase in thrust coefficient.

The engine weight and delivered performance for the staged combustion cycle engine are shown on Figure 85. The staged combustion cycle engine delivers approximately 2 sec greater specific impulse at 10K lb thrust

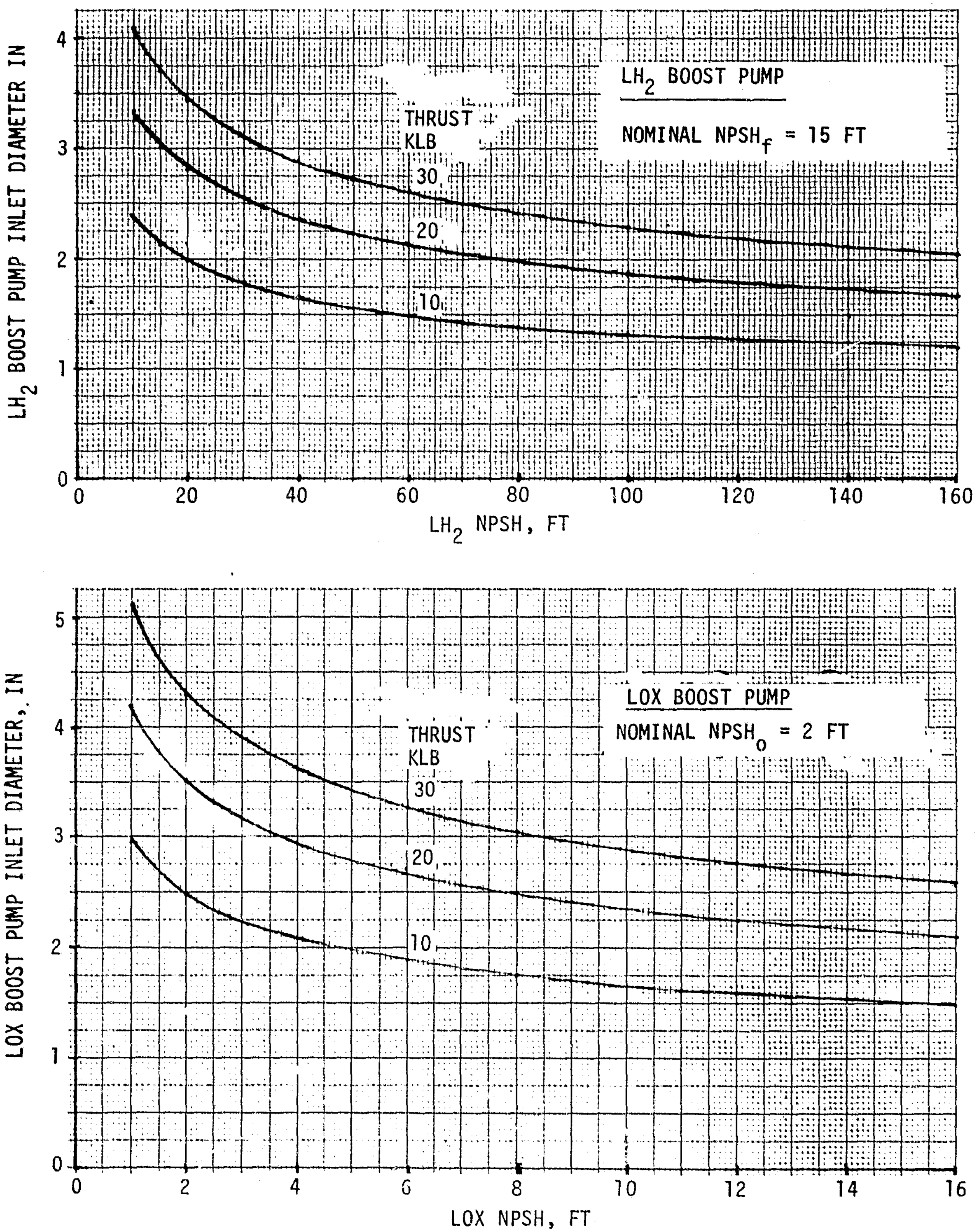


Figure 81. Effect of NPSH and Thrust Upon Boost Pump Inlet Diameters

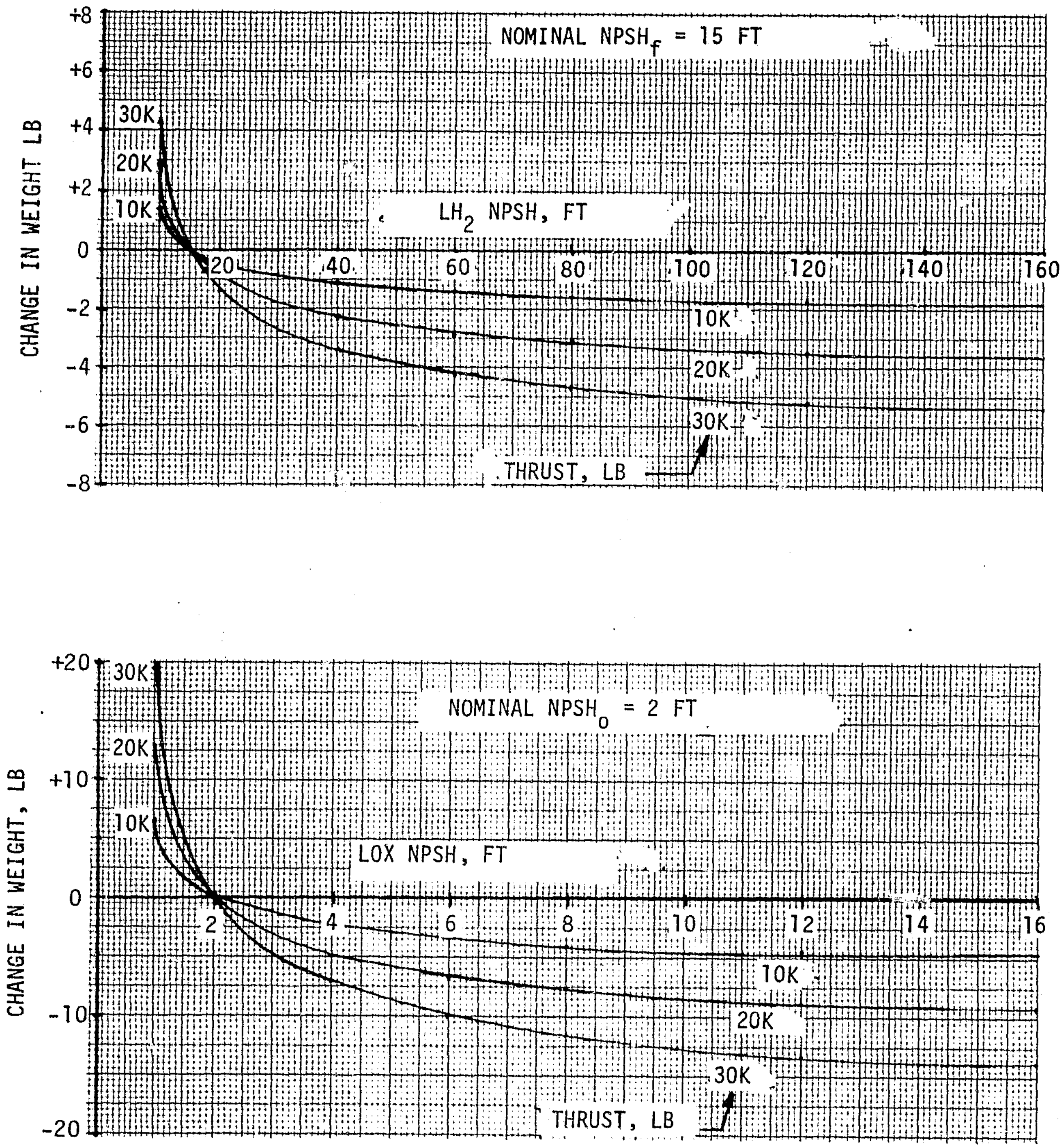


Figure 82. Change in Engine Weight with Net Positive Suction Head (NPSH)

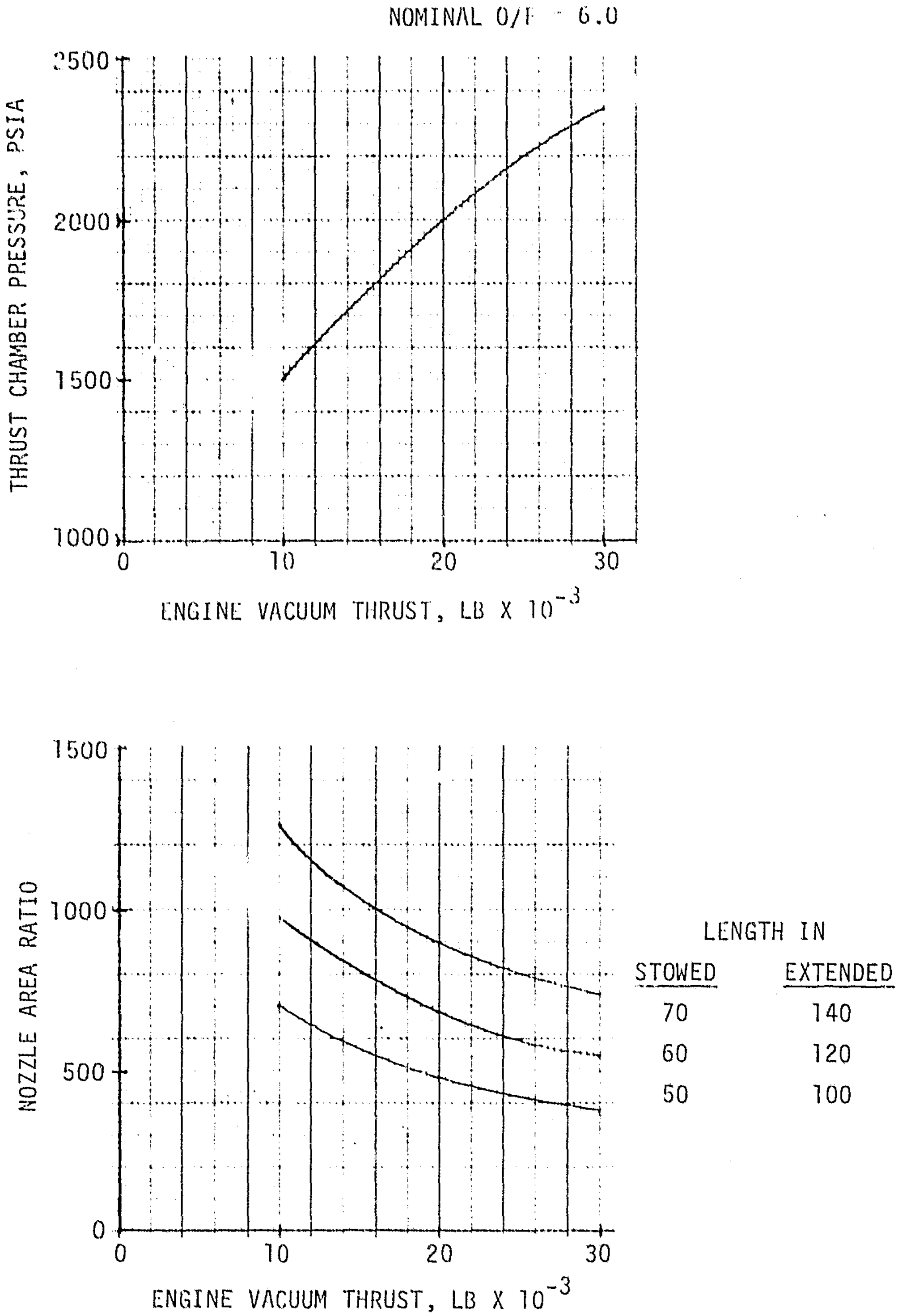
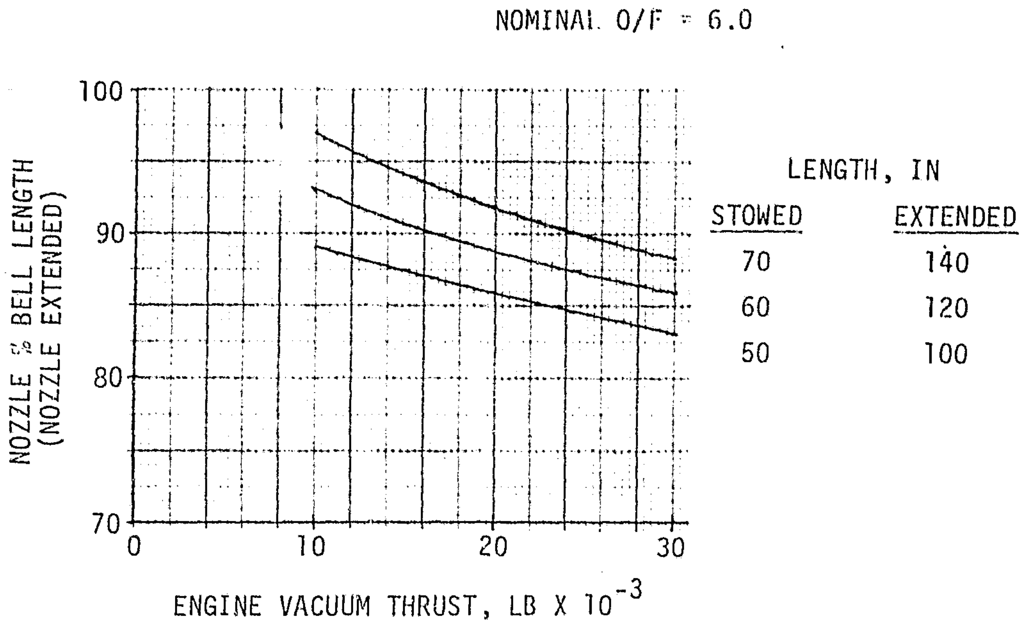


Figure 83. Staged Combustion Cycle Engine Thrust Chamber Pressure and Nozzle Area Ratio Variations with Rated Thrust



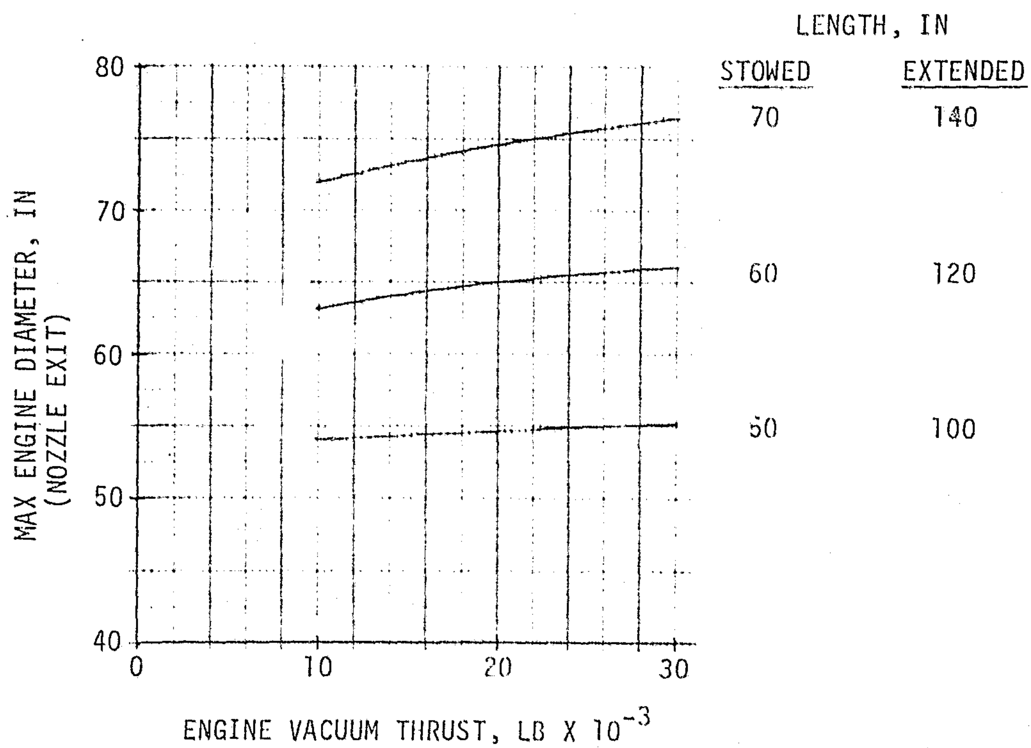


Figure 84. Staged Combustion Cycle Engine % Bell Length and Diameter Variations with Rated Thrust

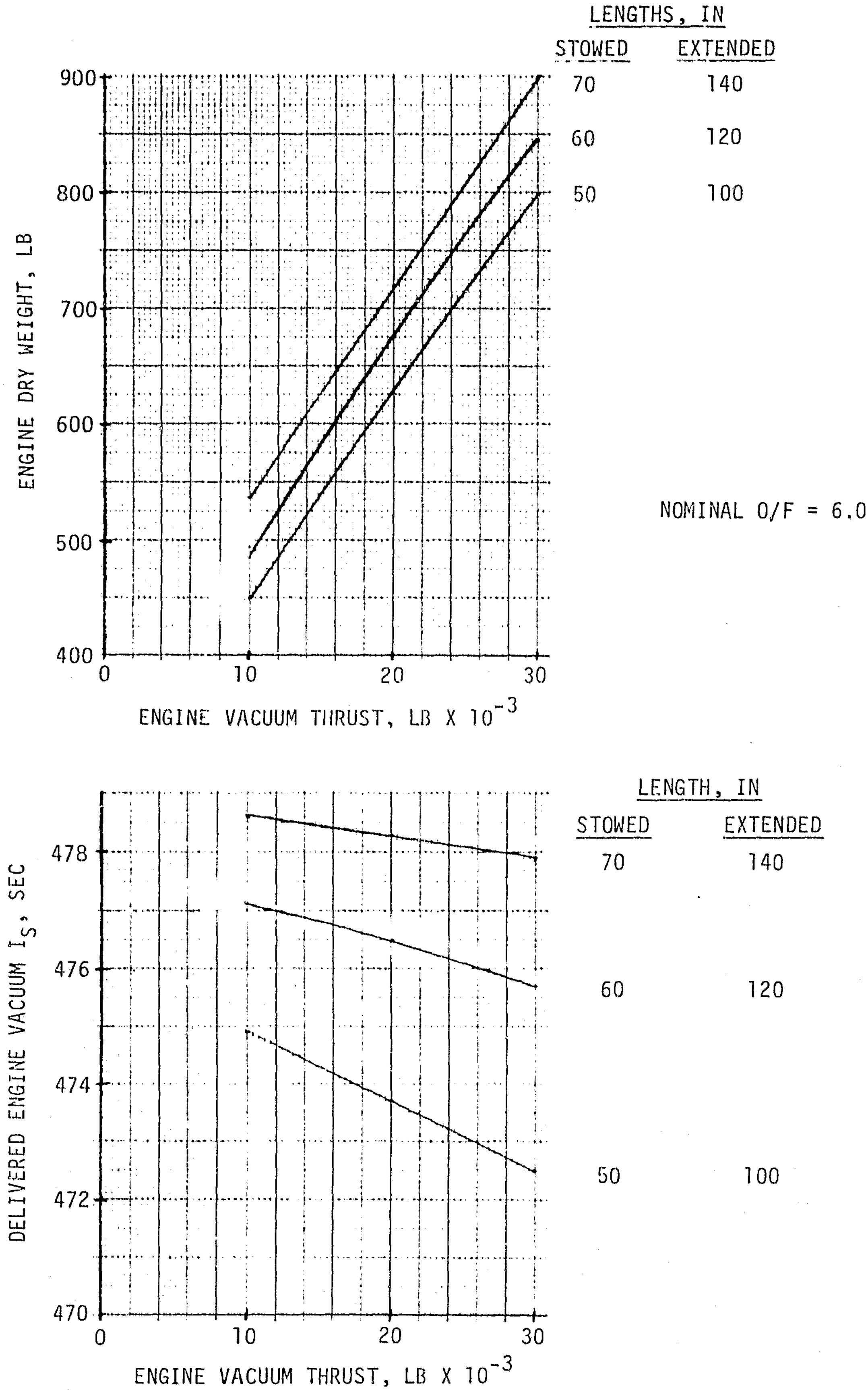


Figure 85. Staged Combustion Cycle Engine Weight and Performance Variations with Thrust

than the expander cycle engine but it also weighs more. Its performance advantage is improved at higher thrust levels because of the higher operating chamber pressures. The weight is higher than the expander cycle engine because of the higher component operating pressures and additional components.

The staged combustion cycle engine performance and critical operating parameters such as, thrust chamber pressure, turbine inlet temperature and pump discharge pressures are shown over a mixture ratio range from 6.0:1 to 7.0:1 on Figures 86, 87, 88 and 89.

Figure 86 shows that peak performance occurs between 6.0 and 6.5 O/F because of the extremely high area ratios obtained. The change of I_S with increasing thrust is not as great as that shown for the expander cycle engine because chamber pressure increases with increasing thrust for the staged combustion cycle. (The expander cycle chamber pressure decreases with thrust increases.) This means that the staged cycle area ratios remain high over the total thrust range.

The pump discharge pressure requirements for the staged combustion cycle engine do not vary with the stowed engine length because the turbine inlet temperature is not dependent upon length, as is the case for the expander cycle. The turbine inlet temperature does increase slightly with increasing mixture ratio. This could make off-design O/F operation of the staged combustion cycle critical in terms of turbine life.

3. Gas Generator Cycle

The gas generator cycle engine analyzed in this task was described in Section IV,A and depicted schematically on Figure 18.

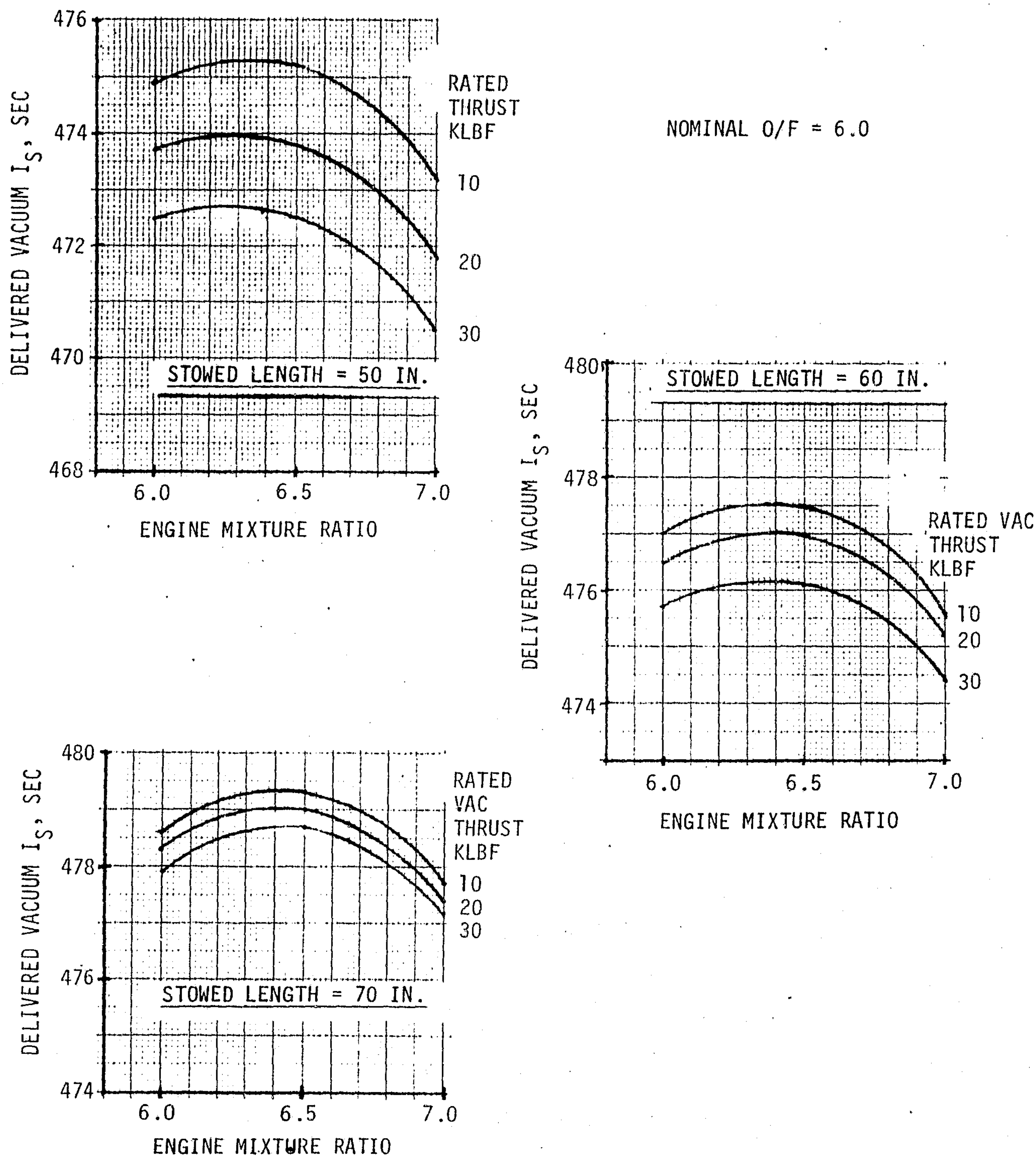


Figure 86. Staged Combustion Cycle Engine Performance at Design and Off-Design O/F

NOMINAL O/F = 6.0

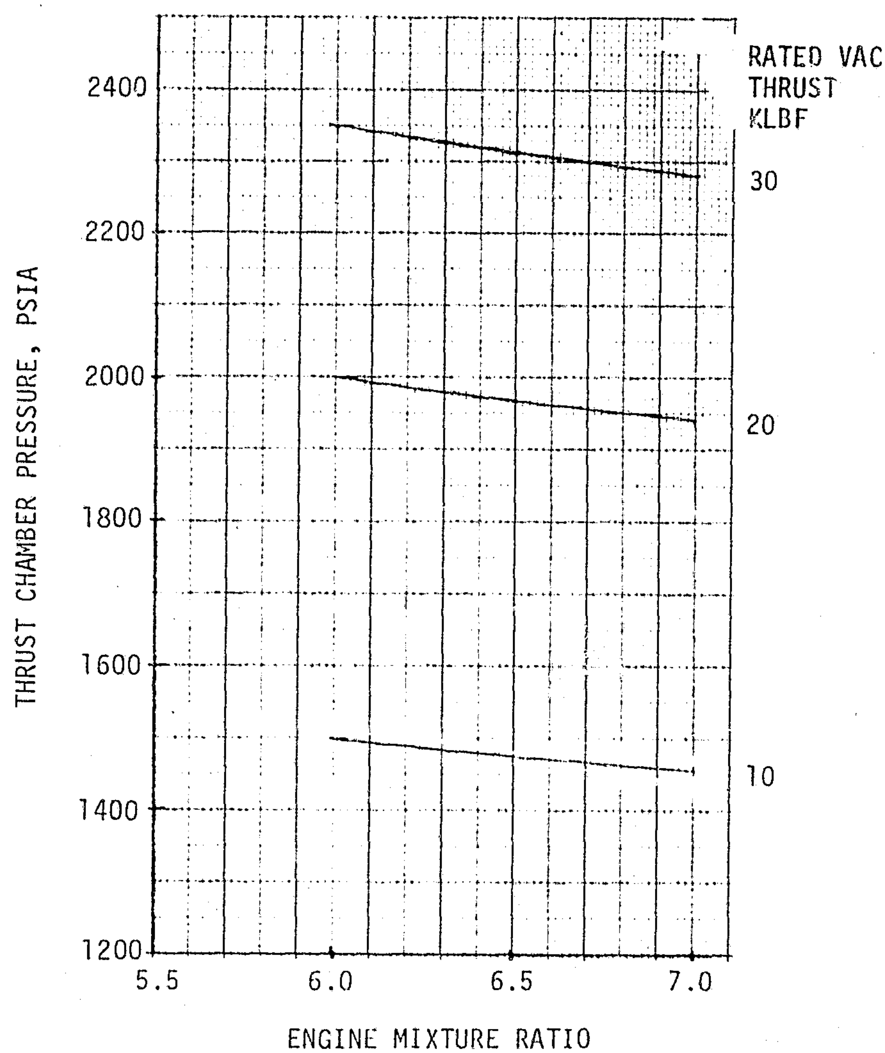


Figure 87. Staged Combustion Cycle Engine Thrust Chamber Pressure at Design and Off-Design O/F

NOMINAL O/F = 6.0

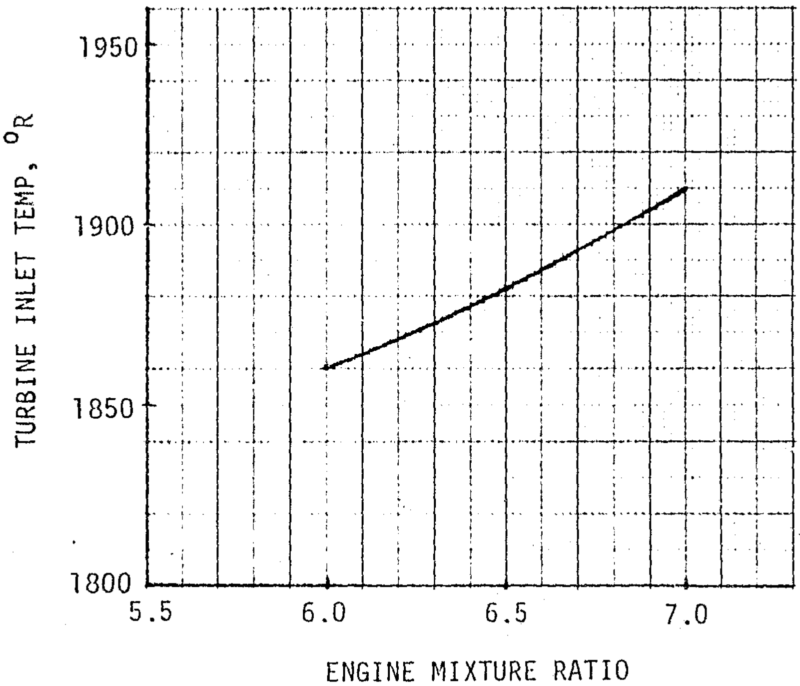


Figure 88. Staged Combustion Cycle Engine Turbine Inlet Temperature Requirements at Design and Off-Design O/F

C-3

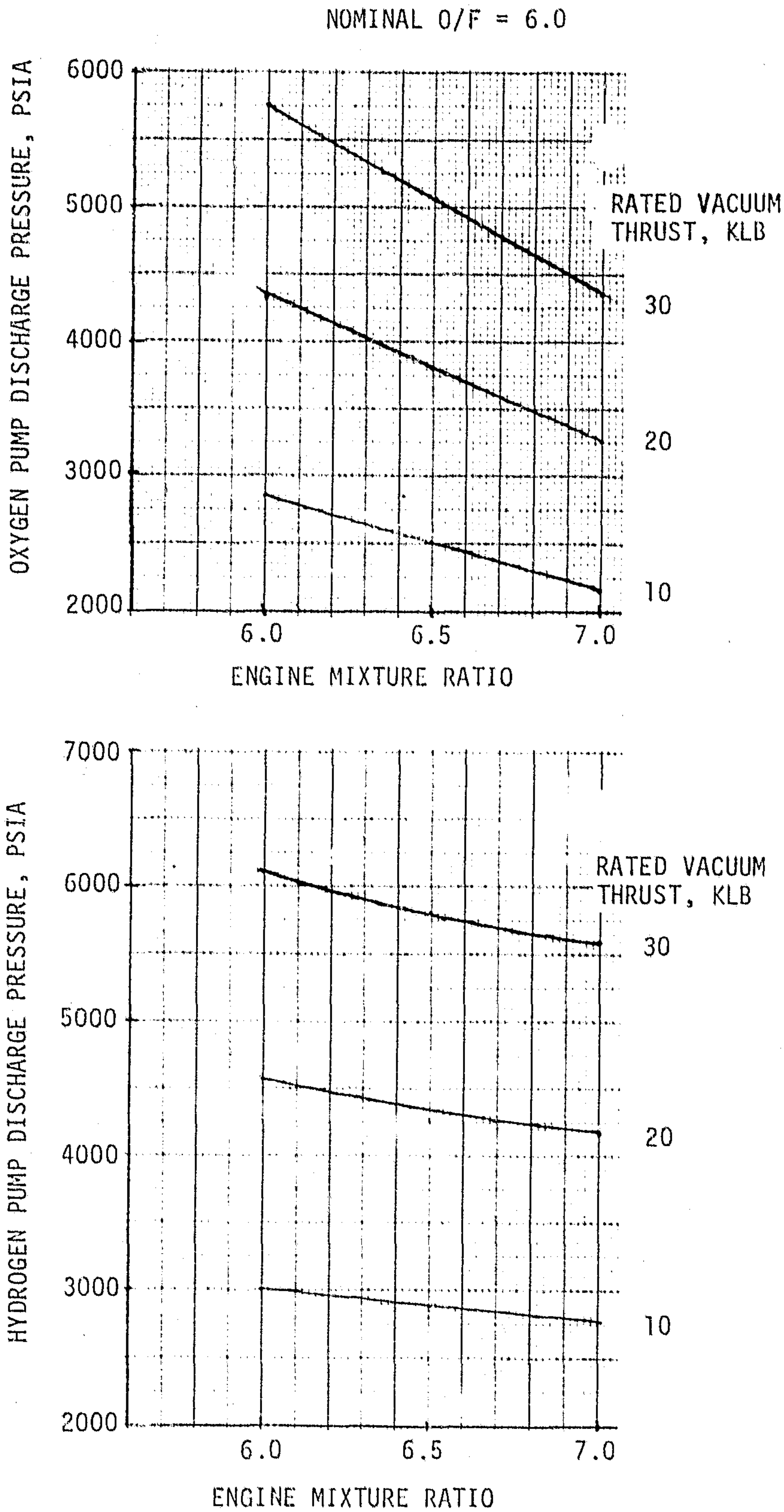
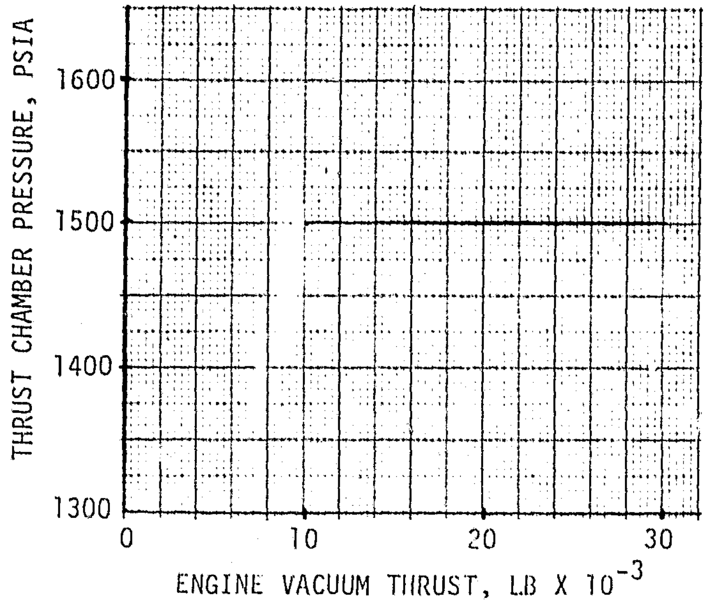


Figure 89. Staged Combustion Cycle Engine Pump Discharge Pressure Requirements at Design and Off-Design O/F

The operating chamber pressure for this engine is 1500 psia over the 10K to 30K thrust range. The nozzle percent bell length is shown as a function of the rated thrust on Figure 90 for a stowed length of 60 in. The nozzle area ratios and nozzle exit diameters are presented on Figure 91. The resulting performance and engine weight data are shown on Figure 92. The engine weight is slightly heavier than the expander cycle because of additional components and higher operating pressures. The performance is significantly lower than either the expander or staged combustion cycles because of the turbine exhaust loss.

Off-design operation over a range of engine mixture ratios from 6.0 to 7.0 is shown on Figures 93, 94 and 95. The turbine inlet temperature was held constant at 1860°R for the off-design calculations. Figure 93 shows that the peak performance again occurs at higher mixture ratios for the lower thrust engines. The thrust chamber mixture ratio shift caused by the turbine drive flow requirements is shown on Figure 94. The pump discharge pressure requirements are not significantly affected by the design thrust level, as shown by Figure 95, because this cycle is not power balance sensitive. Fuel pump discharge pressure requirements decrease slightly with increasing thrust becuase the higher thrust engines are easier to cool (i.e., less coolant jacket pressure drop).

NOMINAL O/F = 6.0



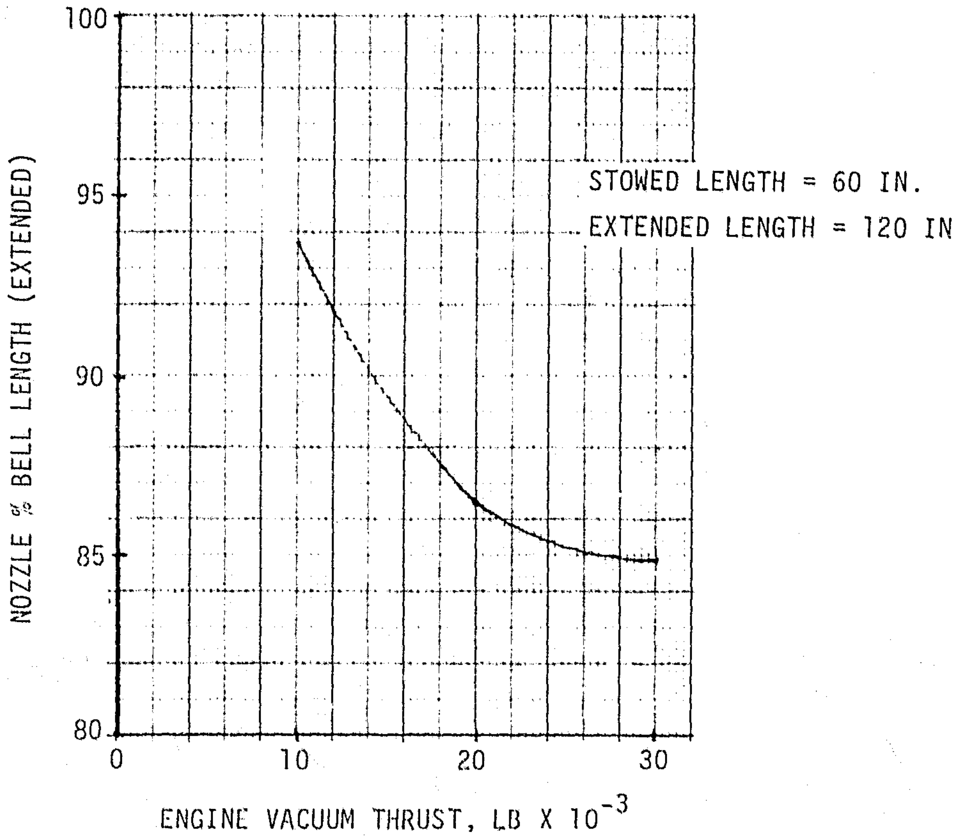
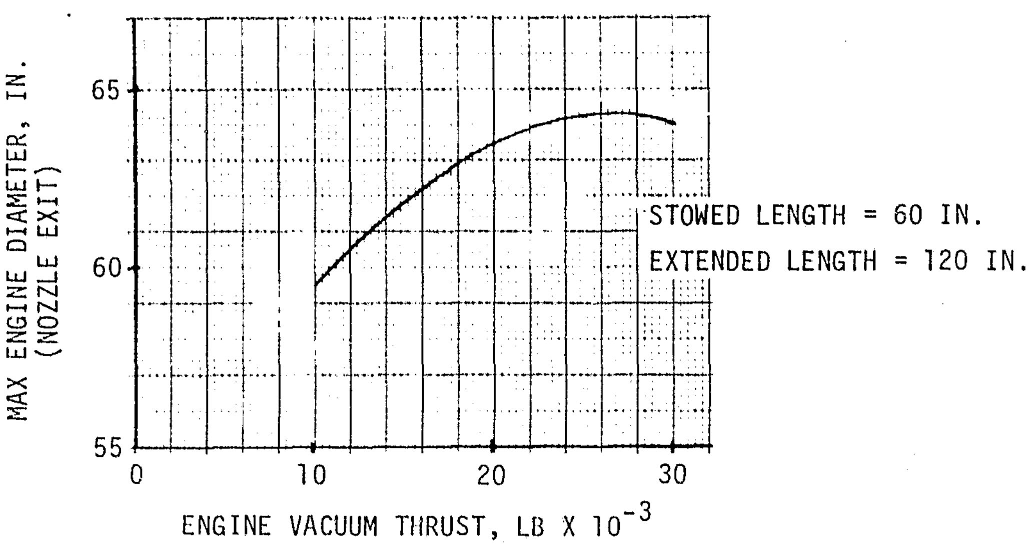


Figure 90. Gas Generator Cycle Engine Chamber Pressure and % Bell Length Variations with Rated Vacuum Thrust

NOMINAL O/F = 6.0



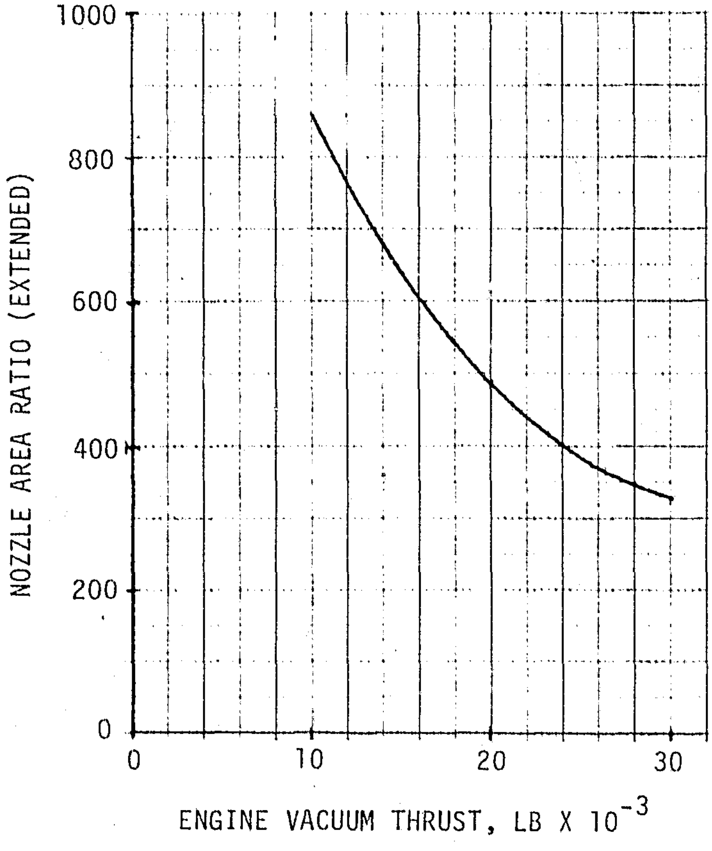


Figure 91. Gas Generator Cycle Engine Diameter and Nozzle Area Ratio Variations with Rated Vacuum Thrust

NOMINAL O/F = 6.0

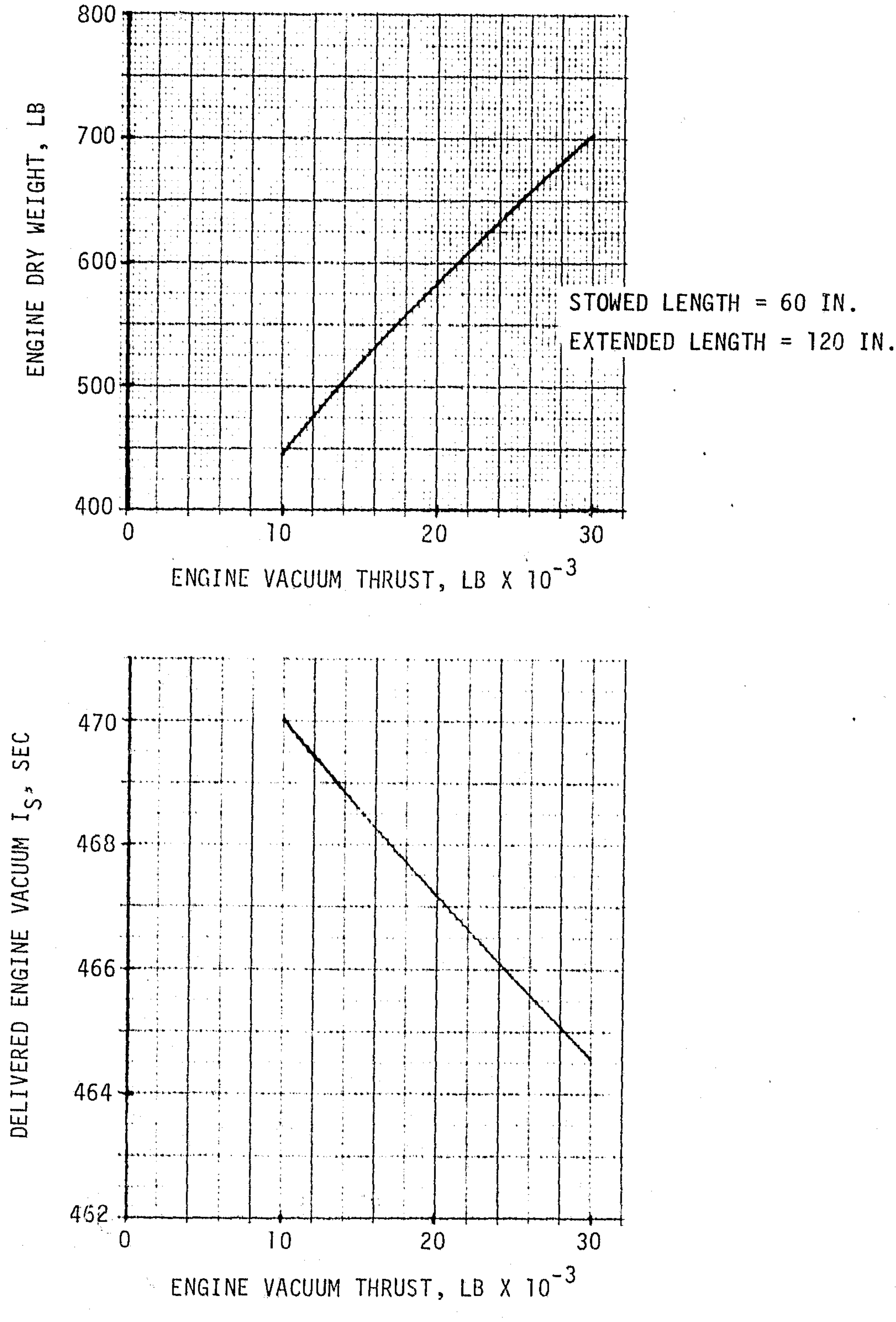


Figure 92. Gas Generator Cycle Engine Weight and Performance Variations with Rated Thrust

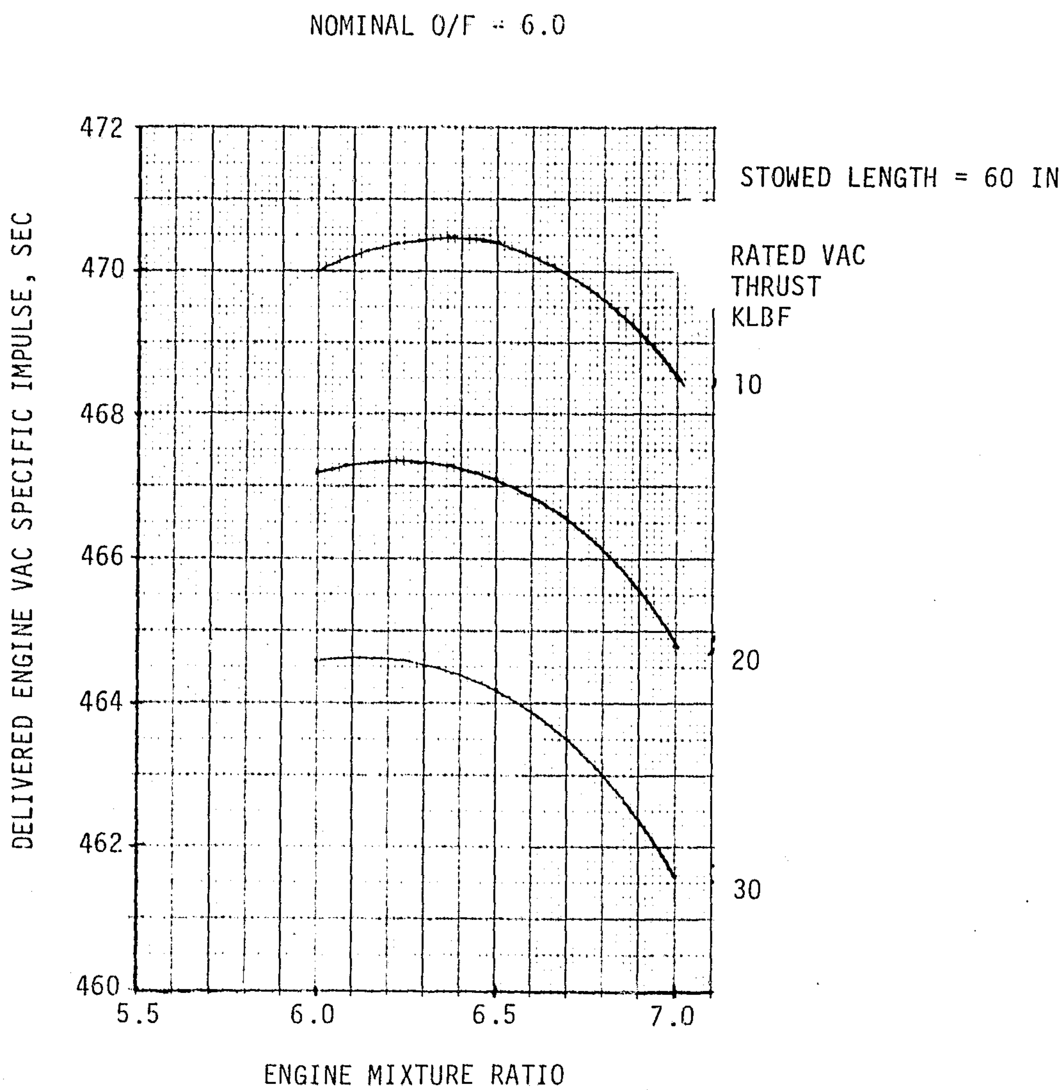
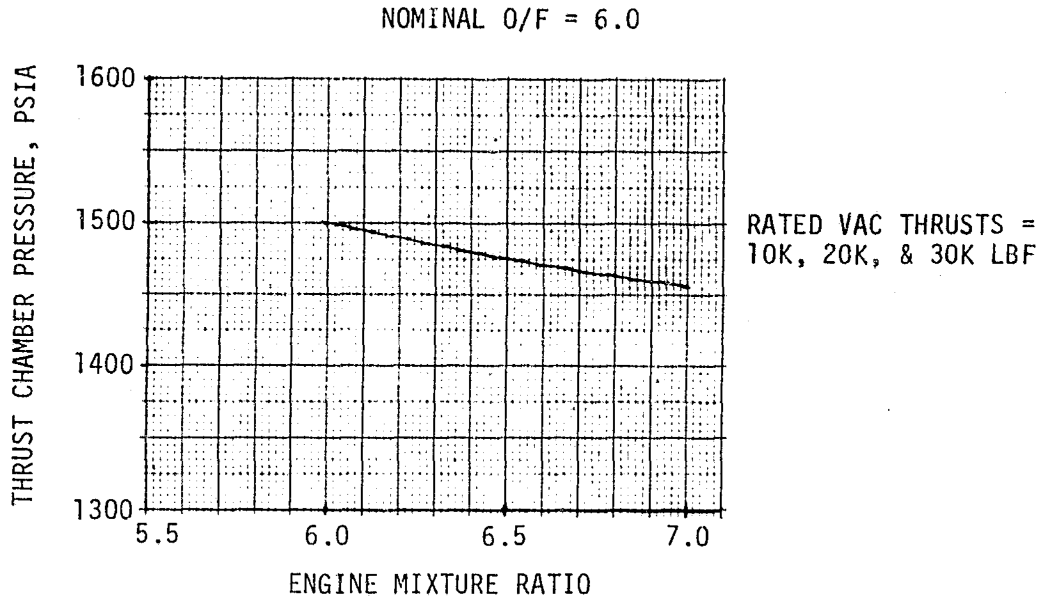


Figure 93. Gas Generator Cycle Engine Performance at Design and Off-Design O/F



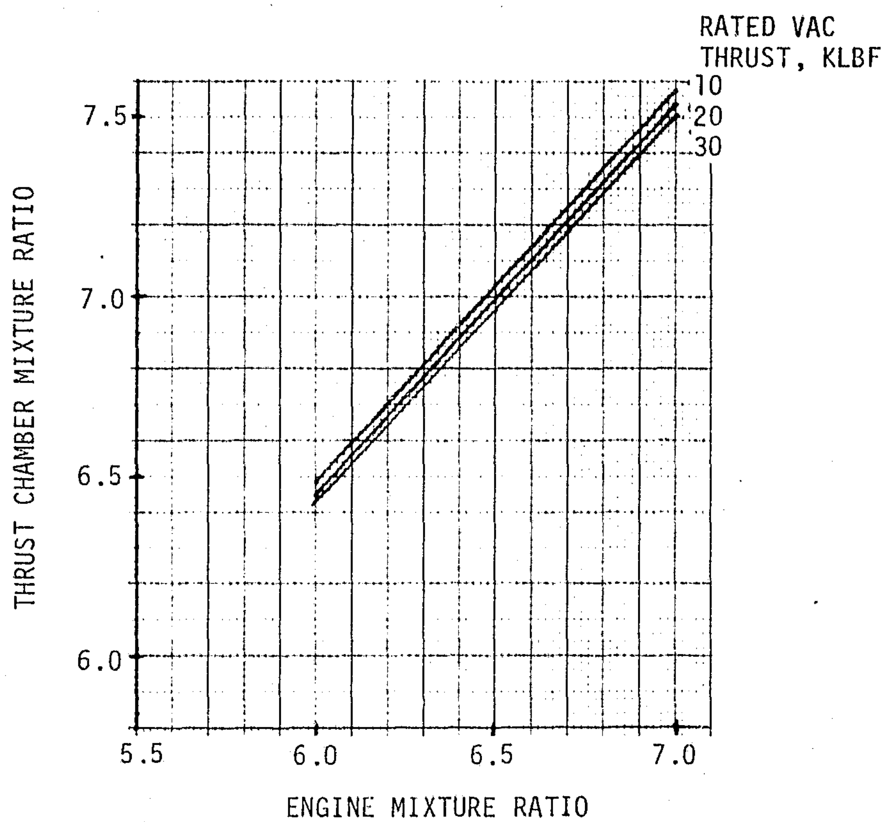
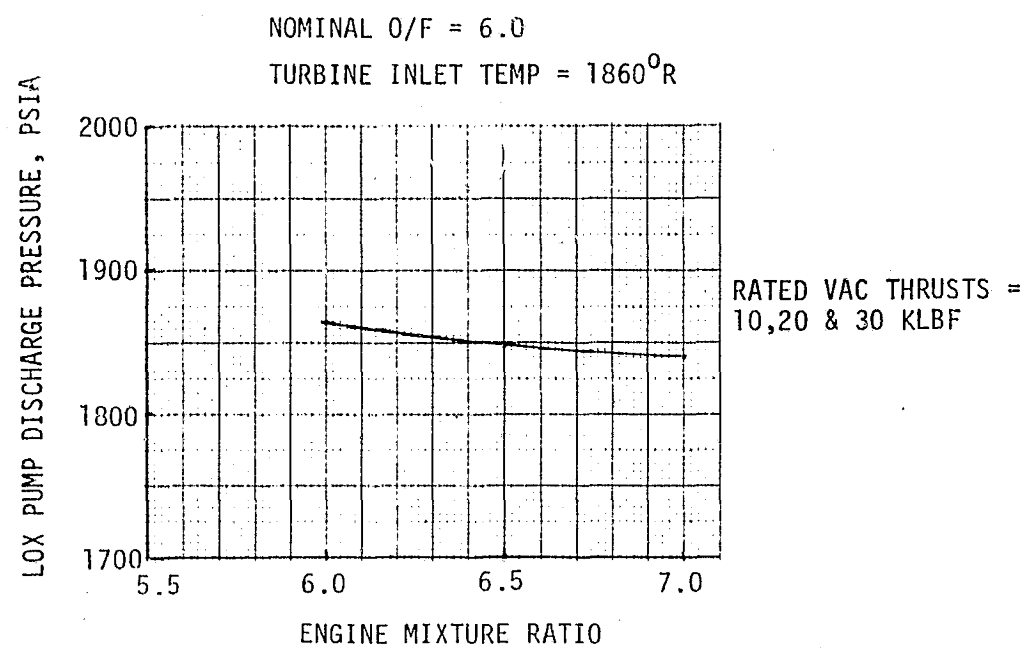


Figure 94. Gas Generator Cycle Engine Thrust Chamber Pressure and Thrust Chamber Mixture Ratio at Design and Off-Design O/F



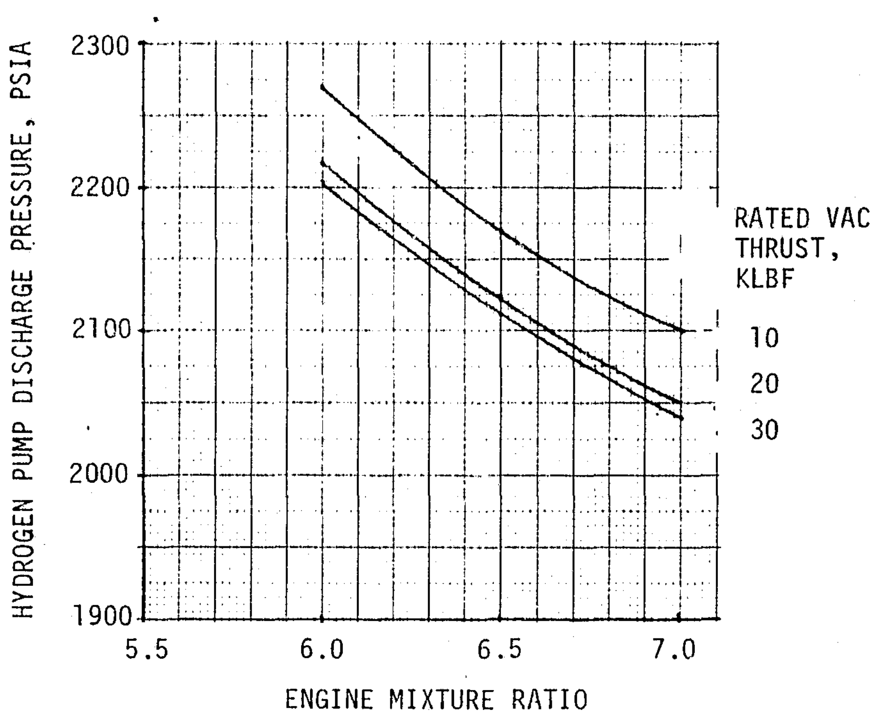


Figure 95. Gas Generator Cycle Engine Pump Discharge Pressure Requirements at Design and Off-Design O/F

VI. TASK IV - ENGINE OFF - DESIGN OPERATION

The objective of this task was to evaluate the impact of requirements for operation at minimum thrust levels (tank-head idle and pumped idle) and for one-time emergency operation at a mixture ratio of 10.0:1. These evaluations were performed for the recommended 10K lb thrust advanced expander cycle engine.

A. TANK-HEAD IDLE MODE

The tank-head idle mode is a pressure fed mode of operation with saturated propellants in the tanks. Its purpose is to thermally condition the engine without nonpropulsive dumping of the propellants. In operation, the turbine control valve would be closed and the turbine by-pass valve would be fully opened. 100% of the hydrogen flow by-passes the turbines so that the pumps are not rotating. The OOS and RL-10 Derivative studies have shown the desirability of placing an oxygen heat exchanger in the turbine by-pass line. This gasifies the LOX during tank-head operation. This heat exchanger is also necessary to provide the gaseous oxygen for LOX tank pressurization in the other modes of operation. The estimated engine performance during this mode of operation is shown on Table XXIX. JANNAF procedures were again used to calculate the performance. Both the kinetics and boundary layer losses for the operating conditions are extremely high. The performance loss summary follows:

ODE Specific Impulse, sec	481.0	(O/F = 4.0)
Boundary Layer Loss, sec	-48.0	
Kinetics Loss, sec	-29.0	
Divergence Loss, sec	-2.9	
ERL Loss, sec	-2.5	
Delivered I_S, sec	398.6	

TABLE XXIX

ADVANCED EXPANDER CYCLE ENGINE TANK-HEAD IDLE MODE SUMMARY

Thrust, lb	37.3
Chamber Pressure, psia	6.0
Mixture Ratio	4.0
Vacuum Specific Impulse, sec	399.0
Flow Rate, lb/sec	0.0935
Hydrogen Flow, lb/sec	0.0187
Oxygen Flow, lb/sec	0.0748

A requirement for the tank-head idle mode operation would increase the number of engine and component tests required during the DDT&E phase to verify performance and operation in this mode. It is estimated that this would increase the DDT&E cost by approximately 0.5% and stretch out the program by about 2 months.

B. PUMPED IDLE MODE

The pumped-idle mode is a pump-fed mode of operation at reduced thrust with initially saturated propellants in the tanks. A primary purpose is to provide gaseous oxygen and hydrogen for pressurization of the vehicle tanks to a level sufficient to permit acceleration of the engine to full thrust. All propellants are expended propulsively during this mode of operation. A thrust level of 25% of rated thrust was selected after review of the OOS, ASE and RL-10 Derivative studies which showed that lower thrust operation could result in chugging instability. In operation, approximately 50% of the hydrogen flow by-passes the turbines and heats the oxygen in a LOX heat exchanger. The estimated performance in this mode of operation is shown on Table XXX. Performance loss estimates using the simplified JANNAF procedures are:

ODE Specific Impulse, sec	490.7
Boundary Layer Loss, sec	-21.0
Kinetics Loss, sec	-7.0
Divergence Loss, sec	-2.9
Energy Release Loss, sec	-2.5
Delivered Performance, sec	457.3

The fuel autogenous flow can be tapped off from the chamber coolant exit manifolding. Potential locations for the oxygen heat exchanger are; (1) the turbine by-pass flow line, (2) the turbine exhaust, and (3) the

TABLE XXX

ADVANCED EXPANDER CYCLE ENGINE PUMPED IDLE MODE SUMMARY

Thrust, lb	2500
Chamber Pressure, psia	334
Mixture Ratio	6.0
Vacuum Specific Impulse, sec	457.3
Flow Rate, lb/sec	5.47
Hydrogen Flow, lb/sec	0.78
Oxygen Flow, lb/sec	4.69

fixed nozzle. Further definition of the autogenous system is planned in point design studies.

The impact of this requirement upon the DDT&E cost and schedules is estimated to be approximately 6% and 12 months, respectively.

C. EMERGENCY ONE-TIME OPERATION

Engine cycle power balances and performance estimates were made for operating the 10K expander cycle engine at an O/F of 10.0. Thermal and turbomachinery analysis results previously discussed were used in these calculations. The results are summarized on Table XXXI. Pump discharge pressure for this operating mode are:

Fuel Pump Discharge Pressure = 2875 psia
Oxidizer Pump Discharge Pressure = 1537 psia

Calculated performance and losses are:

ODE Specific Impulse, sec	444.0
Boundary Layer Loss, sec	-6.8
Kinetics Loss, sec	-6.1
Divergence Loss, sec	-2.7
Energy Release Loss, sec	-2.2
Delivered Performance, sec	426.2

The delivered performance is very low because the ODE specific impulse is 47.3 seconds lower than that at a nominal O/F of 6.0.

TABLE XXXI

ADVANCED EXPANDER CYCLE ENGINE
EMERGENCY ONE-TIME OPERATION SUMMARY

(Stowed Length = 60 in.)

Thrust, lb	10,000
Chamber Pressure, psia	1,260
Mixture Ratio	10.0
Vacuum Specific Impulse	426.2
Flow Rate, lb/sec	23.46
Chamber Coolant Pressure Drop, psi	90.0
Turbine Inlet Temperature, °R	856
Pump Efficiencies	
Fuel Pump, %	60.4
Oxidizer Pump, %	59.5
Turbine Efficiencies	
Fuel TPA, %	66.0
Oxidizer TPA, %	39.3

Pertinent engine operating parameters for this mode of operation are shown as a function of stowed length on Figure 96.

The structural and thermal analysis results (Sections V,B and V,C) have shown that this mode of operation is less severe than operation at the nominal O/F. No degradation in the engine service life is anticipated.

The impact of this requirement upon the DDT&E cost and schedules is estimated to be approximately 3% and 6 months, respectively.

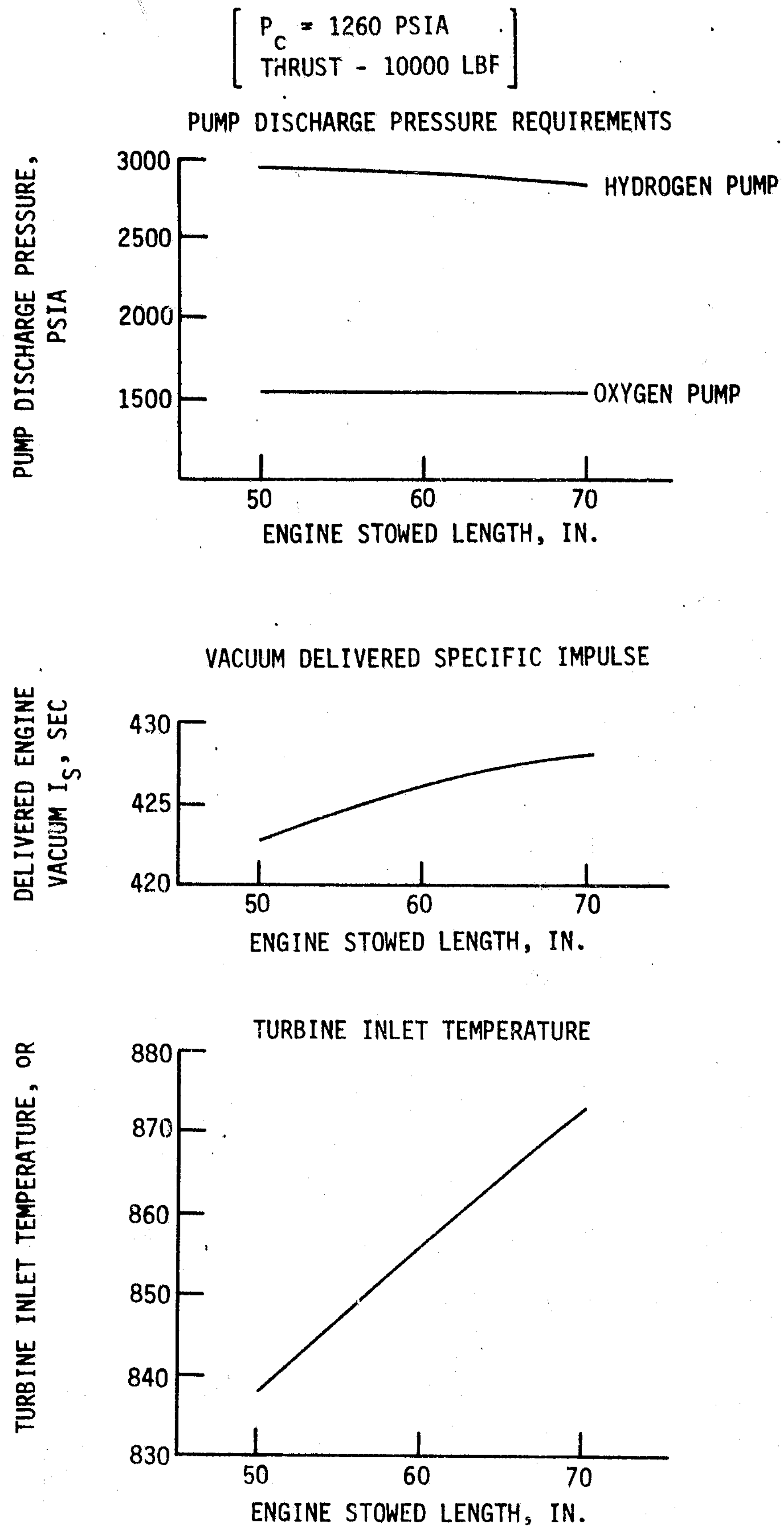


Figure 96. Advanced Expander Cycle Engine Emergency One-Time Operation (O/F = 10)

VII. TASK V - WORK BREAKDOWN STRUCTURE

The objective of this task was to establish a work breakdown structure (WBS) for use in the Task VII, Cost Estimates (Volume III). The major program elements are summarized on Figure 97. The WBS first level is the OTV main engine. The second WBS levels are DDT&E (Design, Development, Test and Evaluation), Production, and Operations.

The detailed WBS was structured in concert with NASA/MSFC and is shown on Table XXXII. Cost estimates were made to the fourth WBS level, summarized to the third level and spread over the program duration to the second level. The same WBS structure was used for each engine cycle candidate except that DDT&E item 1.1.3 is not applicable for an expander cycle engine. This WBS provided a consistent set of guidelines for cost estimation on each engine concept.

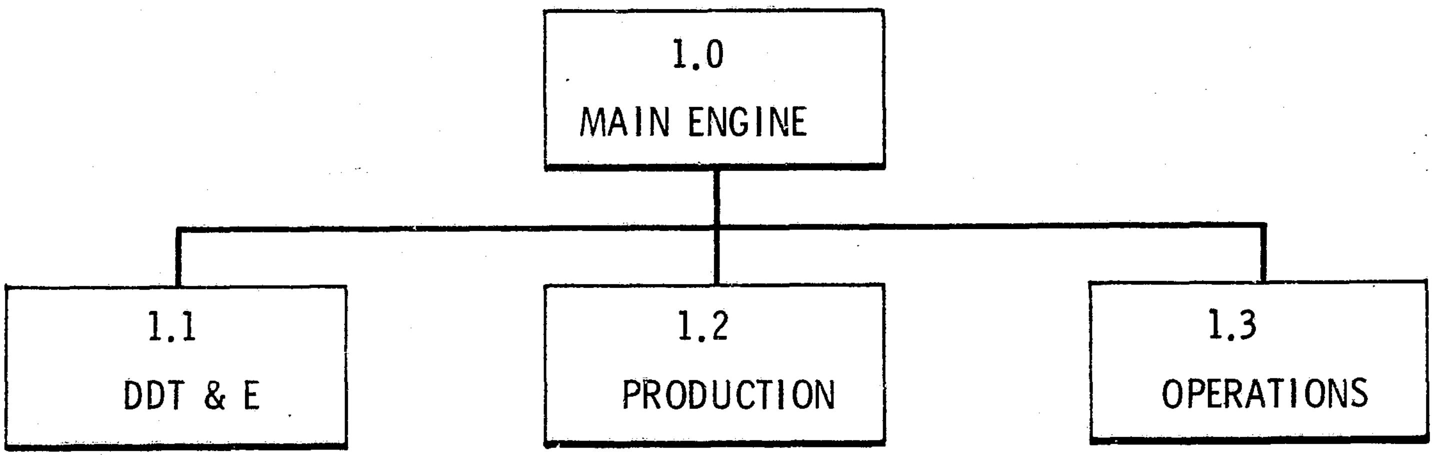


Figure 97. Work Breakdown Structure (WBS) Summary

TABLE XXXII

WORK BREAKDOWN STRUCTURE (WBS)

1.0	Main Engine
1.1	DDT&E
1.1.1	Turbomachinery
1.1.1.1	Main Fuel Pump
1.1.1.2	Main Oxidizer Pump
1.1.1.3	Fuel Boost Pump
1.1.1.4	Oxidizer Boost Pump
1.1.1.5	Assembly and Checkout
1.1.2	Main Combustion Chamber
1.1.2.1	Injector
1.1.2.2	Chamber
1.1.2.3	Upper Nozzle (fixed)
1.1.2.4	Igniter
1.1.2.5	Gimbal Assembly
1.1.2.6	Assembly and Checkout
1.1.3	Preburner/Gas Generator*
1.1.3.1	Injector
1.1.3.2	Combustor
1.1.3.3	Igniter
1.1.3.4	Assembly and Checkout
1.1.4	Nozzle Assembly
1.1.4.1	Lower Nozzle (Extendible)
1.1.4.2	Extension/Retraction Mechanisms
1.1.4.3	Assembly and Checkout
1.1.5	Controls
1.1.5.1	Engine Controller and Electrical Harness
1.1.5.2	Control Valves
1.1.5.3	Instrumentation and Electrical Harness
1.1.5.4	Assembly and Checkout
1.1.6	Pressurization
1.1.6.1	Heat Exchangers
1.1.6.2	Assembly and Checkout
1.1.7	Propellant Systems
1.1.7.1	Feed, Fill, Vent, Abort Dump, and Drain
1.1.7.2	Assembly and Checkout

*Staged Combustion/Gas Generator Cycles Only.

TABLE XXXII (cont.)

1.1.8	Initial Tooling
1.1.9	Ground Support Equipment
1.1.9.1	Handling and Protective Equipment
1.1.9.2	Checkout and Maintenance Equipment
1.1.9.3	Assembly and Checkout
1.1.10	Test
1.1.10.1	Development Testing
1.1.10.2	PFC Testing
1.1.10.3	FFC Testing
1.1.11	System Engineering and Integration
1.1.11.1	Integration of DDT&E Activities
1.1.11.2	Engine Assembly and Checkout
1.1.11.3	Engine/Vehicle Interface
1.1.12	Project Management
1.1.13	Facilities
1.1.14	Consumables
1.2	<u>Production</u>
1.2.1	Main Engines
1.2.1.1	Turbomachinery
1.2.1.2	Combustion Devices
1.2.1.3	Controls
1.2.1.4	Pressurization
1.2.1.5	Propellant Systems
1.2.1.6	Engine Assembly
1.2.2	Initial Spares
1.2.3	Facility Maintenance
1.2.3.1	Manufacturing and Test Facilities
1.2.3.2	Sustaining Tooling
1.2.3.3	GSE
1.2.4	Sustaining Engineering
1.2.5	Project Management
1.2.6	Consumables

TABLE XXXII (cont.)

1.3	Operations
1.3.1	Inplant Support
1.3.2	Field Support
1.3.2.1	Launch Support
1.3.2.2	Flight Support
1.3.2.3	Refurbishment and Maintenance
1.3.2.4	Checkout
1.3.3	Major Engine Overhaul
1.3.4	Facility Maintenance
1.3.5	Follow-on Spares
1.3.6	Project Management
1.3.7	Consumables

VIII. TASK VI - PROGRAMMATIC ANALYSIS AND PLANNING

The objectives of this task were to formulate preliminary project planning information, prepare schedules, conduct risk assessments and develop a post flight maintenance and refurbishment philosophy. This information was prepared for the recommended 10K lb thrust advanced expander cycle engine.

A. SCHEDULES AND PLANS

The OTV engine and vehicle development schedule provided by NASA/MSFC for the programmatic analysis and planning task is shown on Figure 98. Key milestones on the figure are the authority to proceed (ATP) date for the engine DDT&E phase on 1 January 1982 and the initial operating capability (IOC) date for the OTV on 31 December 1987. The overall schedule is, of course, subject to revision. However, this schedule was used as a baseline to conduct this study.

The overall engine schedule, shown on Figure 99, was structured to meet the NASA OTV development schedule. Prior to the engine development phase, the engine concept definition studies (Phase A), engine point design, critical technology and Phase B design efforts are planned. The engine DDT&E phase is 4-1/2 years. This is the maximum amount of time that appears to be available to meet the flight engine delivery dates and the vehicle IOC date which were derived from the previous figure. The first prototype flight engine, which is defined as a Pre-flight Certification (PFC) engine that can be reused, is to be delivered in the 3rd quarter of 1984. The first flight engine need date is 31 March 1985. This engine would incorporate modifications based upon PFC testing but would not have completed Final Flight Certification (FFC). The FFC date is 30 June 1986. The production program for the AMOTV would go through the final quarter of 1990. The OTV flight program, for planning purposes, is 10 years and carries through the last quarter of 1997.

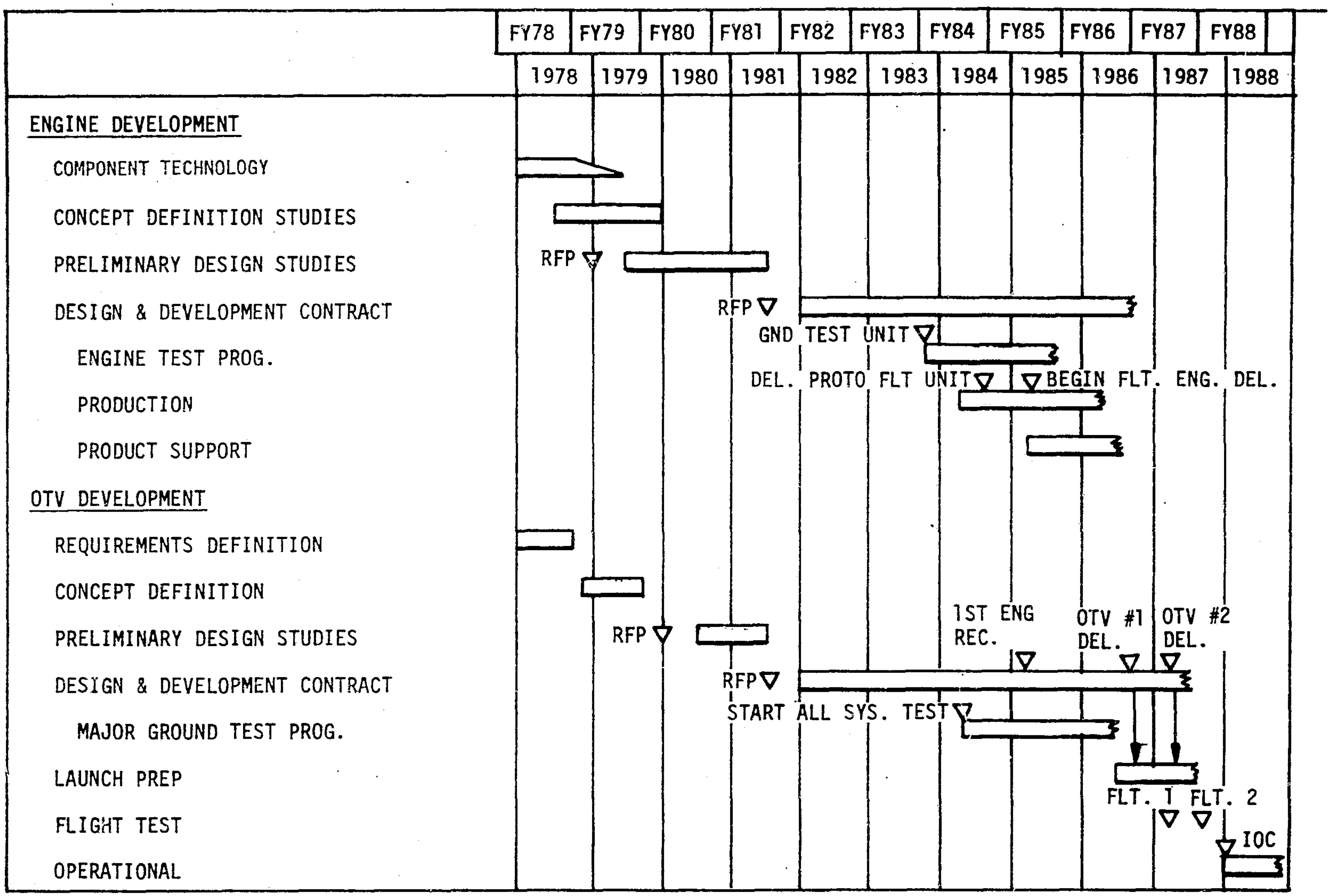


Figure 98. NASA Orbit Transfer Vehicle (OTV) Development Schedule

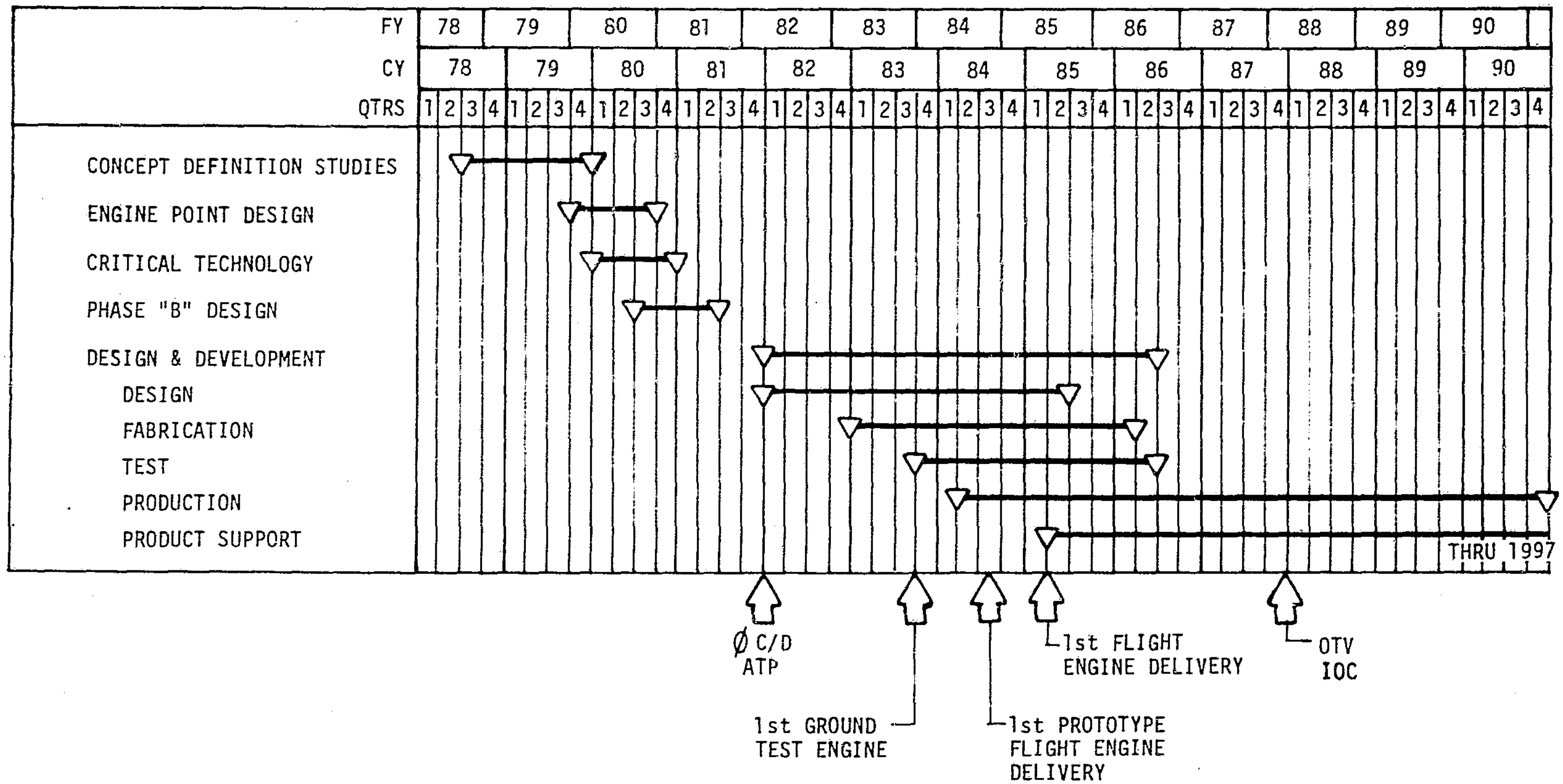


Figure 99. Overall OTV Engine Schedule

The engine DDT&E schedule is shown on Figure 100 for 4.5 years which was derived to meet the OTV schedule requirements. ATP is assumed as 1 January 1982. Major engine component design, fabrication and test are shown to the fourth WBS level. Component testing is scheduled for completion at the end of the second program year. The Initial Design Review (IDR) on the engine is scheduled prior to starting the ground test engine fabrication. Engine development testing is scheduled for completion mid-way through the second quarter of 1984, at which time the Preliminary Design Review (PDR) is scheduled. PFC testing is scheduled for completion on 31 March 1985 and the Critical Design Review (CDR) is scheduled to be held immediately thereafter. FFC testing is scheduled to be completed on 30 June 1986. This program is ambitious but can be accomplished if the technology programs precede ATP. It should also be noted that the program shown is success oriented.

A preliminary risk assessment of the candidate engine cycles was conducted in terms of potential problems that might occur during development and their impact upon the DDT&E cost and schedule. This analysis is summarized on Table XXXIII. The table shows that the staged combustion cycle has the highest development risk in terms of both cost and schedule. Two levels of risk on cost were evaluated. The first considered only the program uncertainty, neglecting differences in engine complexity in the comparisons. The second considered both the program uncertainties and the differences in engine complexity. Complexity was assumed to be proportional to the component pressure levels. The predicted change in DDT&E cost for the various cycle candidates is summarized on Table XXXIV. The probable cases were calculated as the square root of the sum of individual cost uncertainties squared. The maximum exposure cases were the sum of the individual uncertainties. The table shows that the expander cycle has potentially the lowest development cost risk. Further risk analyses are planned in the Phase A extension to this contract. To support this risk analysis, more detailed DDT&E schedules will be prepared to coincide with the lowest WBS level. This will also be accomplished in the extension to this contract.

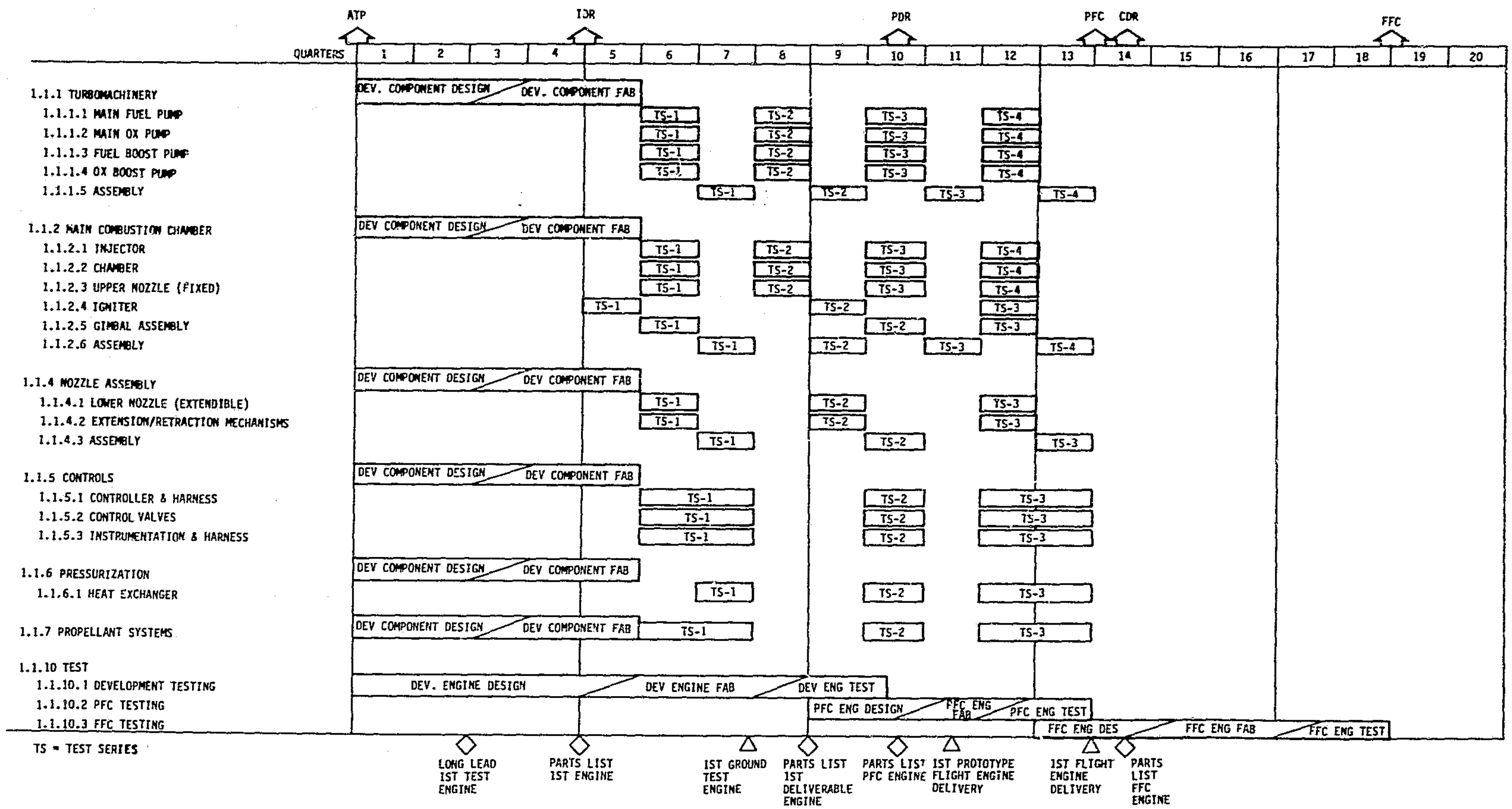


Figure 100. Advanced Expander Cycle Engine DDT&E Schedule

TABLE XXXIII

PROBLEM RECOVERY - DDT&E COST AND SCHEDULE IMPACT

PROBLEM	CYCLE	SCHEDULE SLIP MONTHS	COST INCREASE, MILLION $ (1)	(2)	ASSUMED SOLUTION
PREBURNER STABILITY	STAGED COMBUSTION	8	21	34	RESONATOR RETUNE
	EXPANDER	0	0	0	
	GAS GENERATOR	8	18	18	
MAIN BURNER STABILITY	S. C.	8	14	20	RESONATOR RETUNE
	EXPANDER	8	12	12	
	G.G.	8	12	15	
PREBURNER GAS UNIFORMITY	S. C.	10	26	42	INJECTOR REDESIGN
	EXPANDER	0	0	0	
	G.G.	10	22	22	
MAIN BURNER LOW PERFORMANCE	S.C.	7	12	17	INJECTOR ELEMENT MODIFICATION
	EXPANDER	7	10	10	
	G.G.	7	10	12	
LOW TURBINE POWER	S. C.	0	0	0	LONGER CHAMBER
	EXPANDER	9	14	14	
	G.G.	0	0	0	
LOW TURBOPUMP EFFICIENCY	S.C.	10	14	19	REDESIGN IMPELLER
	EXPANDER	10	13	13	
	G.G.	10	13	13	

(1) Program Technical Uncertainty Considered.
(2) Engine Complexity Considered

TABLE XXXIII (cont.)

PROBLEM	CYCLE	SCHEDULE SLIP MONTHS	COST INCREASE, MILLION $ (1)	(2)	ASSUMED SOLUTION
CYCLE BALANCE LOW POWER	S.C.	17	24	44	REDESIGN PUMP
	EXPANDER	14	16	16	REDESIGN PUMP
	G.G.	0	0	0	MORE FLOW
TURBINE CYCLE LIFE	S.C.	17	24	44	REDESIGN TURBINE
	EXPANDER	0	0	0	
	G.G.	0	0	0	LOWER TEMP INCREASE FLOW
CHAMBER CYCLE LIFE	S.C.	21	30	43	NEW CHAMBER
	EXPANDER	21	24	24	
	G.G.	21	25	30	
VALVE LEAKAGE	S.C.	7	6	11	REDESIGN SEALS
	EXPANDER	7	4	4	
	G.G.	7	6	6	
ONE FAILURE OTHER COMPONENTS	S.C.	17	15	21	APPROPRIATE SOLUTION
	EXPANDER	17	14	14	
	G.G.	17	14	17	

TABLE XXXIV

POTENTIAL DDT&E COST OVERRUN RISK

		PROGRAM TECHNICAL UNCERTAINTY CONSIDERED		ENGINE COMPLEXITY CONSIDERED	
	SUCCESS ORIENTED	PROBABLE CASE	MAX EXPOSURE CASE	PROBABLE CASE	MAX EXPOSURE CASE
CYCLE	DELTA COST (MILLIONS OF DOLLARS)				
STAGED COMBUSTION	0	62	186	101	295
EXPANDER	0	41	107	41	107
GAS GENERATOR	0	45	120	48	133

The estimated hardware requirements to support the advanced expander cycle DDT&E program are shown on Table XXXV. The lead time estimates are shown on Table XXXVI. The hardware and long lead estimates are based upon historical data from engine development programs as well as, the information available from the OOS, Space Tug and ASE preliminary design studies.

The estimated engine and component test requirements are shown on Table XXXVII. These estimates are also based upon historical data and past study results.

The preliminary cost estimate for activation and modification of existing ALRC facilities to test the OTV engine and components is 5 million calendar year 1979 dollars.

Engine production was planned to support two mission models; one for the APOTV and one for the AMOTV. The nominal mission models used in this study are shown on Tables 33 and 34 of NASA TMX-73394 for the APOTV and AMOTV, respectively. Two engines are assumed to be required per OTV with a total thrust level of 20,000 lb (10,000 lb per subassembly).

It was estimated that 40 engines (20 sets) are required to support the AMOTV mission model. In addition, 4 engines (2 sets) are required for the first two OTV's for a total of 44 engines. The nominal AMOTV flight program is 252 missions, of which 236 are reusable. The expendable planetary missions (16) govern the number of engine sets required. The AMOTV engine production schedule is shown on Figure 101. The engine production phase is scheduled for start on 31 March 1984 and runs through the last quarter of 1990. Go-ahead for long lead occurs in the first quarter of 1983. The first flight qualified engines would be delivered for two OTV's on 31 December 1987. The production is then planned at the rate of one engine per month for 36 months. This production rate was chosen to minimize the production cost. A lower rate would increase the cost of the production program.

TABLE XXXV

ADVANCED EXPANDER CYCLE ENGINE DDT&E HARDWARE REQUIREMENTS

Components	Number
Main Fuel Pumps	10
Main Oxidizer Pumps	10
Fuel Boost Pumps	10
Oxidizer Boost Pumps	10
Injectors	24
Combustion Chambers	12
Upper Nozzles (Fixed)	10
Igniters	12
Gimbal Assemblies	8
Lower Nozzles	10
Extension/Retraction Mechanisms	10
Controllers and Harness	8
Control Valves (sets)	10
Instrumentation and Harness	8
Heat Exchangers	8
Propellant Systems	10
Engines	
Development Engines	7(1)
PFC Engines	4(2)
FFC Engines	4

(1) Assumes 50 starts per engine. Development engine buildups from component residuals.

(2) These engines are dedicated to the PFC test program. In addition, four engines (2 sets) are deliverable. One set is a preprototype engine and the second set an FFC design.

TABLE XXXVI

LONG LEAD SUMMARY

Item	Lead Time Estimate, Months
Main Fuel Pump	12
Main Oxidizer Pump	12
Fuel Boost Pump	12
Oxidizer Boost Pump	12
Injector, Chamber and Fixed Nozzle	12
Igniter	8
Nozzle Assembly (Extendible)	14
Main Control Valves	10
Engine Controller and Electrical Harness	14
Instrumentation and Harness	12
Heat Exchangers	8
GSE	12

TABLE XXXVII

ADVANCED EXPANDER CYCLE ENGINE
DDT&E TEST PLAN

		No. of Tests	Total
1.1.1	Turbomachinery		400
1.1.1.1	Main Fuel Pump	75	
1.1.1.2	Main Ox Pump	75	
1.1.1.3	Fuel Boost Pump	50	
1.1.1.4	Oxidizer Boost Pump	50	
1.1.1.5	Assembly	150	
1.1.2	Main Combustion Chamber		650
1.1.2.1	Injector	200	
1.1.2.2	Chamber	100	
1.1.2.3	Upper Nozzle (Fixed)	50	
1.1.2.4	Igniter	150	
1.1.2.5	Gimbal Assembly	50	
1.1.2.6	Assembly	100	
1.1.4	Nozzle Assembly		150
1.1.4.1	Lower Nozzle (Extendible)	50	
1.1.4.2	Extension/Retraction Mechanisms	50	
1.1.4.3	Assembly	50	
1.1.5	Controls		500
1.1.5.1	Controller and Harness	200	
1.1.5.2	Control Valves	200	
1.1.5.3	Instrumentation and Harness	100	
1.1.6	Pressurization		100
1.1.6.1	Heat Exchangers	100	
1.1.7	Propellant Systems	50	50
1.1.10	Engine Assembly		500
1.1.10.1	Development	350	
1.1.10.2	PFC	50	
1.1.10.3	FFC	100	

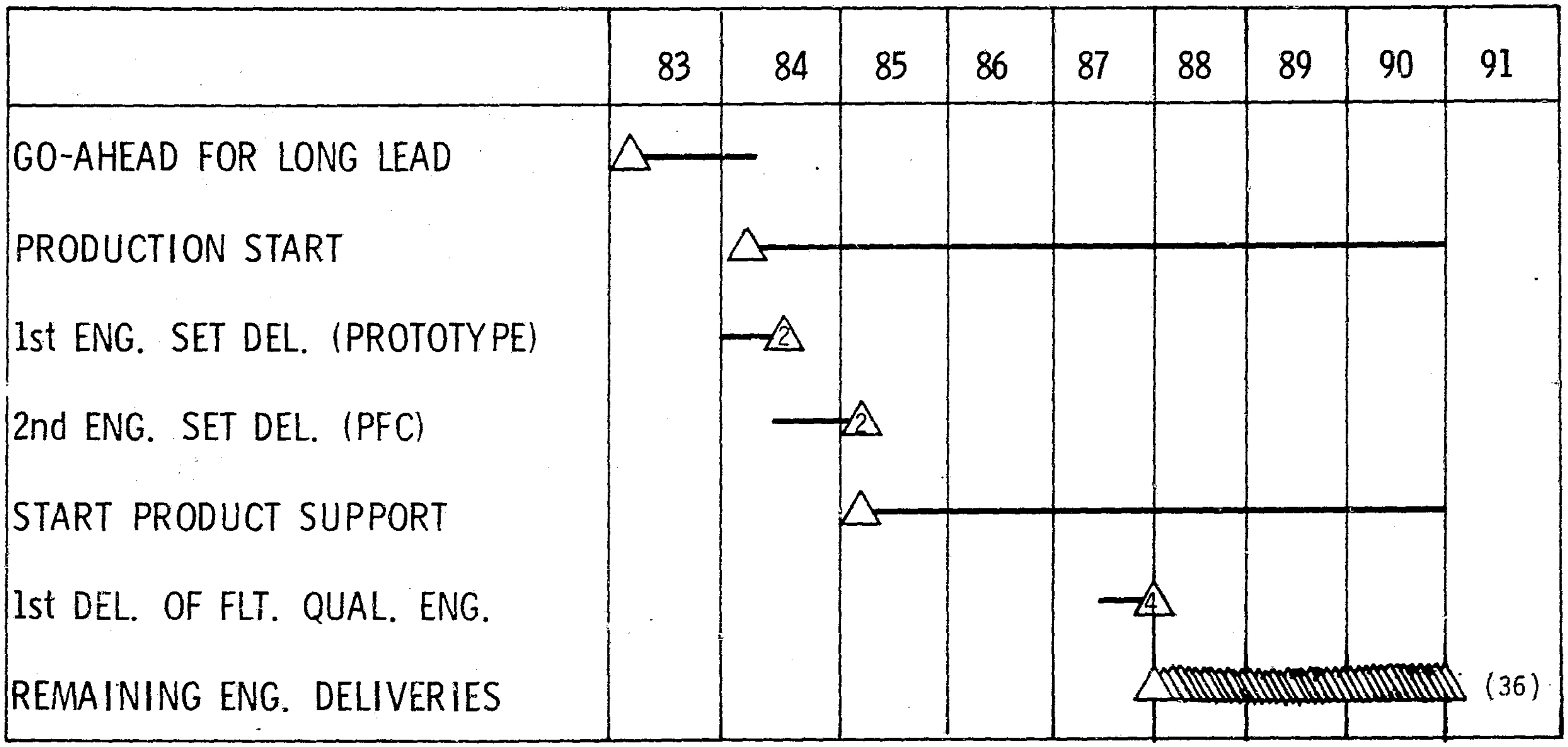


Figure 101. AMOTV Engine Production Schedule

It was estimated that 52 engines (26 sets) would be required to support the nominal APOTV mission model. In addition, 4 engines (2 sets) are required for the first two OTV's for a total of 56 engines. The APOTV nominal mission model has 450 reusable missions and 16 expendable planetary missions. To accommodate this mission model with 52 engines, it was assumed that the expendable planetary missions are flown with an engine which is at the end of its service life. The 10 hour accumulated run time requirement was calculated to result in a useful engine life of 18 missions before requiring a major engine overhaul. The APOTV engine production is also at a rate of 1 per month which extends this production program 12 months, through the last quarter of 1991, when compared to the AMOTV.

Items which would be deliverable are shown on Table XXXVIII. This deliverable item summary is patterned after the Titan III. The engine hardware on this list would be assigned a configuration item identification (CII) number, a configuration item specification number, a part number and a group of serial numbers for all items delivered during production. This identification is outlined on Table XXXIX. A strawman of a specification "tree" for the deliverable items and major subcomponents is shown on Table XL.

B. OTV ENGINE MAINTENANCE CONCEPT

The primary objective of the OTV engine maintenance concept is to maintain safety, reliability, and economy required by the operational objectives. To achieve the objective, the maintenance concept emphasizes minimum scheduled maintenance, short turnaround and reaction times, and cost effectiveness. The maintenance concept for the OTV engine resulted from utilizing maintainability studies and maintenance engineering analysis, generated from past programs, that were updated to be compatible with the OTV engine concept. The documents reviewed were (1) Space Tug Storable

TABLE XXXVIII

DELIVERABLE ITEM SUMMARY

Rocket Engine Assembly

1.1.1	Turbomachinery
1.1.2	Main Combustion Chamber
1.1.5.2	Control Valves
1.1.6.1	Heat Exchangers
1.1.7	Propellant Systems

Nozzle Assembly

1.1.4.1	Lower Nozzle (Extendible)
1.1.4.2	Extension/Retraction Mechanisms

Engine Controller and Electrical Harness
(1.1.5.1)

Instrumentation and Electrical Harness
(1.1.5.3)

Handling and Protective Equipment

Checkout and Maintenance Equipment

Technical Manuals (OTV Engine Procedures)

TABLE XXXIX

CONFIGURATION ITEM IDENTIFICATION[1]

OTV Engine AJ23-XXX

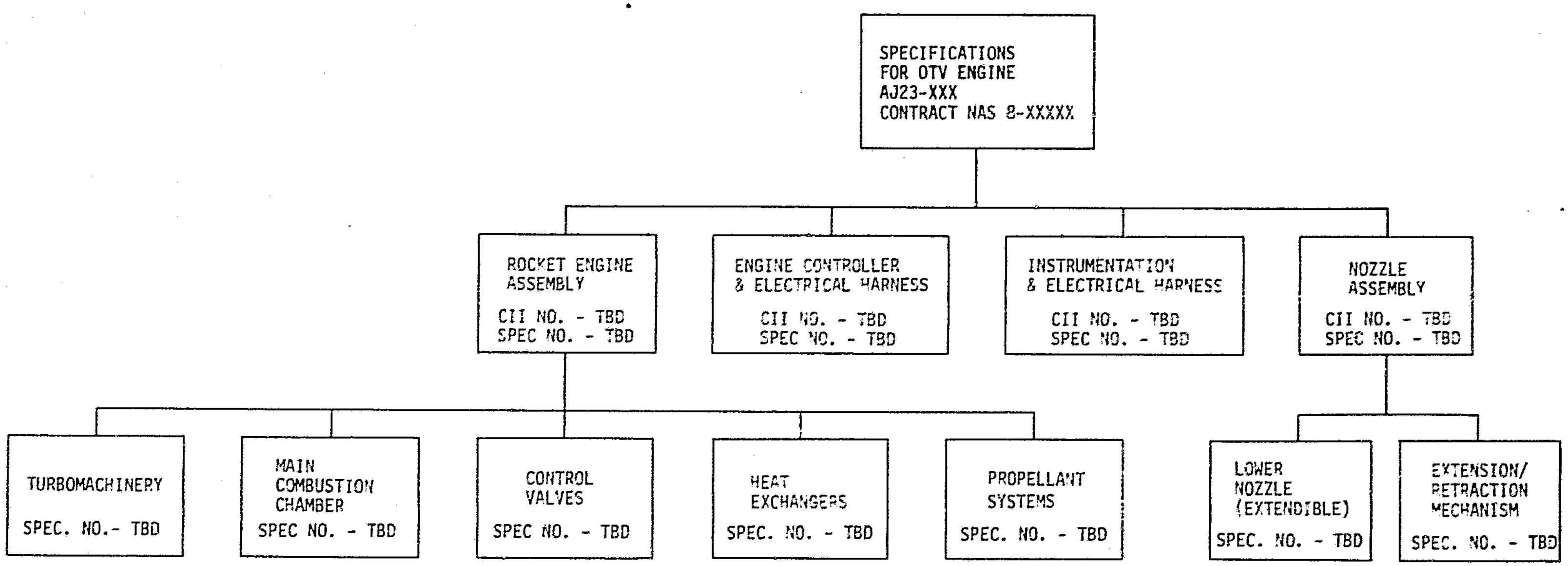
Contract NAS 8-XXXXX

Description	CII No.	CII Spec	Part Number	Serial Numbers
Rocket Engine Assembly	TBD	TBD	XXXX TBD	0000001 through 00000XX
Nozzle Assembly	TBD	TBD	XXXX TBD	0000001 through 00000XX
Engine Controller and Electrical Harness	TBD	TBD	XXXX TBD	0000001 through 00000XX
Instrumentation and Electrical Harness	TBD	TBD	XXXX TBD	0000001 through 00000XX

(1) Deliverable Hardware

TABLE XL

PRELIMINARY SPECIFICATION "TREE"



CII = CONFIGURATION ITEM IDENTIFICATION

Engine Study, issued by Aerojet Liquid Rocket Company, dated 31 January 1974, (2) Orbit-to-Orbit Shuttle Design Study, issued by Aerojet Liquid Rocket Company, dated December 1971, (3) Shuttle Main Engine Definition Study, Phase B, issued by Aerojet Liquid Rocket Company, dated 11 December 1970, (4) Engine Study for Interim Upper Stage (IUS) System, issued by Aerojet Liquid Rocket Company, dated 28 February 1975, (5) Design Study of RL-10 Derivatives, issued by Pratt & Whitney Aircraft, dated 15 December 1973, and (6) Orbital Maneuvering Subsystem (OMS) Rocket Engine Maintainability Report, issued by Aerojet Liquid Rocket Company, dated 28 October 1977.

The maintenance concept is defined by the following:

- Maximum utilization of an on-board checkout system for engine monitoring during flight to identify significant trends, and for fault isolation either during flight or on the ground.
- Capability of fault isolation to the Line Replaceable Unit (LRU) level by selecting the location and number of data acquisition sensors located on the engine.
- Design of Ground Support Equipment (GSE) to interface with and complement the on-board checkout system.
- On-the-Vehicle maintenance of the engine by removing and replacing LRU's.
- Capability at the launch site to remove and replace selected components of the engine system that do not qualify as LRU's.
- Field maintenance capability at the launch site for selected elements of the engine system.

1. Maintenance and Refurbishment Summary

Three basic types of maintenance actions have been considered; routine servicing, corrective or unscheduled maintenance performed to repair failures, and preventive or scheduled maintenance performed to prevent failure by timely replacements and refurbishment or overhaul of parts or components.

The OTV Engine maintenance philosophy recommended for the determination of repair or replace actions is based on an evaluation of flight data, routine maintenance data, and inspections.

Initially, ground test data would be utilized and as flight data is accumulated, engine criteria would be adjusted to extend the operating time limits on parts or components.

Flight data, routine maintenance data, and inspection results would be stored in a data bank to be used by the maintenance trend analysis program. This analysis would have the capability of comparing flight-to-flight, engine-to-engine performance to establish a band width of acceptable limits. This philosophy is in keeping with current maintenance practices employed by United Air Lines where degradation in performance compared to acceptable criteria determines maintenance as opposed to a total time or number of cycles criteria. The data input and services provided by the maintenance trend analysis program is shown schematically in Figure 102.

2. Maintenance

The OTV engine turnaround cycle consists of safe and purge, maintenance and launch operations. As shown on Figure 103, the longest portion of the cycle is required for turnaround maintenance which averages

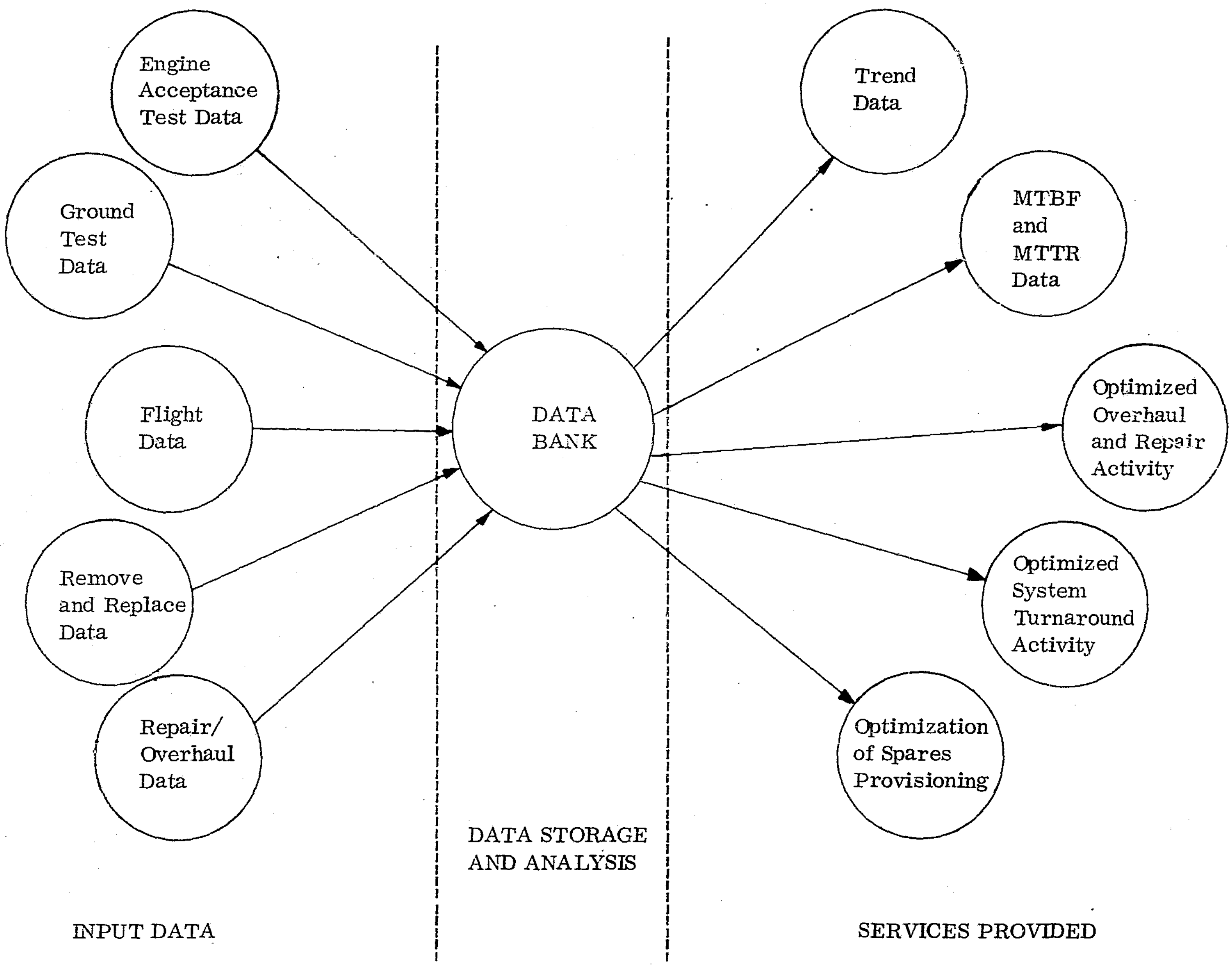


Figure 102. Maintenance Trend Analysis Program

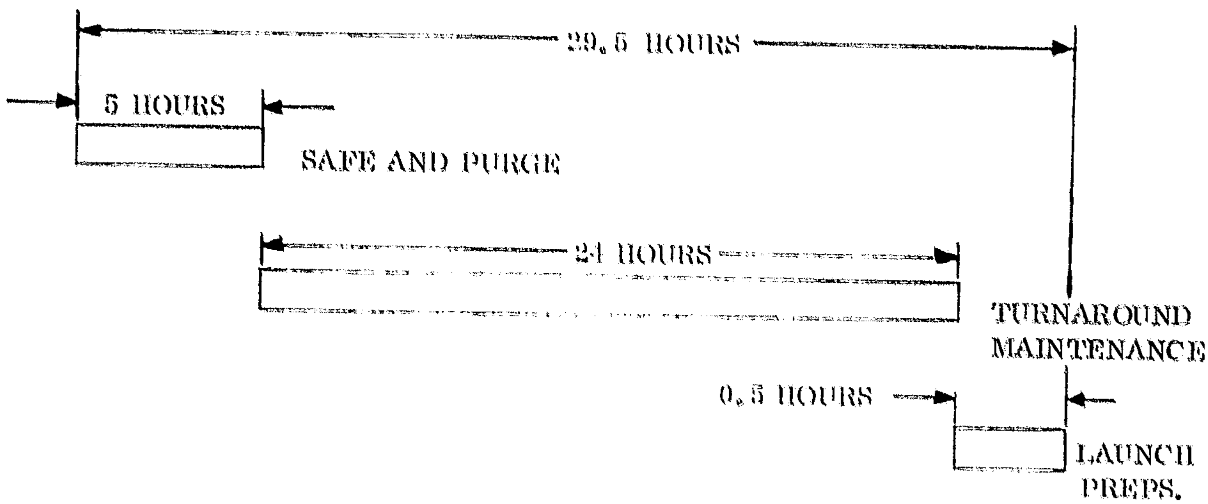


Figure 103. OTV Engine Turnaround Cycle Time

24 hours. During this 24-hour period routine maintenance is performed to determine engine condition. The routine maintenance actions require 11.5 hours. The remaining turnaround maintenance time of 12.5 hours is allocated for corrective maintenance, if required. Flight data and routine maintenance data are analyzed to determine any corrective maintenance required. Figure 104 shows the tasks and time for the OTV engine turnaround cycle.

The engine maintenance plan is shown on Figure 105 and the maintenance tasks will start when the OTV is removed from the shuttle payload bay and placed in a safe and purge area.

a. Safe and Purge

The Safe and Purge tasks consist of draining and venting residual propellants and pressurants, purging through the system until a dry condition is indicated, and performing a visual safety inspection. External moisture must be removed to prevent the possibility of cryogenic system contamination during later maintenance.

The findings of the initial visual inspection are relayed to the maintenance area to be used with other data to determine if corrective maintenance is necessary.

b. Turnaround Maintenance

The OTV engine, safe and dry with protective covers installed, is moved to the Maintenance Area where the turnaround maintenance begins. The operation consists of routine maintenance, which is identical between each flight, and corrective maintenance.

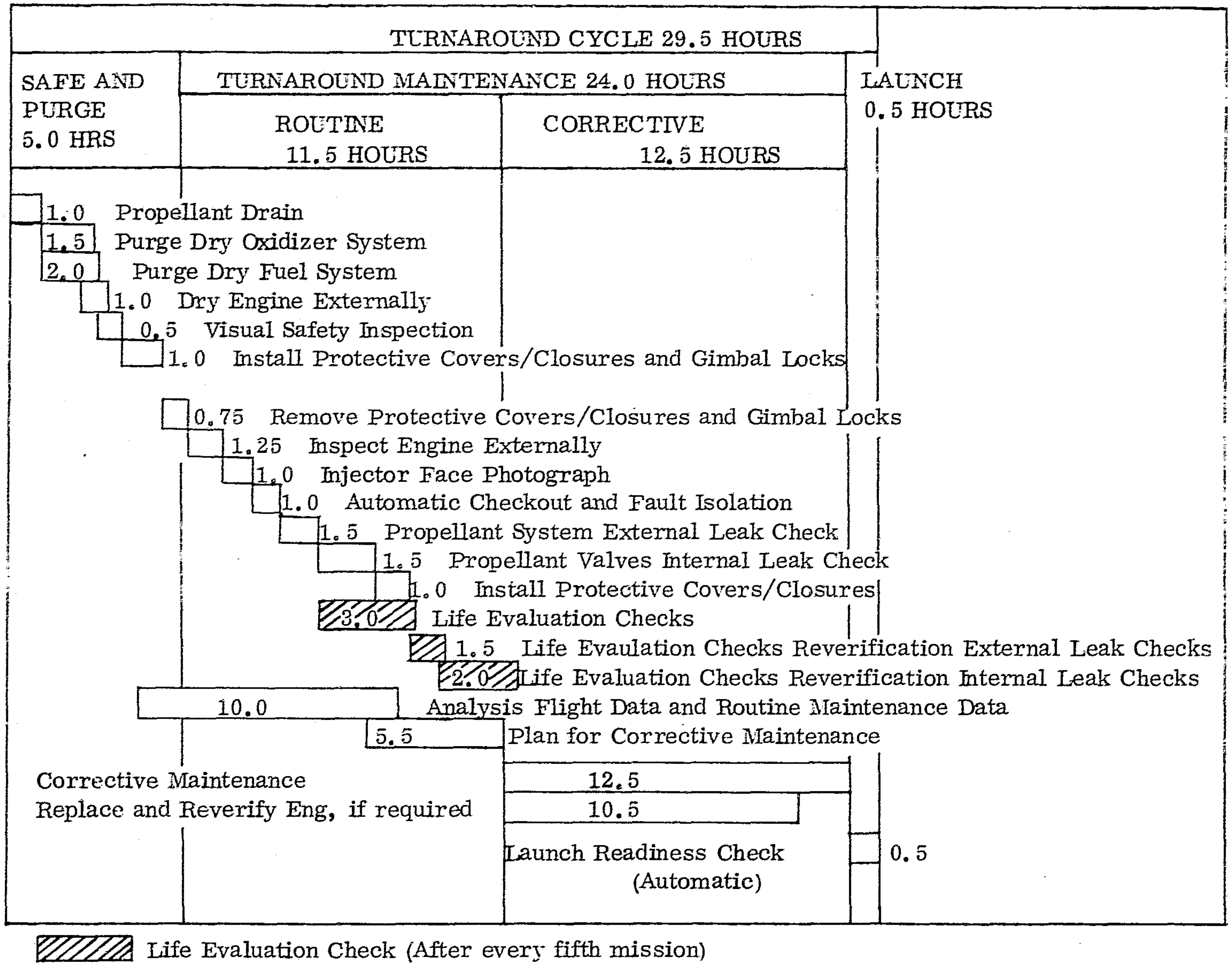


Figure 104. Turnaround Cycle Timeline

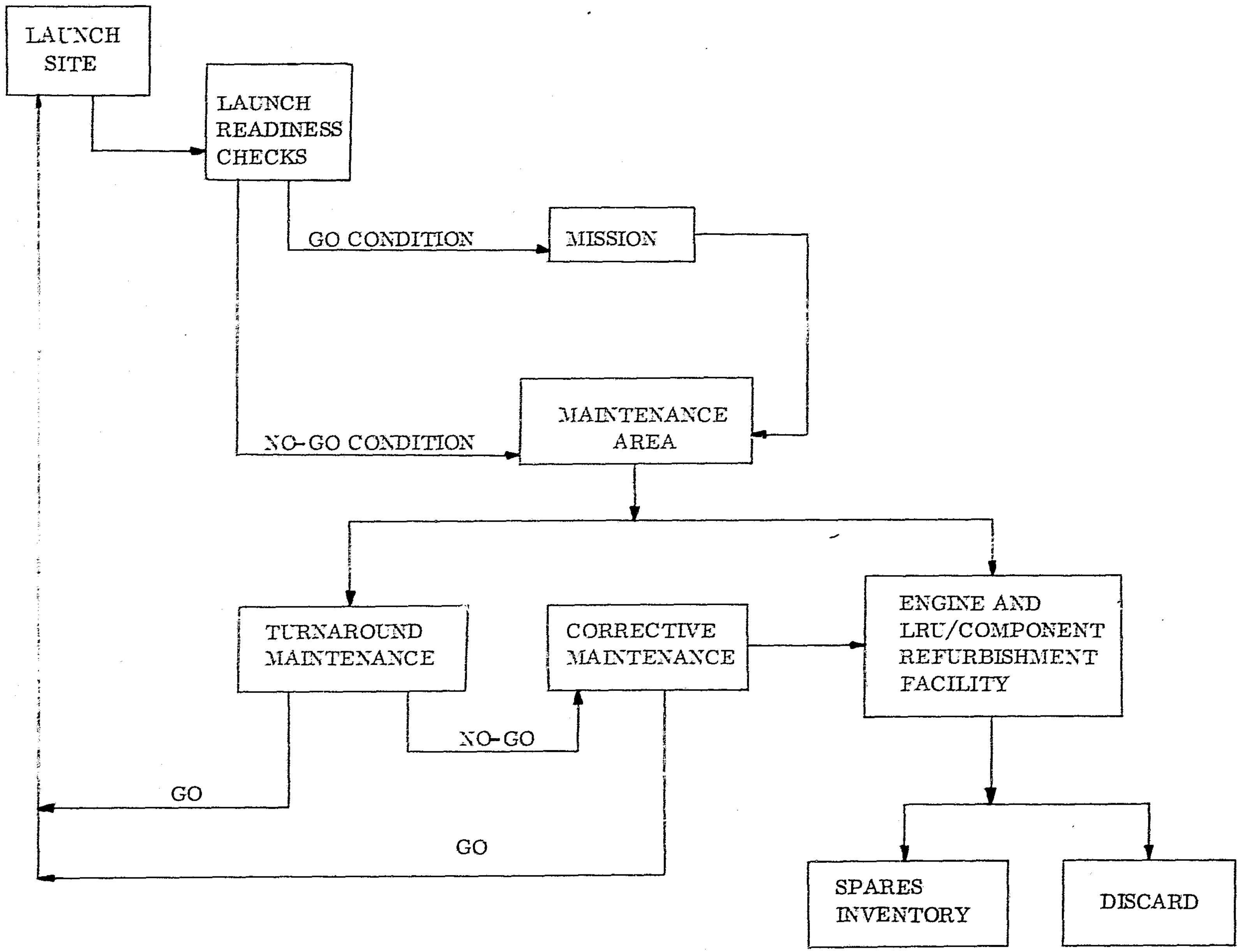


Figure 105. Engine Maintenance Plan

Routine maintenance consists of an external inspection of engine hardware, including taking photographs of the injector face, automatic checkout, and leak checks.

(1) External Engine Inspection

A 100 percent external visual inspection will be accomplished after each flight. Table XLI identifies the inspections to be performed.

In addition to the visual inspection of the thrust chamber assembly, photographs will be taken of the injector face. A baseline photograph of the injector face, taken following engine acceptance test, will be included in the engine log book. All subsequent photographs will be compared against the baseline to determine the amount and rate, if any, of injector degradation such as cracks in the face plate, orifice erosion or plugged orifices.

The inspection results will be analyzed to determine if any corrective maintenance is required and the inspection results will be logged in the historical data bank and utilized for maintenance trend analysis.

(2) Automatic Checkout and Fault Isolation

The OTV on-board checkout system will be primarily responsible for inflight checkout and monitoring of the OTV engine performance. However, its capability to perform checkout functions will be utilized during ground routine maintenance activities. The first automatic check out will provide a maintenance baseline for which all subsequent checks will be compared to establish trend analysis data. During the automatic checkout, should

TABLE XLI

OTV ENGINE INSPECTION

INSPECTION AREA	TYPE OF INSPECTION	TYPE OF FAULT	INSPECTION TECHNIQUE
Extendable Nozzle	External	Signs of thermal damage	Visual
Combustion Chamber	Internal - Combustion Chamber wall	Hot spots, streaking, erosion, cracks, dents, scratches	Visual
Injector	Internal - Injector Face	Cracks, scratches, erosion, plugged orifices	Visual and injector inspection kit
Turbo-machinery	External - Housing Flanges and weldments	General Damage	Visual
Propellant Systems	External - Bellows, Weldments, Manifolds, Lines	General Damage	Visual
Controls	External - Electrical Harness, Control Valves, Sensors	General Damage	Visual
Pressurization System	External - Weldments, Manifolds, and Lines	General Damage	Visual

a test be unsuccessful, the fault would be isolated to a Line Replaceable Unit (LRU) level. All redundancy would be verified during the automatic checkout.

Components checked automatically include the checkout unit itself, sensors, cables, actuators, solenoids, propellant valves, check valves and spark igniters. The functional check of valves would include an opening and closing time evaluation. Following the automatic checkout, the data would be analyzed along with flight data to determine if corrective maintenance is required. The data would be logged in the historical data bank for future comparison.

The engine parameters selected to be instrumented to ascertain flight performance and also utilized during ground automatic checkout are shown on Table XLII.

(3) Leak Checks

Certain faults can exist that are not detectable by either a visual inspection or automatic checkout and data analysis. As part of the routine maintenance activity, propellant valves would be checked for internal leakage. External leak checks would be accomplished on those systems that were pressurized during propellant valve internal leak checks. The results of the leak test data would be analyzed to determine if any corrective action is required and the data would be logged in the historical data bank.

(4) Engine Life Evaluation Check

Engine life evaluation checks would be scheduled after every fifth mission as part of the routine maintenance activity. They

TABLE XLII

OTV ENGINE INSTRUMENTATION REQUIREMENTS

RECOMMENDED ENGINE SUPPLIED	RECOMMENDED VEHICLE SUPPLIED
Chamber Pressure	Fuel Pump Inlet Pressure
Oxidizer Boost Pump Discharge Pressure	Fuel Pump Inlet Temperature
Oxidizer Boost Pump Housing Temperature	Oxidizer Pump Inlet Pressure
Oxidizer Boost Pump Rotational Speed	Oxidizer Pump Inlet Temperature
Oxidizer Main Pump Discharge Pressure	Vehicle Supply Voltage
Oxidizer Main Pump Housing Temperature	
Oxidizer Main Pump Rotational Speed	
Fuel Boost Pump Discharge Pressure	
Fuel Boost Pump Housing Temperature	
Fuel Boost Pump Rotational Speed	
Fuel Main Pump Discharge Pressure	
Fuel Main Pump Housing Temperature	
Fuel Main Pump Rotational Speed	
Autogenous System Pressure	
Autogenous System Temperature	
By-Pass Valve Position Switch	
Oxidizer Control Valve Position Switch	
Fuel Control Valve Position Switch	
Ignition Exciter Voltage	
Injector Oxidizer Pressure	

TABLE XLII (cont.)

RECOMMENDED ENGINE SUPPLIED	RECOMMENDED VEHICLE SUPPLIED
Injector Oxidizer Temperature	
Injector Fuel Pressure	
Injector Fuel Temperature	
Vibration Accelerometers	
Nozzle Extension Contact Switch	
Control Valves Position Transducer	
By-Pass Valve Position Transducer	
Fuel Flowmeter	
Oxidizer Flowmeter	
Turbomachinery Bearing(s) Temperature	

would be conducted to monitor engine wear progression and to provide a more accurate and cost effective scheduling of overhaul. The type of life evaluation checks that would be conducted are shown on Table XLIII.

Following the Life Evaluation Checks, complete engine internal and external leak checks will be performed to verify engine integrity. The installation of a throat plug into the combustion chamber will be required during the performance of the leak checks.

The results of Life Evaluation Checks and leak checks will be analyzed to determine if any corrective maintenance is required and the data will be logged in the historical data bank and utilized for maintenance trend analysis.

(5) Corrective Maintenance

Routine maintenance data, along with flight data, will be analyzed to detect discrepancies. The effort required to correct the discrepancies takes place during the corrective maintenance time span. Corrective maintenance represents the largest expenditure of time and resources during the turnaround cycle and consists of in-the-OTV repair of engine and in a limited number of cases, engine removal.

Although an engine can sometimes be changed faster than a component, the engine change would involve more reverification. To reduce cost and improve integrity it is preferable to replace Line Replaceable Units (LRU's). However, corrective actions requiring more than 12.5 hours would be cause for engine replacement.

TABLE XLIII

LIFE EVALUATION CHECKS

Assembly/Component	Type of Check
Turbomachinery	. Internal Bearing Damage . Internal Seal Wear . Torque Check
Thrust Chamber Assembly	. Deformation and Structural Integrity
Extendible Nozzle	. External Structure and Thermal Damage
Controls	. Functional Check of all Valves

c. Launch Pad Activity

As part of the launch countdown sequence, an OTV engine readiness check will be performed by the orbiter on-board automatic checkout system. The readiness check will verify the engine electrical system continuity and that critical engine parameters are within a specified range.

3. Refurbishment

The recommended turnaround maintenance concept provides for engine checkout, fault isolation and corrective action. The corrective action may be the replacement of a seal, an LRU, or a complete engine.

The refurbishment activity takes place outside of the turnaround maintenance cycle and with few exceptions will be accomplished at the depot level as shown on Table XLIV for an engine and Table XLV for an LRU/component.

4. Ground Support Equipment

Engine Peculiar Ground Support Equipment (GSE) will be required for storage, handling and turnaround maintenance activity. A preliminary listing of the equipment and function is as follows:

TABLE XLIV

ENGINE REPAIR/OVERHAUL AT DEPOT LEVEL

A. ENGINE RECEIVING

1. Removal from shipping container
2. Receiving inspection (visual)

B. PRE-DISASSEMBLY CHECKS

1. Leak checks
2. Functional checks
3. Electrical checks
4. Check to aid in anomaly investigation

C. ENGINE DISASSEMBLY

1. Disassemble engine
2. Disassemble components

D. INSPECTION

1. Dimensional inspection of critical surfaces
2. Visual inspection of noncritical surfaces
3. X-ray of structural critical areas

E. PARTS REPLACEMENT

1. Automatically replaced
    a. Gaskets, seals and packings
    b. Bearings
    c. Tablocks
    d. Key washers
2. Other parts as dictated by inspection

F. CLEANING

1. Clean reusable and new parts per applicable specifications

G. ASSEMBLY AND TEST

1. Reassemble components
    a. Retest components per applicable procedures
2. Reassemble engine
3. Perform engine reacceptance test per applicable requirements and procedures

H. PACKING

1. Prepare engine for shipment

TABLE XLV

LRU/COMPONENT REPAIR/OVERHAUL AT DEPOT LEVEL

A. RECEIVING LRU/COMPONENT

1. Receiving Inspection (visual)

B. PRE-DISASSEMBLY CHECKS

1. Individual checks to aid in anomaly investigation

C. DISASSEMBLY

1. Disassemble LRU/component

D. INSPECTION

1. Dimensional inspection of critical areas
2. Visual inspection of non-critical areas
3. X-ray of parts where applicable

E. PARTS REPLACEMENT

1. Automatically replaced -
   a. Gaskets, seals, and packings
   b. Tablocks
   c. Key washers
2. Other parts as dictated by anomaly investigation and inspection

F. CLEANING

1. Clean reusable and new parts per applicable specifications

G. ASSEMBLE AND TEST

1. Reassemble LRU/component
2. Calibrate mechanical and electrical components as required
3. Perform reacceptance test per applicable requirements and procedures.

H. PACKING

1. Prepare LRU/component for replacement in the spares inventory.

Nomenclature	Function
Engine Shipping and Storage Unit	Shipping and storage container.
Engine Handling Unit	Remove engine from shipping unit and used for engine to OTV installation or removal.
Engine Combustion Chamber Protective Cover Set	Protect external surfaces of chamber and provide closure for combustion chamber exit.
Throat Plug	Leak check of propellant systems.
Leak Test Adapter	Leak check of propellant systems.
Engine Tool Kit	Provides special tools such as gimbal locks, stiff links, torque adapters, etc.
Injector Inspection Kit	Provide camera, modified to provide a light source at the injector face and extend the lens into the combustion chamber with the camera remaining outside.

5. OTV Engine Procedures

Procedures will be required to support the engine maintenance and refurbishment activities. The procedural format recommended for the OTV engine is the format that has proven successful in existing programs, such as the Titan Engine System operational procedures. The procedures are sectionalized and described below:

Section 0 - General Information

Procedures that are referenced in other procedures, i.e., Damage Limits, Torque Tables, Index, etc.

Section 1 - Handling, Shipping and Storage

Procedures necessary for handling, shipping and storage of the OTV engine.

Section 2 - Inspection

Procedures for routine and special engine and/or component inspections.

Section 3 - Operation and Servicing

Step-by-step procedures for routine checkout and maintenance.

Section 4 - Component Removal and Replacement

Step-by-step procedures for corrective maintenance requiring removal and replacement of components.

Section 5 - Component Disassembly, Repair and Reassembly

Step-by-step procedures for performing depot maintenance.

VIII, B, OTV Engine Maintenance Concept (cont.)

Section 6 - Illustrated Parts Breakdown

Provides indexed illustrations and complete breakdown of parts.

Section 7 - GSE

Provides operation and maintenance instructions for GSE and also an illustrated parts breakdown of the GSE item.

IX. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The conclusions which were derived from the results of this study are discussed herein and shown on Figure 106. These conclusions cover the results of all study tasks.

Crew safety was found to be a major concept selection and engine design driver. A minimum of two engines are required since single engine installations result in unacceptable crew losses. Series redundant main propellant valves are required to assure that the engine will shutdown. This is the same as the design philosophy used for the OMS twin engines. Redundant ignition systems are required to assure that the engine will start.

Another major design driver was the high engine minimum performance requirement. This requirement dictates high nozzle area ratios and hence, small throat sizes because the engines are length constrained. The throat size can be reduced through high chamber pressure operation, by going to multiple engine installations, or both. In any case, a new engine is needed to meet the minimum performance, man-rating, reusability and long life requirements.

The staged combustion cycle and advanced expander cycle engines have approximately the same payload capability when used in multiple (two or more) engine installations. Therefore, a choice between these two engine cycles cannot be made on purely a performance basis. The gas generator cycle engine concept cannot meet the minimum performance requirements in single or multiple installations and results in significant payload penalties (greater than 3%) compared to the expander and staged combustion cycle engines.

- CREW SAFETY DICTATES:
  - (1) MINIMUM OF TWO ENGINES
  - (2) SERIES REDUNDANT MAIN PROPELLANT VALVES
  - (3) REDUNDANT IGNITION
- A NEW ENGINE IS REQUIRED TO MEET THE MAN-RATING, HIGH PERFORMANCE, REUSABILITY AND LONG LIFE REQUIREMENTS
- EXPANDER AND STAGED COMBUSTION CYCLE ENGINES HAVE NEARLY EQUAL PAYLOAD CAPABILITY (TECHNICAL STALEMATE)
- GAS GENERATOR CYCLE ENGINES CANNOT MEET PERFORMANCE REQUIREMENTS RESULTING IN PAYLOAD PENALTIES
- A TOTAL ENGINE THRUST OF 20K LBF APPEARS TO BE ABOUT OPTIMUM FROM A PAYLOAD BASIS
- DDT&E AND PRODUCTION COSTS OF EXPANDER CYCLE ENGINES ARE LOWER THAN OTHER CANDIDATES
- EXPANDER CYCLE PROVIDES LESS DEVELOPMENT RISK

Figure 106. Conclusions

A total engine thrust level of 20,000 lb is approximately optimum on a payload basis. This and the crew safety results make two 10,000 lb thrust engines an attractive choice.

The DDT&E and production cost analyses results, reported in Volume III, show that the expander cycle engine costs are lower than either the staged combustion or the gas generator engine cycles. In addition, the expander cycle engine provides less risk in terms of potential DDT&E cost and schedule overruns. These benefits are obtained with an expander cycle because it has fewer components and does not have a high temperature, fuel-rich hot gas turbine drive. The expander cycle turbines operate in a benign environment.

Based upon the foregoing, a new advanced expander cycle engine is the best choice for the OTV.

B. RECOMMENDATIONS

The recommendations derived from this study are shown on Figure 107. The list includes OTV engine design recommendations, recommendations for future study and advanced technology program recommendations.

Two advanced expander cycle engines of 10K lb thrust each are the recommended baseline configuration for the OTV. The gas generator engine cycle should be dropped from all future study efforts because of low performance capability.

The merits of two vs three engines should be investigated further in vehicle studies. These studies should consider impacts upon both vehicle design and maintenance. While three engines will increase maintenance

- BASELINE AN INSTALLATION OF TWO 10K LB THRUST ENGINES.
- INVESTIGATE THE MERITS OF TWO VS. THREE ENGINES FROM A VEHICLE DESIGN AND MAINTENANCE VIEWPOINT.
- GAS GENERATOR CYCLE SHOULD BE DROPPED FROM FURTHER STUDY EFFORTS.
- DETAILED SAFETY, RELIABILITY, DEVELOPMENT RISK AND LIFE CYCLE COST ANALYSES OF EXPANDER AND STAGED COMBUSTION CYCLE ENGINES SHOULD BE CONDUCTED.
- DESIGN OPTIMIZATION OF AN ADVANCED EXPANDER CYCLE ENGINE (AEC) SHOULD BE UNDERTAKEN.
- COMPONENT REDUNDANCY REQUIREMENTS SHOULD BE EXAMINED FURTHER AND DESIGNED INTO THE ENGINE.
- CONTINUE TO EVALUATE THE IMPACT OF CREW SAFETY ON AEC DESIGN.
- EXPANDER CYCLE ENGINE COMPONENT TECHNOLOGY PROGRAMS SHOULD BE INITIATED.

Figure 107. Recommendations

costs compared to two engines, the mission losses may be reduced. Therefore, total life cycle costs for delivering the payloads required by a mission model need to be evaluated.

Further analyses should be undertaken to compare the advanced expander cycle engine to a staged combustion cycle engine in terms of safety, reliability, development risk and life cycle cost. The Phase A extension to this contract will evaluate this. Failure mode and effects analyses (FMEA) should also be conducted on both the advanced expander and staged combustion cycle engines. The objective of the FMEA analyses would be to identify any further component redundancy requirements to aid in future cycle comparisons and selections. For example, there is a high probability that redundant preburner valves would also be required on a staged combustion cycle to assure safe shutdown.

Design optimization of an advanced expander cycle engine at the 10,000 lb thrust level should be initiated. A proposal was recently submitted to NASA/MSFC to conduct such a point design evaluation.

The impact of crew safety upon the advanced expander cycle engine should also be continually evaluated. To be effective, safety considerations must be incorporated into the initial design concept and not come as an "after-thought." Instrumentation and features that the astronaut would like to see on the engine should be identified early.

Component technology programs should be initiated on the advanced expander cycle. Critical components and items are: (1) the combustion chamber and the heat input into the coolant, and (2) small turbomachinery design, efficiencies and parasite flows. These component technology programs should culminate in an experimental engine program. This phisosophy was followed on the OMS engine. The OMS technology work aided the engine development phase immensely. Problems encountered during the development phase were recognized rapidly and solutions found quickly because of the experience gained from the technology efforts. The experimental engine program would minimize the development program risk.

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