Report No. 32999-F

NASA

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

FINAL REPORT

VOLUME I: EXECUTIVE SUMMARY

(NASA-CR-161297) ORBIT TRANSFER VEHICLE (OTV) ENGINE, PHASE A STUDY. VOLUME 1: EXECUTIVE SUMMARY Final Report (Aerojet Liquid Rocket Co.) 100 p HC A05/MF A01 CSCL 22B G3/16

N79-33235

Unclas 38815

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA

CONTRACT NAS 8-32999

29 JUNE 1979

AEROJET LIQUID ROCKET COMPANY

REPORT 32999-F

29 JUNE 1979

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

FINAL REPORT
VOLUME I: EXECUTIVE SUMMARY

CONTRACT NAS 8-32999

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA 35812

PREPARED BY

J. A. MELLISH
STUDY MANAGER
ALRC ENGINEERING

AEROJET LIQUID ROCKET COMPANY
P.O. BOX 13222
SACRAMENTO, CALIFORNIA 95813

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 of this study was from 10 July 1978 to 4 June 1979.

The final report is submitted in three volumes:

Volume I: Executive Summary

Volume II: Study Results

Volume III: Study Cost Estimates

Volume I: Executive Summary

Table of Contents

TABLE LIST

FIGURE LIST

FIGURE LIST (cont.)

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 "Orbit Transfer Systems with Emphasis on Shuttle Applications - 1986-1991."

II. 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 technical and cost data and provide this data in suitable format for use by NASA and 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

II, Study Objectives and Scope (cont.)

- 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

III. METHOD OF APPROACH AND PRINCIPLE ASSUMPTIONS

A. APPROACH

The overall study logic flow diagram depicting the seven major study tasks, their interrelationships, and the 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.

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, a engine concept that was capable of meeting the requirements was defined in Task II. Engine cycles considered in Task II, Engine Concept Definition, were the staged combustion, expander and gas generator. Parametric analyses were conducted during Task III on all candidate cycles over a thrust range of 10K to 30K lb thrust and detailed supporting analytical studies were conducted on the engine concept selected in Task II. Tank-head and pumped idle mode operation were evaluated for the selected concept in Task IV, along with one-time emergency operation at a mixture ratio of 10. A work breakdown structure (WBS) was structured in Task V in concert with NASA/MSFC. 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.

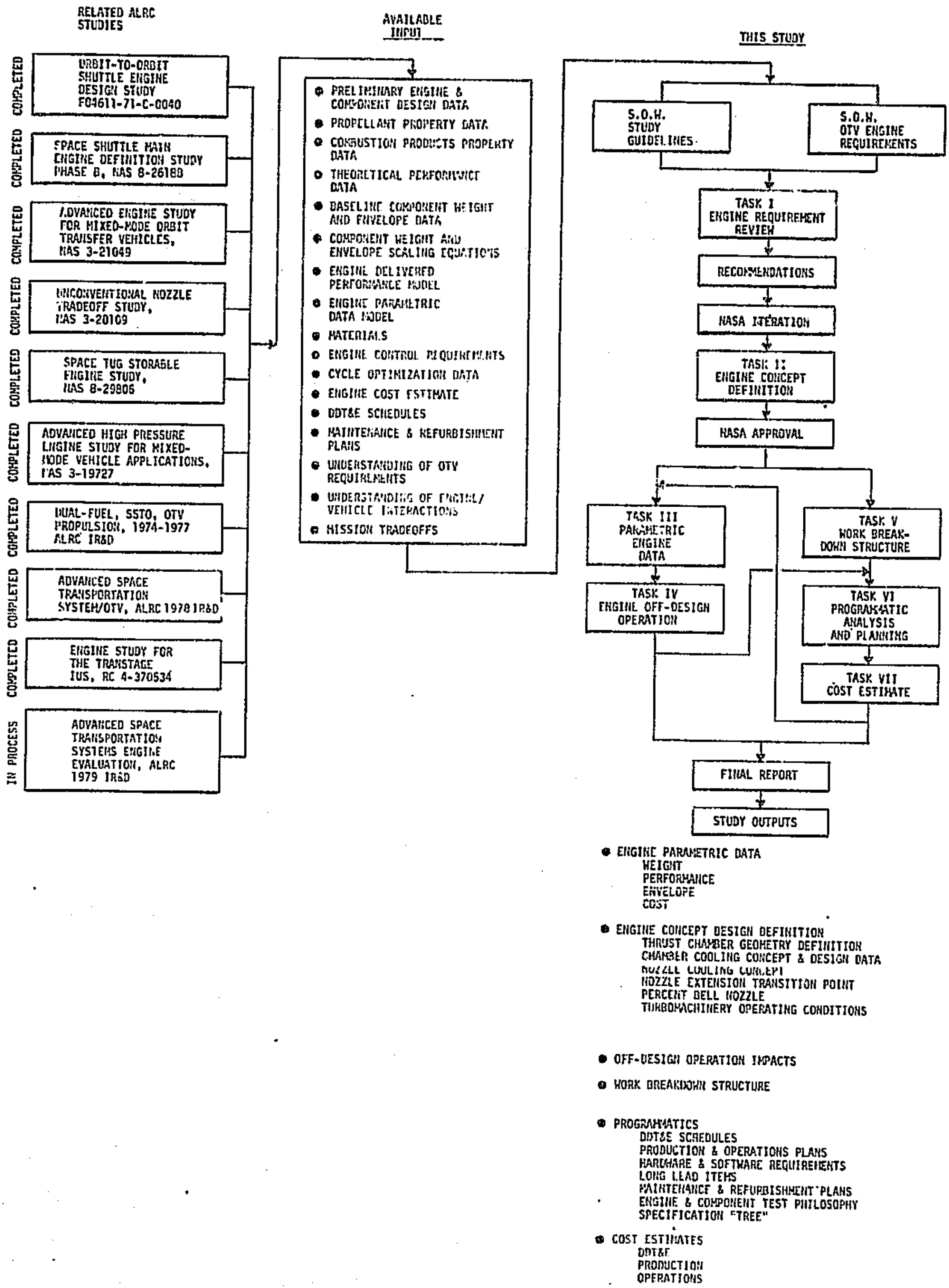


Figure 1. Study Flow Logic Diagram

B. PRINCIPLE ASSUMPTIONS

The following principle assumptions and guidelines were provided by NASA/MSFC and used to conduct this engine study program.

1. All engine designs and characteristics will be compatible with the OTV requirements and schedules 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.

5. 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.).

III, B, Principle Assumption (cont.)

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-rountine) 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.

IV. 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.

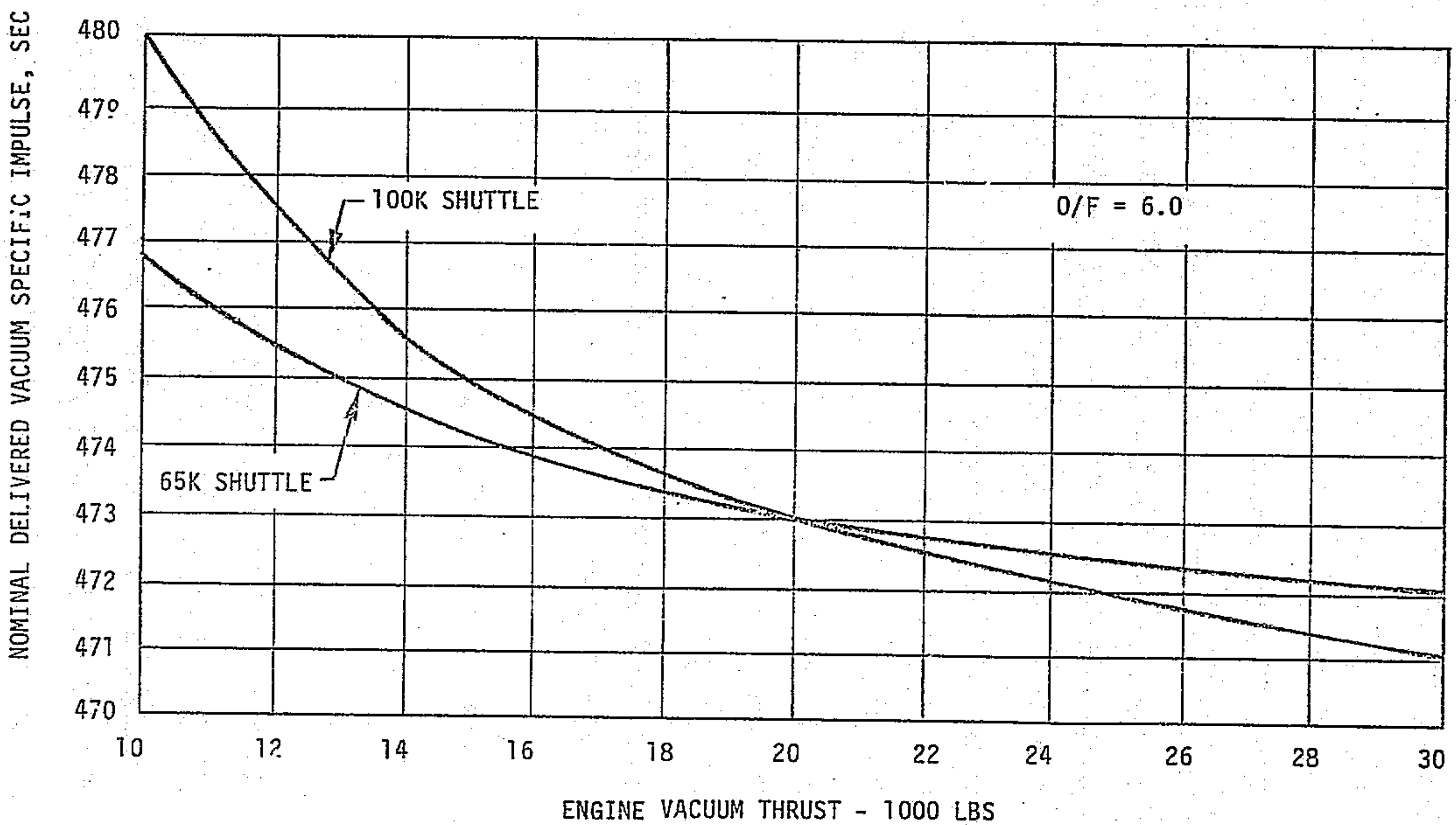
2. Engine design and materials technology are based on 1980 state-of-the-art, or start of phase C/D contract.

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 2. 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 2. OTV Engine Minimum Nominal Specific Impulse Requirements vs Engine Vacuum Thrust

IV, A, OTV Engine Requirements (cont.)

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 ± 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 ± 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.

IV, A, OTV Engine Requirements (cont.)

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

The foregoing requirements, particularly the performance, man-rating, reusability and service life requirements, dictate the development of a new engine. 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 engine different from the engine studies conducted in support of the Space Tug studies performed in 1973.

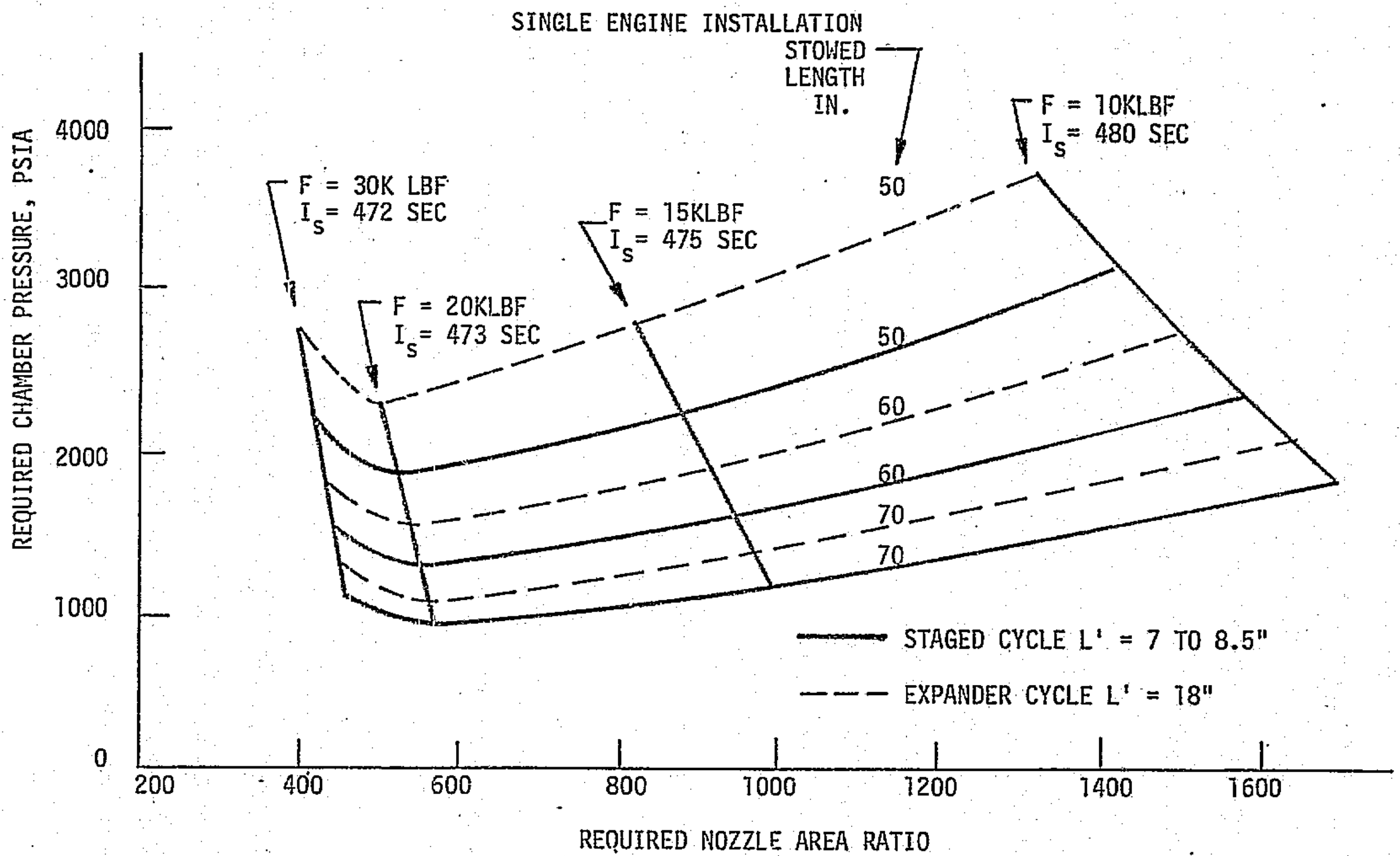
B. PERFORMANCE REQUIREMENT IMPACTS

The minimum required nominal specific impulse was shown on Figure 2. 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 on the figure, high nozzle area ratios are required. Because the engine was also length contrained, 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.

Figure 3 shows the thrust chamber pressures and nozzle area ratios that are necessary to achieve the minimum specific impulse requirements with staged combustion and expander cycle engines. It should be noted that a single engine installation was assumed for this analysis.

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. 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. It should also be noted that not all operating points on the figure are feasible for either the expander or staged combustion cycles and some performance penalty must be accepted, particularly at 10 K lb thrust.

The results of this performance requirement analysis, inpart, lead to the consideration of multiple engines (2 or 3) to achieve the



PERFORMANCE GOALS MORE DIFFICULT TO ACHIEVE WITH EXPANDER CYCLE

Figure 3. Chamber Pressures and Nozzle Area Ratios Required to Meet Minimum Performance Requirements

performance goals. By reducing the thrust per chamber, small throat sizes and hence, high area ratios are possible. The multiple engine analysis is reported in Section V.B. of this report, as part of the concept definition task.

C. MAN-RATING REQUIREMENT IMPACTS

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.

Mission reliability is the successful delivery of a payload and the return of the OTV to the Shuttle Orbiter. The goal is to minimize mission losses. Crew safety is the safe return of the crew to the orbiter regardless of the payload status. The objective is to eliminate crew losses.

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.

An acceptable crew risk (ACR) of about 5×10^{-4} was assumed to be tolerable based upon historical precedence and the following logic.

ACR Estimation

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

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

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 the various engine cycle candidates. Engine failure (shutdown) was assumed to occur anytime there is a malfunction in a critical component. 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. Additional guidelines and assumptions are discussed in Vol. II, Section III,C.

Figure 4 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. 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 figure also 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 failures 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 for all burns. This occurs 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 and was selected.

Figure 5 shows that all candidate concepts satisfy the reliability and safety requirements in the 2/1 concept. The expander cycle is slightly better than the other two concepts because of the other component count.

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

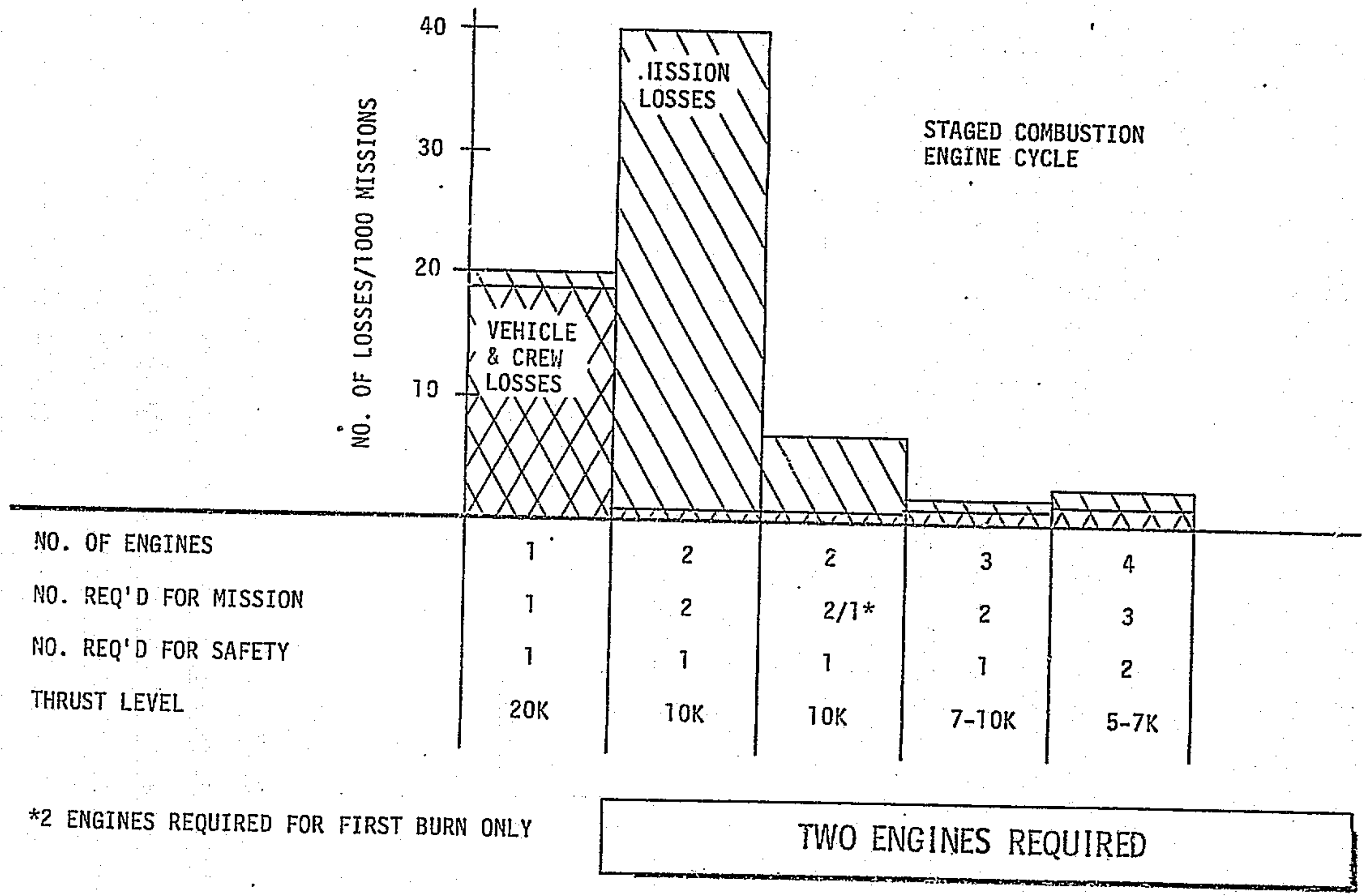
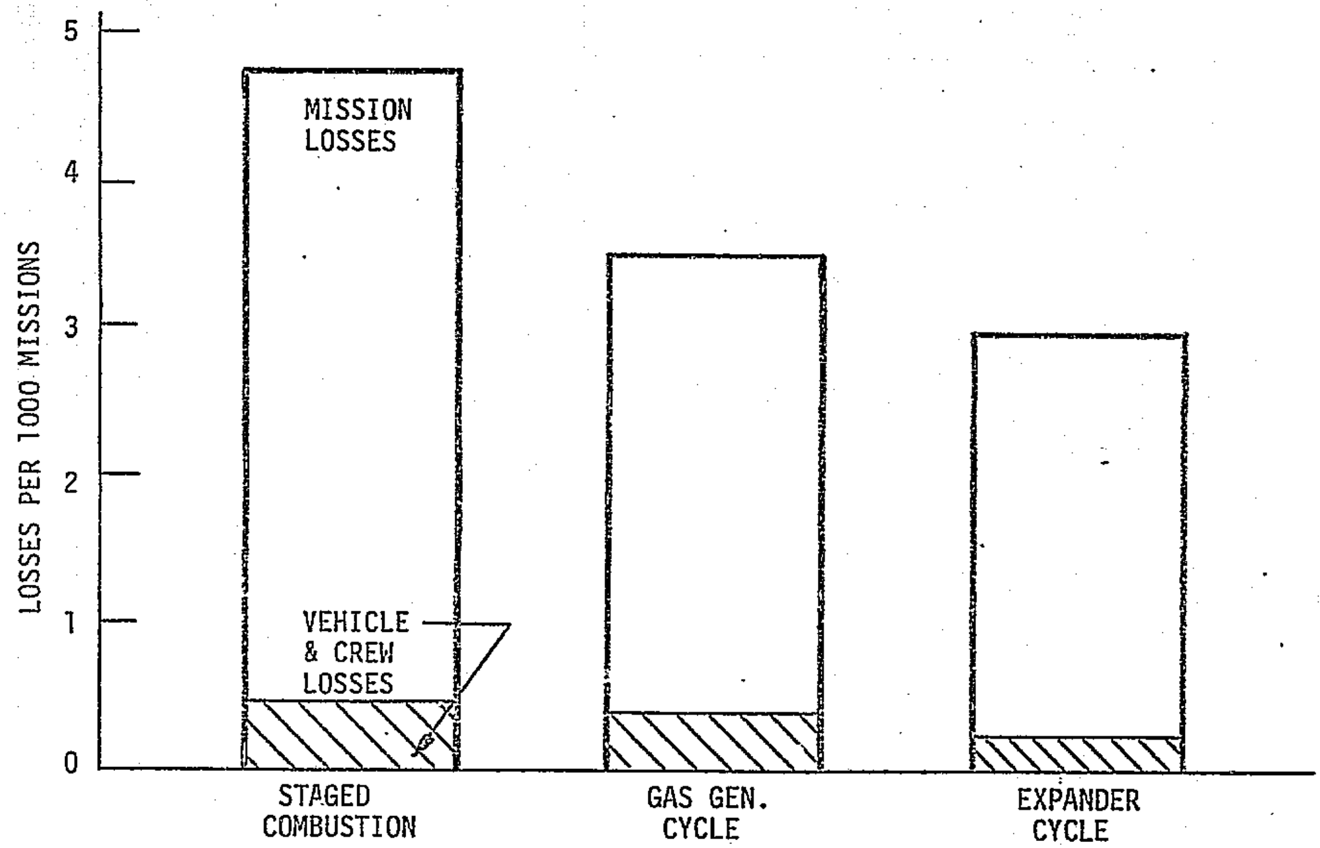


Figure 4. OTV-Mission and Vehicle/Crew Risk



ALL 2/1 CONCEPTS APPEAR TO BE ACCEPTABLE

Figure 5. 2/1 Engine System Concept

- A minimum of two engines is mandatory
- Series redundant main propellant valves are required. (Engine must shutdown)
- Redundant spart ingiter is required (engine must start)
- 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 trade-off studies of Task II.

V. TASK II: ENGINE CONCEPT DEFINITION

A. CYCLE ANALYSIS

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.

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 6. 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.

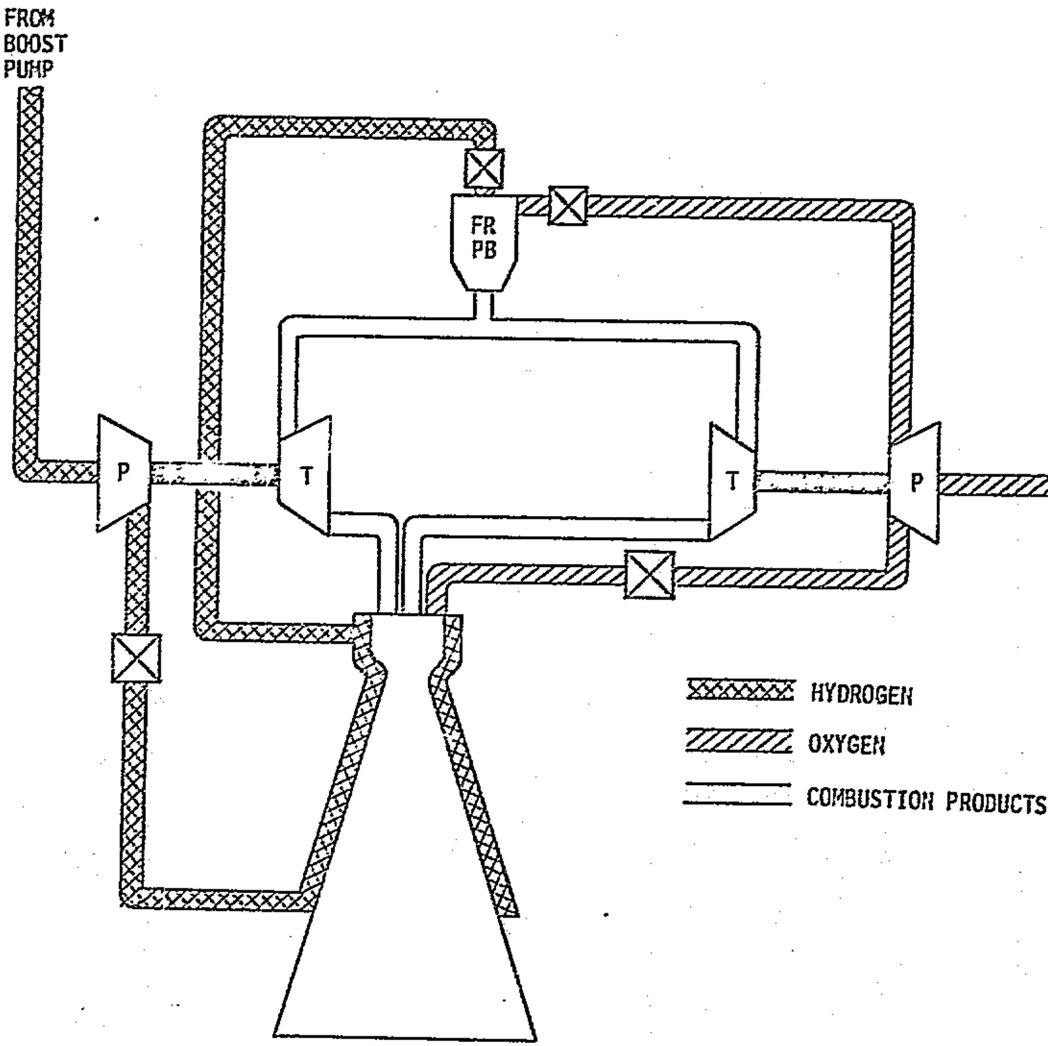


Figure 6. Single Fuel-Rich Preburner Staged Combustion Cycle

The expander cycle engine evaluated in the concept definition phase is a parallel turbine drive concept shown on Figure 7. 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 8) 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.

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-off evaluations.

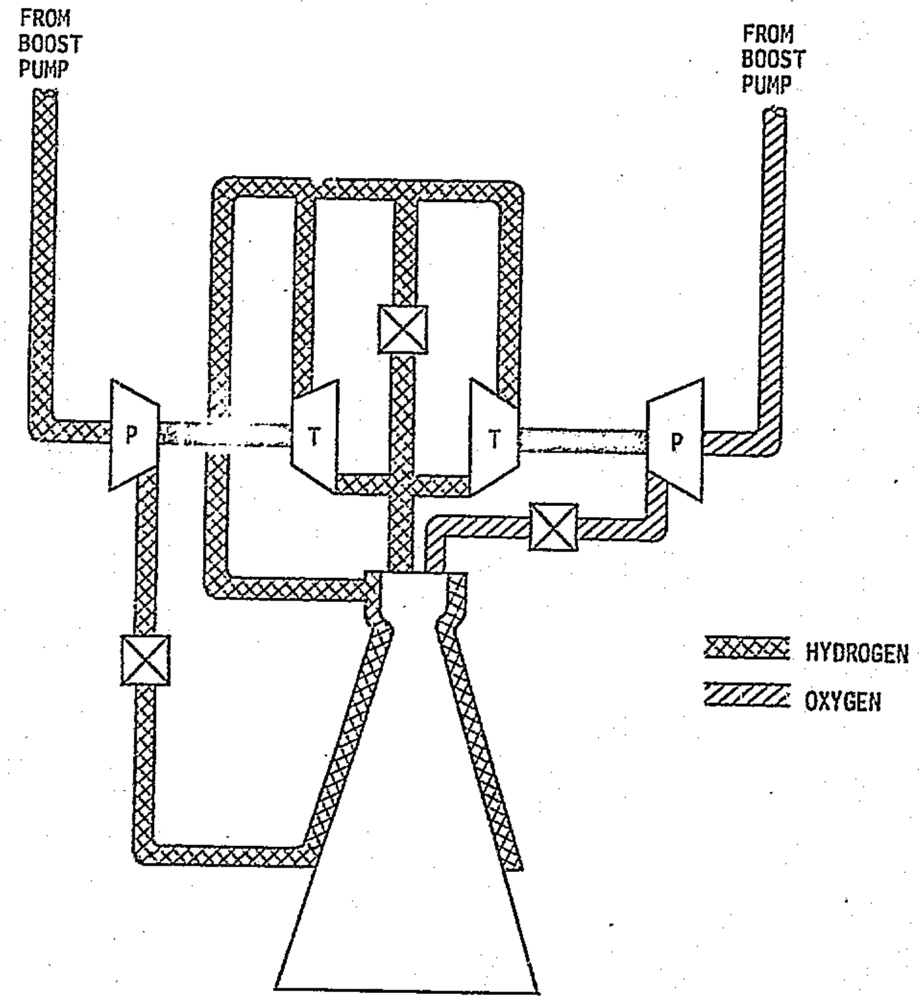


Figure 7. Expander Cycle

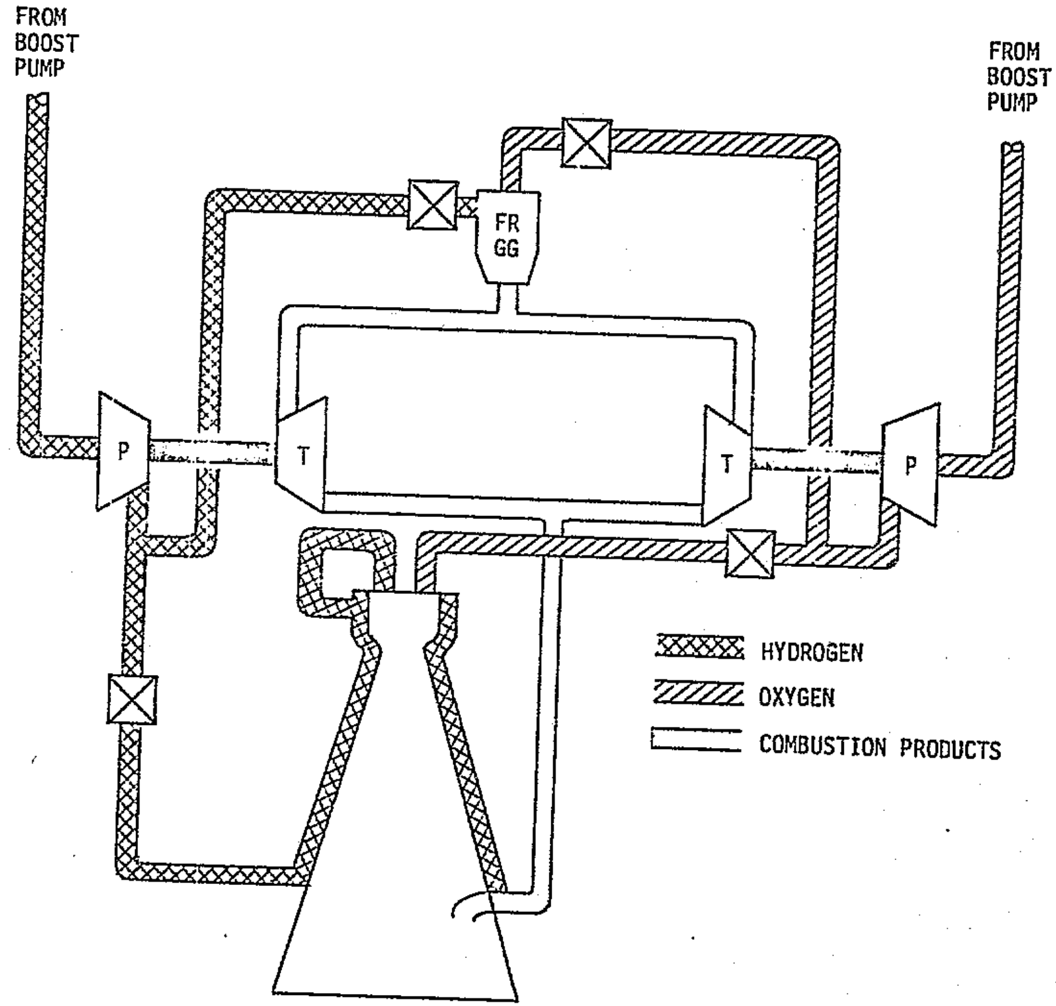


Figure 8. Gas Generator Cycle

The expander cycle engine was found to be power balance limited because of the low (650°R or less) turbine inlet temperatures. The staged combustion cycle engine is ultimately power balance limited but not in the range of chamber pressures investigated in this study. 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 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 operating chamber pressure. Either pressure drops become too high because of high coolant velocities or the coolant bulk temperature rise limit reached.

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. 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. Performance/weight trade-offs were performed and resulted in a maximum recommended chamber pressure of 1500 psia for the gas generator cycle.

The maximum recommended operating pressures for the engine cycle candidates investigated are summarized on Figure 9 along with the limiting criteria for each. All cycles appear to be life cycle limited at a thrust level of approximately 8K lb thrust and lower.

B. CANDIDATE CYCLE PERFORMANCE, WEIGHT AND TRADEOFF ANALYSIS

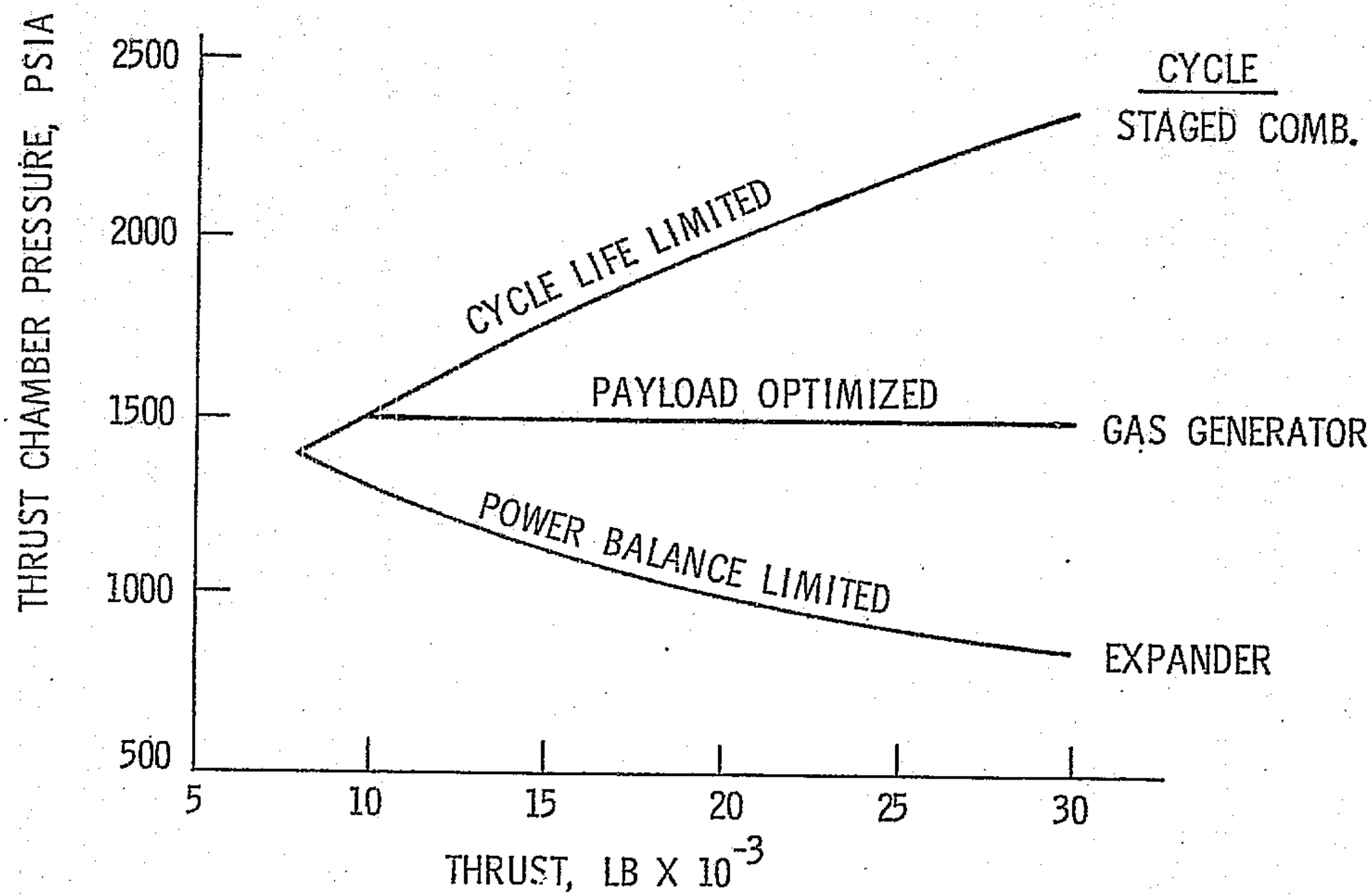
The performance and weight data for the various cycle candidates were calculated in order to perform the system tradeoffs. The data was evaluated as a function of thrust, at the chamber pressure established by the cycle analysis and for a maximum engine length with the extendible nozzle retracted (stowed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

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:

	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 AMOTV payload partials were used because of the importance of high specific impulse.

Because the reliability and safety analysis concluded that a minimum of two engines are required, the data presented emphasizes the twin engine installation.



ALL CONCEPTS CYCLE LIFE LIMITED BELOW 8K LBF

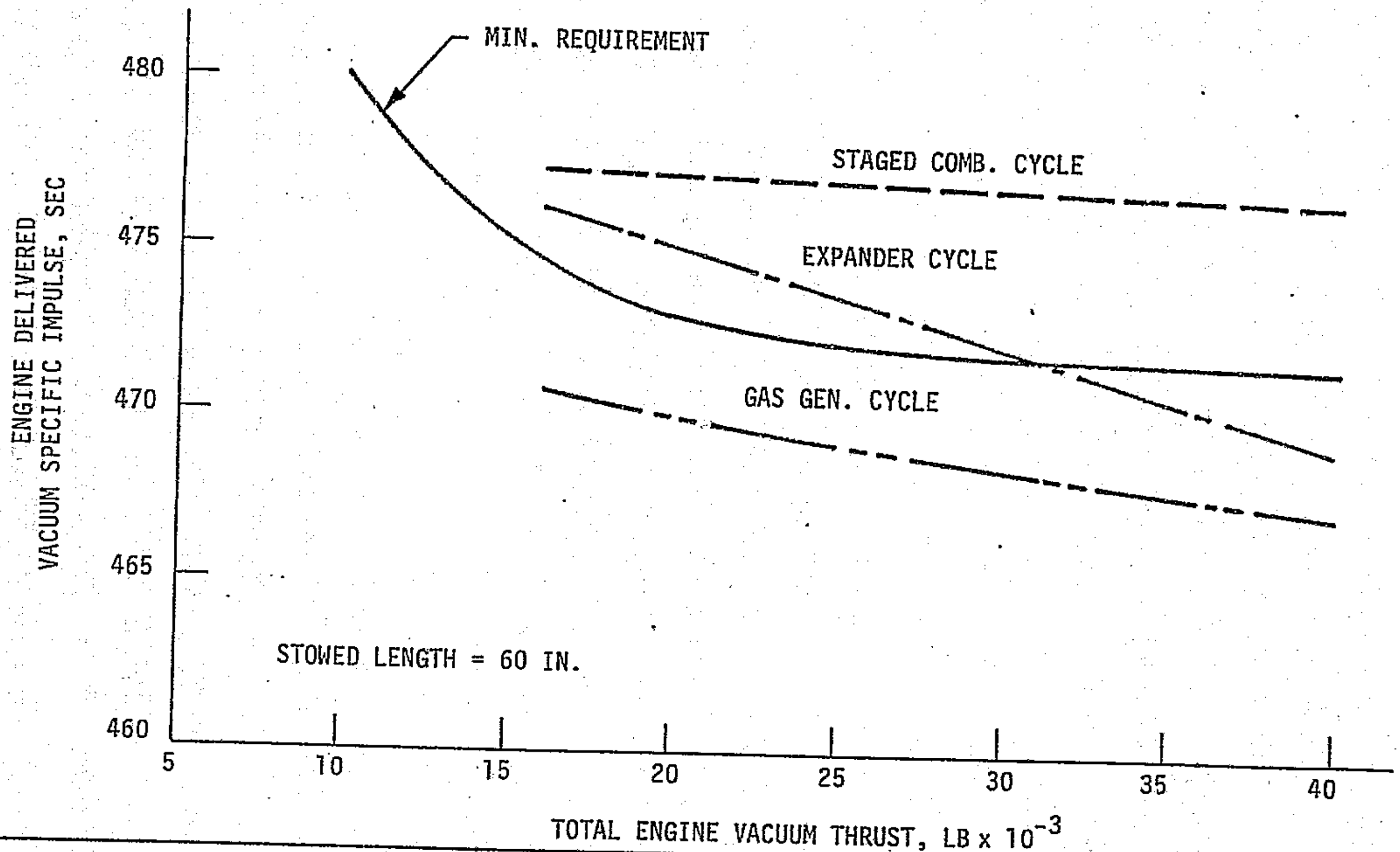
Figure 9. Maximum Engine Operating Pressures

Engine performance for a twin engine installation is shown on Figure 10. Both the staged and expander cycle engines are capable of meeting the minimum specific impulse requirements in this configuration in a total thrust range of approximately 15,000 to 30,000 lbs. As expected, the staged combustion cycle has the highest performance. The gas generator cycle fails to meet the minimum performance requirements over the total thrust range.

Twin engine installation weight data is shown on Figure 11. The weight data shown includes series valve redundancy and redundant igniters which were major reliability and safety analyses recommendations. 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 cycles require more components than an expander cycle engine.

The results of the trade-off analyses for twin engines using the AMOTV payload partials are shown on Figure 12. The payload changes were computed against the minimum specific impulse requirement and a single (20K lb thrust) staged combustion cycle 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.

The results of all trade-off analyses for one, two and three engines are shown on Figure 13. A staged combustion cycle engine is superior only in a single engine installation. However, a single engine is unacceptable for crew safety. Multiple engine installation payload capabilities of staged combustion and expander cycle engines are essentially a "draw".



STAGED AND EXPANDER CYCLES MEET MIN. I_S REQUIREMENTS

Figure 10. Multiple Engine Installation Performance, Two Engines

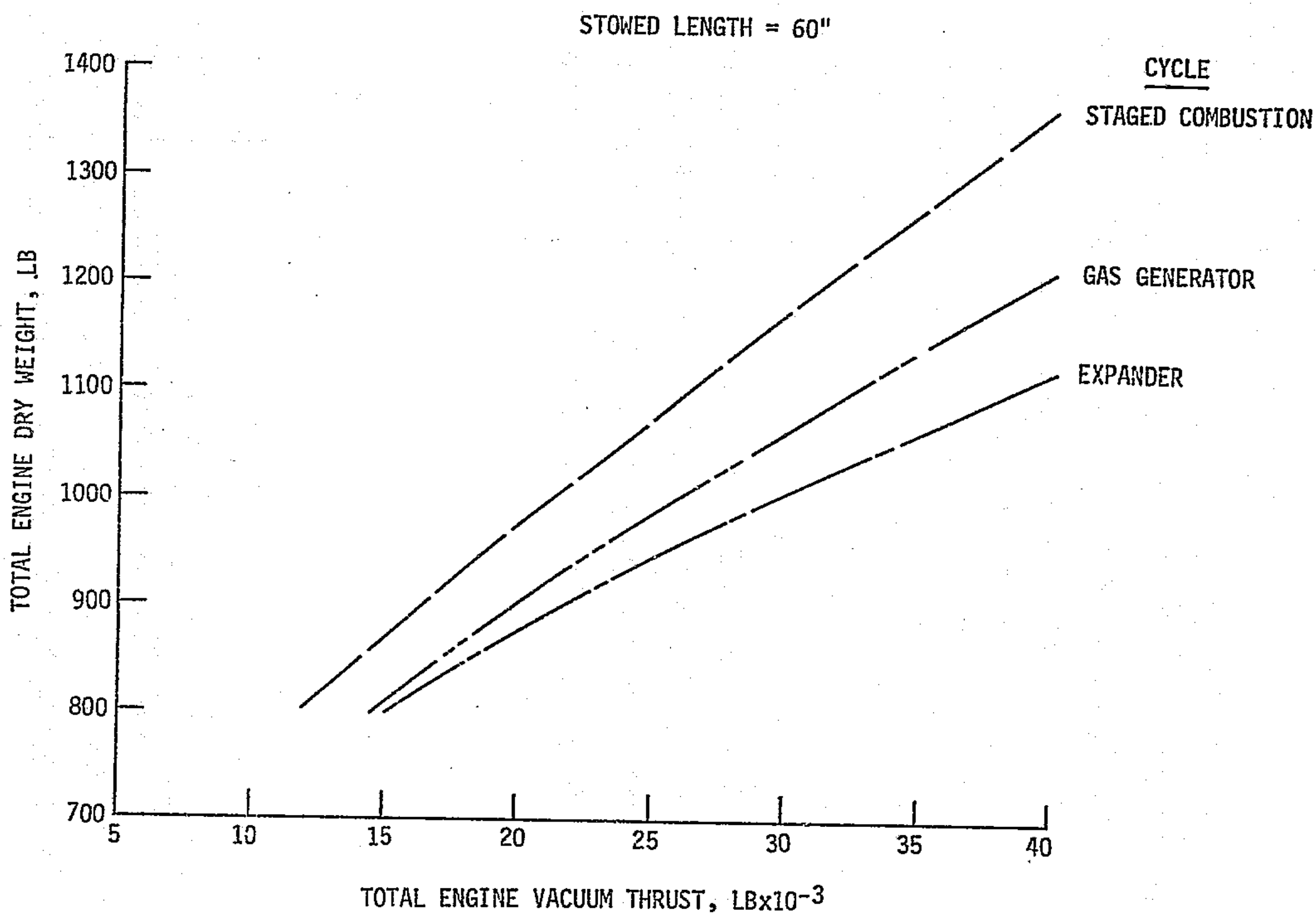
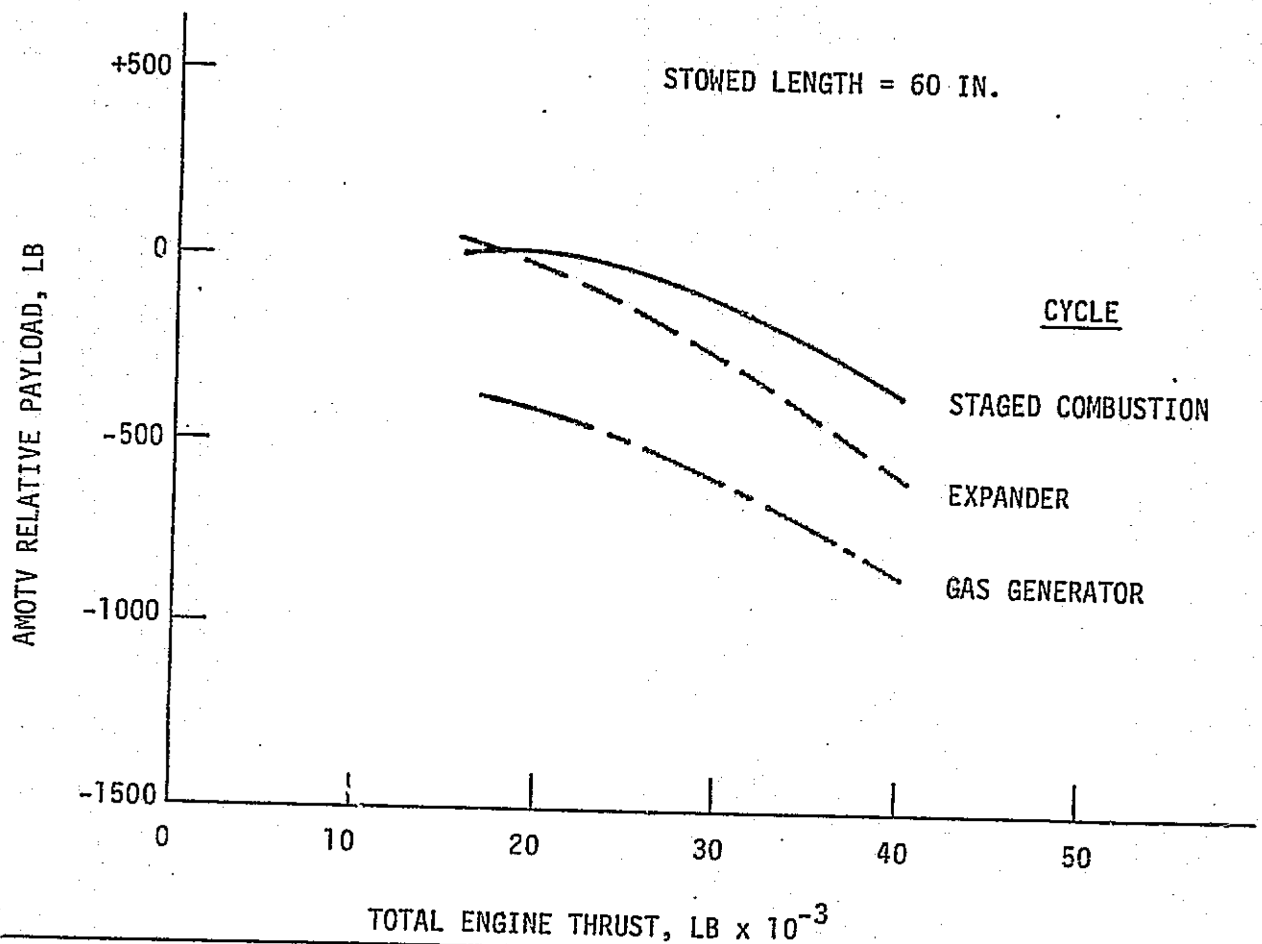


Figure 11. Twin Engine Installation Weight Comparisons



STAGED AND EXPANDER CYCLE PAYLOAD CAPABILITIES ABOUT EQUAL

Figure 12. Engine Cycle Payload Comparisons, Two Engine Installation

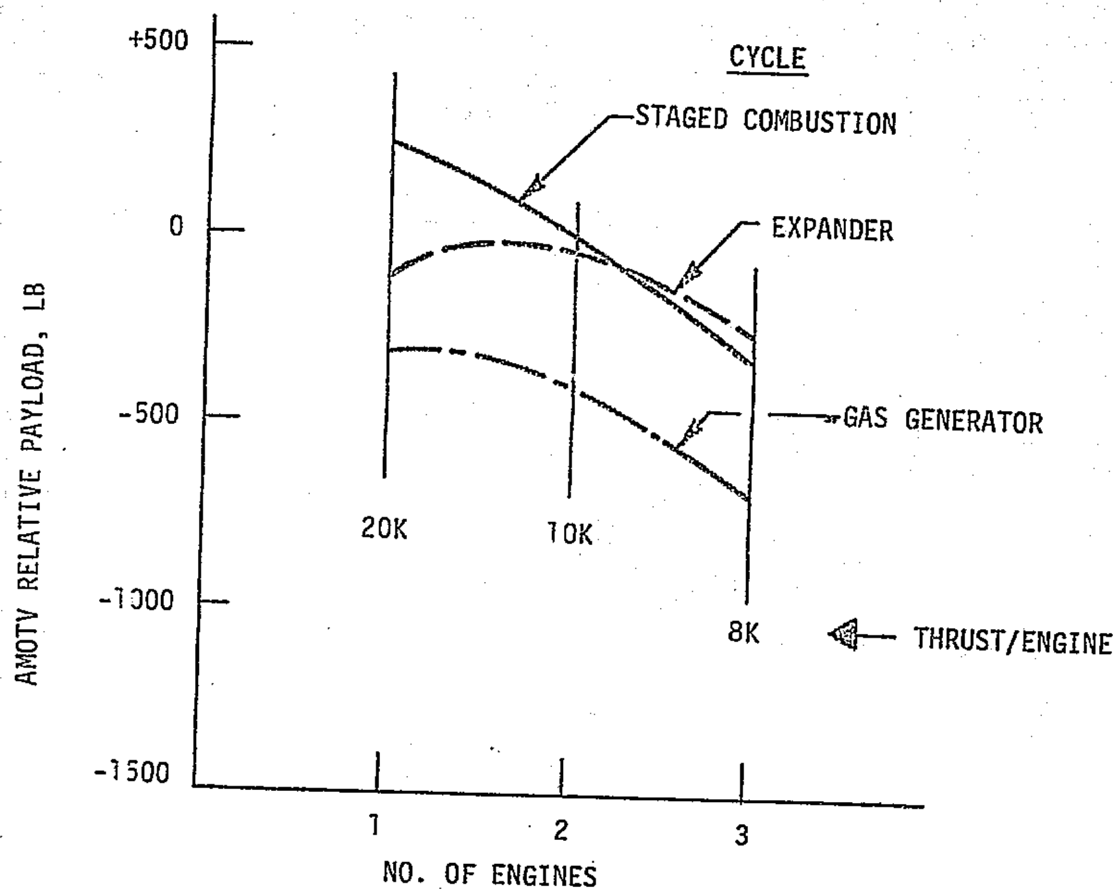


Figure 13. Effect of Number of Engines Upon Relative Payload Capability

The payload capability of multiple gas generator cycle engines is approximately 500 lb less than the other two candidates. The payload capability for all engine concepts is reduced as the number of engines are increased because of the weight penalties incurred with multiple engines.

C. CONCEPT SELECTION

Because the staged combustion and expander cycle engine payload comparisons resulted in a technical stalemate, other factors had to be considered in the concept selection. These factors were development risk and life cycle cost. A preliminary evaluation of these factors was made in ALRC in-house studies. Further assessments are planned in an extension to the contract. The results of the initial analyses showed that the expander cycle development risk is lower than either the staged combustion or gas generator cycles primarily because a critical combustion device and hot gas turbine drive is eliminated. This also results in substantial cost savings. These cost results are supported by the data generated in Task VII, Cost Estimate.

Based upon the engine requirements review and the safety, reliability, relative payload, risk and cost evaluations, an engine concept was selected as a baseline for the remaining study efforts. The recommended configuration was a twin, 10K lb thrust per subassembly, expander cycle engine. This recommendation was approved by NASA/MSFC at the conclusion of Task II. The conclusions are summarized on Figures 14 and 15.

Crew safety dictates a minimum of two engines. Single engines are not acceptable for the manned missions. The payload trades show that 8K to 12K lb thrust engines provide the best multiple engine installation. In this thrust range, the expander and staged combustion cycle engines have approximately the same payload capability. A total thrust of 20,000 lbs is about optimum hence, two 10K lb thrust engines is a good design point. The

- 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 14. 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 15. Engine Requirements and Concept Selection Study Conclusions

gas generator cycle engine would appear to be eliminated because of its low performance and resulting payload penalties.

The expander cycle offers the least development risk in terms of potential schedule and cost overruns because of the elimination of a critical combustion component. This also means that the expander cycle turbines operate in a benign environment. The reduced component count and lower risk obtained with an expander cycle also mean that large savings in total life cycle costs are possible.

The development of a new advanced expander cycle is necessary because of the man-rating, high performance, reusability and long life requirements identified for the OTV engine.

VI TASK III: PARAMETRIC DATA

The primary objective of this task was to provide engine performance, weight, and envelope parametric data for the three engine cycle concepts. The parametric ranges considered in this study are shown on Table I. 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 power balance, cycle life or payload trade-off study results. 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. Some of these analyses and their results are discussed briefly in this section. Primary emphasis was placed upon the recommended expander cycle engine concept.

A. SUPPORTING ANALYSES

1. Performance Analysis

Engine delivered performance data were calculated for an energy release efficiency (ERE) goal of 99.5% and minimum chamber length requirements were established accordingly. A chamber length of 18 inches was selected for the expander cycle engine for power balance reasons. Figure 16 shows that this length is more then adequate to meet the ERE goal. 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. Chamber lengths of approximately 8 and 13 inches were established for the staged combustion and gas generator cycle engines, respectively. The chamber length is longer for the gas generator cycle because the state of the propellants is liquid-liquid rather than liquid-gas.

TABLE I

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

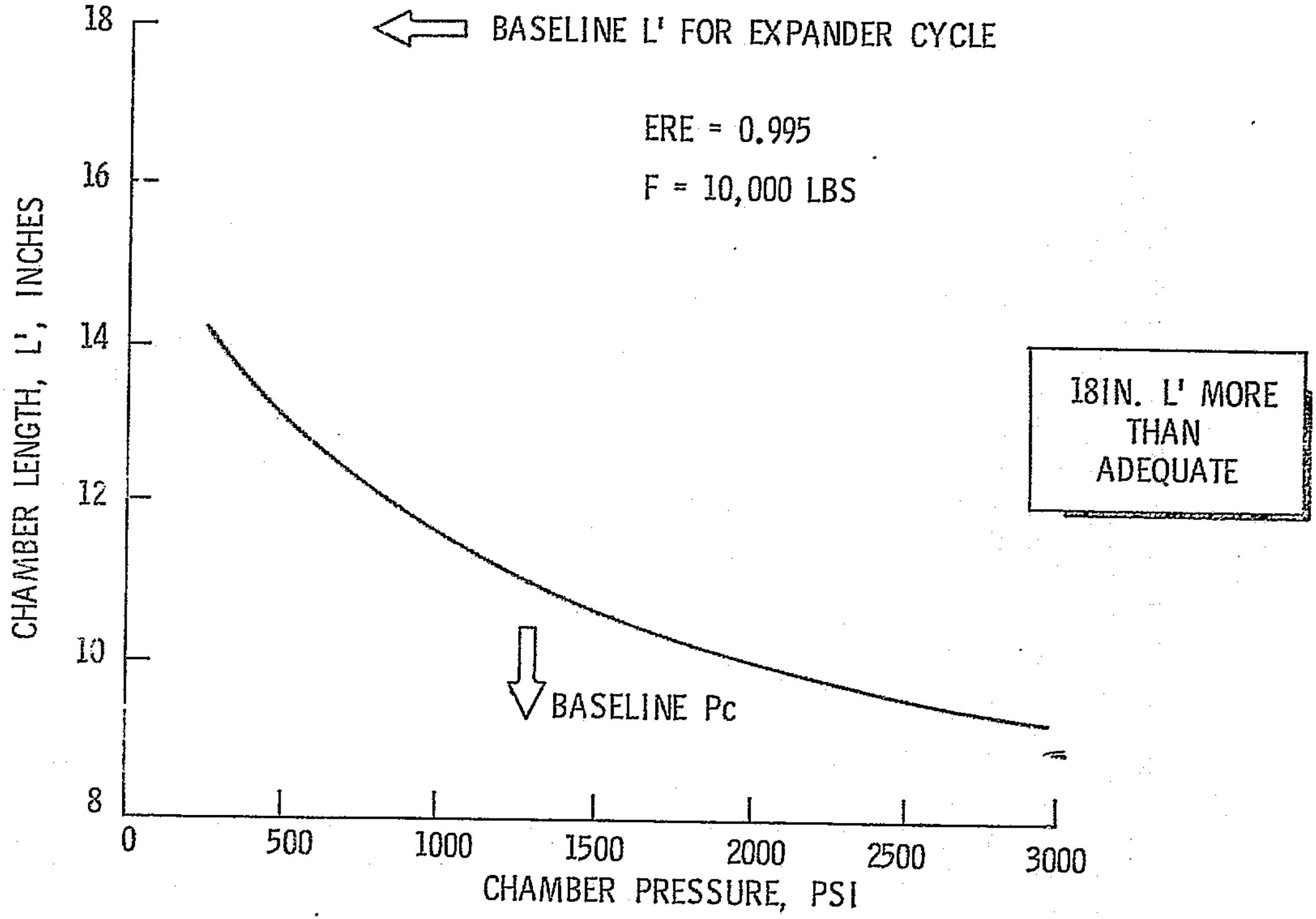


Figure 16. Effect of Chamber Pressure Upon Chamber Length Requirements

Simplified JANNAF performance prediction techniques (CPIA Publication 246) 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 for the ASE at an area ratio of 400:1, a thrust level of 20,000 lb and a chamber pressure of 2000 psia. For these test conditions, the experimental data indicates that the old procedures predicted a boundary layer loss approximately 4 seconds too high.

2. 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 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.

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 17 which was established by computer solutions for many other designs. The detailed analyses results fit the historical base well, as shown on the figure. Therefore, the design curve was used to preduct strain at other points in the chamber and for the parametric studies.

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 18 for a range of allowable strains. A strain of 1.86% was predicted for the initial analysis. This allowable temperature differential data was used in the thermal analyses 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 19. Similar figures were constructed 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.

3. 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.

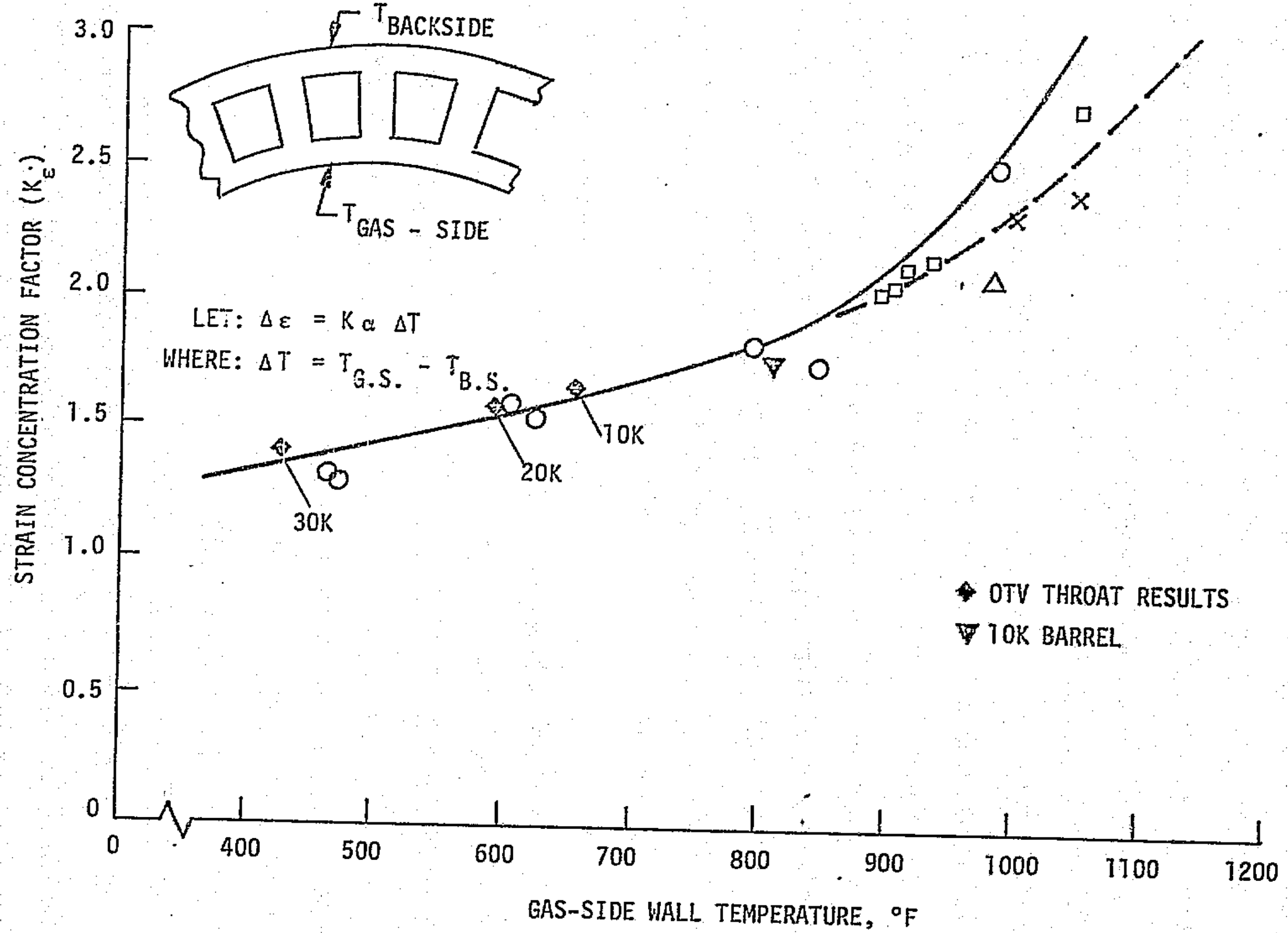


Figure 17. Predicted Strain Concentration Factor vs Gas-Side Wall Temperature

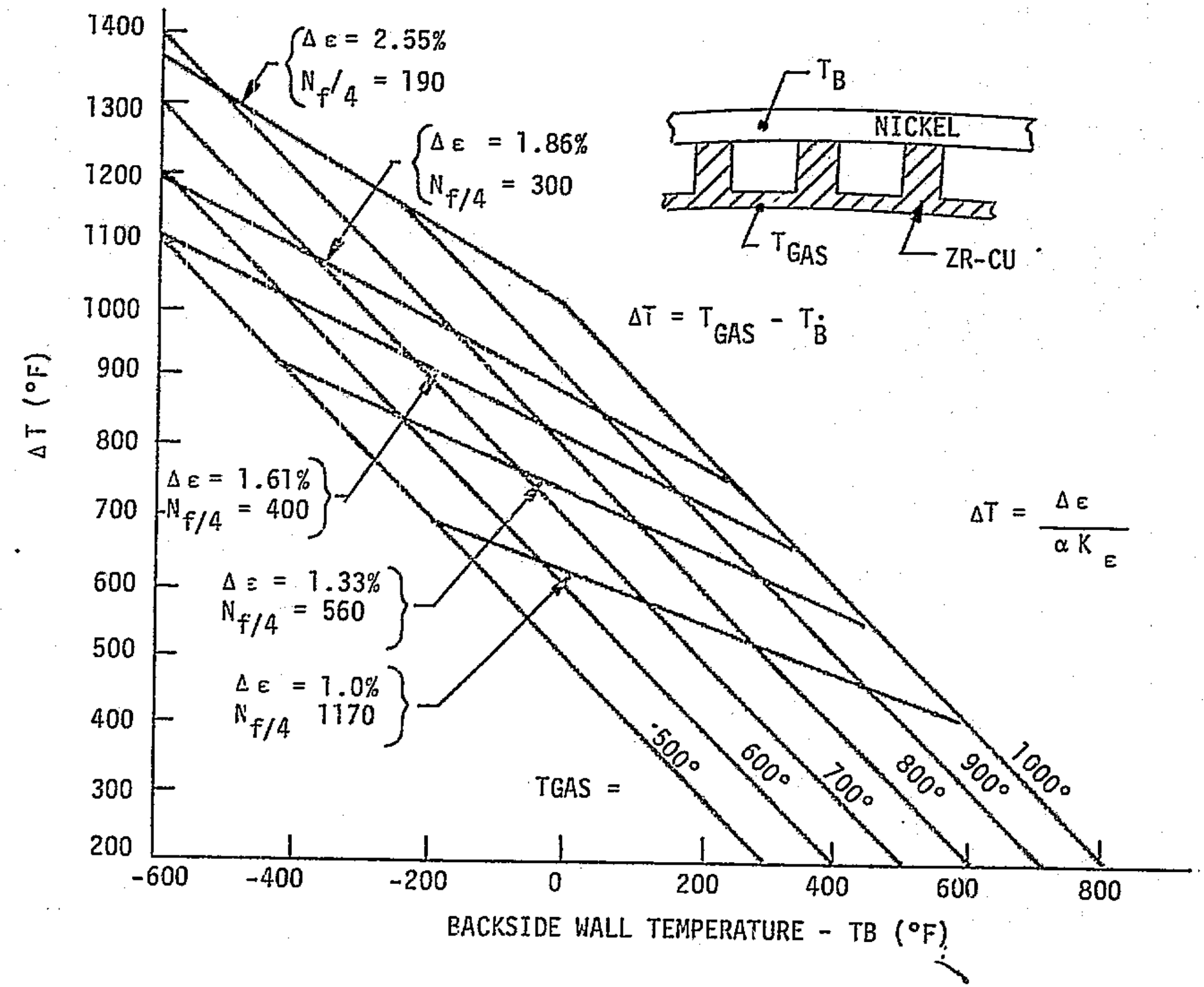


Figure 18. Allowable Temperature Differential

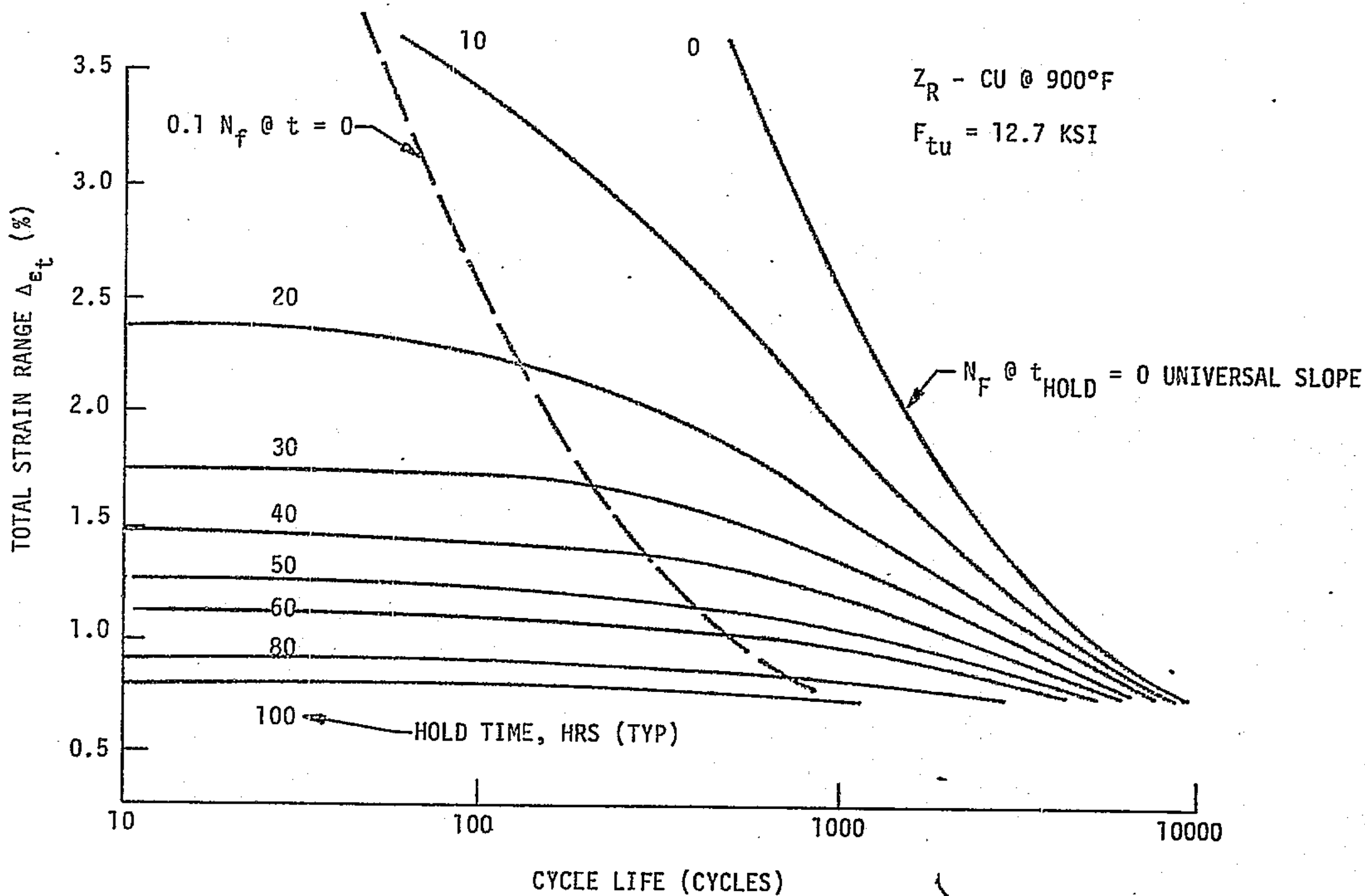


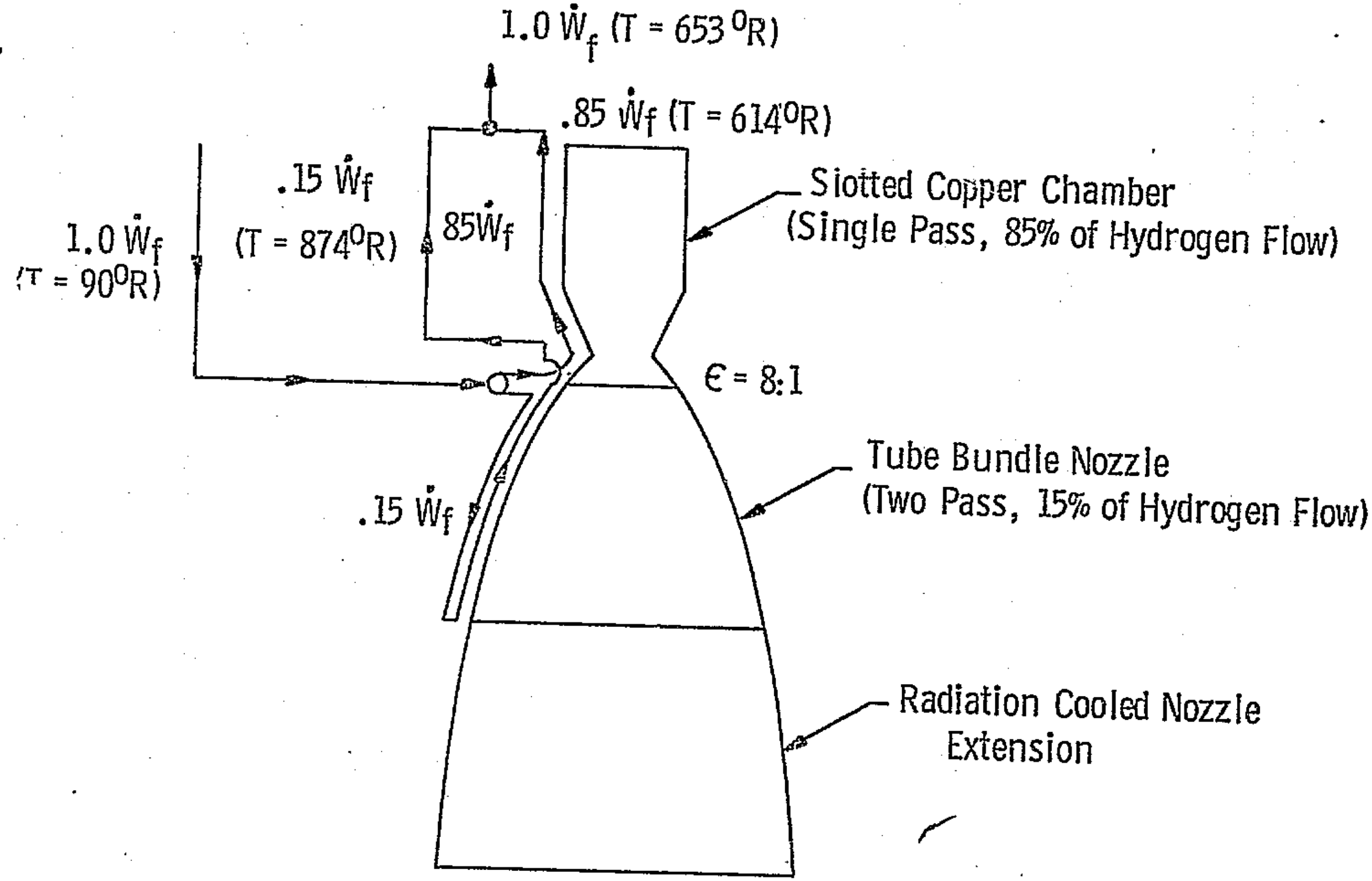
Figure 19. Total Strain Range vs Cycle Life

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 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 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. Cooling of the nozzle with 15% of the flow, that is in parallel with the chamber coolant, results in approximately 14 psi less chamber pressure drop than cooling the chamber with the total hydrogen flow.

The selected coolant flow schematic used in the Task III evaluations is shown on Figure 20. 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 results of thermal analyses conducted to support the power balance efforts are summarized on Figure 21. The figure shows that the turbine inlet temperature increases with decreasing thrust level and increasing



Note: Temperatures are shown for 10KLBF baseline engine.

Figure 20. Advanced Expander Cycle Engine Coolant Flow Schematic

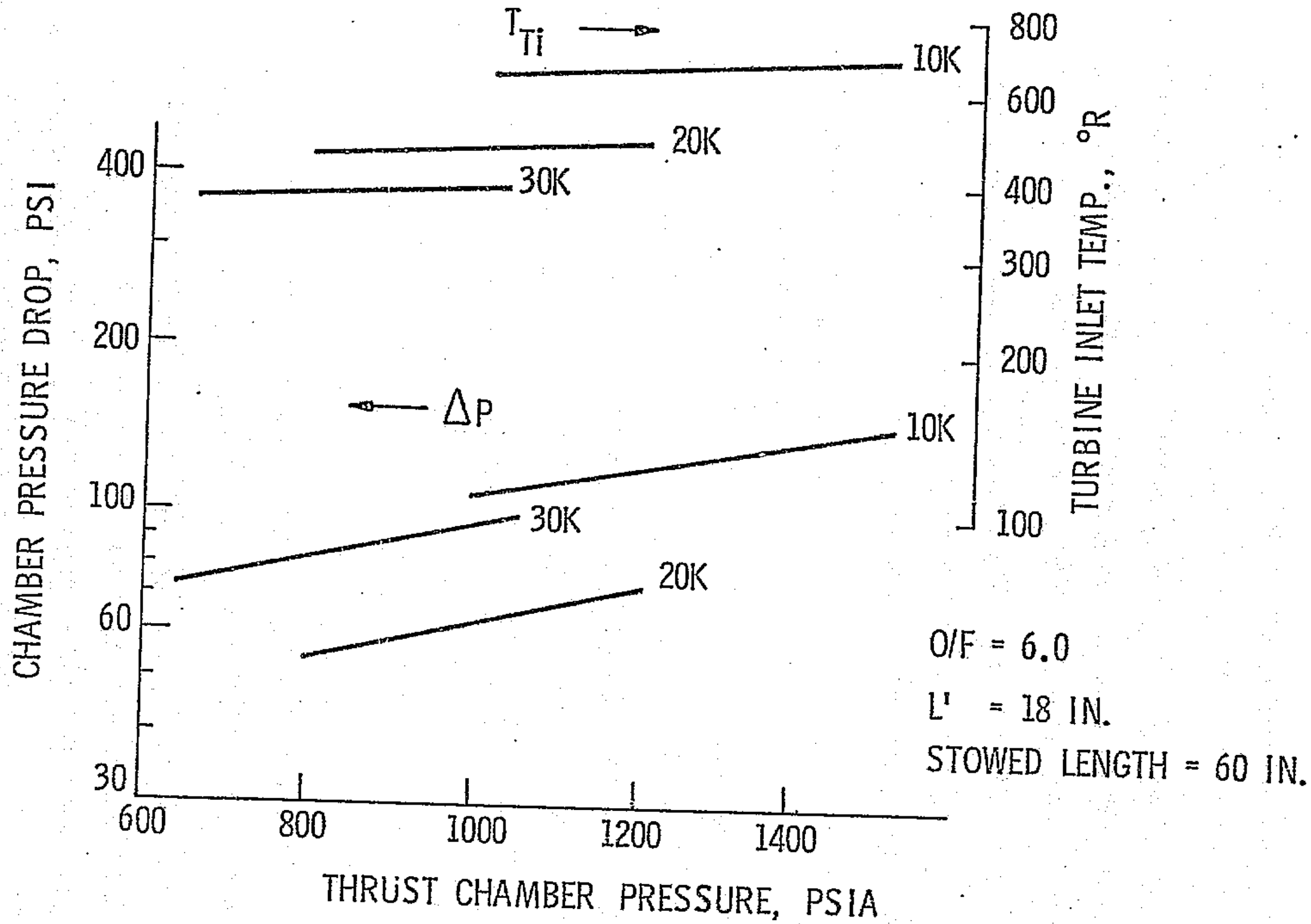


Figure 21. Expander Cycle Parametric Thermal Analysis

chamber pressure. The chamber coolant jacket pressure drop is also the highest at 10K lb thrust. 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.

The thermal analysis data results for the ultimately selected Task III operating chamber pressures are summarized on Table II. 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 to minimize the stowed engine length. This occurs because the fixed nozzle area ratio for longer chamber lengths becomes smaller than the radiation cooled nozzle minimum attachment area ratio criteria. These cases are noted on the table. For engine stowed lengths of 60 and 70 inches, an 18 inch long chamber can be used at all thrust levels. The shorter chamber lengths result in much lower turbine inlet temperatures and affect the power balance and parametric data at these points.

The engine is required to operate over a mixture ratio range of 6.0 to 7.0. In addition, one time emergency operation at a mixture ratio of 10.0 was evaluated. The thermal analysis results for these operating conditions are summarized on Table III. Similar data was generated at other stowed lengths and thrust levels for the O/F's of 6.0 to 7.0 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

TABLE II

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 ΔP, psia	Nozzle ΔT, deg.	Coolant Outlet Temp., °R
10,000	1300	50	18	131	524	19	661	635
	1300	60	18	131	524	12	784	653
	1300	70	18	131	524	14	896	670
20,000	1100	50	16	73	310	9	460	422*
	1100	60	18	76	340	9	563	463
	1100	70	18	76	340	8	659	477
30,000	950	50	12	87	205	7	378	321*
	950	60	18	91	269	6	423	382
	950	70	18	91	269	7	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.

TABLE III

OFF-DESIGN OPERATION THERMAL ANALYSIS SUMMARY (EXPANDER CYCLE)

F = 10,000 lbs

Stowed Length - 60 in.

Mixture Ratio	Chamber Pressure, psia	Chamber Pressure Drop, psi	Chamber Coolant ΔT, Deg.	Turbine Inlet Temp., °R
6.0	1300	131	524	653
6.5	1278	120	542	668
7.0	1261	108	556	680
10.0	1260	90	763	856

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. This is also true for the emergency one-time operation at a mixture ratio of 10. The results of the low cycle thermal fatigue analysis show that the off-design operation at this O/F would have a negligible affect on the chamber service life.

4. 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 main oxygen and hydrogen pump operating specifications are shown on Table IV. A three stage hydrogen pump is used to achieve a reasonable specific speed (N_S) and efficiency. Studies have shown that pump efficiencies drop off rapidly at N_S values less than 600. The hydrogen turbopump is bearing DN limited. The hydrogen bearing DN limit used in this study was 2×10^6 (RPM) (MM). A minimum bearing size of 20 MM was used. The oxygen turbopump operates a maximum suction specific speed limit of 20,000. A single stage oxygen pump is used. The rotating speed was reduced from a maximum of 75,000 RPM (DN limit = 1.5×10^6 RPM x MM) in order to obtain a reasonable impeller size and efficiency. Similar data was established in the study at thrust levels of 20 and 30K lbf.

The expander cycle engine turbine operating specifications and efficiencies are shown on Table V. The hydrogen pump turbine is a partial admission (13% admission) machine with a blade diameter of 3.55 in. and a mean blade height of 0.3 in. The oxygen pump uses a Terry turbine. This is

TABLE IV

MAIN PUMP OPERATING CHARACTERISTICS SUMMARY (EXPANDER CYCLE)

F = 10,000 lbs

	OXYGEN	HYDROGEN
INLET PRESSURE, PSIA	46.6	51.0
NPSH, FT	64.1	1080
VOLUMETRIC FLOW, GPM	114.0	307.1
SUCTION SPECIFIC SPEED, $(RPM)(GPM)^{1/2}/(FT)^{3/4}$	20000	9300
SPEED, RPM	42,440	100,000
DISCHARGE PRESSURE, PSIA	1585	3130
NUMBER OF STAGES	1	1
SPECIFIC SPEED, $(RPM)(GPM)^{1/2}/(FT)^{3/4}$	1085	706
IMPELLER TIP DIA., IN.	2.50	3.33
EFFICIENCY, %	62.0	64.7

TABLE V

EXPANDER CYCLE TURBINE OPERATING

CHARACTERISTICS SUMMARY

$F = 10{,}000$ LB

	LOX TPA	LH_2 TPA
INLET TEMPERATURE, °R	653	653
INLET PRESSURE, PSIA	2850	2850
FLOW RATE, LB/SEC	0.735	2.095
SHAFT HORSEPOWER	173	878
GAS PROPERTIES		
C_p, BTU/LB - °R	3.53	3.53
γ	1.405	1.405
PRESSURE RATIO	2.02	2.02
EFFICIENCY, %	39.4	70.0
TURBINE BYPASS FLOW, LB/SEC (%)	0.18(6.0)	

a very low flow and low efficiency machine. The LOX TPA horsepower is low and therefore, the low efficiency does not have as big an impact on the power balance as the hydrogen TPA efficiencies have. The LOX turbine mean diameter is 4.38 in. Similar turbine operating specifications have also been prepared at 20 and 30K lbf in the study.

All the turbopump efficiency data established to support the Task III expander cycle power balance analyses is summarized on Figure 22. The figure shows that the efficiencies improve as the turbopumps get larger. The hydrogen pump efficiency levels off because of a change from three-stage to two-stage pumps between 20 and 30K lb thrust.

5. 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 23. 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.

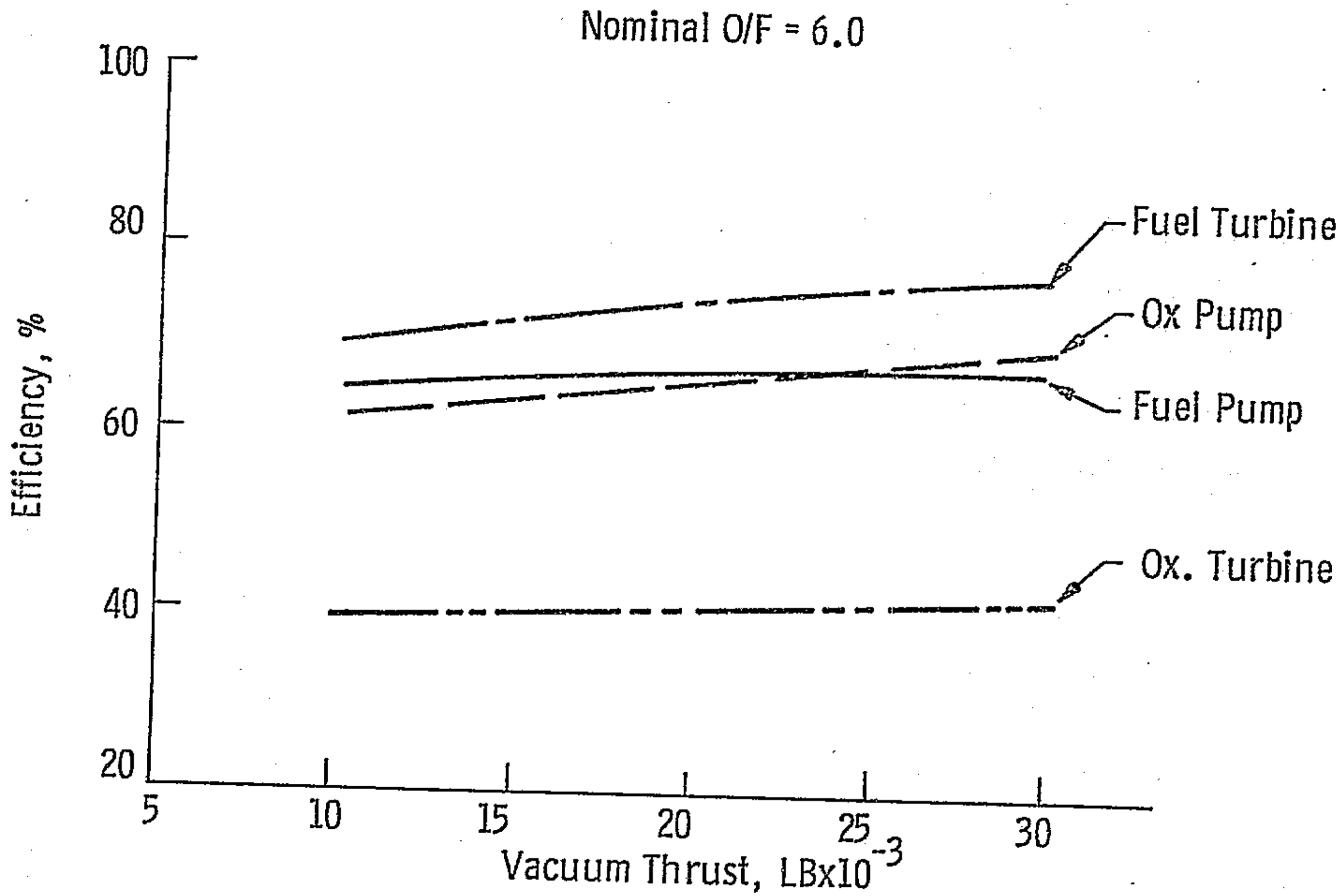


Figure 22. Advanced Expander Cycle Engine Turbomachinery Efficiencies

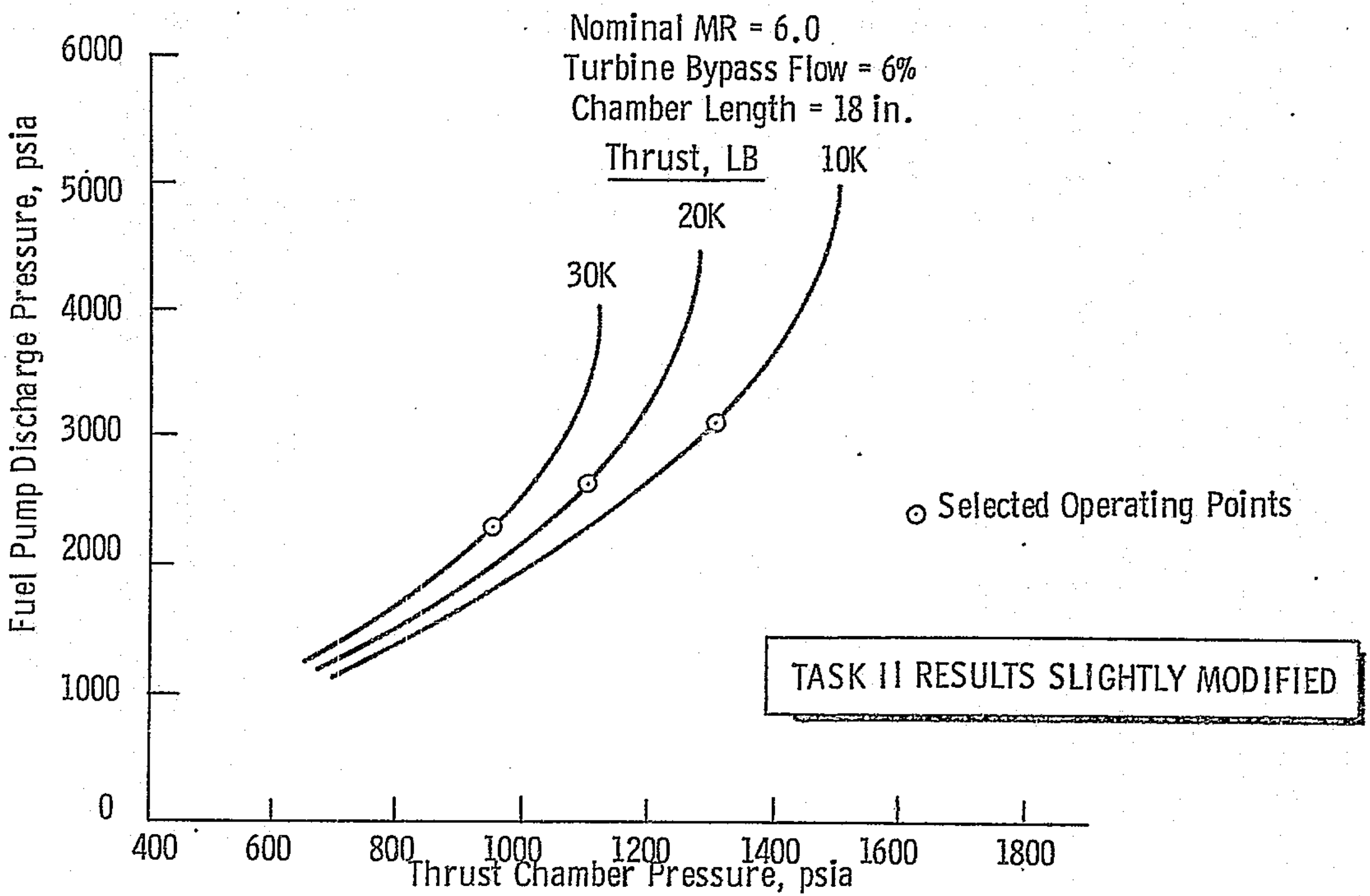


Figure 23. Task III Expander Cycle Power Balances

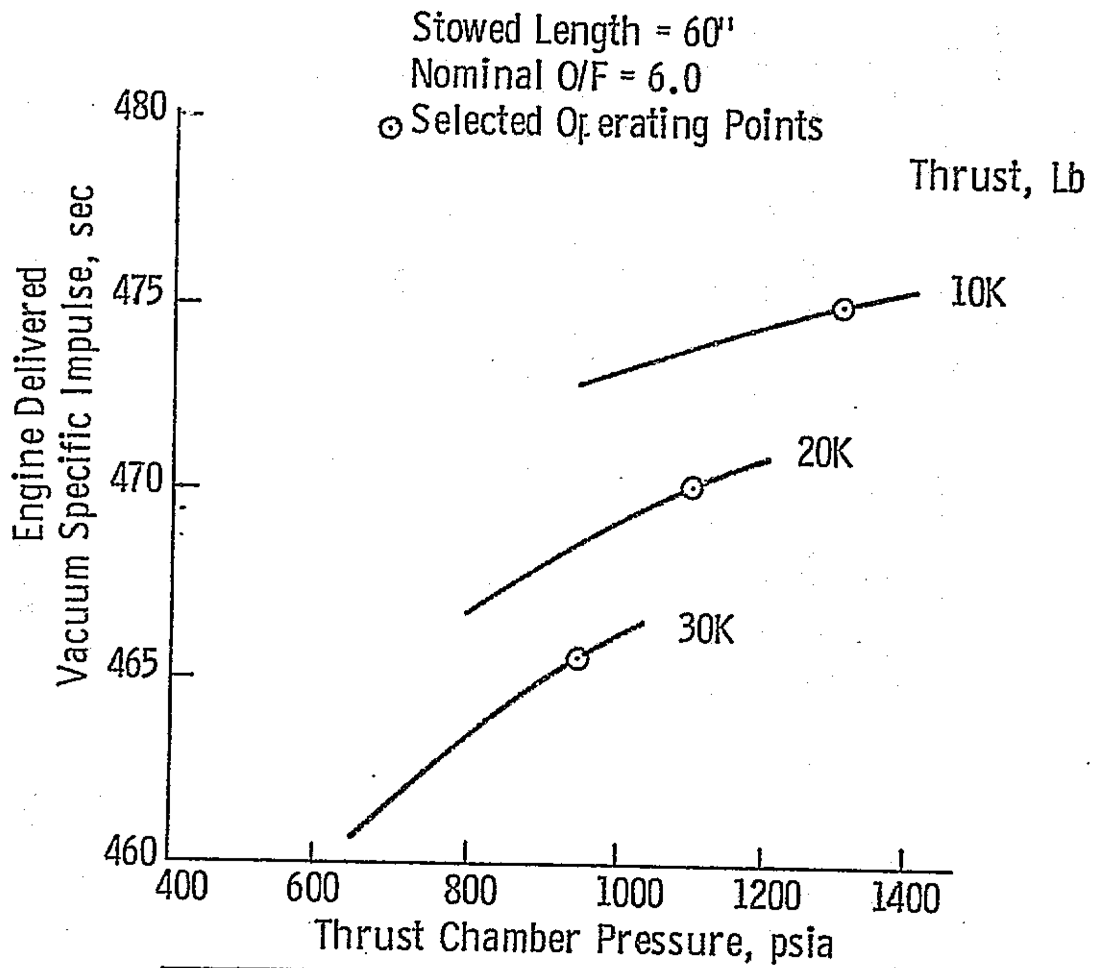
The operating thrust chamber pressures were selected at a point where the rate of change in the hydrogen pump discharge pressure with chamber pressure is small. The selected pressures have the same sensitivity at all thrust levels.

Further sensitivity analyses were conducted for the expander cycle by calculating the delivered performance as a function of thrust and thrust chamber pressure. Figure 24 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 of ODE specific impulse at these high area ratios (i.e., ∿800:1) 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 recommended baseline advanced expander cycle engine characteristics that evolved from this study are shown on Figure 25. 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 (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



SENSITIVITY LEAST AT 10KLB THRUST

Figure 24. Expander Cycle Performance Sensitivity to Operating Chamber Pressure

- 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 25. Recommended Baseline Advanced Expander Cycle Engine Characteristics for Twin Engine Installation

The expander cycle engine schematic is shown on Figure 26. 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 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.

B. TYPICAL PARAMETRIC DATA

The engine performance and weight parametric data is summarized in this section for each of the three cycle candidates. The operating chamber pressures selected to conduct the parametric studies for each candidate engine concept are shown on Table VI. The expander cycle engine is power balance limited and pressures were selected on the basis of the Task III cycle analysis results (Section VI,A,5). The staged combustion and gas generator cycle operating pressure levels were established in Task II (Section V,A) and used for the parametric analysis. Because the Task II, Concept Definition, results showed that the low performance of the gas generator cycle engine would preclude its use, data on this cycle was only generated for a nominal 60 inch stowed length.

The parametric engine data presented herein, 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 data presented at 10K should be doubled.

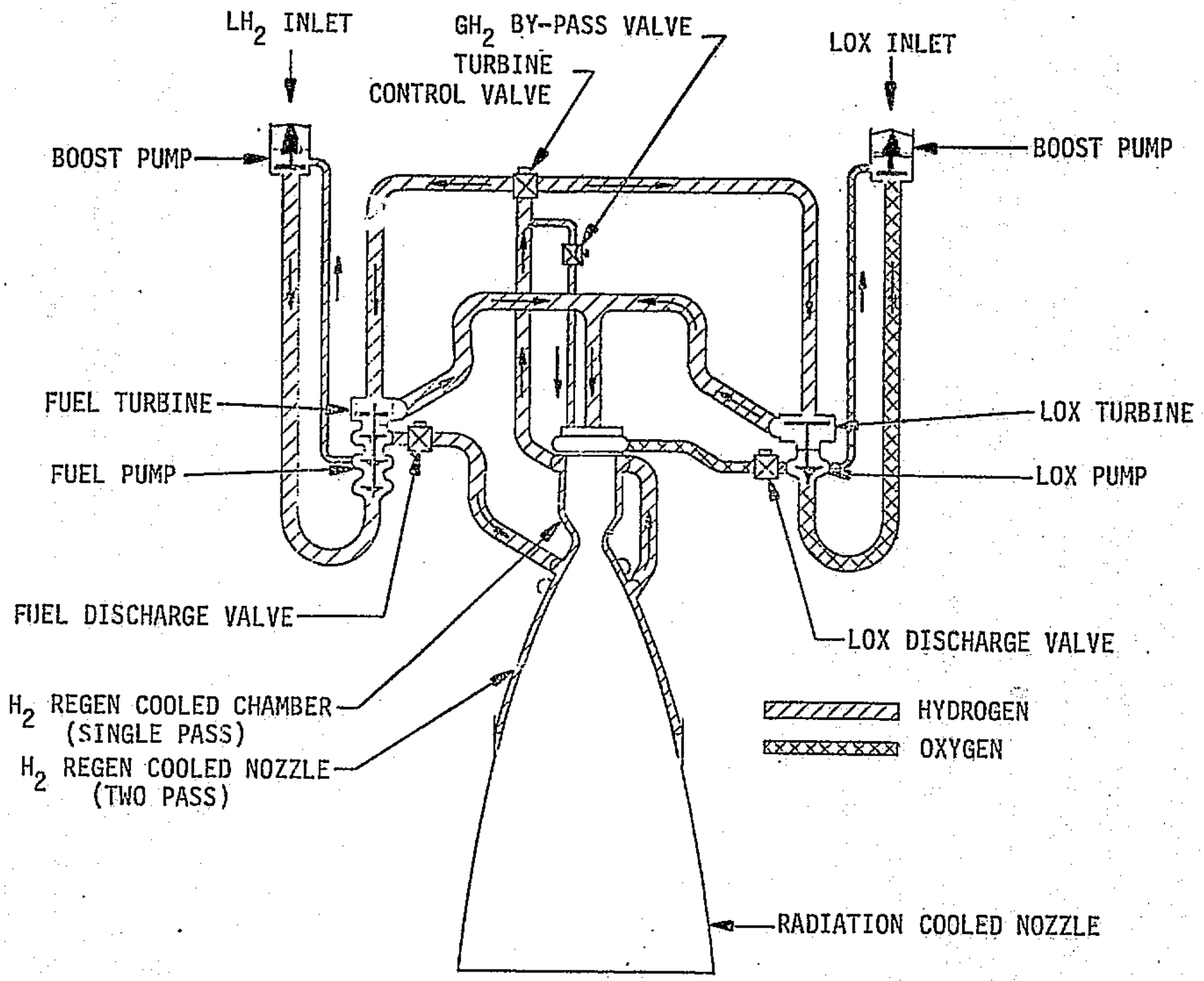


Figure 26. Task III Advanced Expander Cycle Engine Schematic

TABLE VI

TASK III CANDIDATE CYCLE OPERATING PRESSURES

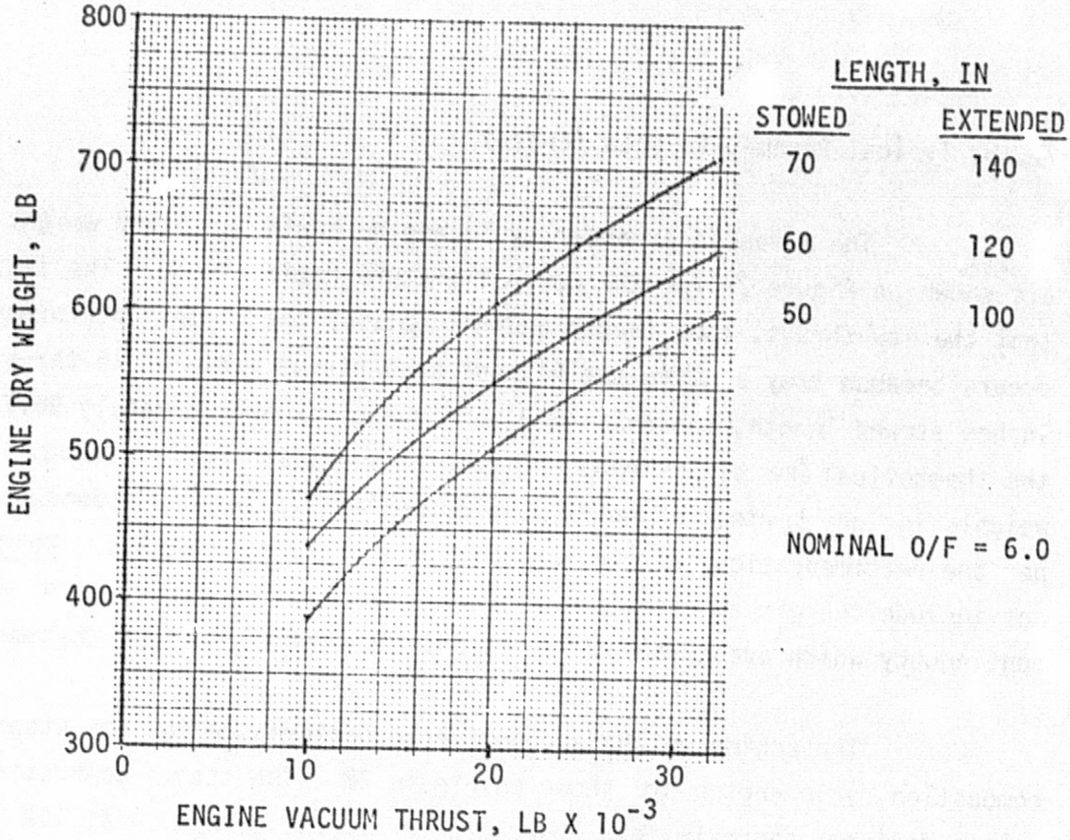
Cycle	Vacuum Thrust, Klb	Thrust Chamber Pressure, psia	Criteria
Expander	10	1300	Task III Power Balance
Expander	20	1100	Task III Power Balance
Expander	30	950	Task III Power Balance
Staged Combustion	10	1500	Cycle Life*
Staged Combustion	20	2000	Cycle Life*
Staged Combustion	30	2300	Cycle Life*
Gas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

*Same as Task II.

The advanced expander cycle engine performance and weight parametrics are shown on Figure 27 for the nominal mixture ratio of 6.0. The figure shows that the low thrust, long length engines deliver the highest performance. This occurs because they achieve the highest area ratios. At 10K lb thrust and 60 inches stowed length, the delivered performance is approximately 96.7% of the theoretical one dimensional equilibrium specific impulse value. The engine weights include series redundant main propellant valves and redundant igniters per the recommendations of the reliability and safety analyses. Weights do not include the gimbal actuators and actuation system, prevalves or a contingency which are normally included in the vehicle weight statement.

The engine weight and delivered performance for the staged combustion cycle engine are shown on Figure 28. The staged combustion cycle engine delivers approximately 2 sec greater specific impulse at 10K lb 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 gas generator cycle engine performance and weight data are shown on Figure 29. 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.



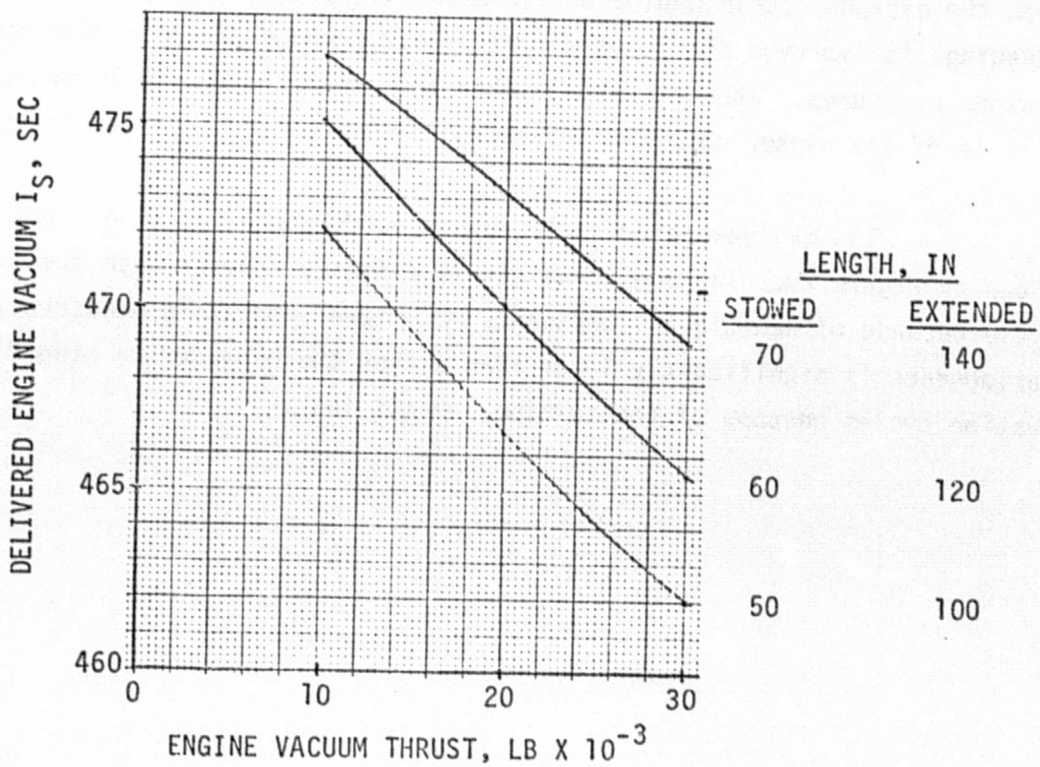
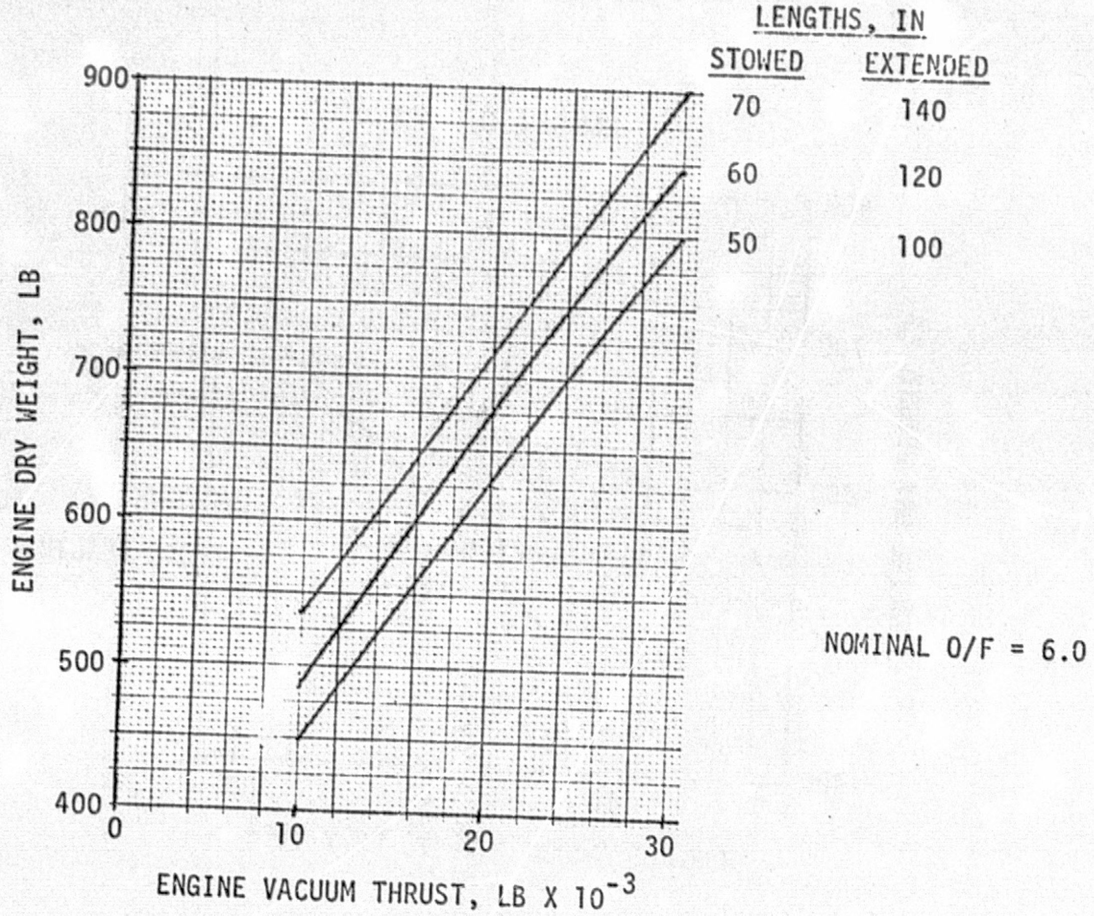


Figure 27. Advanced Expander Cycle Engine Weight and Performance Variations with Rated Thrust



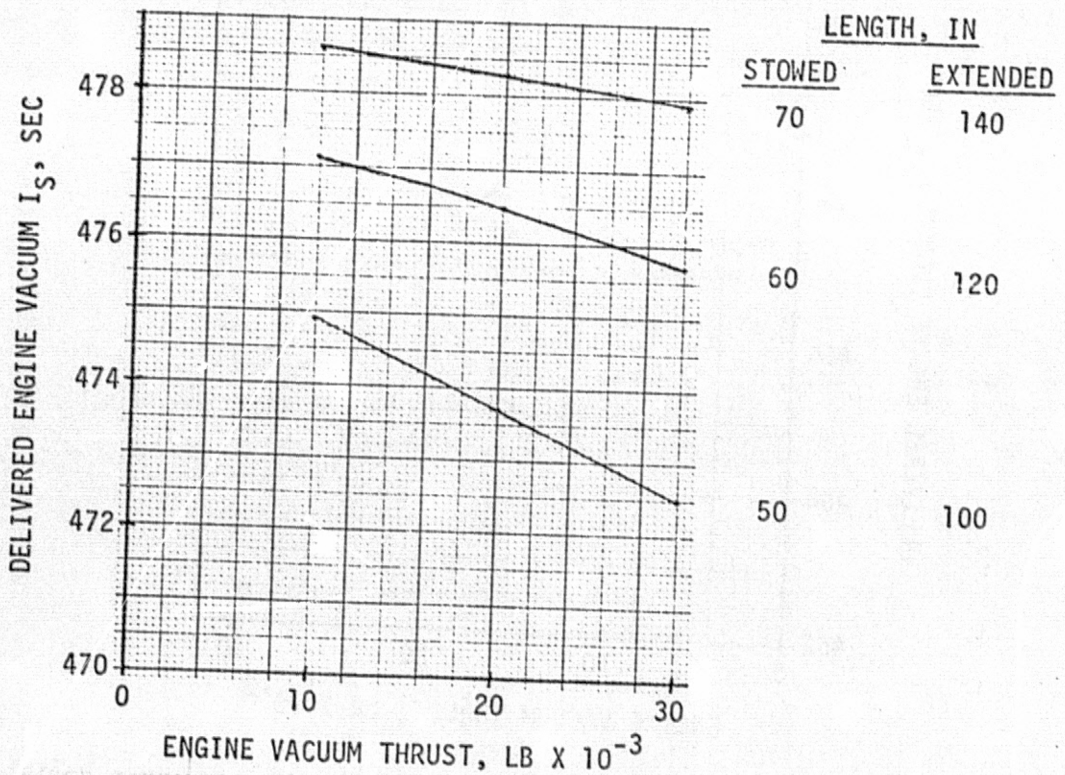


Figure 28. Staged Combustion Cycle Engine Weight and Performance Variations with Thrust

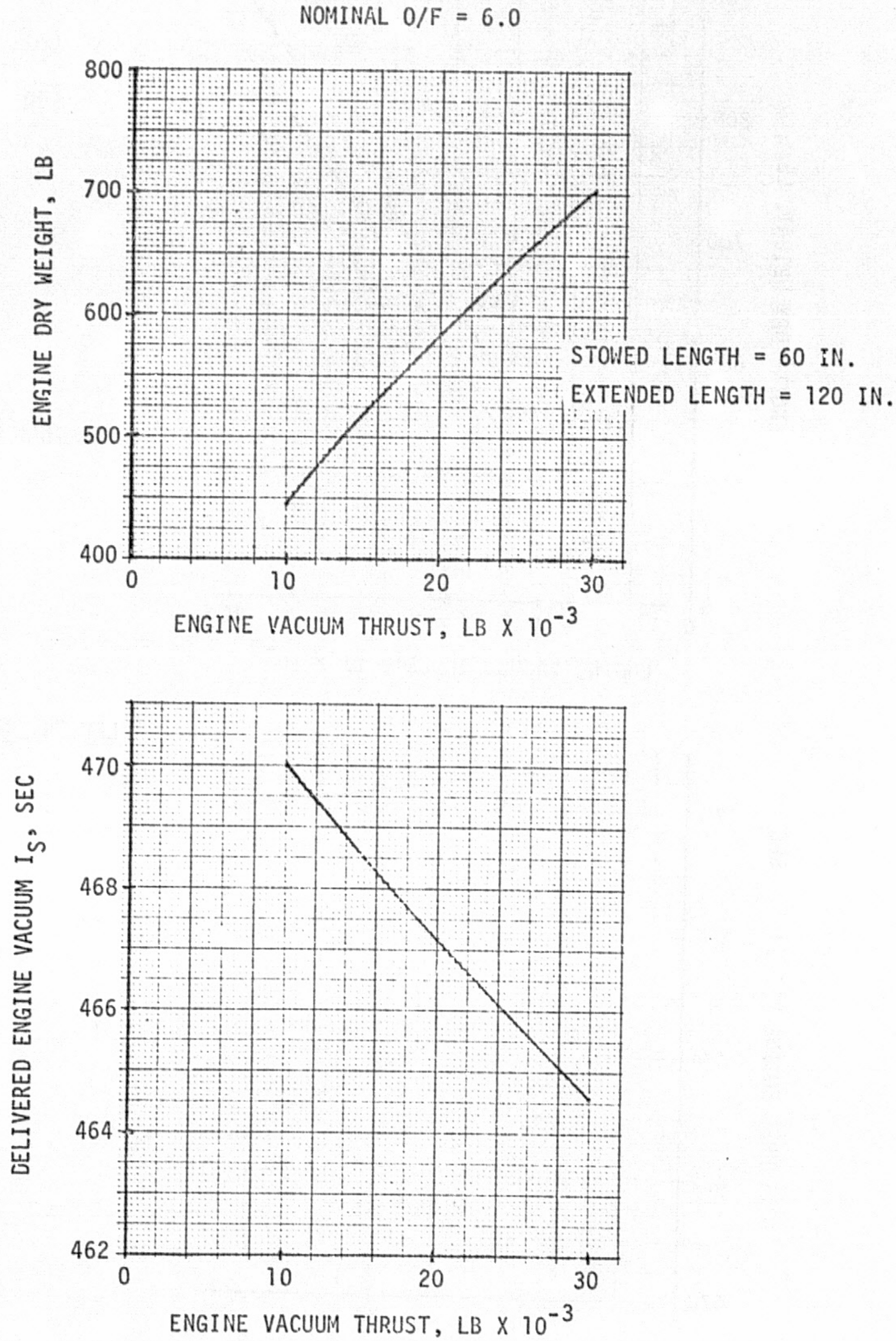


Figure 29. Gas Generator Cycle Engine Weight and Performance Variations with Rated Thrust

VII. 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.

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 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.

Engine cycle power balances and performance estimates were also 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. These analyses showed that the engine could operate at an O/F of 10 without degrading its service life.

VII, Task IV: Engine Off-Design Operation (cont.)

The engine performance data for all three off-design operating modes is summarized on Table VII.

TABLE VII

ADVANCED EXPANDER CYCLE ENGINE OFF-DESIGN OPERATION SUMMARY

	TANK-HEAD IDLE	PUMPED IDLE	EMERGENCY
THRUST, LB	37.3	2500	10000
CHAMBER PRESSURE, PSIA	6.0	334	1260
MIXTURE RATIO	4.0	6.0	10.0
VACUUM SPECIFIC IMPULSE, SEC	399.0	457.3	426.2
FLOW RATE, LB/SEC	0.0935	5.47	23.46

VIII TASK V: WORK BREAKDOWN STRUCTURE

The objective of this task was to establish a work breakdown structure (WBS) for use in Task VII, Cost Estimates. This was accomplished in concert with NASA/MSFC. 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 engine DDT&E phase is shown on Table VIII at the third WBS level. Cost estimates were made to the fourth WBS level and accumulated to this third level to summarize the information. The same WBS structure was used for each engine candidate except that item 1.1.3 is not applicable for an expander cycle engine. Therefore, costing was conducted for a consistent set of guidelines.

The items included in the production phase for costing purposes are shown on Table IX to the third WBS level. Costs were accumulated at the fourth level for items 1.2.1 and 1.2.3 but summarized to the third WBS level.

Items included in the engine operations phase are shown on Table X at the third WBS level. Operations costs were also estimated to the fourth WBS level for item 1.3.2.

TABLE VIII

WBS - ENGINE DDT&E (1.1)

1.1.1	Turbomachinery
1.1.2	Main Combustion Chamber
1.1.3	Preburner/Gas Generator*
1.1.4	Nozzle Assembly
1.1.5	Controls
1.1.6	Pressurization
1.1.7	Propellant Systems
1.1.8	Initial Tooling
1.1.9	Gound Support Equipment
1.1.10	Test
1.1.11	System Engineering
1.1.12	Project Management
1.1.13	Facilities
1.1.14	Consumables

*Staged/Gas Gen. Cycles Only

TABLE IX

WBS - ENGINE PRODUCTION (1.2)

1.2.1	Main Engines
1.2.2	Initial Spares
1.2.3	Facility Maintenance
1.2.4	Sustaining Engineering
1.2.5	Project Management
1.2.6	Consumables

TABLE X

WBS - ENGINE OPERATIONS (1.3)

1.3.1	Inplant Support
1.3.2	Field Support
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

IX. TASK VI: PROGRAMMATIC ANALYSIS AND PLANNING

The primary objectives of this task were to formulate preliminary project planning information, prepare schedules 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 30. 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 31, 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.

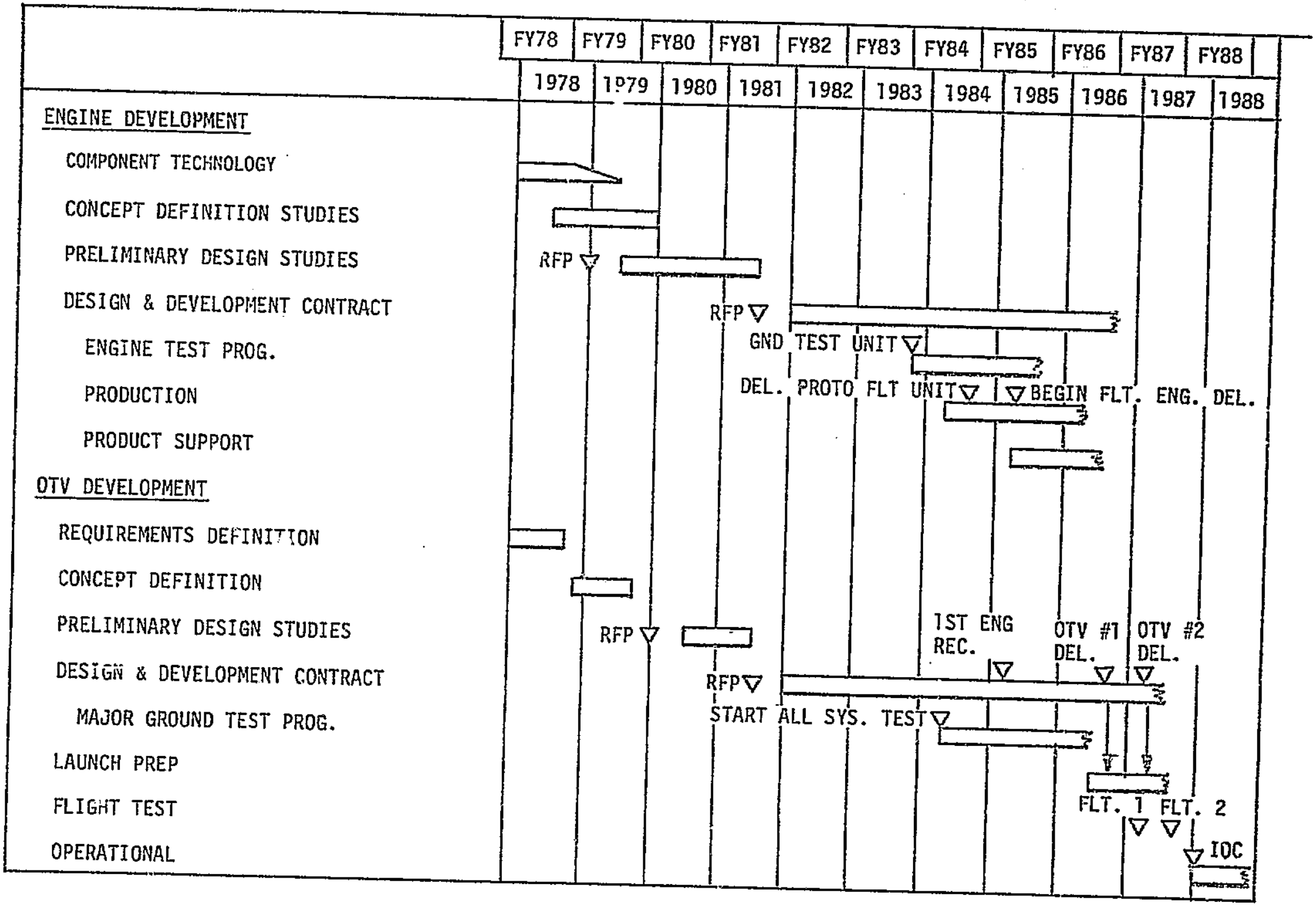


Figure 30. NASA Orbit Transfer Vehicle (OTV) Development Schedule

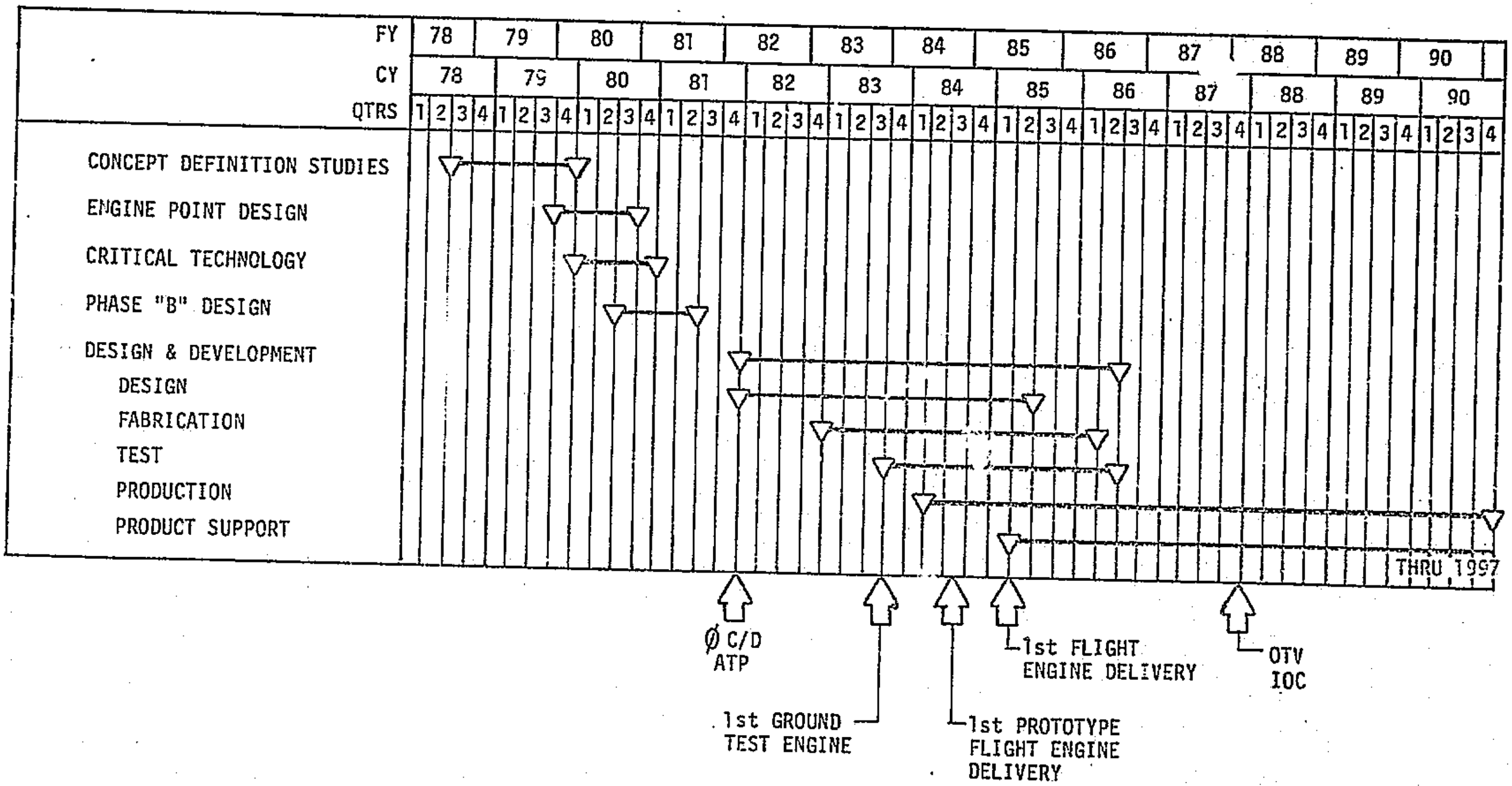


Figure 31. Overall OTV Engine Schedule

The engine DDT&E schedule is shown on Figure 32 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 third 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. Risk analysis and schedule impacts are planned in the Phase A extension to this contract.

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 such as, the OOS, Space Tug and OMS, that were updated to be compatible with the OTV engine concept.

The OTV engine turnaround cycle consists of safe and purge, maintenance and launch operations. As shown on Figure 33, the longest portion of the cycle is required for turnaround maintenance which averages 24 hours. During this 24-hour period routine maintenance is performed to determine engine condition. The routine maintenance actions require 11.5 hours. The

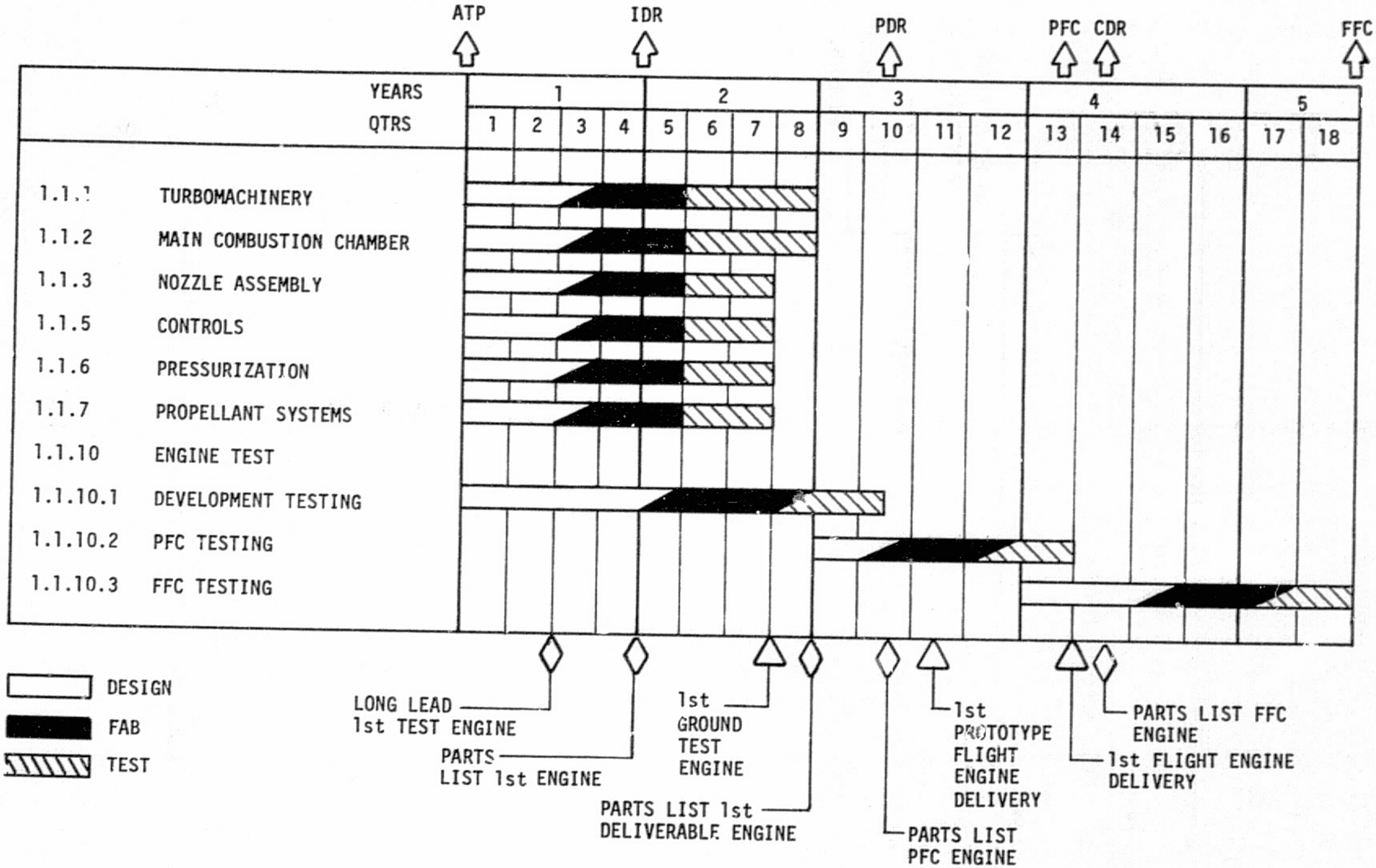


Figure 32. Advanced Expander Cycle Engine DDT&E Schedule

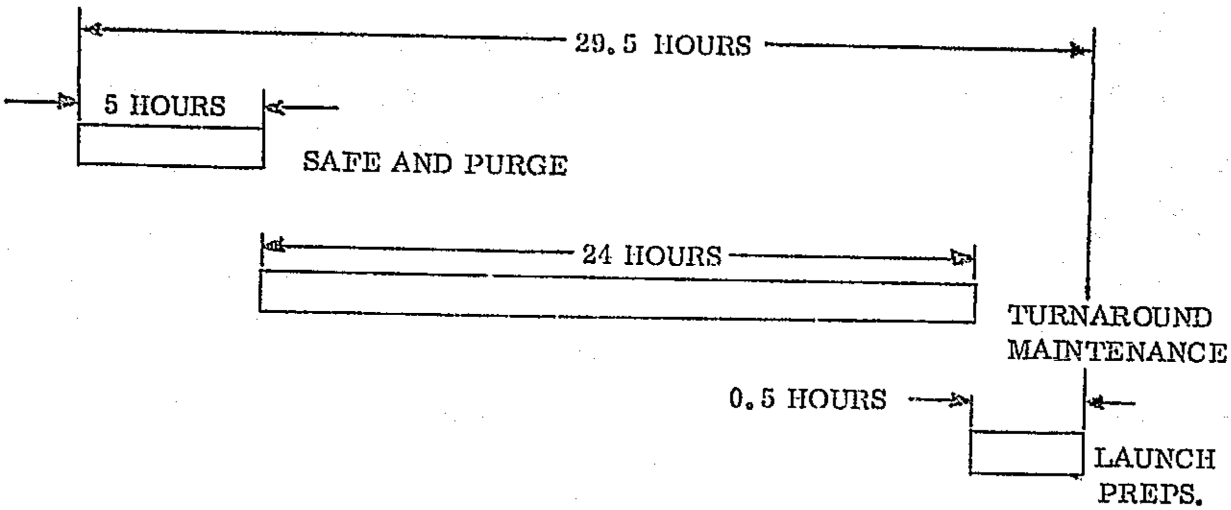


Figure 33. OTV Engine Turnaround Cycle Time

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.

The engine maintenance plan is summarized on Figure 34. 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. The maintenance tasks start when the OTV is removed from the shuttle payload bay and placed in the safe and purge area either after a mission or after a mission cancellation.

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 corrective maintenance.

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

Routine maintenance after every mission consists of an external inspection of engine hardware, including taking photographs of the injector face, automatic checkout and leak checks. Life evaluation checks are conducted after every fifth mission. Routine maintenance data, along with flight

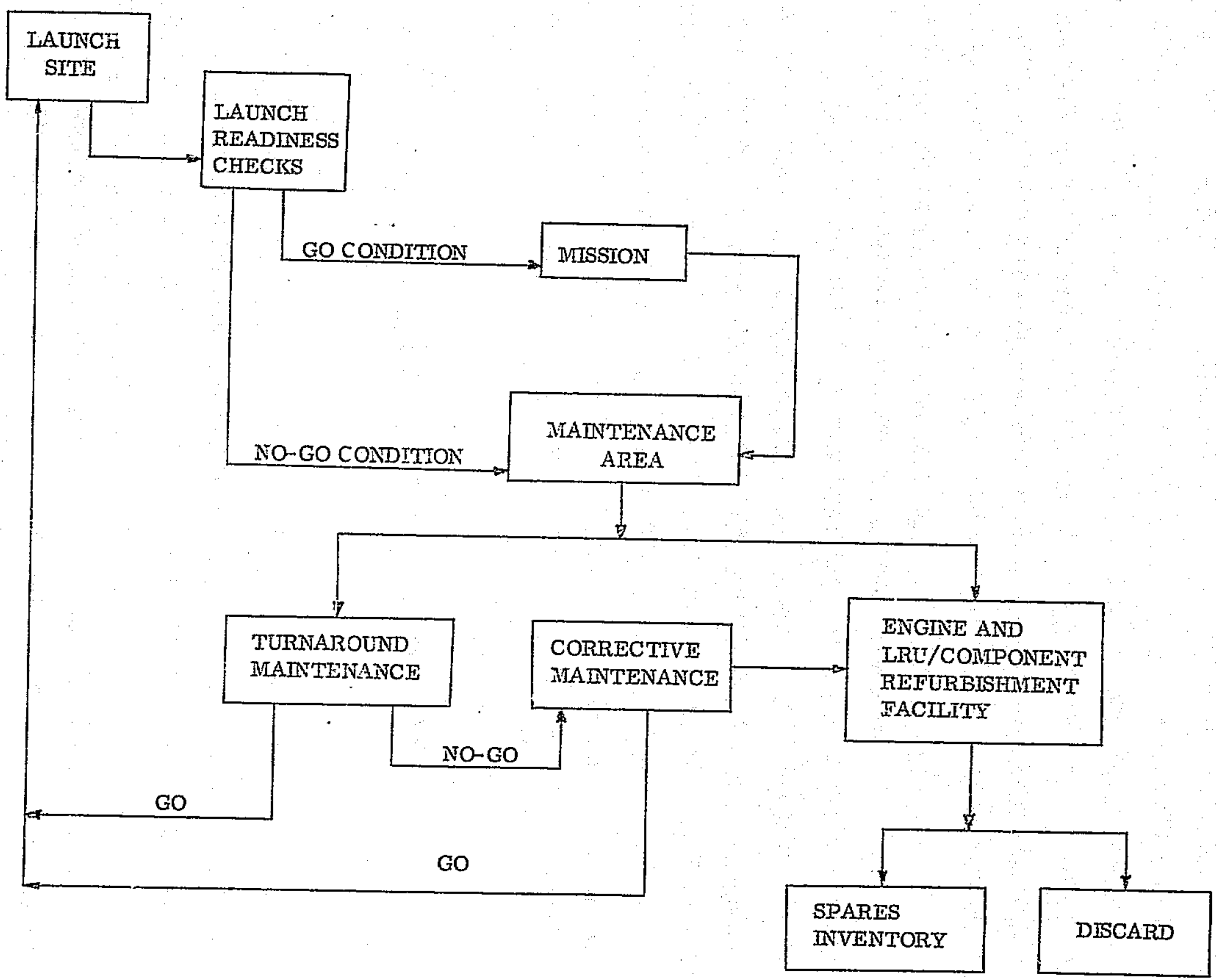


Figure 34. Engine Maintenance Plan

data, will be analyzed to detect discrepancies. The effort required to correct the discrepancies takes place during the corrective maintenance time span shown on the figure. 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.

The refurbishment activity takes place outside of the turnaround maintenance cycle and with few exceptions will be accomplished at depot level.

X. TASK VII: COST ESTIMATE

The primary objective of this task was to provide DDT&E, Production and Operations cost estimates per the WBS. This was accomplished at the recommended 10K lbf engine thrust level. Parametric cost data was also established for each of the engine candidates and reported in Volume III. Volume III also presents the costing approach, methodology and rationale and the cost estimates spread by yearly increments. The data summarized in this section is for the AMOTV. Engine production and operations costs for the APOTV are also presented in Volume III.

The DDT&E cost estimates for the various engine cycle candidates are compared at the third WBS level on Table XI. The costs are shown in millions of 1979 dollars and do not include the contractor's fee. The consumables are assumed to be government furnished propellants and, at the direction NASA, are shown in millions of pounds. The costs shown in all cases are for a 10K lb thrust engine. The expander cycle engine DDT&E cost is less primarily because of the elimination of a critical combustion device (preburner or gas generator). This affects costs of other items because of the component integration development that is also necessary. All costs are based upon a success oriented program. The affect of development risk upon the costs will be assessed in the Phase A extension to this contract.

The costs associated with the engine production necessary to support the AMOTV nominal mission model shown on Table 34 of NASA TMX-73394 are compared for the various cycle candidates on Table XII. It was estimated that 40 engines (20 sets) are required to support th's mission model. In addition 4 engines (2 sets) are required for the first two OTV's for a total of 44 engines. The baseline thrust level used in computing these costs is 10,000 lbs per engine or 20K lbf total. The primary difference in the production costs of the cycle candidates is the average cost of an engine. Except for the initial spares, other costs are the same for all candidates. The initial

TABLE XI

DDT&E WBS COSTS INDICATE
EXPANDER CYCLE SAVINGS

MILLIONS OF DOLLARS

		STAGED COMBUSTION	EXPANDER	GAS GENERATOR
1.1.1	TURBOMACHINERY	30.2	30.2	30.2
1.1.2	MAIN COMBUSTION CHAMBER	28.8	22.4	22.4
1.1.3	PREBURNER/GAS GENERATOR	23.1	0.0	18.5
1.1.4	NOZZLE ASSEMBLY	5.6	5.6	5.6
1.1.5	CONTROLS	20.2	13.6	20.2
1.1.6	PRESSURIZATION	10.2	10.2	10.2
1.1.7	PROPELLANT SYSTEMS	8.2	6.6	7.4
1.1.8	INITIAL TOOLING	15.7	12.6	14.9
1.1.9	GROUND SUPPORT EQUIPMENT	12.7	12.7	12.7
1.1.10	TEST	46.6	39.6	44.3
1.1.11	SYSTEM ENGINEERING	13.1	11.8	11.8
1.1.12	PROJECT MANAGEMENT	19.4	17.5	18.4
1.1.13	FACILITIES	12.9	11.6	12.3
1.1.14	CONSUMABLES (IN MILLIONS OF LBS.)	(27.5)	(21.7)	(25.8)
	TOTAL COST	246.7	194.4	228.9

ELIMINATION OF HOT GAS TURBINE DRIVE LOWERS DDT&E COST

TABLE XII

AMOTV PRODUCTION WBS COSTS COMPARED

- 2 ENGINE VEHICLE
- 44 ENGINE PRODUCTION RUN

		MILLIONS OF DOLLARS		
		STAGED COMBUSTION	EXPANDER	GAS GENERATOR
1.2.1	MAIN ENGINES	101.4	76.8	82.9
1.2.2	INITIAL SPARES	1.8	1.3	1.4
1.2.3	FACILITY MAINTENANCE	3.5	3.5	3.5
1.2.4	SUSTAINING ENGINEERING	3.7	3.7	3.7
1.2.5	PROJECT MANAGEMENT	6.0	6.0	6.0
1.2.6	CONSUMABLES (IN MILLIONS OF LBS.)	(1.5)	(1.5)	(1.5)
	TOTAL COST	116.4	91.3	97.5

FEWER COMPONENTS LOWER EXPANDER CYCLE PRODUCTION COSTS spares cost was computed as 50% of the theoretical first unit cost which was a guideline provided by NASA. A 90% learning curve was used to project the costs. A production rate of one engine subassembly per month was assumed.

The operations costs to support the AMOTV mission model are shown on Table XIII. The operations costs are not expected to vary as a function of the engine cycle type. It was estimated that six engines (3 sets) per year would be overhauled. Costs are shown for one year only and are assumed to be spread evenly over 10 years. The follow-on spares cost was computed as 4.5 times the theoretical first unit cost of an expander cycle engine and divided by 10 to obtain an average cost for one year. The 4.5 factor was a guideline provided by NASA. The engine thrust level is not expected to affect the operations cost.

TABLE XIII

AMOTV OPERATIONS WBS COSTS

- 44 ENGINE FLEET
- SERVICE 6 ENGINES/YEAR

1.3.1	INPLANT SUPPORT	0.87
1.3.2	FIELD SUPPORT	1.14
1.3.3	MAJOR ENGINE OVERHAUL	1.77
1.3.4	FACILITY MAINTENANCE	0.15
1.3.5	FOLLOW-ON SPARES	1.20
1.3.6	PROJECT MANAGEMENT	0.20
1.3.7	CONSUMABLES (IN MILLIONS OF LBS)	(0.20)

TOTAL PROGRAM 5.33 M x 10 YEARS = 53.3 M/YEAR

XI. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The conclusions which were derived from the results of this study are discussed herein and shown on Figure 35. 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 35. 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 show that the expander cycle engine costs are lower than either the staged combustion or the gas generator engine cycles. 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 36. 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 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.

C-2

- 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 36. Recommendations

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 philosophy 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.